The Lunar Module - Reference

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The Lunar Module Descent Stage

Descent Stage

The descent stage is the unmanned portion of the LM; it represents approximately two-thirds of the weight of the LM at the earth-launch phase. This is because the descent engine is larger than the ascent engine and it requires a much larger propellant load. Additionally, its larger proportion of weight results from necessity of the descent stage to:

Support the entire ascent stage.

Provide for attachment of the landing gear.

Support the complete LM in the SLA.

Provide structure to support the scientific and communications equipment to be used on the lunar surface.

Act as a launching platform of the ascent stage.

The main structure of the descent stage consists of two pairs of parallel beams arranged in a cruciform, with a deck on the upper and lower surfaces, approximately 65 inches apart. The ends of the beams, aproximately 81 inches from the center, are closed off by aluminum beams to provide five equally sized compartments: a center compartment, one forward and one aft of the center compartment, and one right and one left of the center compartment. A four-legged truss (outrigger) at the end of each pair of beams serves as a support for the LM in the SLA and as the attachment point for the upper end of the landing gear primary strut. Two of the four interstage fittings for attachment of the ascent stage are mounted on the forward compartment beams. The other two fittings are on the aft beam of the side compartments. The five compartments formed by the main beams house the Main Propulsion Subsystem components. The center compartment houses the descent engine, which is supported by truss members and an engine gimbal ring. Descent engine fuel and oxidizer tanks are in the remaining compartments.

The Descent Stage

Struts between the ends of all main beams form triangular bays, or quadrants, to give the descent stage its octagon shape. The quadrants are designated 1 through 4, beginning at the left of the forward compartment and continuing counterclockwise (as viewed from the top) around the center. The quadrants house components from the various subsystems. In addition, the modularized equipment stowage assembly (MESA), in quad No. 4 and a pallet assembly is stowed in quad No. 3.

The MESA consists of television equipment for obtaining and stowing lunar samples, and PLSS components to be used by the astronauts during the lunar stay.

The MESA

The quad No. 3 pallet assembly contains two pallets, a Lunar Roving Vehicle (LRV) pallet, and a pallet holding the Lunar Retro-Ranging Reflector. The LRV pallet contains a lunar geological exploration tool carrier, a lunar dust brush, a gnomen, a recording penetrometer, tongs, a trenching tool, collection bags, and other items needed during lunar exploration.

Four plume deflectors, which deep the plumes of the downward firing RCS thrusters from impinging upon the descent stage, are truss mounted to the descent stage.

Thermal and Micrometeoroid Shield

The entire descent stage structure is enveloped in a thermal and micrometeoroid shield similar to that used on the ascent stage. Because the top deck and side panels of the descent stage are subjected to engine exhaust, these areas are extensively protected with a nickel inconel mesh sandwich outboard of the mylar and H-film blankets. A teflon-coated titanium blast shield that deflects the ascent engine exhaust out of and away from the descent engine compartment is secured to the upper side of the compartment, below the thermal blanket. Layers of H-film, joined to the blast deflector, act as an ablative membrane which protects the descent stage from ascent engine exhaust gases that are deflected outward, between the stages, during lift-off from the lunar surface. The engine compartment and the bottom of the descent stage are subjected to temperatures in excess of 1800 degrees F when the descent engine is fired. A special base heat shield protects the descent stage structure and internal components. It consists of a titanium shield attached to descent stage structure. The heat shield supports a thermal blanket on each of its sides. The thermal blanket that faces the enine nozzle consists of multiple layers of nickel foil and glass wool and an outer layer of H-film. This blanket acts as a protective membrane to withstand engine exhaust gas back pressure at lunar touchdown and prevent heat, absorbed by the lunar surface during LM landing, from radiating back into the descent stage. Twenty-five layers of H-film make up the blanket on the other side of the titanium. A flange-like ring of columbium backed with a fibrous (Min-K) insulation is attached directly to the engine nozzle extension and joined to the base heat shield by an annular bellows of 25-layer H-film. This bellows arrangement permits descent engine gimbaling, but prevents engine heat from leaking into the engine compartment.

Landing Gear

The landing gear provides the impact attenuation required to land the LM on the lunar surface, prevents tip over of the LM on a lunar surface with a 6 degree slope having 24-inch depressions or protuberances, and supports the LM during lunar stay and lunar launch. Landing impact is attenuated to load levels that preserves the LM structural integrity. At earth launch, the landing gear is retracted to reduce the overall size. It remains retracted until the docked CSM and LM attain lunar orbit and the astronauts have transferred to the LM. Before the LM is separated from the CSM, the Commander in the LM operates the landing gear deployment switch to extend the gear. At this time landing gear uplocks are explosively released, allowing springs in deployment mechanisms to extend the gear. Once extended, the landing hear is locked in place by down lock mechanisms.

The Landing Gear Assembly

The cantilever landing gear consists of four assemblies, each connected to an outrigger that extends from the ends of the structural parallel beams. The landing gear assemblies extend from the front, rear, and both sides of the descent stage. Each assembly consists of struts, trusses, a footpad, lock and deployment mechanisms, and, on all but the forward gear assembly, a lunar surface sensing probe. A ladder is affixed to the forward gear assembly.

The landing gear can withstand: (1) a 10-foot/second vertical velocity of the LM when the horizonal velocity is zero feet/second, (2) a 7-foot/second vertical velocity with a horizontal velocity not exceeding 4 feet/second, and (3) a vehicle attitude within 6 degrees of the local horizontal when the rate of attitude change is 2 degrees/second or less.

Primary Strut

The upper end of the primary strut is attached to the outboard end of the outrigger; the lower end has a ball joint for the footpad. The strut is of the piston-cylinder type; it absorbs the compression load of the lunar landing and supports the LM on the lunar surface. Compression load are attenuated by a crushable aluminum-honeycomb cartridge in each strut. Maximum compression length of the primary strut is 32 inches. The aluminum honeycomb has the shock-absorbing capability of accepting one lunar landing. This may include one or two bounces of the LM, but after the full weight of the LM is on the gear, the shock-absorbing medium is expended. Use of compressible honeycomb cartridges eliminated the need for thick-walled, heavy weight, pnuedraulic-type struts.

Landing Gear Primary Strut

The footpad, attached to the strut by a ball socket fitting, is aluminum-honeycomb; its diameter is 37 inches. This large diameter ensures minimal penetration of the LM on low load-bearing-strength lunar surface. During earth launch, four restraining straps hold the pads in a fixed position on the strut. The straps shear or bend on pad contact with the lunar surface, permitting the pad to conform to surface irregularities.

Lunar Surface Sensing Probe

The lunar surface sensing probe attached to each landing gear footpad, except the forward one, is an electromechanical device. The probes are retained in the stowed position, against the primary strut, until landing gear deployment. During deployment, mechanical interlocks are released permitting spring energy to extend the probes so that the probe head is approximately 5 feet below the footpad. When any probe touches the lunar surface, pressure on the probe head will complete the circuit that advises the astronauts to shut down the descent engine. This shutdown point which determines LM velocity at impact, is a tradeoff between landing gear design weight and the thermal and thrust reactions caused by the descent engine operating near the lunar surface. Each probe has indicator plates attached to it, which, when aligned, indicate that the probes are fully extended.

Secondary Struts

Each landing gear assembly has two secondary struts. The outboard end of each strut is attached to the primary strut; the inboard ends are attached to a deployment truss assembly. Each strut is a piston-cylinder-type device that contains compressible aluminum honeycomb capable of absorbing compression and tension loads. The design and the location of the secondary struts in relation to the primary strut enables the LM to land on an un-symmetrical surface or to land when the LM is moving laterally over the lunar surface.

Uplock Assembly

One uplock assembly is attached to each landing gear assembly. It consists of a fixed link (strap) and two end detonator cartridges in a single case. The fixed link, attached between the primary strut and the descent stage structure, holds the landing gear in its retracted position. When the Commander operates the landing gear deployment switch, it activates an electrical circuit which explosively severs the fixed link to permit the deployment mechanism to extend the landing gear. When detonated, either end cartridge has sufficient energy to sever the fixed link.

Deployment and Down lock Mechanism

The deployment portion of the deployment and down lock mechanism consists of a truss assembly, two clock-type deployment springs, and connecting linkage. The truss, connecting the secondary struts and descent stage structure, comprises two side frame assemblies separated by a crossmember. The deployment springs are attached, indirectly, to the side frame assemblies through connecting linkages. The down lock portion of the mechanism consists of a spring-loaded lock and a cam follower. The follower rides on a cam attached to the deployment portion of the mechanism. When the fixed link of the uplock assembly is severed, the deployment springs pull the connecting linkage and, indirectly, the deployment truss. This action drives the landing gear from the stowed to the fully deployed position. At full gear deployment, the cam follower reaches a point that permits the spring-loaded lock to snap over a roller on the truss assembly. The lock cannot be opened. A landing gear deployment talkback advises the astronauts that the landing gear is fully deployed.

The Down lock Mechanism

Ladder

The ladder affixed to the primary strut of the forward landing gear assembly has nine rungs between two railings. The rungs are spaced nine inches apart; the railings have approximately 20 inches between centers. The top of the ladder is approximately 18 inches below the forward end of the platform on the outrigger; the ladder extends down to within 30 inches of the footpad. This allows the primary strut to telescope when the LM impacts the lunar surface.

Platform

An external platform, approximately 32 inches wide and 45 inches long is mounted over the forward landing gear outrigger. The platform is just below the forward hatch. The upper surface is corrugated to facilitate hand and foot holds. The platform, in conjunction with the ladder below it provides the astronauts with a means of access between the vehicle and the lunar surface and between the LM interior and free space for EVA.

Interfaces

At earth launch, the LM is housed within the SLA, which has an upper and a lower section. The upper section has deployable panels, which are jettisoned; the lower section has fixed panels. The upper panels are deployed and jettisoned when the CSM is separated from the SLA. During this separation phase, an explosive charge separates an umbilical line that connects the LM, SLA, and launch umbilical tower. Before earth launch, this umbilical enables monitoring, purging, and control of the LM environment.

After transposition, the CSM docks with the LM. A ring at the top of the ascent stage docking tunnel provides a structural interface for joining the CSM to the LM. The ring is compatible with a clamping mechanism in the CSM docking ring. A drogue, which mates with the CSM docking probe, is installed in the docking tunnel, just below the ring. The probe provides initial vehicle soft docking and attenuates impact imposed by contact of the CSM and LM. After the CSM probe and drogue have joined, latches around the periphery of the CSM docking ring engage to effect full structural continuity and pressure-tight seal between the vehicles. After docking has been completed, the astronauts connect electrical umbilical's in the CSM and the LM. These umbilical's provide electrical power to the LM, for separation from the SLA.

Crew Personal Equipment

Crew personal equipment includes a variety of mission-oriented equipment required for life support and astronaut safety and accessories related to successful completion of the mission.

These equipment's range from astronaut space suits and docking aids to personal items stored throughout the cabin. The Modularized Equipment Stowage Assembly (MESA), Apollo lunar scientific experiments payload (ALSEP), Quad 3 pallet assembly, and the Lunar Roving Vehicle (LRV) are stored in the descent stage.

This equipment is used for sample and data collecting and scientific experimenting. The resultant data will be used to derive information on the atmosphere and distance between earth and moon.

The portable life support system (PLSS) interfaces with the Environmental Control Subsystem (ECS) for refills of oxygen and water. The pressure garment assembly (PGA) interfaces with the ECS for conditioned oxygen, through oxygen umbilical's, and with the Communications and Instrumentation Subsystems for communications and bioinstrumentation, through the electrical umbilical.

EXTRAVEHICULAR MOBILITY UNIT

The extravehicular mobility unit (EMU) provides life support in a pressurized or unpressurized cabin, and up to 7 hours of extravehicular life support (depending on astronaut's metabolic rate).

In its extravehicular configuration, the EMU is a closed-circuit pressure vessel that envelops the astronaut. The environment inside the pressure vessel consists of 100% oxygen at a nominal pressure of 3.75 psia. The oxygen is provided at a flow rate of 6 cfm. The extravehicular life support equipment configuration includes the following:

Liquid cooling garment (LCG)

Pressure garment assembly (PGA)

Integrated thermal micrometeoroid garment (ITMG)

Portable life support system (PLSS)

Oxygen purge system (OPS)

Communications carrier

EMU waste management system

EMU maintenance kit

PLSS remote control unit

Lunar extravehicular visor assembly (LEVA)

Biomedical belt

LIQUID COOLING GARMENT

The liquid cooling garment (LCG) is worn by the astronauts while in the LM and during all extravehicular activity. It cools the astronaut's body during extravehicular activity by absorbing body heat and transferring excessive heat to the sublimator in the PLSS. The LCG is a one-piece, long-sleeved, integrated-stocking undergarment of netting material. It consists of an inner liner of Beta cloth, to facilitate donning, and an outer layer of Beta cloth into which a network of Tygon tubing is woven. The tubing does not pass through the stocking area. A double connector for incoming and outgoing water is located on the front of the garment. Cooled water, supplied from the PLSS, is pumped through the tubing. Pockets for bioinstrumentation signal conditioners are located around the waist. A zipper that runs up the front is used for donning and doffing the LCG; an opening at the crotch is used for urinating. Dosimeter pockets and snaps for attaching a biomedical belt are part of the LCG.

PRESSURE GARMENT ASSEMBLY

The pressure garment assembly (PGA) is the basic pressure vessel of the EMU. It provides a mobile life-support chamber if cabin pressure is lost due to leaks or puncture of the vehicle. The PGA consists of a helmet, torso and limb suit, intravehicular gloves, and various controls and instrumentation to provide the crewman with a controlled environment. The PGA is designed to be worn for 115 hours, in an emergency, at a regulated pressure of 3.75+ 0.25 psig, in conjunction with the LCG.

The torso and limb suit is a flexible pressure garment that encompasses the entire body, except the head and hands. It has four gas connectors, a PGA multiple water receptacle, a PGA electrical connector, and a PGA urine transfer connector for the PLSS/PGA and ECS/PGA interface. The PGA connectors have positive locking devices and can be connected and disconnected without assistance. The gas connectors comprise an oxygen inlet and outlet connector, on each side of the suit front torso. Each oxygen inlet connector has an integral ventilation diverter valve. The PGA multiple water receptacle, mounted on the suit torso, serves as the interface between the LCG multiple water connector and PLSS multiple water connector. A protective external cover provides PGA pressure integrity when the LCG multiple water connector is removed from the PGA water receptacle. The PGA electrical connector, provides a communications, instrumentation, and power interface to the PGA. The PGA urine transfer connector on the suit right leg is used to transfer urine from the urine collection transfer assembly (UCTA) to the waste management system.

The urine transfer connector, permits dumping the urine collection bag without depressurizing the PGA. A pressure relief valve on the right leg thigh vents the suit in the event of over pressurization. If the valve does not open, it can be manually over-ridden. A pressure gauge on the left sleeve indicates suit pressure.

The helmet is a Lexan (polycarbonate) shell with a bubble-type visor, a vent-pad assembly, and a helmet attaching ring. The vent-pad assembly permits a constant flow of oxygen over the inner front surface of the helmet. The astronaut can turn his head within the helmet neck-ring area. The helmet does not turn independently of the torso and limb suit. The helmet has provisions on each side for mounting a lunar extravehicular visor assembly (LEVA). When the LM is unoccupied, the helmet protective bags are stowed on the cabin floor at the crew flight stations. Each bag has a hollow-shell plastic base with a circular channel for the helmet and the LEVA, two recessed holes for glove connector rings, and a slot for the EMU maintenance kit. The bag is made of Beta cloth, with a circumferential zipper; it folds toward the plastic base when empty.

The intravehicular gloves are worn during operations in the LM cabin. The gloves are secured to the wrist rings of the torso and limb suit with a slide lock; they rotate by means of a ball-bearing race. Freedom of rotation, along with convoluted bladders at the wrists and adjustable anti-ballooning restraints on the knuckle areas, permits manual operations while wearing the gloves.

All PGA controls are accessible to the crewman during intravehicular and extravehicular operations. The PGA controls comprise two ventilation diverter valves, a pressure relief valve with manual override, and a manual purge valve. For intravehicular operations, the ventilation diverter valves are open, dividing the PGA inlet oxygen flow equally between the torso and helmet of the PGA. During extravehicular operation, the ventilation diverter valves are closed and the entire oxygen flow enters the helmet. The pressure relief valve accommodates flow from a failed- open primary oxygen pressure regulator. If the pressure relief valve fails open, it may be manually closed. The purge valve interfaces with the PGA through the PGA oxygen outlet connector. Manual operation of this valve initiates an 8 pound/hour purge flow, providing CO2 washout and minimum cooling during contingency or emergency operations.

A pressure transducer on the right cuff indicates pressure within the PGA. Biomedical instrumentation comprises an EKG (heart) sensor, ZPN (respiration rate) sensor, dc-to-dc converter, and wiring harness. A personal radiation dosimeter (active) is attached to the integrated thermal micrometeoroid garment for continuous accumulative radiation readout. A chronograph wristwatch (elapsed-time indicator) is readily accessible to the crewman for monitoring.

COMMUNICATIONS CARRIER

The communications carrier (cap) is a polyurethane-foam headpiece with two independent earphones and microphones, which are connected to the suit 21-pin communications electrical connector. The communications carrier is worn with or without the helmet during intravehicular operations. It is worn with the helmet during extravehicular operations.

INTEGRATED THERMAL MICROMETEOROID GARMENT

The ITMG, worn over the PGA, protects the astronaut from harmful radiation, heat transfer, and micrometeoroid activity. It is a one-piece, form-fitting, multilayered garment that is laced over the PGA and remains with it. The LEVA, gloves, and boots are donned separately. From the outer layer in, the ITMG is made of a protective cover, a micrometeoroid-shielding layer, a thermal-barrier blanket (multiple layers of aluminized mylar), and a protective liner. A zipper on the ITMG permits connecting or disconnecting umbilical hoses. For extravehicular activity, the PGA gloves are replaced with the extravehicular gloves. The extravehicular gloves are made of the same material as the ITMG to permit handling intensely hot or cold objects outside the cabin and for protection against lunar temperatures. The extravehicular boots (lunar overshoes) are worn over the PGA boots for extravehicular activity. They are made of the same material as the ITMG. The soles have additional insulation for protection against intense temperatures.

The LEVA, which fits over the clamps around the base of the helmet; provides added protection against solar heat, space particles, solar glare, ultraviolet rays, and accidental damage to the helmet. The LEVA is comprised of a plastic shell, cover, hinge assemblies, three eyeshades, and two visors (protective and sun visors). The protective visor provides impact, infrared and ultra-violet ray protection. The sun visor has a gold coating which provides protection against light and reduces heat gain within the helmet. The eyeshades, two located on each side and one in the center, reduces low angle solar glare by preventing light penetration at the sides and overhead viewing area. When the LM is occupied, the LEVA's are stowed in helmet stowage bags and secured on the ascent engine cover.

PORTABLE LIFE SUPPORT SYSTEM

The PLSS is a self-contained, self-powered, rechargeable environmental control system. In the extravehicular configuration of the EMU, the PLSS is worn on the astronaut's back. The PLSS supplies pressurized oxygen to the PGA, cleans and cools the expired gas, circulates cool liquid in the LOG through the liquid transport loop, transmits astronaut biomedical data, and functions as a duel VHF transceiver for communication.

The PLSS has a contoured fiberglass shell to fit the back, and a thermal micrometeoroid protective cover. It has three control valves, and, on a separate remote control unit, two control switches, a volume control, and a five- position switch for the dual VHF transceiver. The remote control unit is set on the chest.

The PLSS attaches to the astronaut's back, over the ITMG; it is connected by a shoulder harness assembly. When not in use, it is stowed on the floor or in the left-hand midsection. To don the PLSS, it is first hooked to the overhead attachments in the left-hand midsection ceiling. The astronaut backs against the pack, makes PGA and harness connections, and unhooks the PLSS straps from the overhead attachment.

The PLSS can operate for 7 hours, depending upon the astronauts metabolic rate, before oxygen and feedwater must be replenished and the battery replaced. The basic systems and loops of the PLSS are primary oxygen subsystem, oxygen ventilation loop, feedwater loop, liquid transport loop, and electrical system.

The space suit communicator (SSC) in the PLSS provides primary and secondary duplex voice communication and physiological and environmental telemetry. All EMU data and voice must be relayed through the LM and CM and transmitted to MSFN via S-band. The VHF antenna is permanently mounted on the oxygen purge system (OPS). Two tone generators in the SSC generate audible 3- and 1.5-kHz warning tones to the communications cap receivers. The generators are automatically turned on by high oxygen flow, low vent flow, or low PGA pressure. Both tones are readily distinguishable.

PLSS REMOTE CONTROL UNIT

The PLSS remote control unit is a chest-mounted instrumentation and control unit. It has a fan switch, pump switch, SSC mode selector switch, volume control, PLSS oxygen quantity indicator, five status indicators, and an interface for the OPS actuator.

OXYGEN PURGE SYSTEM

The OPS is a self-contained, independently powered, high-pressure, non-rechargeable emergency oxygen system that provides 30 minutes of regulated purge flow. The OPS consists of two interconnected spherical high-pressure oxygen bottles, an automatic temperature control module, an oxygen pressure regulator assembly, a battery, an oxygen connector, and checkout instrumentation. In the normal extravehicular configuration, the OPS is mounted on top of the PLSS and is used with PLSS systems during emergency operations. In the contingency extravehicular configuration, the OPS is attached to the PGA front lower torso and functions independently of the PLSS. The OPS has no communications capability, but provides a hard mount for the SSC VHF antenna. Two OPS's are stowed in the LM.

UMBILICAL ASSEMBLY

The umbilical assembly consists of hoses and connectors for securing the PGA to the ECS, Communications Subsystem (CS), and Instrumentation Subsystem (IS). Separate oxygen and electrical umbilical’s connect to each astronaut.

The oxygen umbilical consists of Flourel hoses (1.25-inch inside diameter) with wire reinforcement. The connectors are of the quick-disconnect type, with a 1.24-inch, 90 degree elbow at the PGA end. Each assembly is made up of two hoses and a dual-passage connector at the ECS end and two separate hoses (supply and exhaust) at the PGA end. When not connected to the PGA, the ECS connector end remains attached and the hoses stowed.

The electrical umbilical carries voice communications and biomedical data, and electrical power for warning-tone impulses.

CREW LIFE SUPPORT

The crew life support equipment includes food and water, a waste management system, personal hygiene items, and pills for in-flight emergencies. A potable-water unit and food packages contain sufficient life- sustaining supply for completion of the LM mission.

CREW WATER SYSTEM

The water dispenser assembly consists of a mounting bracket, a coiled hose, and a trigger-actuated water dispenser. The hose and dispenser extend approximately 72 inches to dispense water from the ECS water feed control assembly. The ECS water feed control valve is opened to permit water flow. The dispenser assembly supplies water at +50 to +90 degrees for drinking or food preparation and fire extinguishing. The water for drinking and food preparation is filtered through a bacteria filter. The water dispenser is inserted directly into the mouth for drinking. Pressing the trigger-type control supplies a thin stream of water for drinking and food preparation. For firefighting, a valve on the dispenser is opened. The valve provides a greater volume of water than that required for drinking and food preparation.

FOOD PREPARATION AND CONSUMPTION

The astronaut's food supply (approximately 3,500 calories per man per day) includes liquids and solids with adequate nutritional value and low waste content. Food packages are stowed in the LM midsection, on the shelf above PLSS No. 1 and the right-hand stowage compartment and the MESA.

The food is vacuum packed in plastic bags that have one-way poppet valves into which the water dispenser can be inserted. Another valve allows food passage for eating. The food bags are packaged in aluminum-foil-backed plastic bags for stowage and are color coded: red (breakfast), white (lunch), and blue (snacks).

Food preparation involves reconstituting the food with water. The food bag poppet-valve cover is cut with scissors and pushed over the water dispenser nozzle after its protective cover is removed. Pressing the water dispenser trigger releases water. The desired consistency of the food determines the quantity of water added. After withdrawing the water dispenser nozzle, the protective cover is replaced and the dispenser returned to its stowage position. The food bag is kneaded for approximately 3 minutes, after which the food is considered reconstituted. After cutting off the neck of the food bag, food can be squeezed into the mouth through the food-passage valve. A germicide tablet, attached to the outside of the food bag, is inserted into the bag after food consumption, to prevent fermentation and gas formation. The bag is rolled to its smallest size, banded, and placed in the waste disposal compartment.

EMU WASTE MANAGEMENT SYSTEM The EMU waste management system provides for the disposal of body waste through use of a fecal containment system and a urine collection and transfer assembly, and for neutralizing odors. Personal hygiene items are stowed in the left-hand stowage compartment.

Waste fluids are transferred to a waste fluid collector assembly by a controlled difference in pressures between the PGA and cabin (ambient). The primary waste fluid collector consists of a long transfer hose, control valve, short transfer hose, and a 8,900 cc multilaminate bag. The long transfer hose is stowed on a connector plate when not in use. To empty his in-suit urine container, the astronaut attaches the hose to the PGA quick-disconnect, which has a visual flow indicator. Rotating the handle of the spring-loaded waste control valve controls passage of urine to the assembly. The 8,900-cc bag is in the PLSS LiOH storage unit, the short transfer hose is connected between the waste control valve and the bag.

With cabin pressure normal (4.8 psia), the long transfer hose is removed from the connector stowage plate and attached to the PGA male disconnect. The PGA is over pressurized by 0.8 +/- 0.2 psia and the waste control valve is opened. Urine flows from the PGA to the collector assembly at a rate of approximately 200 cc per minute. When bubbles appear in flow indicator, the valve indicator is released and allowed to close.

A secondary waste fluid collector system provides 900cc waste fluid containers, which attach directly to the PGA. Urine is transferred directly from the PGA, through the connectors, to the bags. These bags can then be emptied into the 8,900 cc collector assembly.

FECAL DEVICE

The fecal containment system consists of an outer fecal/emesis bag (one layer of Aclar) and a smaller inner bag. The inner bag has waxed tissue on its inner surface. Polyethylene-backed toilet tissue and a disinfectant package are stored in the inner bag.

To use, the astronaut removes the inner bag from the outer bag. After unfastening the PGA and removing undergarments, the waxed tissue is peeled off the bag's inner surface and the bag is placed securely on the buttocks. After use, the used toilet tissue is deposited in the used bag and the disinfectant package is pinched and broken inside the bag. The bag is then closed, kneaded, and inserted in the outer bag. The wax paper is removed from the adhesive on the fecal/emesis bag and the bag is sealed then placed in the waste disposal compartment.

PERSONAL HYGIENE ITEMS

Personal hygiene items consist of wet and dry cleaning cloths, chemically treated and sealed in plastic covers. The cloths measure 4 by 4 inches and are folded into 2-inch squares. They are stored in the food package container.

MEDICAL EQUIPMENT

The medical equipment consists of biomedical sensors, personal radiation dosimeters, and emergency medical equipment.

Biomedical sensors gather physiological data for telemetry. Impedance pneumographs continuously record heart beat (EKG) and respiration rate. Each assembly (one for each astronaut) has four electrodes which contain electrolyte paste; they are attached with tape to the astronaut's body.

Six personal radiation dosimeters are provided for each astronaut. They contain thermoluminescent powder, nuclear emulsions, and film that is sensitive to beta, gamma, and neutron radiation. They are placed on the forehead or right temple, chest, wrist, thigh, and ankle to detect radiation to eyes, bone marrow, and skin. Serious, perhaps critical, damage results if radiation dosage exceeds a predetermined level. For quick, easy reference each astronaut has a dosimeter mounted on his EMU.

The emergency medical equipment consists of a kit of six capsules: four are pain killers (Darvon) and two are pep pills (Dexedrine). The kit is attached to the interior of the flight data file, readily accessible to both astronauts.

CREW SUPPORT AND RESTRAINT EQUIPMENT

The crew support and restraint equipment includes armrests, handholds (grips), Velcro on the floor to interface with the boots, and a restraint assembly operated by a rope-and-pulley arrangement that secures the astronauts in an upright position under zero-g conditions.

The armrests, at each astronaut position, provide stability for operation of the thrust/ translation controller assembly and the attitude controller assembly, and restrain the astronaut laterally. They are adjustable (four positions) to accommodate the astronaut; they also have stowed (fully up) and docking (fully down) positions. The armrests, held in position by spring-loaded detents, can be moved from the stowed position by grasping them and applying downward force. Other positions are selected by pressing latch buttons on the armrest forward area. Shock attenuators are built into the armrests for protection against positive 9 forces (lunar landing). The maximum energy absorption of the armrest assembly is a 300-pound force, which will cause a 4-inch armrest deflection.

The handholds, at each astronaut station and at various locations around the cabin, provide support for the upper torso when activity involves turning, reaching, or bending; they attenuate movement in any direction. The forward panel handholds are single upright, peg-type, metal grips. They are fitted into the forward bulkhead, directly ahead of the astronauts, and can be grasped with the left or right hand.

The restraint assembly consists of cables, restraint rings, and a constant-force reel system. The cables attach to D- rings on the PGA sides, waist high. The constant-force reel provides a downward force of approximately 30 pounds, it is locked during landing or docking operations. When the constant-force reel is locked, the cables are free to reel in. A ratchet stop prevents paying out of the cables and thus provides zero-g restraint. During docking maneuvers, the Commander uses pin adjustments to enable him to use the crewmen optical alignment sight (COAS) at the overhead (docking) window.

DOCKING AIDS AND TUNNEL HARDWARE

Docking operations require special equipment and tunnel hardware to effect linkup of the LM with the CSM. Docking equipment includes the crewman's optical alignment sight (COAS) and a docking target. A drogue assembly, probe assembly; the CSM forward hatch, and hardware inside the LM tunnel enable completion of the docking maneuver.

The COAS provides the Commander with gross range cues and closing rate cues during the docking maneuver. The closing operation, from 150 feet to contact, is an ocular, kinesthetic coordination that requires control with minimal use of fuel and time. The COAS provides the Commander with a fixed line-of-sight attitude reference image, which appears to be the same distance away as the target.

The COAS is a collimating instrument. It weighs approximately 1 .5 pounds, is 8 inches long, and operates from a 28-volt d-c power source. The COAS consists of a lamp with an intensity control, a reticle, a barrel-shaped housing and mounting track, and a combiner and power receptacle. The reticle has vertical and horizontal 10 degree gradations in a 10 degree segment of the circular combiner glass, on an elevation scale (right side) of -10 to +31.5 degrees. The COAS is capped and secured to its mount above the left window (position No. 1).

To use the COAS, it is moved from position No. 1 to its mount on the overhead docking window frame (position No. 2) and the panel switch is set from OFF to OVHD. The intensity control is turned clockwise until the reticle appears on the combiner glass; it is adjusted for required brightness.

The docking target permits docking to be accomplished on a three-dimensional alignment basis. The target consists of an inner circle and a standoff cross of black with self-illuminating disks within an outer circumference of white. The target-base diameter is 17.68 inches. The standoff cross is centered 15 inches higher than the base and, as seen at the intercept, is parallel to the X-axis and perpendicular to the Y-axis and the Z-axis.

The drogue assembly consists of a conical structure mounted within the LM docking tunnel. It is secured at three points on the periphery of the tunnel, below the LM docking ring. The LM docking ring is part of the LM midsection outer structure, concentric with the X-axis. The drogue assembly can be removed from the CSM end or LM end of the tunnel.

Basically, the assembly is a three-section aluminum cone secured with mounting lugs to the LM tunnel ring structure. A lock and release mechanism on the probe, controls capture of the CSM probe at CSM-LM contact. Handles are provided to release the drogue from its tunnel mounts.

The tunnel contains hardware essential to final docking operations. This includes connectors for the electrical umbilical’s, docking latches, probe-mounting lugs, tunnel lights, and dead facing switches.

The probe assembly provides initial CSM-LM coupling and attenuates impact energy imposed by vehicle contact. The probe assembly may be folded for removal and for stowage within either end of the CSM transfer tunnel.

CREW MISCELLANEOUS EQUIPMENT

Miscellaneous equipment required for completion of crew operations consists of in-flight data with checklists, emergency tool B and window shades.

The in-flight data are provided in a container in the left-hand midsection. The Commander's checklist is stowed at his station. The in-flight data kit is stowed in a stowage compartment. The packages include the flight plan, experiments data and checklist, mission log and data book, systems data book and star charts.

Tool B (emergency wrench) is a modified Allen head L-wrench. It is 6.25 inches long and has a 4.250-inch drive shaft with a 7/16-inch drive. The wrench can apply a torque of 4,175 inch-pounds; it has a ball-lock device to lock the head of the drive shaft. The wrench is stowed on the right side stowage area inside the cabin. It is a contingency tool for use with the probe and drogue, and for opening the CM hatch from outside.

Window shades are used for the overhead (docking) window and forward windows. The window shade material is Aclar. The surface facing outside the cabin has a highly reflective metallic coating. The shade is secured at the bottom (rolled position). To cover the window, the shade is unrolled, flattened against the frame area and secured with snap fasteners.

MODULARIZED EQUIPMENT STOWAGE ASSEMBLY

The MESA pallet is located in quad 4 of the descent stage. The pallet is deployed by the extravehicular astronaut when the LM is on the lunar surface. It contains fresh PLSS batteries and LiOH cartridges, a TV camera and cable, still camera, tools for obtaining lunar geological samples, food, film, and containers in which to store the samples. It also has a folding table on which to place the sample return containers. Pallets are provided and are used to transfer the PLSS batteries and the cartridges to the cabin.

QUAD 3 PALLET ASSEMBLY

The quad 3 pallet assembly contains two pallets, a Lunar Roving Vehicle (LRV) pallet, and a palleet holding the Lunar Retro Ranging Reflector. The LRV pallet contains a lunar geological exploration tool carrier, a lunar dust brush, a gnomen, a recording penetrometer, tongs, a trenching tool, collection bags and other items needed during lunar exploration.

APOLLO LUNAR SURFACE EXPERIMENT PACKAGE

The Apollo Lunar Surface Experiment Package (ASLEP) consists of two packages of scientific instruments an supporting subsystems capable of transmitting scientific data to earth for one year. These data will be used to derive information regarding the composition and structure of the lunar body, its magnetic field, atmosphere and solar wind. Two packages are stowed in quad 2 of the descent stage. The packages are deployed on the lunar surface by the extravehicular astronaut.

ALSEP power is supplied by a radioisotope thermoelectric generator (RTG). Electrical energy is developed through thermoelectric action. The RTG provides a minimum of 16 volts at 56.2 watts to a power-conditioning unit. The radioisotopes fuel capsule emits nuclear radiation and approximately 1,500 thermal watts continuously. The surface temperature of the fuel capsule is approximately 1,400 degrees F. The capsule is stowed in a graphite cask, which is externally mounted on the descent stage. The capsule is removed from the cask and installed in the RTG.

LASER RANGING RETRO-REFLECTOR The laser ranging retro-reflector is a passive experiment with an array of optical reflectors that serve as targets for laser-pointing systems on earth. The experiment is designed to accurately measure the distance between earth and the moon.

Controls and Displays

The controls and displays enable astronauts to monitor and manage the LM subsystems and to control the LM manually during separation, docking, and landing.

In general, the controls and displays are in sub- systems groupings located in accordance with astronaut responsibilities. Certain controls and displays are duplicated to satisfy mission and/or safety requirements; a system of interlocks prevents simultaneous operation of these controls and displays that enable either astronaut to control the LM are centrally located; these are accessible from both flight stations. Controls that could be operated inadvertently are appropriately guarded.

Annunciator displays go on if malfunctions occur in the LM subsystems; at the same time, two flashing master alarm lights and an alarm tone (in the astronaut headsets) are activated. Digital and analog displays provide the astronauts with subsystem-status information such as gas and liquid quantities, pressures, temperatures, and voltages. There are 12 control and display panels. The main control and display panels (1 and 2) are canted and centered between the flight stations. Panels 3 and 4 are below these panels, within convenient reach and scan of both astronauts. Panels 5, 8, and 11 are located for use by the Commander. Panels 6, 12, 14, and 16 are located for use by the LM pilot.

Panel 1, directly in view of the Commander, contains various controls and displays, including warning lights, digital counters, navigational instruments, engine thrust control switches, and engine, fuel, and altitude indicators. Panel 2, directly in view of the LM Pilot, contains caution lights, reaction control indicators, environmental control indicators, navigational instruments, and various other indicators and switches. Panel 3 contains controls and displays for radar, stabilization and control, heater control, an event timer, and lighting. Panel 4 contains a display and keyboard assembly. The display and keyboard provides a two-way communications link between the astronauts and the LM guidance computer. The panel contains indicator lights, pushbuttons, data displays, and toggle switches. In front of the Commander's and LM Pilot's stations, at waist height, are panels 5 and 6, respectively. Panel 5 contains lighting and mission timer controls, engine start and stop pushbuttons, and an X- translation pushbutton. Abort guidance controls are on panel 6.

At the left of the Commander's station is panel 8, which is canted up 15 degrees from the horizontal. This panel contains controls and displays for explosive devices and descent propulsion, and audio controls. The orbital rate display earth and lunar (ORDEAL) panel aft and on top of panel 8, is an electromechanical device that provides an alternative to the pitch display of the flight director attitude indicator on panels 1 and 2. When selected, the ORDEAL produces a flight director attitude indicator display of computer local vertical attitude indicator display of computed local vertical attitude during earth or lunar circular orbits. Panel 11, directly above panel 8, has five angled surfaces that contain circuit breakers. Each row of circuit breakers is canted 15 degrees to the line of sight, so that a white band around the circuit breakers is visible when the breakers are open.

At the right of the LM Pilot's station is panel 12, which is canted up 15 degrees from the horizontal. This panel contains audio, communications, and communications antenna controls and displays. Directly above panel 12 is panel 14. It is canted up 36.5 degrees from the horizontal and contains controls and displays for electrical power distribution and monitoring. Panel 16, directly above panel 14, has four angled surfaces that contain circuit breakers.

This makes a white band visible around the circuit breakers when the breakers are open.

At the right of each flight station is a pistol-grip control (attitude controller assembly), used to control LM attitude changes.

At the left of each flight station is a T-handle control (thrust/transtation controller assembly). This assembly, an integrated translation and thrust controller, is used to command vehicle translations by firing thrusters in the Reaction Control Subsystem, and to throttle the descent engine.

An alignment optical telescope is located between and above the flight stations; it is a manually operated, unity- power, periscope-type device. It is used to determine the position of the LM, using a catalog of stars stored in the LM guidance computer and celestial measurements made by the astronauts. A utility light control centered above the flight stations comprises two switches and two portable tight fixtures, one for the Commander and one for the LM Pilot.

Environmental Control Subsystem controls are grouped together on the aft bulkhead, behind the LM Pilot. These controls include an oxygen control module, a suit gas diverter, suit flow controls, suit circuit and canister controls, cabin/suit temperature controls, and water management controls. Cabin relief and dump valve controls are located on the forward and overhead hatch. The Environmental Control Subsystem controls enable the astronauts to maintain a habitable environment, decompress and repressurize the cabin, and regulate water flow for drinking, cooling, food preparation, and firefighting.

Guidance, Navigation and Control

The LM is designed to take two astronauts the orbiting CSM to the lunar surface and back again. The primary function of the Guidance, Navigation, and Control Subsystem (GN&CS) is accumulation, analysis, and processing of data to ensure the LM follows a predetermined flight plan at all times. To perform these functions, the guidance portion must know present position and velocity with respect to the guidance goal. The GN&CS provides navigation, guidance, and flight control to accomplish the specific guidance goal.

The astronaut is an active and controlling element of the LM. He can monitor information to and from the various LM subsystems and can manually duplicate the various control functions. During completely automatic flight, the astronaut functions as a monitor and decision maker; during semiautomatic flight, he is a controlling influence on the automatic system; and during manually controlled flight he may perform all GN&CS functions himself. The astronaut can also initiate an optical sighting program, utilizing celestial objects to align the guidance equipment.

Using cabin displays and controls, the astronaut can select modes of operation necessary to perform a desired function. In some mission phases, sequencing of modes of operation is automatically controlled by a computer. As calculations are performed by the computer, the results are displayed for astronaut evaluation and verification with ground-calculated data.

In the event of failure of automatic control, the astronaut manually controls the LM and performs vehicle flight control normally performed by the computer. He does this with a pair of hand controller, which control attitude and translation, and with other controls on the cabin panels.

For purposes of the following discussion, a distinction is made between guidance (orbital alteration or redirection of the LM) and navigation (accumulation and processing of data to define the proper guidance to be accomplished).

NAVIGATION AND THE LUNAR MODULE

LM navigation involves the determination of the vehicle's present position and velocity so that the guidance function can plot the trajectory that the LM must follow.

When flying an aircraft between two points on earth, both points remain fixed with respect to each other. In spaceflight, however, the origin of the spacecraft's path and its destination or target are moving rapidly with respect to each other.

To determine the present position of the LM, celestial navigation is used to align the guidance system. This is accomplished by determining the vehicle's position in relation to certain fixed stars. Even though the stars may be moving, the distance that they move in relation to the total distance of the stars from the vehicle is so small that the stars can be thought of as being stationary.

The optical device which the astronauts use for navigation is an alignment optical telescope (AOT) protruding through the top of the vehicle and functioning as a sextant. The astronauts use rt to take direct visual sightings and precise angular measurements of pairs of celestial objects. These measurements are transferred by the astronaut to the guidance elements to compute the position of the vehicle and to perform alignment of an inertial guidance system. There is a direct relationship between the angular measurements taken with the telescope and the mounting position of the telescope. The computer program knows the telescope's mounting position which is in alignment with the LM body axes and from this knowledge and astronaut generated information, the computer is able to calculate the LM position.

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, landing radar). 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.

GUIDANCE AND THE LUNAR MODULE

After the position and velocity of the LM are determined, the guidance function establishes the steering for the predetermined flight path. Since objects in space are moving targets (as compared to those on earth, which are stationary), the guidance problem involves aiming not at the target's present position but at the position in which it will be when the vehicle path intersects the target path. On earth, the guidance problem is a two dimensional one; it involves only longitude and latitude. In space, a third dimension is introduced; position cannot be plotted in earth terms.

To calculate the guidance parameters, a reference coordinate frame must be determined. A three-axis, right-hand, orthogonal, coordinate frame (inertial reference frame) is used. It is fixed in space and has an unchanging angular relationship with the stars. Its dimensional axes are designated as X, Y, and Z, and all spacecraft positions and velocities are related to this frame. The astronaut establishes this frame by sighting of celestial objects using the AOT. The vertical axis is designated as the X-axis. Its positive direction is from the descent stage to the ascent stage, passing through the overhead hatch. The lateral axis is designated as the Y-axis. Its positive direction is from left to right across the astronauts shoulders when they are facing the windows in the LM cabin. To complete the three-axis orthogonal system, the Z-axis is perpendicular to the X and Y axes This axis is referred to as the forward axis, because +Z-axis direction is through the forward hatch. The +Z-axis is also used as the zero reference line for all angular measurements.

The guidance system based on this coordinate frame is referred to as an inertial guidance system. Inertial guidance provides information about the actual path of the vehicle in relation to a predetermined path. All deviations are transmitted to a flight control system. The inertial guidance system performs these functions without information from outside the vehicle. The system stores the predetermined flight plan, then automatically but not continuously, computes distance and velocity for a given mission time (called the state vector) of the vehicle to compensate, through vehicle control, for changes in direction.

Inertial guidance systems are based on measurements made by accelerometers mounted on a structure called the stable member or platform. The stable member, in turn, is mounted inside three spherical gimbals, one for each principal axis of motion. Gyroscopes mounted on the stable member drive the gimbals to isolate the stable member from changes in LM attitude and hold the stable member in a fixed inertial position.

During flight, the stable member's axes must be held in fixed relation to the inertial reference frame regardless of the LM motion; other wise resolvers mounted on each gimbal issue error signals. These error signals are used by the computer to generate commands to correct the attitude of the LM. The rotational axes of the LM are designated as yaw, pitch, and roll. Yaw rotation, about the X-axis affects the vehicle in the Y-Z plane. The effect is analogous to spinning around one's heels. Pitch rotation, about the Y-axis, affects the vehicle in the X-Z plane. The effect is analogous to a gymnast performing a somersault. Roll rotation, about the Z-axis, affects the vehicle in the X-Y plane. The effect is analogous to a person doing a cartwheel. Positive rotation is determined by the right-hand rule. This involves placing the thumb of the right hand in the positive direction of the axis about which rotation is to be determined. Then the remainder of the fingers are curled around the axis. The direction in which the fingers point is considered the direction of positive rotation.

FLIGHT CONTROL AND THE LUNAR MODULE

Flight control involves controlling the LM trajectory (flight path) and attitude. Flight path control depends on the motion of the LM center of gravity; attitude control primarily involves rotations about the center of gravity.

In controlling the LM in its flight path, the thrust of its engines must be directed so that it produces a desired variation in either magnitude or direction to place the LM in some particular orbit, position, or attitude. The major velocity changes associated with the lunar orbit, injection, landing, and ascent phases of the mission are accomplished by either the descent propulsion section or ascent propulsion section of the Main Propulsion Subsystem (MPS). The engines can produce high thrust in specific directions in inertial space.

During the descent phase, the LM must be slowed (braked) to place it in a transfer orbit from which it can make a soft landing on the lunar surface. To accomplish braking, descent engine thrust is controllable so that the precise velocity (feet per second) necessary to alter the vehicle's trajectory can be achieved. For a soft landing on the lunar surface, the weight of the LM must be matched by an upward force so that a state of equilibrium exists, and from this point, the descent engine is shut off and the LM free falls to the lunar surface. The thrust of the descent engine provides this upward force, and since the weight of the vehicle is a variable (due to consumption of expendables) this is another reason why the magnitude of the engine thrust is controllable. In addition, the center of gravity is also variable and the thrust must be such that it is in line with the LM center of gravity. This is accomplished by gimbaling (tilting) the descent engine.

During the lunar ascent phase, the flight control portion of the GN&CS commands the ascent engine. In this phase, control of the thrust direction is not achieved by gimbaling the engine, but by attitude control, using the Reaction Control Subsystem (RCS) thrusters. This is necessary during ascent to keep the vehicle stabilized, because the center of gravity changes due to propellant depletion. The ascent engine is not throttleable, since the function of this engine is to lift the ascent stage from the lunar surface and conduct rendezvous. The proper orbit for rendezvous is achieved by means of a midcourse correction (if necessary) in which thrust is directed by attitude control, and thrust magnitude is controlled by controlling the duration of the burn.

It is apparent then for flight control, that some measure of the LM velocity vector and its position must be determined at all times for purposes of comparison with a desired (predetermined) velocity vector, at any particular instant, to generate an error signal if the two are not equal. The flight control portion of the primary guidance and navigation section then directs the thrust to reduce the error to zero.

Attitude control maintains the LM body axes in a fixed relationship to the inertial reference axes. Any pitch, roll, or yaw rotations of the vehicle produce a misalignment between the LM axes and where the

LM axes should be. This is called attitude error and is detected by the inertial guidance system, which, in turn, routes the errors to the computer. The computer generates on and off commands for the RCS to reduce the error to zero. Attitude control is implemented through 16 rocket engine thrusters (100 pounds thrust each) equally distributed in clusters of four around the ascent stage. Each cluster is located so that it will exert efficient torque to rotate the LM about its center of gravity. The thrusters are capable of repeated starts and very short (fraction of second) firing times. The appropriate thrusters are selected by the computer during automatic operation and manually by the astronaut during manual operation.

GUIDANCE, NAVIGATION, AND CONTROL SUBSYSTEM

To accomplish guidance, navigation, and control, the astronauts use 55 switches, 45 circuit breakers, and 13 indicators which interface with the various GN&CS equipment. This equipment is functionally contained in a primary guidance and navigation section, an abort guidance section, a control electronics section, and in the landing and rendezvous radars.

The primary guidance and navigation section (PGNS) provides, as the name implies,, the primary means for implementing inertial guidance and optical navigation for the LM. When aided by either the rendezvous radar or the landing radar, the section provides for radar navigation. The section when used in conjunction with the control electronics section (CES) provides automatic flight control. The astronauts can supplement or override automatic control, with manual inputs.

The abort guidance section (AGS) is primarily used only if the primary guidance and navigation section malfunctions. If the primary guidance and navigation section is functioning properly when a mission is aborted, it is used to control the LM. Should the primary section fail, the lunar mission would have to be aborted; thus, the term "abort guidance section." Abort guidance provides only guidance to place the LM in a rendezvous trajectory with the CSM or in a parking orbit for CSM-active rendezvous. The navigation function is performed by the primary section, but the navigation information also is supplied to the abort section. In case of a primary guidance malfunction, the abort guidance section uses the last navigation data provided to it. The astronaut can update the navigation data by manually inserting rendezvous radar data into the abort guidance section.

These integrated sections allow the astronauts to operate the LM in fully automatic, several semi-automatic, and manual control modes.

Because the astronauts frequently become part of the control loop in this highly flexible system, a great deal of information must be displayed for their use. These displays include attitude and velocity, radar data, fuel and oxidizer parameters, caution and warning information, total velocity change information, timing and other information to assist them in completing their mission.

PRIMARY GUIDANCE AND NAVIGATION SECTION

The primary guidance and navigation section acts as an autopilot in controlling the LM throughout the mission. Normal guidance requirements include transferring the LM 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 which results in terminal rendezvous with the CSM. If the mission is to be aborted, the primary guidance and navigation section performs guidance maneuvers that place the LM in a parking orbit or in a trajectory that intercepts the CSM.

The navigational functional requirement of the section is that it provides the navigational data required for LM guidance. These data include line-of-sight (LOS) data from the AOT for inertial reference alignment, signals for initializing and aligning the abort guidance section, and data to the astronauts for determining the location of the computed landing site.

The primary guidance and navigation section includes three major subsections: optical, and computer. Individually or in combination they perform all the functions mentioned previously.

The inertial subsection establishes the inertial reference frame that is used as the central coordinate system from which all measurements and computations are made. The inertial subsection 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 subsection is energized and aligned to the inertial reference, any LM rotation (attitude change) is sensed by the stable member. All inertial measurements (velocity and attitude) are with respect to the stable member. These data are used by the computer in determining solutions to the guidance problems.

The optical subsection is used to determine the position of the LM, 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 uses this data along with prestored data to compute position and velocity and to align the inertial components.

The computer subsection, as the control and data processing center of the LM, performs all the guidance and navigation functions necessary for automatic control of the 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 member and positions both radar antennas. It also provides control commands to both radars, the ascent engine, the descent engine, the RCS thrusters, and the LM cabin displays. As a general purpose computer, it solves guidance problems required for the mission.

ABORT GUIDANCE SECTION

The abort guidance section is used as backup for the primary guidance and navigation section during a LM mission abort. It determines the LM trajectory or trajectories required for rendezvous with the CSM and can guide the LM from any point in the mission, from LM-CSM separation to LM-CSM rendezvous and docking, including ascending from the lunar surface. It can provide data for altitude displays, for making explicit guidance computations and also issue commands for firing and shutting down engines. Guidance can be accomplished automatically or manually by the astronauts, based on data from the abort guidance section.

The abort guidance section is an inertial system rigidly strapped to the LM rather than mounted on a stabilized platform. Use of the strapped-down inertial system, rather than a gimbaled system, offers sufficient accuracy for LM missions, at savings in size and weight. Another feature is that it can be updated with radar and optical aids.

CONTROL ELECTRONICS SECTION

The control electronics section processes RCS and MPS control signals for vehicle stabilization and control. To stabilize the LM during all phases of the mission the control electronics section 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 primary guidance and navigation section during normal automatic operation from two hand controllers during manual operations, or from the abort guidance section during certain abort situations.

The control electronics section 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 LM center of gravity.

LANDING RADAR

The landing radar, located in the descent stage, provides altitude and velocity data during lunar descent. The primary guidance and navigation section calculates control signals for descent rate, hovering, and soft landing. Slant range data begins at approximately 40,000 feet above the lunar surface; velocity data at approximately 35,000 feet.

The landing radar uses four microwave beams; three to measure velocity by Doppler shift continuous wave, one to measure altitude by continuous-wave frequency modulation.

RENDEZVOUS RADAR

The rendezvous radar, operated in conjunction with a CSM transponder, acquires and tracks the CSM before and during rendezvous and docking. The radar, located in the ascent stage, tracks the CSM during the descent phase of the mission to supply tracking data for any required abort maneuver and during the ascent phase to supply data for rendezvous and docking. When the radar tracks the CSM, continuous measurements of range, range rate, angle, and angle rate (with respect to the LM) are provided simultaneously to the primary guidance and navigation section and to LM cabin displays. This allows rendezvous to be performed automatically under computer control, or manually by the astronauts. During the rendezvous phase, rendezvous radar performance is evaluated by comparing radar range and range rate tracking values with MSFN tracking values.

The CSM transponder receives an X-band three-tone phase-modulated, continuous-wave signal from the rendezvous radar, offsets the signal by a specified amount, and then transmits a phase-coherent carrier frequency for acquisition by the radar. This return signal makes the CSM appear as the only object in the radar field of view. The transponder provides the long range (400 nm) required for the mission.

The transponder and the radar use solid-state varactor frequency-multiplier chains as transmitters, to provide high reliability. The radar antenna is space stabilized to negate the effect of LM motion on the line-of-sight angle. The gyros used for this purpose are rate-integrating types; in the manual mode they also supply accurate line of-sight, angle-rate data for the astronauts. Range rate is determined by measuring the two-way Doppler frequency shift on the signal received from the transponder. Range is determined by measuring the time delay between the received and the transmitted three-tone phase modulated waveform.

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 the functions required to complete the LM mission. If a failure occurs in this path the abort guidance path can be substituted. To understand these two loops, the function of each major component of GN&CS equipment must be known.

PRIMARY GUIDANCE AND NAVIGATION SECTION

INERTIAL SUBSECTION

The inertial subsection consists of a navigation base, an inertial measurement unit, a coupling data unit, pulse torque assembly, power and servo assembly, and signal conditioner assembly.

The navigation base is a lightweight mount that supports, in accurate alignment, the inertial measurement unit (IMU), the AOT, and an abort sensor assembly (part of the abort guidance section). Structurally, it consists of a center ring with four legs that extend from either side of the ring. The inertial measurement unit is mounted to the legs on one end and the telescope and the abort sensor assembly are mounted on the opposite side.

The inertial measurement unit is the primary inertial sensing device of the LM. It is a three-degree-of-freedom, stabilized device that maintains an orthogonal, inertially referenced coordinate system for LM attitude control and maintains three accelerometers in the reference coordinate system for accurate measurement of velocity changes.

The coupling data unit converts and transfers angular information between the navigation and guidance hardware. The unit is an electronic device that performs analog-to-digital and digital-to-analog conversions. The coupling data unit processes the three attitude angles associated with the inertial reference and the two angles associated with the rendezvous radar antenna.

The pulse torque assembly supplies inputs to, and processes outputs from, the inertial components in the inertial subsection.

The power and servo assembly contains electronic equipment in support of the primary guidance and navigation section: power supplies for generation of internal power required by the section, servomechanisms for the inertial measurement unit, and failure detection circuitry for the inertial measurement unit.

The signal conditioner assembly provides an interface between the primary guidance and navigation section, and the Instrumentation Subsystem (IS).

OPTICAL SUBSECTION

The optical subsection consists of the alignment optical telescope and a computer control and reticle dimmer assembly.

The alignment optical telescope, an L-shaped periscope approximately 36 inches long, is used by the astronaut to take angular measurements of celestial objects. These angular measurements are required for orienting the stable member during certain periods while the LM is in flight and during prelaunch preparations while on the lunar surface. Sightings taken with the telescope are transferred to the computer by the astronaut using the computer control and reticle dimmer assembly. This assembly also controls the brightness of the telescope reticle pattern.

COMPUTER SUBSECTION

The computer subsection consists of the LM guidance computer (LGC) and a display and keyboard, which is a computer control panel. The display and keyboard is commonly referred to as "the DSKY" (pronounced "disky").

The guidance computer processes data and issues discrete control signals for various subsystems. It is a control computer with many of the features of a general-purpose computer As a control computer, it aligns the inertial measurement unit stable member and provides rendezvous radar antenna drive commands. The LGC also provides control commands to the landing and rendezvous radars, the ascent and descent engines, the RCS thrusters, and the cabin displays. As a general purpose computer, it solves guidance problems required for the mission. In addition, the guidance computer monitors the operation of the primary guidance and navigation section.

The guidance computer stores data pertinent to the ascent and descent flight profiles that the LM must assume to complete its mission. These data (position, velocity, and trajectory information) are used by the computer to solve flight equations. The results of various equations are used to determine the required magnitude and direction of thrust. The computer establishes corrections to be made. The LM engines are turned on at the correct time, and steering commands are controlled by the computer to orient the LM to a new trajectory, if required. The inertial subsection senses acceleration and supplies velocity changes to the computer for calculating total velocity. Drive signals are supplied from the computer to the coupling data unit and stabilization gyros in the inertial subsection to align the gimbal angles in the inertial measurement unit. Stable-member position signals are supplied to the computer to indicate attitude changes.

The computer provides drive signals to the rendezvous radar for antenna positioning and receives, from the rendezvous radar channels of the coupling data unit, antenna angle information. The computer uses this information in the antenna-positioning calculations. During lunar-landing operations, star-sighting information is manually loaded into the computer, using the DSKY. This information is used to calculate alignment commands for the inertial measurement unit. The LM guidance computer and its programming help meet the functional requirements of the mission. The functions performed in the various mission phases include both automatic and semiautomatic operations that are implemented mostly through the execution of the programs stored in the computer memory.

The DSKY provides a two-way communications link between the astronauts and the LM guidance computer. The astronauts are able to insert various parameters into the computer, display data from the computer, and to monitor data in the computer's memory.

ABORT GUIDANCE SECTION

The abort guidance section consists of an abort sensor assembly, a data entry and display assembly (DEDA), and an abort electronics assembly. The data entry and display assembly is commonly referred to as "the DEDA" (pronounced "deeda").

The abort sensor assembly, by means of gyros and accelerometers, provides incremental attitude information around the LM X, Y, and Z axes and incremental velocity changes along the LM X, Y, and Z axes. Data pulses are routed to the abort electronic assembly, which uses the LM attitude and velocity data for computation of steering errors.

The DEDA is used by the astronauts to select the desired mode of operation, insert the desired targeting parameters, and monitor related data throughout the mission. To select a mode of operation or insert data, three digits (word address) then a plus (+) or minus (-), and finally, a five digit code must be entered. If this sequence is not followed, an operator error light goes on when the enter pushbutton is pressed. To read out any parameter, three digits (address of the desired word) must be entered and a readout pushbutton pressed.

The abort electronics assembly, by means of special input-output subassemblies, interfaces the abort guidance section with the other LM subsystems and displays. This assembly is basically a general-purpose digital computer, which solves guidance and navigation problems mode and sub-mode entries coupled from the data entry and display assembly determine the operation of the computer. The computer uses incremental velocity and attitude inputs from the abort sensor assembly to calculate LM position, attitude, and velocity in the inertial reference frame. It routes altitude and altitude-rate data to altitude and altitude rate indicators; out of plane velocity data, to X-pointer indicators. Also, roll, pitch and yaw steering error signals are routed to flight director altitude indicators.

Engine-on commands are routed to the appropriate engine via the control electronics section when the following occur: an abort or abort stage pushbutton is pressed, appropriate switches are set, necessary data are entered into the DEDA, and velocity-to-be-gained exceeds a predetermined threshold (currently 2.1 fps). At the appropriate time, as determined by velocity-to-be gained, an engine-off command is sent.

CONTROL ELECTRONICS SECTION

The control electronics section comprises two attitude controller assemblies, two thrust/ translation controller assemblies, an attitude and translation control assembly, a rate gyro assembly, descent engine control assembly, three stabilization and control (S&C) control assemblies and two gimbal drive actuators.

The attitude controller assemblies are right-hand pistol grip controllers, which the astronauts use to command changes in LM attitude. These controllers function in a manner similar to an aircraft's "control stick". Each is installed with its longitudinal axis approximately parallel to LM X axis; vehicle rotations correspond to astronaut hand movements.

The thrust/translation controller assemblies are left-hand controllers used by the astronauts to control LM translation in any axis. Vehicle translations correspond approximately to the astronauts hand movements.

The attitude and translation control assembly routes the RCS thruster on and off commands from the guidance computer to the thrusters, in the primary control mode. During abort guidance control, the assembly acts as a computer in determining which RCS thrusters are to be fired.

The rate gyro assembly is used during abort guidance control to supply the attitude and translation control assembly with damping signals to limit vehicle rotation rates and to facilitate manual rate control.

The descent engine control assembly processes engine throttling commands from the astronauts (manual control) and the guidance computer (automatic control), gimbal commands for thrust vector control, preignition (arming) commands, and on and off commands to control descent engine ignition and shutdown.

The S&C control assemblies are three similar assemblies. They process, switch, and/or distribute the various signals associated with the GN&CS.

The gimbal drive actuators position the descent engine in roll and pitch in response to DECA outputs.

LANDING RADAR

The landing radar 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 single-beam radar altimeter. Velocity and range data are made available to the LM guidance computer as 15-bit binary words; forward and lateral velocity data, to the LM displays as d-c analog voltages; and range and range rate data, to the LM displays as pulse-repetition frequencies.

The landing radar consists of an antenna assembly and an electronics assembly. The antenna assembly forms, directs, transmits, and receives the four microwave beams. Two interlaced phased arrays transmit the velocity- and altimeter-beam energy. Four broadside arrays receive the reflected energy of the three velocity beams and the altimeter beam. The electronics assembly processes the Doppler and continuous-wave FM returns, which provide the velocity and slant range data for the LM guidance computer and the LM displays.

The antenna assembly transmits velocity beams (10.51 gHz) and an altimeter-beam (9.58 gHz) to the lunar surface.

When the electronics assembly is receiving and processing the returned microwave beam, data-good signals are sent to the LGC. When the electronics assembly is not operating properly, data-nogood signals are sent to the pulse code modulation timing electronics assembly of the Instrumentation Subsystem for telemetry.

Using LM controls and indicators, the astronauts can monitor LM velocity, altitude, and radar transmitter power and temperatures; apply power to energize the radar; initiate radar self-test; and place the antenna in descent or hover position. Self-test permits operational checks of the radar without radar returns from external sources. An antenna temperature control circuit, energized at earth launch, protects antenna components against the low temperatures of space environment while the radar is not operating.

The radar is first turned on and self-tested during LM checkout before separation from the CSM. The self-test circuits apply simulated Doppler signals to radar velocity sensors, and simulated lunar range signals to an altimeter sensor. The radar is self-tested again immediately before LM powered descent, approximately 70,000 feet above the lunar surface. The radar operates from approximately 50,000 feet until lunar touchdown.

Altitude (derived from slant range) is available to the LGC and is displayed on a cabin indicator at or above 25,000 feet. Slant range data are continuously updated to provide true altitude above the lunar surface at, or above 18,000 feet, forward and lateral velocities are available to the LM guidance computer and cabin indicators.

At approximately 200 feet above the lunar surface, the LM pitches to orient its X-axis perpendicular to the surface; all velocity vectors are near zero. Final visual selection of the landing site is followed by touchdown under automatic or manual control. During this phase, the astronauts monitor altitude and velocity data from the radar.

The landing radar antenna has a descent position and a hover position. In the descent position, the antenna boresight-angle is 24 degrees from the LM X-axis. In the hover position, the antenna boresight is parallel to the X- axis and perpendicular to the Z-axis. Antenna position is selected by the astronaut during manual operation and by the LM guidance computer during automatic operation. During automatic operation, the LM guidance computer commands the antenna to the hover position 8,000 to 9,000 feet above the lunar surface.

RENDEZVOUS RADAR

The rendezvous radar has two assemblies, the antenna assembly and the electronics assembly. The antenna assembly automatically tracks the transponder signal after the electronics assembly acquires the transponder carrier frequency. The return signal from the transponder is received by a four-port feedhorn. The feedhorn, arranged in a simultaneous lobing configuration, is located at the focus of a Cassegrainian antenna. If the transponder is directly in line with the antenna bore-sight, the transponder signal energy is equally distributed to each port of the feedhorn. If the transponder is not directly in line, the signal energy is unequally distributed among the four ports.

The signal passes through a polarization diplexer to a comparator, which processes the signal to develop sum and difference signals. The sum signal represents the sum of energy received by all feedhorn ports (A + B + C + D). The difference signals, representing the difference in energy received by the feedhorn ports, are processed along two channels: a shaft-difference channel and a trunnion-difference channel. The shaft difference signal represents the vectoral sum of the energy received by adjacent ports (A + D)-(B + C) of the feedhorn. The trunnion difference signal represents the vectoral sum of the energy received by adjacent ports (A + B) - (C + D). The comparator outputs are heterodyned with the transmitter frequency to obtain three intermediate-frequency signals. After further processing, these signals provide unambiguous range, range rate, and direction of the CSM. This information is fed to the LGC and to cabin displays.

The rendezvous radar operates in three modes; automatic tracking, slew (manual), or LM guidance computer control.

Automatic Tracking Mode. This mode enables the radar to track the CSM automatically after it has been acquired; tracking is independent of LM guidance computer 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. This mode enables an astronaut to position the antenna manually to acquire the CSM.

LM Guidance Computer Control Mode. In this mode, the computer automatically controls antenna positioning, initiates automatic tracking once the CSM is acquired, and controls change in antenna orientation. The primary guidance and navigation section, which transmits computer-derived commands to position the radar antenna, provides automatic control of radar search and acquisition.

PRIMARY GUIDANCE PATH

The primary guidance path comprises the primary guidance and navigation section, control electronics section, landing radar, and rendezvous radar and the selected propulsion section required to perform the desired maneuvers. The control electronics section routes flight control commands from the primary guidance and navigation section and applies them to the descent or ascent engine, and the appropriate thrusters.

INERTIAL ALIGNMENT

Inertial subsection operation can be initiated automatically by the primary guidance computer or manually by the astronaut, using DSKY entries to command the computer. The inertial subsection status or mode of operation is displayed on the DSKY as determined by a computer program. When the inertial subsection is powered up, the gimbals of the inertial measurement unit are driven to zero by a reference voltage and the coupling data unit is initialized to accept inertial subsection data. During this period, there is a 90-second delay before power is applied to the gyro and accelerometer torquing loops This is to prevent them from torquing before the gyros reach synchronous rotor speed.

The stable member of the inertial measurement unit must be aligned with respect to the reference coordinate frame each time the inertial subsection is powered up During flight the stable member may be periodically realigned because it may deviate from its alignment, due to gyro drift. 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 program within the computer.

Inertial subsection alignment is accomplished in two steps: coarse alignment and fine alignment. To initiate coarse alignment, the astronaut selects, by a DSKY entry, a program that determines stable member orientation, and a coarse-alignment routine. The computer sends digital pulses, representing the required amount of change in gimbal angle, to the coupling data unit. The coupling data unit converts these digital pulses to analog signals which drive torque motors in the inertial measurement unit. As the gimbal angle changes, a gimbal resolver signal is applied to the coupling data unit, where it is converted to digital pulses. These digital pulses cancel the computer pulses stored in the coupling data unit. When this is accomplished, coarse alignment is completed and the astronaut can now select an in-flight fine-alignment routine.

To perform the fine-alignment routine, the astronaut must use the alignment optical telescope to sight on at least two stars; The gimbals, having been coarse aligned, are relatively close to their preferred angles. The computer issues fine alignment torquing signals to the inertial measurement unit after it processes star-sighting data that have been combined with known gimbal angles.

Once the inertial subsection is energized and aligned, 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 attitude with respect to the stable member. These angular measurements are displayed to the astronauts by the flight director attitude indicators, and angular changes of the inertial reference are sent to the computer.

Inertial stability of the stable member in the inertial measurement unit is maintained with a stabilization loop which uses the IMU gyro outputs as inputs to amplifiers in the power and servo assy. The amplifier outputs drive torquers on each of the three IMU gimbals to null out the gyro errors.

ATTITUDE CONTROL

Desired attitude is calculated in the primary guidance computer and compared with the actual gimbal angles. If there is a difference between the actual and calculated angles, the inertial subsection channels of the coupling data unit generate attitude error signals, which are sent to the attitude indicators for display. These error signals are used by the digital autopilot program in the primary guidance computer to activate RCS thrusters for LM attitude correction. LM acceleration due to thrusting is sensed by three accelerometers, 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 computer, which calculates the total LM velocity.

Two normal modes of operation achieve attitude control: automatic and attitude hold. In addition to these two modes, there is a minimum impulse mode and a four-jet manual override mode. Either of the two normal modes may be selected on the primary guidance mode control switch.

In automatic mode, all navigation, guidance, and flight control is handled by the primary guidance computer. The computer calculates the desired or preferred attitude, generates the required thruster commands and routes them to the attitude and translation control assembly which fires the selected thruster.

Attitude hold mode is a semiautomatic mode in which either astronaut can command attitude change at an angular rate proportional to the displacement of his attitude controller. The LM holds the new attitude when the controller is brought back to its neutral (detent) position. During primary guidance control, rate commands proportional to controller displacement are sent to the computer. The computer processes these commands and generates thruster commands for the attitude and translation control assembly.

Minimum impulse mode enables the astronaut to control the LM with a minimum of fuel consumption. Each movement of the attitude controller out of its detent position causes the primary guidance computer to issue commands to the appropriate thrusters. The controller must be returned to the neutral position between each impulse command. This mode is selected by DSKY entry only while the control electronics section is in attitude hold. In this mode, the astronaut must perform his own rate damping and attitude steering.

Manual override also is known as the hard-over mode. In certain contingencies that may require an abrupt attitude maneuver, the attitude controller can be displaced to the maximum limit (hard-over position) to command an immediate attitude change about any axis. This displacement applies signals directly to the RCS solenoids to fire four thrusters that provide the desired maneuver. This maneuver can override any other attitude control mode.

TRANSLATION CONTROL

Automatic and manual translation control is available in all three axes, using the RCS. Automatic control consists of thruster commands from the primary guidance computer to the attitude and translation control assembly. These commands are used for translations of small velocity increments and for ullage maneuvers (to settle propellant in the tanks) before ascent or descent engine ignition after coasting phases. Manual control during primary guidance control consists of on and off commands generated by the astronaut using his thrust/translation controller. These commands are routed through the computer to the attitude and translation control assembly to fire the proper thrusters. Translation along the +X-axis can also be initiated by the astronaut using a pushbutton switch that actuates the secondary solenoid coils of the four downward firing thrusters.

DESCENT ENGINE CONTROL

Descent engine ignition is controlled either automatically by the primary guidance and navigation section, or manually through the control electronics section. Before ignition can occur, the engine arm switch must be set to the descent engine position. This opens the pre-valves to allow fuel and oxidizer to reach the propellant shutoff valves, arming the descent engine.

Engine-on commands from either computer are routed to the descent engine control assembly which commands the descent engine on by opening the propellant shutoff valves. The engine remains on until an engine-off discrete is initiated by the astronauts with either of two engine stop pushbuttons or by the computer. When the LM reaches the hover point where the lunar contact probes touch the lunar surface, a blue lunar contact light is illuminated. This indicates to the astronauts that the engine should be shut down. From this point (approximately 5 feet above the lunar surface), the LM free-falls to the lunar surface.

Descent engine throttling can be controlled by the primary guidance and navigation section and/or the astronauts. Automatic increase or decrease signals from the guidance computer are sent to the descent engine control assembly. An analog output from the control assembly corresponds to the percentage of thrust desired. The engine is controllable from 10 percent of thrust to a maximum of 92.5 percent. There are two thrust control modes: automatic and manual. In the automatic mode, the astronaut can use the selected thrust/translation controller to increase descent engine thrust only. During this mode, manual commands by the astronaut are used to override the throttle commands generated by the computer. In the manual mode, the astronauts have complete control over descent engine thrust.

Descent engine trim is automatically controlled during primary control, to compensate for center-of-gravity offsets due to propellant depletion and, in some cases for attitude control. The primary guidance computer routes trim commands for the pitch and roll axes. These signals drive a pair of gimbal drive actuators. These actuators, which are screw jack devices, tilt the descent engine about the Y-axis and Z-axis a maximum of +6 degrees or -6 degrees from the X-axis.

ASCENT ENGINE CONTROL

Ascent engine ignition and shutdown can be initiated automatically by the primary guidance computer or manually by the astronauts. 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 control assemblies are enabled when the astronauts select the ascent engine position of the engine arm switch.

In an abort stage situation while the descent engine is firing, the control assemblies provide a time delay before commanding staging and ascent engine ignition. The time delay ensures that descent engine thrusting has completely stopped before staging occurs.

ABORT GUIDANCE PATH

The abort guidance path comprises the abort guidance section, control electronics section, and the selected propulsion section. The abort guidance path performs all inertial guidance and navigation functions necessary to effect a safe orbit or rendezvous with the CSM. The stabilization and control functions are performed by analog computation techniques, in the control electronics section.

The control electronics section functions as an autopilot when the abort guidance path is selected. It uses inputs from the abort guidance section and from the astronauts to provide the following: on, off, and manual throttling commands for the descent engine descent engine gimbal drive actuator commands; ascent engine on and off commands; engine sequencer logic to ensure proper arming and staging before engine startup and shutdown; RCS on and off commands; RCS jet-select logic to select the proper thruster for the various maneuvers; and modes of control, ranging from automatic to manual.

ATTITUDE CONTROL

The abort guidance path operates in the automatic mode or the attitude hold mode. In automatic, navigation and guidance functions are controlled by the abort guidance section, attitude by the control electronics section. The abort electronics assembly (abort guidance computer) generates roll, pitch, and yaw attitude error signals, which are summed with rate damping and attitude rate signals in the attitude and translation control assembly. A jet-select logic circuit selects the thruster to be fired and issues the appropriate thruster command.

In attitude hold, the astronaut uses manual control. In this mode, a pulse sub-mode and a two-jet direct sub-mode are available in addition to manual override (hard-over). The pulse and two-jet direct sub-modes are selectable on an individual axis basis only. The attitude controller generates attitude rate, pulse, direct, and hard-over commands.

During abort guidance control, with the attitude controller in the neutral position, attitude is held by means of attitude error signals detected by the abort electronics assembly. When either controller is moved out of the neutral position, the attitude error signals from the abort guidance section are zero. Rate commands, proportional to controller displacement, are processed in the attitude and translation control assembly, and the thrusters are fired until the desired vehicle rate is achieved. When the controller is returned to the neutral position, the vehicle rate is reduced to zero and the abort guidance section holds the LM in the new attitude.

The pulse sub-mode is selected by the astronaut, using the appropriate attitude control switch Automatic attitude control about the selected axis is then disabled and a fixed train of pulses is generated when the attitude controller is displaced from its neutral position. To change vehicle attitude in this sub-mode, the attitude controller must be moved out of neutral. This commands acceleration about the selected axis through low-frequency thruster pulsing. The pulse sub-mode uses the primary solenoid coils of the thrusters; the direct sub-mode, the secondary solenoid coils. To terminate rotation, an opposite acceleration about the selected axis must be commanded.

The direct sub-mode is selected by the astronaut, using the attitude control switches that are used for the pulse sub-mode. When selected, automatic control about the selected axes is disabled and direct commands are routed to the RCS secondary solenoids to two thrusters when the attitude controller is displaced from the neutral position. The thrusters under direct control fire continuously until the controller is returned to the neutral position.

TRANSLATION CONTROL

During abort guidance control, only manual translation is available because the abort programs do not require lateral or forward translation maneuvers. Translation control consists of on and off commands from a thrust/translation controller to the jet select logic of the attitude and translation control assembly. RCS thrust along the +X-axis is accomplished the same way as during primary guidance control when the astronaut uses the +X-axis translation pushbutton.

DESCENT ENGINE CONTROL

Descent engine ignition is automatically controlled by programs stored in the abort electronics assembly. This assembly computes the abort guidance trajectory and required steering. If the primary guidance and navigation section fails while the descent engine is being used, the astronaut initiates abort guidance descent engine control through a DEDA entry. The abort electronics assembly can only control descent engine ignition and shutdown. Descent engine throttling and gimbaling are not under computer control when operating with the abort guidance section. As with the primary guidance path, the abort path generates an engine-off command when the required velocity is attained. This velocity depends upon whether the program used will place the LM in a rendezvous trajectory or in a parking orbit. Manual on and off control also is available. In all cases, the S&C control assemblies receive engine on and off commands. As in the primary guidance path, these assemblies route the commands to the descent engine control assembly which routes them to the engine.

The astronaut uses the thrust/translation controller to control descent engine throttling and translation maneuvers. The manual throttle commands are supplied to the descent engine control assembly, which generates analog signals driving the throttle valve actuator.

Descent engine trim control under abort guidance, is achieved by using attitude errors from the abort electronics assembly These errors are used by the attitude and translation control assembly for attitude control and steering calculation. The roll and pitch attitude errors are routed to the descent engine control assembly as trim commands.

ASCENT ENGINE CONTROL

Ascent engine control during abort guidance is similar to that of the primary guidance. During abort guidance control, automatic ascent engine ignition and shutdown are controlled by the abort electronics assembly.

If the descent stage is attached, the LM can be staged manually through use of the appropriate switches on the explosive devices panel. The astronaut has the option of using an abort stage pushbutton to start an automatic ascent engine ignition sequence. If the ascent engine-on command is lost, the ascent engine latching device memory circuit keeps issuing the command.

EQUIPMENT

PRIMARY GUIDANCE AND NAVIGATION SECTION

NAVIGATION BASE

The navigation base is a lightweight mount (about 3 pounds) bolted to the LM structure above the astronauts heads, with three mounting pads on a center ring. The center ring is approximately 14 inches in diameter and each of the four legs, which are part of the base, is approximately 10 inches long.

INERTIAL MEASUREMENT UNIT

The inertial measurement unit contains the stable member, gyroscopes, and accelerometers necessary to establish the inertial reference.

The stable member serves as the space-fixed reference for the inertial subsection. It is supported by three gimbal rings (outer, middle, and inner) for complete freedom of motion.

The outer gimbal is mounted to the case of the unit its axis is parallel to the LM X-axis. The middle gimbal is mounted to and perpendicular with the outer; its axis is parallel to the LM Z-axis. The inner gimbal supports the stable member; its axis is parallel to the LM Y-axis. The inner gimbal is mounted to the middle one. All three gimbals are spherical with 360 degrees of freedom. To overcome the small amount of friction inherent in the support system, small torque motors are mounted on each axis.

The three Apollo inertial reference integrating gyroscopes, used to sense attitude changes, are mounted on the stable member, mutually perpendicular. The gyros are fluid- and magnetically suspended, single-degree-of-freedom types. They sense displacement of the stable member and generate error signals proportional to displacement, the three pulse integrating pendulous accelerometers are fluid and magnetically suspended devices.

Thermostats maintain gyro and accelerometer temperature within their required limits during inertial measurement unit standby and operating modes. Heat is applied to end-mount heaters on the inertial components, by stable member heaters, and by a temperature control anticipatory heater. Heat is removed by convection, conduction, and radiation. The natural convection used during inertial measurement unit standby mode is changed to blower- controlled, forced convection during the operating mode. inertial measurement unit internal pressure is normally between 3.5 and 15 psia, enabling the required forced convection. To aid in removing heat, water-glycol passes through the case. Therefore, heat flow is from the stable member to the case and coolant. The temperature control system consists of the temperature control circuit, the blower control circuits, and temperature alarm circuit.

COUPLING DATA UNIT

The coupling data unit performs analog to digital conversion, digital-to-analog conversion, inertial subsection modeling and failure detection. It consists of a sealed container which encloses 34 modules of 10 different types that make up five almost identical channels: one each for the inner, middle, and outer gimbals of the inertial measurement unit and one each for the rendezvous radar shaft and trunnion gimbals. Several of the modules are shared by all five channels.

The two channels used with the rendezvous radar interface between the antenna and the guidance computer. The computer calculates digital antenna position commands before acquisition of the CSM. These signals are converted to analog form by the coupling data unit and applied to the antenna drive mechanism to aim the antenna. Tracking- angle information in analog form is converted to digital by the unit and applied to the guidance computer.

The three channels used with the inertial measurement unit provide interfaces between it and the guidance computer and between the computer and the abort guidance section. Each of the three IMU gimbal angle resolvers provide its channel with analog gimbal-angle signals that represent LM attitude. The coupling data unit converts these signals to digital form and applies them to the guidance computer. The computer calculates attitude or translation commands and routes them through the control electronics section to the proper thruster. The coupling data unit converts attitude error signals to 800-cps analog signals and applies them to the attitude indicator. Coarse and fine-alignment commands generated by the guidance computer are coupled to the inertial measurement unit through the coupling data unit.

The digital-to-analog converters of the coupling data unit are a-c ladder networks. When the unit is used to position a gimbal, the guidance computer calculates the difference between the desired gimbal angle and the actual gimbal angle. This difference results in a servo error signal that drives the gimbal to the desired angle.

The analog-to-digital converter operates on an incremental basis. Using a digital analog feedback technique which utilizes the resolvers as a reference, the coupling data unit accumulates the proper angular value by accepting increments of the angle to close the feedback loop. These data are applied to counters in the guidance computer for rendezvous radar tracking information, and to counters in the primary and abort guidance computers for the inertial reference gimbal angles. in this manner, the abort guidance section attitude reference is fine aligned simultaneously with that of the primary guidance and navigation section.

PULSE TORQUE ASSEMBLY

The pulse torque assembly consists of 17 electronic modular subassemblies mounted on a common base. There are four binary current switches: one furnishes torquing current to the three gyros; the other three furnish torquing current to the three accelerometers. Four d-c differential amplifier and precision voltage reference subassemblies regulate torquing current supplied through the binary current switches.

Three a-c differential amplifier and interrogator subassemblies amplify accelerometer signal generator signals and convert them to positive and negative torque pulse. The gyro calibration module applies torquing current to the gyros when commanded by the guidance computer. Three accelerometer calibration module compensate for the difference in inductive loading of accelerometer torque generator windings and regulate the balance of positive and negative torque. A pulse torque isolation transformer couples torque commands, data pulses, interrogate pulses, switching pulses, and synchronizing pulses between the guidance computer and the pulse torque assembly. The pulse torque power supply supplies power. for the other 16 subassemblies.

POWER AND SERVO ASSEMBLY

The power and servo assembly provides a central mounting point for the primary guidance and navigation section amplifiers, modular electronic components, and power supplies. The assembly is on the cabin bulkhead behind the astronauts. It consists of 14 subassemblies mounted to a header assembly.

SIGNAL CONDITIONER ASSEMBLY

The signal conditioner assembly preconditions primary guidance and navigation section measurements to a 0- to 5- volt d-c format before the signals are routed to the Instrumentation Subsystem.

ALIGNMENT OPTICAL TELESCOPE

The alignment optical telescope, mounted on the navigation base to provide mechanical alignment and a common reference between the telescope and the inertial measurement unit, is a unity-power, periscope-type device with a 60 degree conical field of view. It is operated manually by the astronauts. The telescope has a movable shaft axis (parallel to the LM X-axis) and a line of sight approximately 45 degrees from the X-axis in the Y-Z plane.

The telescope line of sight is fixed in elevation and movable in azimuth to six detent positions These detent positions are selected by turning a detent selector knob on the telescope; they are located at 60 degree intervals. The forward (F), zero detent position, places the line of sight in the X-Z plane. looking forward and up as one would look from inside the LM. The right (R) position places the line of sight 60 degrees to the right of the X-Z plane; the left (L) position, 60 degrees to the left of the X-Z plane Each of these positions maintains the line of sight at 45 degrees from the LM +X-axis. The remaining three detent positions reverse the prism on top of the telescope.. These positions are right-rear, closed (CL), and left-rear. The CL position (180 degrees from the F position) is the stowed position. The right-rear and left-rear positions have minimal use.

The optics consist of two sections: shaft optics and eyepiece optics. The shaft optics section is a -5 power complex that provides a 60 degree field of view. The eyepiece optics section is a +5 power complex that provides shaft and trunnion angle measurements. The reticle pattern the eye piece optics consists of crosshairs and a pair of Archimedes spirals The vertical crosshair, an orientation line designated the Y-line, is parallel to the LM X-axis when the reticle is at the 0 degree reference position. The horizontal crosshair, an auxiliary line designated the X-line, is perpendicular to the orientation line. The one-turn spirals are superimposed from the center of the field of view to the top of the vertical crosshair. Ten miniature red lamps mounted around the reticle prevent false star indications caused by imperfections in the reticle and illuminate the reticle pattern. Stars will appear white; reticle imperfections, red. Heaters prevent fogging of the mirror due to moisture and low temperatures during the mission.

A reticle control enables manual rotation of the reticle for use in lunar surface alignments. A counter on the left side of the unit, provides angular readout of the reticle rotation. The counter reads in degrees to within +/- 0.02 degrees or +/- 72 seconds. The maximum reading is 359.88 degrees, then the counter returns to 0 degrees. Interpolation is possible to within +/- 0.01 degree.

A rotatable eye guard is fastened to the end of the eyepiece section. The eye guard is axially adjustable for head position. It is used when the astronaut takes sightings with his faceplate open. This eye guard is removed when the astronaut takes sightings with his faceplate closed; a fixed eye guard, permanently cemented to the telescope, is used instead. The fixed eye guard prevents marring of the faceplate by the eyepiece. A high-density filter lens, supplied as auxiliary equipment, prevents damage to the astronaut's eyes due to accidental direct viewing of the sun or if the astronaut chooses to use the sun as a reference.

The alignment optical telescope is used for in-flight and lunar surface sightings.

For in-flight sightings, the telescope may be placed in any of the usable detent positions, However, when the LM is attached to the CSM, only the forward position is used. The astronaut selects a detent and the particular star he wishes to use. He then maneuvers the LM so that the selected star falls within the telescope field of view. The specific detent position and a code associated with the selected star are entered into the guidance computer by the astronaut using 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 a mark pushbutton is pressed, a discrete is sent to the guidance computer. The guidance computer then records the time of mark and the inertial measurement unit gimbal angles at the instant of the mark.

Crossing of a reticle line by the star image defines containing the star. Crossing of the other reticle 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 guidance computer calculates the orientation of the stable member with respect to a predefined reference coordinate system.

On the lunar surface, the LM cannot be maneuvered to obtain a star-image that crosses the reticle crosshairs. The astronaut using the reticle control knob, adjusts the reticle to superimpose the orientation (Y) line on the target star. The reticle angle display on the reticle counter, is then inserted into the computer by the astronaut. This provides the computer with the star orientation angle (shaft angle). The astronaut then continues rotating the reticle until a point on the spirals is superimposed on the target star. This second angular readout (reticle angle) is then entered into the computer along with the detent position and the code of the observed star. The computer 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 spirals, the Delta angle is proportional to the distance of the star from the center of the field of view. Using this angle and a proportionality equation, the computer can calculate the trunnion angle. At least two star sightings are required for determination of the inertial orientation of the stable member.

COMPUTER CONTROL AND RETICLE DIMMER ASSEMBLY

The computer control and reticle dimmer assembly is mounted on the alignment optical telescope guard. The mark X and mark Y push buttons are used by the astronauts to send discrete signals to the primary guidance computer when star sightings are made. The reject push button is used if an invalid mark has been sent to the computer. A thumbwheel on the assembly is used to adjust the brightness of the telescope's reticle lamps.

LM GUIDANCE COMPUTER

The LM guidance computer is the central data-processing device of the GN&CS. It is a parallel fixed-point, one's- complement, general-purpose digital computer with a fixed rope core memory and an erasable ferrite core memory. It has a limited self-check capability inputs to the computer are received from the landing radar and rendezvous radar, from the inertial measurement unit through the inertial channels of the coupling data unit and from an astronaut through the DSKY. The computer performs four major functions: (1) calculates steering signals and generates engine and RCS thruster commands to keep the LM on a required trajectory (2) aligns the stable member (inner gimbal) of the inertial measurement unit to a coordinate system defined by precise optical measurements, (3) conducts limited malfunction isolation for the GN&CS, and (4) computes pertinent navigation information for display to the astronauts. Using information from navigation fixes, the computer determines the amount of deviation from the required trajectory and calculates the necessary attitude and thrust corrective commands. Velocity corrections are measured by the inertial measurement unit and controlled by the computer. During coasting phases of the mission, velocity corrections are not made continuously, but are initiated at predetermined checkpoints.

The computer's memory consists of an erasable and a fixed magnetic core memory with a combined capacity of 38,916 16-bit words. The erasable memory is a coincident-current, ferrite core array with a total capacity of 2,048 words; it is characterized by destructive readout. The fixed memory consists of three magnetic-core rope modules. Each module contains two sections; each section contains 512 magnetic cores. The capacity of each core is 12 words, making a total of 36,864 words in the fixed memory. Readout from the fixed memory is nondestructive.

The logic operations of the computer are mechanized using micrologic elements. in which the necessary resistors are diffused into single silicon wafers. One complete NOR gate, which is the basic building block for all the circuitry, is in a package the size of an aspirin tablet. Flip-flops, registers, counters, etc. are made from these standard NOR elements in different wiring configurations. The computer performs all necessary arithmetic operations by addition, adding two complete words and preparing for the next operation in approximately 24 microseconds. To subtract, the computer adds the complement of the subtrahend. Multiplication is performed by successive additions and shifting; division, by successive addition of complements and shifting.

Functionally, the computer contains a timer, sequence generator, central processor, priority control, an input-output section, and a memory unit.

The timer generates all necessary synchronization pulses to ensure a logical data flow with the LM subsystems. The sequence generator directs the execution of the programs. The central processor performs all arithmetic operations and checks information to and from the computer. Memory stores the computer data and instructions. Priority control establishes a processing priority for operations that must be performed by the computer. The input output section routes and conditions signals between the computer and the other subsystems.

The main functions of the computer are implemented through execution of programs stored in memory. Programs are written in machine language called basic instructions. A basic instruction can be an instruction word or a data word. Instruction words contain a 12-bit address code and a three-bit order code.

The computer operates in an environment in which many parameters and conditions change in a continuous manner. The computer, however, operates in an incremental manner, one item at a time. Therefore, for it to process the parameters, its hardware is time shared. The time sharing is accomplished by assigning priorities to the processing functions. These priorities are used by the computer so that it processes the highest priority processing function first.

In addition, each of the functions has a relative priority with respect to the others; also within each there are a number of processing functions, each having a priority level relative to the other in the group. Most of the processing performed by the computer is in the program controlled processing category. During this processing the computer is controlled by the program stored in its memory.

Real time, which is used in solving guidance and navigation problems, is maintained within the computer's memory. A 745.65-hour (approximately 31 days) clock is provided. The clock is synchronized with ground elapsed time (GET) which is "time zero" at launch This time is transmitted every second by downlink operation for comparison with MSFN elapsed time.

Incremental transmissions occur in the form of pulse bursts from the output channels to the coupling data unit, the gyro fine-alignment electronics, the RCS, and the radars. The number of pulses and the time at which they occur are controlled by the program. Discrete outputs, originating in the output channels under program control, are sent to the DSKY and other subsystems. A continuous pulse train at 1.024 mHz originates in the timing output logic and is sent as a synchronization signal to the timing electronics assembly in the instrumentation Subsystem (IS).

The uplink word from MSFN via the digital up link assembly is supplied as an incremental pulse to the priority control. As this word is received, priority produces the address of the uplink counter in memory and requests the sequence generator to execute the instructions that perform the serial-to-parallel conversion of the input word. When the conversion is completed, the parallel word is transferred to a storage location in memory by the uplink priority program. The uplink priority program also retains the parallel word for subsequent downlink transmission. Another program converts the parallel word to a coded display format and transfers the display information to the DSKY.

The downlink operation is asynchronous with respect to the IS. The IS supplies all the timing signals necessary for the downlink operation.

Through the DSKY, the astronaut can load information into the computer, retrieve and display information contained in the computer, and initiate any program stored in memory. A key code is assigned to each keyboard pushbutton. When a DSKY pushbutton is pressed, the key code is sent to an input channel of the computer. A number of key codes are required to specify an address or a data word. The initiated program also converts the keyboard information to a coded display format, which is transferred by another program to an output channel and to the DSKY for display. The display is a visual indication that the key code was received, decoded, and processed properly.

DISPLAY AND KEYBOARD

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 seven caution indicators, seven status indicators, and three data display indicators. These displays provide visual indications of data being loaded in the computer, the computer's condition and the program being used. The displays also provide the computer with a means of displaying or requesting data.

The caution indicators when on, are yellow; the status indicators, white. The operation and data displays are illuminated green when energized The words "PROG," "VERB," and "NOUN' and the lines separating the three groups of display indicators, and the 19 push buttons of the keyboard are illuminated when the guidance computer is powered-up.

Pushbutton Function

0 through 9 Enters numerical data, noun codes, and verb codes into computer + and - Informs computer that following numerical data are decimal and indicates sign of data.

VERB Indicates to computer that it is going to take some action and conditions computer to interpret the next two numerical characters as a verb code

NOUN Conditions computer to interpret next two numerical characters (noun code) as to what type of action is applied to verb code

CLEAR Clears data contained in data display; pressing this pushbutton clears data display currently being used. Successive pressing clears other two data displays

PRO Commands computer to proceed to standby mode; if in standby mode, commands computer to resume regular operation

KEY REL Releases keyboard displays initiated by keyboard action so that information supplied by computer program may be displayed

ENTR Informs computer that data to be inserted is complete and that requested function is to be executed

RSET Turns off condition indicator lamps after condition has been corrected

The DSKY enables the astronauts to insert data into the guidance computer and to initiate computer operations. The astronauts can also use the keyboard to control the modeling of the inertial subsection. The exchange of data between the astronauts and the computer is usually initiated by an astronaut; however, it can also be initiated by internal computer programs.

The operator of the DSKY can communicate with the computer by pressing a sequence of pushbuttons on the DSKY keyboard. The computer can also initiate a display of information or request the operator for some action, through the processing of its program.

The basic language between the astronaut and the DSKY consists of verb and noun codes. The verb code indicates what action is to be taken (operation). The noun code indicates to what this action is applied (operand). Verb and noun codes may be originated manually or by internal computer sequence. Each verb or noun code contains two numerals. The standard procedure for manual operation involves pressing a sequence of seven pushbuttons:

VERB V1 V2 NOUN N1 N2 ENTR

Pressing the verb pushbutton blanks the verb code display on the display panel and clears the verb code register within the computer. The next two pushbuttons (0 to 9) pressed provide the verb code (V1 and V2). Each numeral of the code is displayed by the verb display as the pushbutton is pressed. The noun pushbutton operates the same as the verb pushbutton, for the noun display and noun code register. The enter pushbutton starts the operation called for. It is not necessary to follow any order in punching in the verb or noun code. It can be done in reverse order, and a previously entered verb or noun may be used without re-punching it.

An error noticed in the verb code or the noun code before pressing the enter pushbutton is corrected by pressing the verb or noun pushbutton and re-punching the erroneous code, without changing the other one. Only when the operator has verified that the desired verb and noun codes are displayed does he press the enter pushbutton.

Decimal data are identified by a plus or minus sign preceding the five digits. If a decimal format is used for loading data, it must be used for all components of the verb. Mixing of decimal and octal data for different components of the same load verb is not permissible. If data are mixed, the OPR ERR condition light goes on.

After any use of the DSKY, the numerals {verb, noun, and data words) remain visible until the next use of the DSKY. If a particular use of the DSKY involves fewer than three data words, the unused data display registers remain unchanged unless blanked by deliberate program action. Some verb-noun codes require additional data to be loaded. If additional data are required after the enter pushbutton is pressed, following the keying of the verb-noun code, the verb and noun displays flash on and off at a 1.5Hz rate. These displays continue to flash until all information associated with the verb-noun code is loaded.

OPERATION UNDER COMPUTER CONTROL

Keyboard operations by the internal computer sequences are the same as those described for manual operation. Computer-initiated verb-noun combinations are displayed as static or flashing displays. A static display identifies data displayed only for astronaut information; no crew response is required. A flashing display calls for appropriate astronaut response as dictated by the verbnoun combination. In this case, the internal sequence is interrupted until the operator responds appropriately, then the flashing stops and the internal sequence resumes. A flashing verb- noun display must receive only one of the proper responses, otherwise, the internal sequence that instructed the display may not resume.

ABORT GUIDANCE SECTION

ABORT SENSOR ASSEMBLY

This assembly contains three floated, pulse-rebalanced, single-degree-of-freedom, rate-integrating gyros and three pendulous reference accelerometers. These six sensors are aligned with the three LM reference ax. and housed in a beryllium block mounted on the navigation base. The assembly is controlled to maintain its internal temperature at +120 degrees F. with external temperatures between -65 degrees and +185 degrees F. This is accomplished by two temperature control circuits, one each for fast warmup and fine temperature control. During fast warmup, temperature can be, raised from 0 degrees to +116 degrees F in 40 minutes. The fine temperature control circuit controls the temperature after +116 degrees F is reached and raises the temperature 4 degrees. This operating temperature (+120 degrees F) is maintained within 0.20 degree F.

DATA ENTRY AND DISPLAY ASSEMBLY

Essentially, 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.

As each numerical pushbutton is pressed, its code is displayed. When the appropriate number of pushbuttons are pressed, the enter or readout pushbutton can be pressed to complete the operation. The logic circuits process octal and decimal data. Octal data consists of a sign and five octal characters Decimal data consists of a sign and five binary-coded decimal characters. The input/ output circuits transfer data to and from the abort electronics assembly (computer). Data transfer occurs when the computer detects the depression of the enter or readout pushbutton.

ABORT ELECTRONICS ASSEMBLY

This assembly is a high-speed, general-purpose computer with special-purpose input/output electronics. It uses a fractional two's complement, parallel arithmetic section and parallel data transfer. Instruction words are 18 bits long they consist of a five-bit order code, an index bit, and a 12-bit operand address. For purposes of explanation, the assembly may be separated into a memory, central computer, and input/output subassembly.

The memory is a coincident-current, parallel, random-access, ferrite core stack with a capacity of 4,096 instruction words. It is divided into two sections: temporary storage and permanent storage. Each section has a capacity of 2,048 instruction words The temporary memory stores replaceable instructions and data. Temporary results may be stored in this memory and may be updated as necessary. The permanent memory stores instructions and constants that are not modified during a mission. The cycle time of the memory is 5 microseconds.

Basically, the central computer consists of eight data and control registers, two timing registers, and associated logic. The data and control registers are interconnected by a parallel data bus. Central computer operations are executed by appropriately timed transfer, controlled by the timing registers, of information between the registers, memory, and input/output subassembly.

The input/output subassembly consists of four basic types of registers: integrator, ripple counter, shift, and static. These registers operate independently of the central computer, except when they are accessed during execution of an input or output instruction. All transfers of data between the central computer and the input-output registers are in parallel.

CONTROL ELECTRONICS SECTION

ATTITUDE CONTROLLER ASSEMBLIES

Each attitude controller assembly supplies attitude rate commands proportional to the displacement of its handle, to the computer and the attitude and translation control assembly; supplies an out-of-detent discrete each time the handle is out of its neutral position; and supplies a followup discrete to the abort guidance section each time the controller is out of detent. A trigger-type push-to-talk switch on the pistol grip handle of the controller assembly is used for communication with the CSM and ground facilities. As the astronaut uses his attitude controller, his 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.

Each assembly consists of position sensing transducers, out-of-detent switches, and limit switches installed about each axis. The transducers provide attitude rate command signals that are proportional to controller displacements. The out-of-detent switches provide pulsed or direct firing of the thrusters when either mode is selected. The limit switches are wired to the secondary solenoid coils of the thrusters. Whenever the controller is displaced to its hard-stops (hard-over position), the limit switches close to provide commands that override automatic attitude control signals from the attitude and translation control assembly.

THRUST/TRANSLATION CONTROLLER ASSEMBLIES

The thrust/translation controller assemblies are functionally integrated translation and thrust controllers. The astronauts use these assemblies to command vehicle translations by firing RCS thruster and to throttle the descent engine between 10% and 92.5% thrust magnitude. The controllers are three axis, T-handle, left-hand controllers; they are mounted with their longitudinal axis approximately 45 degrees from a line parallel to the LM Z-axis (forward axis).

Setting a switch in the LM cabin determines whether the Commander's or LM Pilot's assembly is in command. A lever on the right side of the controller enables the astronaut to select either of two control functions: (1) to control translation in the Y-axis and Z-axis Using the RCS thrusters and throttling of the descent engine to control X-axis translation; and (2) to control translation in all three axes Using the RCS thrusters.

Due to the assembly mounting position, LM translations correspond to astronaut hand movements when operating the controller. Moving the T-handle to the left or right commands translation along the Y-axis. Moving the tee-handle inward or outward commands translation along the Z-axis. Moving the tee handle upward or downward commands translation along the X-axis, using the RCS thrusters when the select lever is in the down position. When the lever is in the up position, upward or downward movement of the controller increases or decreases, respectively, the magnitude of descent engine thrust.

The controller is spring loaded to its neutral position in all axes when the lever is in jets position. When the lever is in the throttle position the Y and Z axes movements are spring loaded to the neutral position but the X-axis throttle commands will remain at the position set by the astronauts.

ATTITUDE AND TRANSLATION CONTROL ASSEMBLY

The attitude and translation control assembly controls LM attitude and translation. In the primary guidance path, attitude and translation commands are generated by the primary guidance computer and applied directly to jet drivers within the assembly. In the abort guidance path, the attitude and translation control assembly receives translation commands from the thrusts translation controller assembly, rate damping signals from the rate gyro assembly, and attitude rate commands and pulse commands from the attitude controller assembly.

The assembly combines attitude and translation commands in its logic network to select the proper thruster to be fired for the desired combination of translation and rotation.

RATE GYRO ASSEMBLY

The rate gyro assembly consists of three single-degree-of-freedom rate gyros mounted so that they sense vehicle roll, pitch, and yaw rates. Each rate gyro senses a rate of turn about its input axis, which is perpendicular to the spin and output axes. The rate of turn is dependent on the gimbal position of the gyro. In abort guidance control, pickoff voltages are routed to the attitude and translation control assembly for rate damping.

DESCENT ENGINE CONTROL ASSEMBLY

The descent engine control assembly accepts engine-on and engine-off commands from the S&C control assemblies, throttle commands from the primary guidance computer and the thrust/ translation controller assembly, and trim commands from the primary guidance computer or the attitude and translation control assembly. Demodulators, comparators, and relay logic circuits convert these inputs to the required descent engine commands. The assembly applies throttle and engine control commands to the descent engine and routes trim commands to the gimbal drive actuators,

Under normal operating conditions with primary guidance in control, the decent engine is manually selected and armed by an astronaut action. The descent engine control assembly responds by routing, through relay logic, 28 volts dc to the actuator isolation solenoids of the descent engine. Once the engine is armed, the assembly receives an automatic descent engine-on command from the primary guidance computer or a descent engine-on command initiated by the Commander pressing the start pushbutton. When the engine is fired, the descent engine control switching and logic latch the engine in the on position until an automatic or manual off command is received by the assembly. When the measured change in velocity reaches a predetermined value, the primary guidance computer generates a descent engine-off command. Manual engine commands are generated by the astronauts and will override the automatic function.

The control assembly accepts manual and automatic throttle commands from the thrust/translation controller assembly and the primary guidance computer, respectively. Manual or automatic thrust control is selected by the astronaut. During manual throttle control, computer throttle commands are interrupted and only manual commands are accepted by the assembly. The astronauts can monitor the response to their manual commands on the thrust indicator. Manual throttle commands consist of 800-Hz ac voltages which are proportional to X-axis displacement of the thrust/translation control assembly. The active controller always provides at least a 10% command. These commands drive a nonlinear circuit to provide the desired thrust level. At an approximately 60% thrust the nonlinear region of the thrust translation controller assembly is reached; it is displaced to its hard stop (92.5% thrust) to prevent erratic descent engine operation.

Automatic throttle increase or decrease commands are generated by the primary guidance computer under program control. These are pre-determined levels of thrust and can be overridden by the astronaut using his thrust controller. No provision is made for automatically throttling the engine, using the abort guidance computer. The automatic commands appear on two separate lines (throttle increase and throttle decrease) as 3,200-Hz pulse inputs to an integrating d-c counter (up-down counter). Each pulse corresponds to a 2.7-pound thrust increment.

During automatic throttle operation, computer-commanded thrust is summed with the output of the thrust/translation controller. When the thrust/translation controller is in its minimum position, the computer-commanded thrust is summed with the fixed 10% output of the controller. When an active controller is displaced from its minimum position, the amount of manual thrust commanded is summed with the computer-commanded thrust to produce the desired resultant. In this case, the controller override the computer's control of decent engine thrust. The total thrust commanded (automatic and/or manual) cannot exceed 92.5%. Automatic thrust commands derived by the computer are always 10% lower than required thrust to compensate for the fixed output of the thrust controller.

Two channels of electronics are provided to control the roll and pitch position of the descent engine thrust vector with respect to the vehicle's center of gravity. When the descent engine is firing, this trim control acts as a low-frequency stabilization system in parallel with the higher frequency RCS. Each channel is driven by either the primary guidance computer when the primary guidance mode is used; by the attitude and translation control assembly when the abort guidance mode is used.

In the primary guidance mode, the computer provides automatic trim control. When the computer determines the required descent engine trim, it provides a trim command to the descent engine control assembly, on a positive or negative trim line for the pitch or roll axis. The trim command is routed to a malfunction logic circuit and to a power- switching circuit, which applies 115-volt, 400-Hz power to the proper gimbal drive actuator. In the abort guidance mode, trim commands are provided by the descent engine control assembly, by using the analog trim signals generated in the pitch and roll error channels of the attitude and translation control assembly.

LANDING RADAR

ELECTRONICS ASSEMBLY

The electronics assembly comprises frequency trackers (one for each velocity beam), a range frequency tracker, velocity converter and computer, range computer, signal data converter, and data good/no-good logic circuit.

ANTENNA ASSEMBLY

The assembly comprises four microwave mixers, four dual audio-frequency preamplifiers, two microwave transmitters, a frequency modulator, and an antenna pedestal tilt mechanism. The antenna consists of six planar arrays: two for transmission and four for reception. They are mounted on the tilt mechanism, beneath the descent stage, and may be placed in one of two fixed positions.

RENDEZVOUS RADAR

ELECTRONICS ASSEMBLY

The electronics assembly comprises a receiver, frequency synthesizer, frequency tracker, range tracker, servo electronics, a signal data converter, self-test circuitry, and a power supply. The assembly furnishes crystal- controlled signals, which drive the antenna assembly transmitter; provide a reference for receiving and processing the return signal; and supplies signals for antenna positioning.

ANTENNA ASSEMBLY

The main portion of the rendezvous radar antenna is a 24-inch parabolic reflector. A 4.65-inch hyperbolic sub-reflector is supported by four converging struts. Before the radar is used, the antenna is manually released from its stowed position. The antenna pedestal and the base of the antenna assembly are mounted on the external structural members of the LM. The antenna pedestal includes rotating assemblies that contain radar components. The rotating assemblies are balanced about a shaft axis and a trunnion axis. The trunnion axis is perpendicular to, and intersects, the shaft axis. The antenna reflectors and the microwave and R F electronics components are assembled at the top of the trunnion axis. This assembly is counterbalanced by the trunnion-axis rotating components (gyroscopes, resolvers, and drive motors) mounted below the shaft axis. Both groups of components, mounted opposite each other on the trunnion axis, revolve about the shaft axis This balanced arrangement requires less driving torque and reduces the overall antenna weight The microwave, radiating, and gimbaling components, and other internally mounted components, have low-frequency flexible cables that connect the outboard antenna components to the inboard electronics assembly.

Main Propulsion

The Main Propulsion Subsystem (MPS) consists of two separate, complete, and independent propulsion sections: the descent propulsion section and the ascent propulsion section. Each propulsion section performs a series of specific tasks during the lunar-landing mission. The descent propulsion section provides propulsion for the LM from the time it separates from the CSM until it lands on the lunar surface, the ascent propulsion section lifts the ascent stage off the lunar surface and boosts it into orbit. Both propulsion sections operate in conjunction with the Reaction Control Subsystem (RCS), which provides propulsion used mainly for precise attitude and translation maneuvers. The ascent propellant tanks are connected to the RCS to supplement its propellant supply during certain mission phases. If a mission abort becomes necessary during the descent trajectory, the ascent or descent engine can be used to return to a rendezvous orbit with the CSM. The choice of engines depends on the cause for abort, the amount of propellant remaining in the descent stage, and the length of time that the descent engine had been firing.

Each propulsion section consists of a liquid propellant, pressure-fed rocket engine and propellant pressurization, and feed components. For reliability, many vital components in each section are redundant. In both propulsion sections, pressurized helium forces the hypergolic propellants from the tanks to the engine injector. Both engine assemblies have control valves that start and stop a metered propellant flow to the combustion chamber upon command, trim orifices, an injector that determines the spray pattern of the propellants as they enter the combustion chamber, and a combustion chamber, where the propellants meet and ignite. The gases produced by combustion pass through a throat area into the engine nozzle, where they expand at an extremely high velocity before being ejected. The momentum of the exhaust gases produces the reactive force that propels the vehicle.

The more complicated tasks required of the descent propulsion section-such as propelling the entire LM and hovering over the lunar surface while the astronauts select a landing site-dictate that the decent propulsion section be the larger and more sophisticated of the two propulsion sections. It has a propellant supply that is more than three times that of the ascent propulsion section. The descent engine is almost twice as large as the ascent engine, produces more thrust almost 10,000 pounds at full throttle), is throttleable for thrust control, and is gimbaled (can be tilted) for thrust vector control. The ascent engine, which cannot be tilted, delivers a fixed thrust of 3,500 pounds, sufficient to launch the ascent stage from the lunar surface and place it into a predetermined orbit.

The primary characteristics demanded of the LM propellants are high performance per weight storability over long periods without undue vaporization or pressure buildup; hypergolicity for easy, closely spaced engine starts; no shock sensitivity; freezing and boiling points within controllable extremes: and chemical stability. The ascent and descent propulsion sections, as well as the RCS, use identical fuel/oxidizer combinations. In the ascent and descent propulsion sections, the injection ratio of oxidizer to fuel is approximately 1.6 to 1, by weight.

The fuel is a blend of hydrazine (N2H4) and unsymmetrical dimethylhydrazine (UDMH), commercially known as Aerozine 50. The proportions, by weight, are approximately 50% hydrazine, and 50% dimethylhydrazine.

The oxidizer is nitrogen tetroxide (N2O4). it has a minimum purity of 99.5% and a maximum water content of 0.1%.

The astronauts monitor the performance and status of the MPS with their panel-mounted pressure, temperature, and quantity indicators; talkbacks (flags indicating open or closed position of vital valves); and caution and warning annunciators (placarded lights that go on when specific out of tolerance conditions occur). These data, originating at sensors and position switches in the MPS, are processed in the Instrumentation Subsystem, and are simultaneously displayed to the astronauts in the LM cabin and transmitted to mission controllers through MSFN via the Communications Subsystem. The MPS obtains 28-volt d-c and 115-volt a-c primary power from the Electrical Power Subsystem.

Before starting either engine, the propellants must be settled to the bottom of the tanks. Under weightless conditions, this requires an ullage maneuver; that is, the LM must be moved in the +X, or upward, direction. To perform this maneuver, an astronaut or the automatic guidance equipment operates the downward-firing thrusters of the RCS.

The MPS is operated by the Guidance, Navigation, and Control Subsystem (GN&CS), which issues automatic (and processes manually initiated) on and off commands to the descent or ascent engine. The GN&CS also furnishes gimbal-drive and thrust-level commands to the descent propulsion section.

DESCENT ENGINE OPERATION AND CONTROL

After initial pressurization of the descent propulsion section, the descent engine start requires two separate and distinct operations: arming and firing. Engine arming is performed by the astronauts; engine firing can be performed by the astronauts, or it can be automatically initiated by the LM guidance computer. When the astronauts set a switch to arm the descent engine, power is simultaneously routed to open the actuator isolation valves in the descent engine, enable the instrumentation circuits in the descent propulsion section, and issue a command to the throttling controls to start the descent at the required 10% thrust level. The LM guidance computer and the abort guidance section receive an engine-armed status signal. This signal enables an automatic engine-on program in the GN&CS, resulting in a descent engine start. A manual start is accomplished when the Commander pushes his engine start pushbutton. (Either astronaut can stop the engine because separate engine-stop pushbuttons are provided at both flight stations.)

The normal start profile for all descent engine starts must be at 10% throttle setting. Because the thrust vector at engine start may not be directed through the LM center of gravity, a low-thrust start (10%) will permit corrective gimbaling. If the engine is started at high thrust, RCS propellants must be used to stabilize the LM.

The astronauts can, with panel controls, select automatic or manual throttle control modes and Commander or LM Pilot thrust/translation controller authority, and can override automatic engine operation. Redundant circuits, under astronaut control, ensure descent engine operation if prime control circuits fail.

Signals from the GN&CS automatically control descent engine gimbal trim a maximum of 6 degrees from the center position in the Y- and Z-axes to compensate for center-of-gravity offsets during descent engine firing. This ensures that the thrust vector passes through the LM center of gravity. The astronauts can control the gimbaling only to the extent that they can interrupt the tilt capability of the descent engine which they would do if a caution light indicates that the gimbal drive actuators are not following the gimbal commands.

DESCENT PROPULSION SECTION FUNCTIONAL DESCRIPTION

The descent propulsion section consists of an ambient and supercritical helium tank 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 as required by the mission. At the full-throttle position, the engine develops a nominal thrust of 9,900 pounds; it can also be operated within a range of 1,280 to 6,400 pounds of thrust. Functionally, the descent propulsion section can be subdivided into a pressurization section, a propellant feed section, and an engine assembly.

PRESSURIZATION SECTION

Before earth launch, all the LM propellant tanks are only partly pressurized (less than 230 psia), so that the tanks will be maintained within a safe pressure level under the temperature changes experienced during launch and earth orbit. Before initial engine start, the ullage space in each propellant tank requires additional pressurization. This initial pressurization is accomplished with a relatively small amount of helium stored at ambient temperature and at an intermediate pressure. To open the path from the ambient helium tank to the propellant tanks, the astronauts fire three explosive valves: an ambient helium isolation valve and the two propellant compatibility valves that prevent backflow of propellant vapors from degrading upstream components. After flowing through a filter, the ambient helium enters a pressure regulator which reduces the helium pressure to approximately 245 psi. The regulated helium then enters parallel paths which lead through quadruple check valves into the propellant tanks, The quadruple check valves, consisting of four valves in a series-parallel arrangement, permit flow in one direction only. This protects upstream components against corrosive propellant vapors and prevents hyperbolic action due to backflow from the propellant tanks.

After initial pressurization, supercritical helium is used to pressurize the propellants The supercritical helium tank is isolated by an explosive valve, which is automatically fired 1.3 seconds after the descent engine is started. The time delay prevents the supercritical helium from entering the fuel/helium heat exchanger until propellant flow is established so that the fuel cannot freeze in the heat exchanger. After the explosive valve opens, the supercritical helium enters the two-pass fuel/ helium heat exchanger where it is slightly warmed by the fuel. The helium then flows back into a heat exchanger in the supercritical helium tank where it increases the temperature of the supercritical helium in the tank, causing a pressure rise and ensuring continuous expulsion of helium throughout the entire period of operation. Finally, the helium flows through the second loop of the fuel/ helium heat exchanger where it is heated to operational temperature before it is regulated and routed to the propellant tanks.

The system that reduces the helium pressure consists of two parallel, redundant regulators. If one pressure regulator fails, the astronauts close the malfunctioning line and open the redundant line, to restore normal propellant tank pressurization.

Each propellant tank is protected against over-pressurization by a relief valve, which opens at approximately 260 psia and reseats after over-pressurization is relieved. A thrust neutralizer prevents the gas from generating unidirectional thrust. Each relief valve is paralleled by two series-connected vent valves, which are operated by panel switches. After landing, the astronauts relieve pressure buildup in the tanks, caused by rising temperatures, to prevent uncontrolled venting through the relief valves. The fuel and oxidizer fumes are vented separately; supercritical helium is vented at the same time.

PROPELLANT FEED SECTION

The descent section propellant supply is contained in two fuel tanks and two oxidizer tanks. Each pair of like propellant tanks is manifolded into a common delivery line.

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 line during negative-g acceleration. 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. Each delivery line contains a trim orifice and a woven, stainless-steel wire-mesh filter. The trim orifices provide engine inlet pressure of approximately 222 psia at full throttle position. The filters prevent debris, originating at the explosive valves or in the propellant tanks, from contaminating downstream components.

ENGINE ASSEMBLY

The descent engine is mounted in the center compartment of the descent stage cruciform. Fuel and oxidizer entering the engine assembly are routed through flow control valves to the propellant shutoff valves. A total of eight propellant shutoff valves are used; they are arranged in series-parallel redundancy, four in the fuel line and four in the oxidizer line. The series redundancy ensures engine shutoff, should one valve fail to close. The parallel redundancy ensures engine start, should one valve fail to open.

To prevent rough engine starts, the engine is designed to allow the oxidizer to reach the injector first. The propellants are then injected into the combustion chamber, where hypergolic action occurs.

The propellant shutoff valves are actuator operated. The actuation line branches off the main fuel line at the engine inlet and passes through the parallel-redundant actuator isolation valves to four solenoid-operated pilot valves. From the pilot valves, the fuel enters the hydraulically operated actuators, which open the propellant shutoff valves. The actuator pistons are connected to rack-and-pinion linkages that rotate the balls of the shutoff valves 90 degrees to the open position. The actuator isolation valves open when the astronauts arm the descent engine. When an engine-on command is initiated, the four pilot valves open simultaneously, permitting the actuation fuel to open the propellant shutoff ball valves, thus routing fuel and oxidizer to the combustion chamber.

The flow control valves, in conjunction with the adjustable orifice sleeve in the injector, control the descent thrust. At full throttle, and during the momentary transition from full throttle to the 65% range, throttling take place primarily in the injector and, to a lesser degree, in the flow control valves.. Below the 65% thrust level, the propellant-metering function is entirely controlled by the flow control valves. The flow control valves and the injector sleeve are adjusted simultaneously by a mechanical linkage. Throttling is controlled by the throttle valve actuator, which positions the linkage in response to electrical input signals.

The fuel and oxidizer are injected into the combustion chamber at velocities and angles compatible with variations in weight flow. The fuel is emitted in the form of a thin cylindrical sheet; the oxidizer sprays break up the fuel stream and establish the injection pattern at all thrust settings. Some fuel is tapped off upstream of the injector and is routed through a trim orifice into the barrier coolant manifold. From here, it is sprayed against the combustion chamber wall through fixed orifices, maintaining the chamber wall at an acceptable temperature.

DESCENT PROPULSION SECTION EQUIPMENT

SUPERCRITICAL HELIUM TANK

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 10 psi per hour maximum), the initial loading pressure is planned so that the supercritical helium will be maintained within a safe pressure/time envelope throughout the mission.

The supercritical helium tank is double walled; it consists of an inner spherical tank and an outer jacket. The void between the tank and the jacket is filled with aluminized mylar insulation and evacuated to minimize ambient heat transfer into the tank. The vessel has fill and vent ports, a burst disk assembly, and an internal helium/helium heat exchanger. The inner tank is initially vented and loaded with cryogenic liquid helium at approximately 8 degrees R (- 452 degrees F) at a pressure of 14.7 psia. The cryogenic liquid becomes supercritical helium when the fill sequence is completed by closing the vent and introducing a high-pressure head of gaseous helium. As the high-pressure, low- temperature gas is introduced, the density and pressure of the cryogenic liquid helium are increased. At the end of pressurization, the density of the stored supercritical helium is approximately 8.7 pounds per cubic foot and the final pressure is approximately 178 psi.

The burst disk assembly prevents hazardous over pressurization within the vessel. It consists of two burst disks in series, with a normally open, low-pressure vent valve between the disks. The burst disks are identical; they burst at a pressure between 1,881 and 1,967 psid to vent the entire supercritical helium supply overboard. A thrust neutralizer at the outlet of the downstream burst disk diverts the escaping gas into opposite directions to prevent unidirectional thrust generation. The vent valve prevents low-pressure buildup between the burst disks if the upstream burst disk leaks slightly. The valve is open at pressures below 150 psia; it closes when the pressure exceeds 150 psia.

FUEL/HELIUM HEAT EXCHANGER

Fuel is routed directly from the fuel tanks to the two-pass fuel/helium heat exchanger, where heat from the fuel is transferred to the supercritical helium. The helium reaches operating temperature after flowing through the second heat exchanger passage. The fuel/helium heat exchanger is of finned tube construction; the first and second helium passages are in parallel crossflow with respect to the fuel. Helium flows in the tubes and fuel flows in the outer shell across the bundle of staggered, straight tubes.

PROPELLANT STORAGE TANKS

The propellant supply is contained in four cylindrical, spherical-ended titanium tanks of identical size and construction. Two tanks contain fuel; the other two, oxidizer. Each pair of tanks containing like propellants is interconnected at the top and all propellant lines downstream of the tanks contain trim orifices, to ensure balanced propellant flow. A diffuser at the helium inlet port (top) of each tank distribute the pressurizing helium uniformly into the tank. An antivortex device in the form of a series of vanes, at each tank outlet, prevents the propellant from swirling into the outlet port, thus precluding inadvertent helium ingestion into the engine. Each tank outlet also has a propellant retention device (negative-g can) that permits unrestricted propellant flow from the tank under normal pressurization, but blocks reverse propellant flow (from the outlet line back into the tank) Under zero-g or negative g conditions. This arrangement ensures that helium does not enter the propellant outlet line as a result of a negative-g or zero-g condition or propellant vortexing; it eliminates the possibility of engine malfunction due to helium ingestion.

PROPELLANT QUANTITY GAGING SYSTEM

The propellant quantity gaging system enables the astronauts to monitor the quantity of propellants remaining in the four descent tanks. The propellant quantity gaging system consists of four quantity-sensing probes with low-level sensors (one for each tank), a control unit, two quantity indicators that display remaining fuel and oxidizer quantity, a switch that permits the astronauts to select a set of tanks (one fuel and one oxidizer) to be monitored, and a descent propellant quantity low-level warning light. The low-level sensors provide a discrete signal to cause the warning light to go on when the propellant level in any tank is down to 9.4 inches (equivalent to 5.6% propellant remaining). When this warning light goes on, the quantity of propellant remaining is sufficient for only 2 minutes of engine burn at hover thrust (approximately 25%).

PROPELLANT SHUTOFF VALVE ASSEMBLIES

Each of the four propellant shutoff valve assemblies consists of a fuel shutoff valve, an oxidizer shutoff valve, a pilot valve, and a shutoff valve actuator. The shutoff valve actuator and the fuel shutoff valve are in a common housing. The four solenoid-operated pilot valves control the fuel that is used as actuation fluid to open the fuel shutoff valves. The oxidizer shutoff valve is actuated by a mechanical linkage driven from the fuel shutoff valve. When the pilot valves are opened, the actuation fluid flow (at approximately 110 psia) acts against the spring-loaded actuator plunger, opening the shutoff valves. When the engine-firing signal is removed, the pilot valves close and seal off the actuation fluid. The propellant shutoff valves are closed by the return action of the actuator piston springs, which expels the fuel entrapped in the cylinders and valve passages through the pilot valve vent port.

The propellant shutoff valves are ball valves. The ball element operates against a spring-loaded soft seat to ensure positive sealing when the valve is closed. The individual valves are rotated by a rack-and-pinion-gear arrangement, which translates the linear displacement of the pistons in the shutoff valve actuators.

THROTTLE VALVE ACTUATOR

The throttle valve actuator is a linear-motion electromechanical servo-actuator which moves the throttle linkage in response to an electrical input command. Moving the throttle linkage simultaneously changes the position of the flow control valve pintles and the injector sleeve, thereby varying the amount of fuel and oxidizer metered into the engine and changing the magnitude of engine thrust. The throttle valve actuator is located between the fuel and oxidizer flow control valves; its housing is rigidly attached to the engine head end and its output shaft is attached to the throttle linkage.

The actuator is controlled by three redundant electronic channels, which power three d-c torque motors. The motor shafts supply the input to a ball screw, which converts rotary motion to the lineal motion of the throttle valve actuator output shaft. All mechanical moving parts of the actuator are within a hermetically sealed portion of the unit, pressurized to 0.25 psia with a 9 to 1 mixture of nitrogen and helium. A leak indicator in the cover provides visual evidence of loss of vacuum within the unit. Five potentiometers are ganged to the torque motor shaft through a single- stage planetary reduction gear. Three of these potentiometers supply position feedback information to the three motor amplifier channels, one to each channel. The other two potentiometers provide throttle actuate shaft position data for telemetry to MSFN. The redundancy within the throttle valve actuator ensures that failure of any electrical component will not cause the actuator to fail. The throttle valve actuator also provides a fail-safe system in the event elective external to the throttle valve actuator occur. If either the primary 28- volt d-c power or the command voltage is lost, the throttle valve actuator causes the descent engine to thrust automatically at full throttle.

FLOW CONTROL VALVES

The oxidizer and fuel flow control valves are on the side of the engine, immediately downstream of the propellant inlet lines They are secured to the throttle valve actuator mounting bracket. The flow control valve pintle assemblies are mechanically linked to the throttle valve actuator by a crossbeam.

The flow control valves are non-redundant cavitating venturis with movable pintle sleeves. Engine throttling is initiated by an electrical signal to the throttle valve actuator, commanding an increase or decrease in engine thrust. Operation of the throttle valve actuator changes the position of the pintles in the flow control valves. This axial movement of the pintles decreases or increases the pintle flow areas to control propellant flow rate and thrust. Below an approximate 70% thrust setting, flow through the valves cavitates, and hydraulically uncouples the propellant transfer system (and thereby, the flow rate) from variations in combustion chamber pressure. In the throttling range between 65% and 92.5% thrust, 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.

VARIABLE-AREA INJECTOR

The variable area injector consists of a pintle assembly, drive assembly, and manifold assembly The pintle assembly introduces the propellant uniformly into the combustion chamber. The drive assembly has a twofold function: first. it serves as a passage for conducting the oxidizer into the pintle assembly; second, it contains the bearing and sealing components that permit accurate positioning of the injector sleeve. The injector sleeve varies the injection area so that near-optimum injector pressure drops and propellant velocities are maintained at each thrust level. The primary function of the manifold assembly is to distribute the fuel uniformly around the outer surface of the sleeve. Fuel enters the manifold assembly at two locations and is passed through a series of distribution plates near the outer diameter of the assembly.

At the center of the manifold, the fuel passes through a series of holes before it is admitted into a narrow passage formed by the manifold body and a faceplate. The passage smoothes out gross fuel discontinuities and assists in cooling the injector face. The fuel then passes onto the outer surface of the sleeve, past a fuel-metering lip. The fuel is injected in the form of a hollow cylinder so that it reaches the impingement zone with a uniform circumferential velocity profile and without atomizing, at all flow rates. The oxidizer is injected through a double-slotted sleeve so that it forms a large number of radial filaments. Each filament partially penetrates the fuel cylinder and is enfolded by fuel in such a way that little separation of oxidizer and fuel can occur. For given propellant densities, overall mixture ratio, and injector geometry, there is a range of propellant injection velocity ratios that result in maximum mixture ratio uniformity throughout the resultant expanding propellant spray. When they occur, the liquid-phase reactions generate gas and vapor that atomize and distribute the remaining liquid oxidizer and fuel uniformly in all directions, resulting in high combustion efficiency.

COMBUSTION CHAMBER AND NOZZLE EXTENSION

The combustion chamber consists of an ablative-cooled chamber section, nozzle throat, and nozzle divergent section. The ablative sections are enclosed in a continuous titanium shell and jacketed in a thermal blanket composed of aluminized nickel foil and glass wool. A seal prevents leakage between the combustion chamber and nozzle extension.

The nozzle extension is a radiation-cooled, crushable skirt: it can collapse a distance of 28 inches on lunar impact so as not to affect the stability of the LM. The nozzle extension is made of columbium coated with aluminide. It is attached to the combustion chamber case at a nozzle area ratio of 16 to 1 and extends to an exit ratio of 54.0 to 1.

GIMBAL RING AND GIMBAL DRIVE ACTUATORS

The gimbal ring is located at the plane of the combustion chamber throat. It consists of a rectangular beam frame and four trunnion subassemblies. The gimbal drive actuators under control of the descent engine control assembly, tilt the descent engine in the gimbal ring along the pitch and roll axes so that the engine thrust vector goes through the LM center of gravity. One actuator controls the pitch gimbal the other, the roll gimbal. The gimbal drive actuators consist of a single-phase motor. a feedback potentiometer, and associated mechanical devices. They can extend or retract 2 inches from the mid-position to tilt the descent engine a maximum of 6 degrees along the Y-axis and Z- axis.

ASCENT ENGINE OPERATION AND CONTROL

Shortly before initial ascent engine use, the astronauts fire explosive valves to pressurize the ascent propulsion section. The ascent engine, like the descent engine, requires manual arming before it can be fired. When the astronauts arm the ascent engine, a shutoff command is sent to the descent engine. Then, enabling signals are sent to the ascent engine control circuitry to permit a manual or computer-initiated ascent engine start. For manual engine on and off commands, the astronauts push the same start and stop pushbuttons used for the descent engine. For automatic commands, the stabilization and control assemblies in the GN&CS provide sequential control of LM staging and ascent engine on and off commands The initial ascent engine firing Ä whether for normal lift-off from the lunar surface or in-flight abort Ä is a fire-in-the-hole (FITH) operation: that is, the engine fires while the ascent and descent stags are still mated although no longer mechanically secured to each other. If, during the descent trajectory, an abort situation necessitates using the ascent engine to return to the CSM, the astronauts abort stage sequence. This results in an immediate descent engine shutdown followed by a time delay to ensure that the engine has stopped thrusting before staging occurs. The next command automatically pressurizes the ascent propellant tanks, after which the staging command is issued. This results in severing of hardware that secures the ascent stage to the descent stage and the interconnecting cables. The ascent engine fire command completes the abort stage sequence.

ASCENT PROPULSION SECTION FUNCTIONAL DESCRIPTION

The ascent propulsion section consists of a constant thrust. pressure-fed rocket engine, one fuel and one oxidizer tank, two helium tanks, and associated propellant feed and helium pressurization components. The engine develops 3,500 pounds of thrust in a vacuum, it can be shut down and restarted, as required by the mission. Like the descent propulsion section, the ascent propulsion section can functionally be sub-divided into a pressurization section, a propellant feed section, and an engine assembly.

PRESSURIZATION SECTION

Before initial ascent engine start, the propellant tanks must be fully pressurized with gaseous helium. This helium is stored in two identical tanks at a nominal pressure of 3,050 psia at a temperature of +70 degrees F. An explosive valve at the outlet of each helium tank prevents the helium from leaving the tanks until shortly before initial ascent engine use. To open the helium paths to the propellant tanks, the astronauts normally fire six explosive valves simultaneously: two helium isolation valves and four propellant compatibility valves (two connected in parallel for redundancy in each pressurization path). Before firing the explosive valves, the astronauts check the pressure in each helium tank. If one tank provides an unusually low reading (indicating leakage), they can exclude the appropriate helium isolation explosive valve from the fire command. This will isolate the faulty tank from the pressurization system and will prevent helium loss through the leaking tank via the helium interconnect line.

Downstream of the interconnect line, the helium flows into the primary and secondary regulating paths, each containing a filter, a normally open solenoid valve and two series connected pressure regulators.

Two downstream regulators are set to a slightly higher output pressure than the upstream regulators; 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 pressurize the propellant tanks. If an upstream regulator fails open, control is obtained through the downstream regulator in the same flow path. If both regulators in the same flow path fail open, pressure in the helium manifold increases above the acceptable limit of 220 psia, causing a caution light to go on. This advises the astronauts that they must identify the failed-open regulators and close the helium isolation solenoid valve in the malfunctioning flow path so that normal pressure can be restored.

Downstream of the regulators, a 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. 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 over pressurization This assembly vents pressure in excess of approximately 245 psia and reseals the flow path after over pressurization is relieved. A thrust neutralizer eliminates unidirectional thrust generated by the escaping gas.

PROPELLANT FEED SECTION

The ascent propulsion section has one oxidizer tank and one fuel tank. Transducers in each tank enable the astronauts to monitor propellant temperature and ullage pressure. A caution light, activated by a low-level sensor in each tank warn the astronauts when the propellant supply has diminished to an amount sufficient for only 10 seconds of engine operation.

Helium flows into the top of the propellant tanks, where diffusers uniformly distribute it throughout the ullage space. The outflow from each propellant tank divides into two paths. The primary path routes each propellant through a trim orifice and a filter to the propellant shutoff valves in the engine assembly. The trim orifice provides an engine inlet pressure of 170 psia for proper propellant use. The secondary path connects the ascent propellant supply to the RCS. This interconnection permits the RCS to burn ascent propellants, providing the ascent tanks are pressurized and the ascent or descent engine is operating when the RCS thrusters are fired, A line branches off the RCS interconnect fuel path and leads to two parallel actuator isolation solenoid valves This line routes fuel to the engine pilot valves that actuate the propellant shutoff valves.

ENGINE ASSEMBLY

The ascent engine is installed in the midsection of the ascent stage; it is tilted so that its centerline is 1.5 degrees from the X-axis, in the +Z-direction. Fuel and oxidizer entering the engine assembly are routed, through the filters, propellant shutoff valves, and trim orifices, to the injector. The propellants are injected into the combustion chamber, where the hypergolic ignition occurs. A separate fuel path leads from the actuator isolation valves to the pilot valves. The fuel in this line enters the actuators, which open the propellant shutoff valves.

Propellant flow into the combustion chamber is controlled by a valve package assembly, trim orifices, and the injector. The valve package assembly is similar to the propellant shutoff valve assemblies in the descent engine. The eight propellant shutoff valves are arranged in series-parallel redundant fuel oxidizer pairs. Each pair is operated from a single crankshaft by its actuator.

When an engine-start command is received, the two actuator isolation valves and the four pilot valves open simultaneously. Fuel then flows through the actuator pressure line and the four pilot valves into the actuator chambers. Hydraulic pressure extends the actuator pistons, cranking the propellant shutoff valves 90 degrees to the fully open position. The propellants now flow through the shutoff valves and a final set of trim orifices to the injector. The orifices trim the pressure differentials of the fuel and oxidizer to determine the mixture ratio of the propellants. The physical characteristics of the injector establish an oxidizer lead of approximately 50 milliseconds. This precludes the possibility of a fuel lead which could result in a rough engine start.

At engine shutdown, the actuator isolation valves are closed, preventing additional fuel from reaching the pilot valves. Simultaneously, the pilot valve solenoids are de-energized. 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 forced back by spring pressure, cranking the propellant shutoff valves to the closed position.

ASCENT PROPULSION SECTION EQUIPMENT

HELIUM PRESSURE REGULATOR ASSEMBLIES

Each helium pressure regulator assembly consists of two individual pressure regulators connected in series. The downstream regulator functions in the same manner as the upstream regulator; however, it is set to produce a higher outlet pressure so that it becomes a secondary unit that will only be in control if the upstream regulator (primary unit) fails open. Each pressure regulator unit consists of a direct-sensing main stage and a pilot stage. The valve in the main stage is controlled by the valve in the pilot stage which senses small changes the regulator outlet pressure and converts these changes to proportionally large changes in control pressure. A rise in outlet pressure decreases the pilot valve output, thereby reducing flow into the main stage chamber. An increase in the downstream demand causes a reduction in outlet pressure; this tends to open the pilot valve. The resultant increase in control pressure causes the main stage valve poppet to open, thus meeting the increased downstream demand.

A flow limiter at the outlet of the main stage valve of the secondary unit restricts maximum flow through the regulator assembly to 5.5 pounds of helium per minute, so that the propellant tanks are protected if the regulator fails open The filter at the inlet of the primary unit prevents particles, which could cause excessive leakage at lockup, from reaching the regulator assembly.

PROPELLANT STORAGE TANKS

The propellant supply is contained in two spherical titanium tanks. The tanks are of identical size and construction. One tank contains fuel; the other, oxidizer. A helium diffuser at the inlet port of each tank distributes the pressurizing helium uniformly into the tank. An antivortex device (a cruciform at each tank outlet) prevents the propellant from swirling into the outlet port, precluding helium ingestion into the engine. Each tank outlet also has a propellant- retention device that permits unrestricted propellant flow from the tank under normal pressurization, but blocks reverse propellant flow (from the outlet line back into the tank) under zero-g or negative-g conditions. This arrangement ensures that helium does not enter the propellant outlet line while the engine is not firing; it eliminates the possibility of engine malfunction due to helium ingestion. A low-level sensor in each tank (approximately 4.4 inches above the tank bottom) supplies a discrete signal that causes a caution light to go on when the propellant remaining in either tank is sufficient for approximately 10 seconds of burn time (48 pounds of fuel, 69 pounds of oxidizer).

VALVE PACKAGE ASSEMBLY

At the propellant feed section/engine assembly interface, the oxidizer and fuel lines lead into the valve package assembly. The individual valves that make up the valve package assembly are in a series-parallel arrangement to provide redundant propellant flow paths and shutoff capability. The valve package assembly consists of eight propellant shutoff valves and four solenoid-operated pilot valve and actuator assemblies. valve assembly consists of one fuel shutoff valve and one oxidizer shutoff valve These are ball valves that are operated by a common shaft. which is connected to its respective pilot valve and actuator assembly. Shaft seals and vented cavities prevent the propellants from coming into contact with each other. Separate overboard vent manifold assemblies drain the fuel and oxidizer that leaks past the valve seals, and the actuation fluid (fuel in the actuators when the pilot valves close), overboard. The eight shutoff valves open simultaneously to permit propellant flow to the engine while it is operating; they close simultaneously to terminate propellant flow at engine shutdown. The four non-latching, solenoid-operated pilot valves control the actuation fluid (fuel).

INJECTOR ASSEMBLY

The injector assembly consists of the propellant inlet lines, a fuel manifold, a fuel reservoir chamber, an oxidizer manifold. and an injector orifice plate assembly. Because it takes longer to fill the fuel manifold and reservoir chamber assembly, the oxidizer reaches the combustion chamber approximately 50 milliseconds before the fuel, resulting in smooth engine starts. The injector orifice plate assembly is of the fixed orifice type, which uses a baffle and a series of perimeter slots (acoustic cavities) for damping induced combustion disturbances. The baffle is Y shaped, with a 120 degree angle between each blade. The baffle is cooled by the propellants, which subsequently enter the combustion chamber through orifices on the baffle blades. The injector face is divided into two combustion zones: primary and baffle. The primary zone uses impinging doublets (one fuel and one oxidizer), which are spaced in concentric radial rings on the injector face The baffle zone (1.75 inches below the injector face) uses impinging doublets placed at an angle to the injector face radius. The combustion chamber wall is cooled by spraying fuel against it through canted orifices, spaced around the perimeter of the injector. The nominal temperature of the propellant is +70 degrees F as it enters the injector; with the fuel temperature within 10ø of the oxidizer temperature. The temperature range at engine start may be +40 degrees to 500 degrees F.

COMBUSTION CHAMBER ASSEMBLY AND NOZZLE EXTENSION

The combustion chamber assembly consists of the engine case and mount assembly and an ablative material (plastic) assembly, which includes the nozzle extension. The two assemblies are bonded and locked together to form an integral unit. The plastic assembly provides ablative cooling for the combustion chamber; it consists of the chamber ablative material, the chamber insulator, the nozzle extension ablative material, and a structural filament winding. The chamber ablative material extends from the injector to an expansion ratio of 4.6. The chamber insulator, between the ablative material and the case, maintains the chamber skin temperature within design requirements. The ablative material of the nozzle extension extends from the expansion ratio of 4.6 to 45.6 (exit plane) and provides ablative cooling in this region. The structural filament winding provides structural support for the plastic assembly and ties the chamber and nozzle extension sections together.

Reaction Control

The Reaction Control Subsystem (RCS) provides thrust impulses that stabilize the LM during the descent and ascent trajectory and controls attitude and translation movement of the LM about and along its three axes during hover, landing, rendezvous, and docking maneuvers. The RCS also provides the thrust required to separate the LM from the CSM and the +X-axis acceleration (ullage maneuver) required to settle Main Propulsion Subsystem (MPS) propellants before a descent or ascent engine start. The RCS accomplishes its task during coasting periods or while the descent or ascent engine is firing: it operates in response to automatic control commands from the Guidance, Navigation, and Control Subsystem (GN&CS) or manual commands from the astronauts.

The 16 thrust chamber assemblies (thrusters) and the propellant and helium sections that comprise the RCS are located in or on the ascent stage. The propellants used in the RCS are identical with those used in the MPS. The fuel-Aerozine 50-is a mixture of approximately 50% each of hydrazine and unsymmetrical dimethylhydrazine. The oxidizer is nitrogen tetroxide. The injection ratio of oxidizer to fuel is approximately 2 to 1. The propellants are hypergolic; that is, they ignite spontaneously when they come in contact with each other.

The thrusters are small rocket engines, each capable of delivering 100 pounds of thrust. They are arranged in clusters of four, mounted on four outriggers equally spaced around the ascent stage. In each cluster, two thrusters are mounted parallel to the LM X-axis, facing in opposite directions; the other two are spaced 90 degrees apart, in a plane normal to the X-axis and parallel to the Y-axis and Z-axis.

The RCS is made up of two parallel, independent systems (A and B), which, under normal conditions, function together to provide complete attitude and translation control. Each system consists of eight thrusters, a helium pressurization section, and a propellant feed section. The two systems are interconnected by a normally closed cross feed arrangement that enables the astronauts to operate all 16 thrusters from a single propellant supply. Complete attitude and translation control is therefore available even if one system's propellant supply is depleted or fails. Functioning alone, either RCS system can control the LM, although with slightly reduced efficiency. This capability is due to the distribution of the thrusters, because each cluster has two thrusters of each system located in a relatively different position.

In addition to the RCS propellant supply, the thrusters can use propellants from the ascent propulsion section. This method of feeding the thrusters, which requires the astronauts to open interconnect lines between the ascent tanks and RCS manifolds, is normally used only during periods of ascent engine thrusting. Use of ascent propulsion section propellants is intended to conserve RCS propellants, which may be needed during docking maneuvers.

The astronauts monitor performance and status of the RCS with their panel-mounted pressure, temperature, and quantity indicators; talkbacks (flags, that indicate open or closed position of certain valves); and caution and warning annunciators (placarded lights that go on when specific out-of-tolerance conditions occur). These data originate at sensors and position switches in the RCS, are processed in the Instrumentation Subsystem, and are simultaneously displayed to the astronauts in the LM cabin and transmitted to mission controllers through MSFN via the Communications Subsystem.

The 28-volt d-c and 115-volt a-c primary power required by the RCS is furnished by the Electrical Power Subsystem. Interconnect plumbing between the RCS thruster propellant manifolds and the ascent propulsion section tanks permit the RCS to use propellants from the Main Propulsion Subsystem (MPS) during certain phases of the mission.

Control of the RCS is provided by the GN&CS. Modes of operation, thruster selection, and firing duration are determined by the GN&CS.

FUNCTIONAL DESCRIPTION

THRUSTER SELECTION, OPERATION, AND CONTROL

The GN&CS provides commands that select thrusters and fine them for durations ranging from a short pulse to steady-state operation The thrusters can be operated in an automatic mode, attitude-hold mode, or a manual override mode.

Normally, the RCS operates in the automatic mode; all navigation, guidance, stabilization, and steering functions are initiated and commanded by the LM guidance computer (primary guidance and navigation section) or the abort electronics assembly (abort guidance section).

The attitude-hold mode is a semiautomatic mode in which either astronaut can institute attitude and translation changes. When an astronaut displaces his attitude controller, an impulse proportional to the amount of displacement is routed to the computer, where it is used to perform steering calculations and to generate the appropriate thruster- on command. An input into the DSKY determines whether the computer commands an angular rate change proportional to attitude controller displacement, or a minimum impulse each time the controller is displaced. When the astronaut returns his attitude controller to the neutral (detent) position, the computer issues a command to maintain attitude. For a translation maneuver, either astronaut displaces his thrust/translation controller. This sends a discrete to the computer to issue a thruster-on command to selected thrusters When this controller is returned to neutral, the thrusters cease to fire.

If the abort guidance section is in control, attitude errors are summed with the proportional rate commands from the attitude controller and a rate-damping signal from the rate gyro assembly. The abort guidance equipment uses this data to perform steering calculations, which result in specific thruster-on commands. The astronauts can select two or four X-axis thrusters for translation maneuvers, and they can inhibit the four upward-firing thrusters during the ascent thrust phase, thus conserving propellants. In the manual mode, the four-jet hard-over maneuver, instituted when either astronaut displaces his attitude controller fully against the hard stop, fires four thrusters simultaneously, overriding any automatic commands.

For the MPS ullage maneuver, the astronauts select whether two or four downward-firing thrusters should be used. Depending on which guidance section is in control, the astronauts enter a DSKY input (primary) or use a 2-jet/4-jet selector switch (abort) to make their selection. Under manual control, a +X-translation pushbutton fires the four downward-firing thrusters continuously until the pushbutton is released. Firing two thrusters conserves RCS propellants; however, it takes longer to settle the MPS propellants.

RCS OPERATION

Functionally, the RCS can be subdivided into pressurization sections, propellant feed sections, and thruster sections Because RCS systems A and B are identical, only one system is described.

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 solenoid valves at each thruster pair. Before separation of the LM from the CSM, the astronauts set switches on the control panel to preheat the thrusters and fire explosive valves to pressurize the propellant tanks. Gaseous helium, reduced to a working pressure, enters the propellant tanks and forces the fuel and oxidizer to the thrusters. Here, the propellants are blocked by fuel and oxidizer valves that remain closed until a thruster-on command is issued. As the selected thruster receives the fire command, its fuel and oxidizer valves open to route the propellants through an injector into the combustion chamber. where they impinge and ignite by hypergolic action. The astronauts can disable malfunctioning thrusters by operating appropriate LGC thruster pair command switches on the control panel. When any of these switches is in the disable position, it issues a signal informing the LM guidance computer that the related thruster pair is disabled and that alternate thrusters must be selected. Talkbacks above each switch informs the astronauts of the status of related thruster pair.

PRESSURIZATION SECTION

The RCS propellants are pressurized with high-pressure gaseous helium, stored at ambient temperature. The helium tank outlet is sealed by parallel-connected, redundant helium isolation explosive valves that maintain the helium in the tank until the astronauts enter the LM and prepare the RCS for operation

When the explosive valves are fired, helium enters the pressurization line and flows through a filter. A restrictor orifice, downstream of the filter, dampens the initial helium surge.

Downstream of the restrictor, the flow path contains a pair of pressure regulators connected in series. 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 is in control and provides proper propellant tank pressurization.

Downstream of the pressure regulators, a manifold divides the helium flow into two paths: one leads to the oxidizer tank; the other, to the fuel tank. Each flow has quadruple check valves that permit flow in one direction only, thus preventing backflow of propellant vapors if seepage occurs in the propellant tank bladders. A relief valve assembly protects each propellant tank against over-pressurization. If helium pressure builds up to 232 psia, the relief valve opens to relieve pressure by venting helium overboard. At 212 psia, the relief valve closes.

PROPELLANT FEED SECTION

Fuel and oxidizer are contained in flexible bladders in the propellant tanks. Helium routed into the void between the bladder and the tank wall squeezes the bladder to positively expel the propellant under zero-gravity conditions. The propellants flow through normally open main shutoff valves into separate fuel and oxidizer manifolds that lead to the thrusters. A switch on the control panel enables the astronauts to simultaneously close a pair of fuel and oxidizer main shutoff valves, thereby isolating a system's propellant tanks from its thrusters, if the propellants of that system are depleted or if the system malfunctions. After shutting off one system, the astronauts can restore operation of all 16 thrusters by opening the cross-feed valves between the system A and B manifolds.

During ascent engine firing, the astronauts may open the normally closed ascent propulsion section/RCS interconnect lines if the LM is accelerating in the +X-axis (upward) direction; closing the interconnect lines shortly before ascent engine shutdown ensures that no ascent helium enters the RCS propellant lines. Control panel switches open the interconnect valves in fuel-oxidizer pairs, for an individual RCS system, or for both systems simultaneously.

Transducers in the propellant tanks sense helium pressure and fuel temperature. Due to the proximity of the fuel tank to the oxidizer tank, the fuel temperature is representative of propellant temperature. Quantity indicators for system A and B display the summed quantities of fuel and oxidizer remaining in the tanks.

THRUSTER SECTION

Each of the four RCS clusters consists of a frame, four thrusters, eight heating elements, and associated sensors and plumbing. The clusters are diametrically opposed, evenly distributed around the ascent stage. The frame is an aluminum-alloy casting, shaped like a hollow cylinder, to which the four thrusters are attached; the entire cluster assembly is connected to the ascent stage by hollow struts. The vertical-firing thrusters are at the top and bottom of the cluster frame, the horizontal-firing thrusters are at each side. Each cluster is enclosed in a thermal shield; part of the four thruster combustion chambers and the extension nozzles protrude from the shield. The thermal shields aid in maintaining a temperature-controlled environment for the propellant lines from the ascent stage to the thrusters, minimize heat loss, and reflect radiated engine heat and solar heat.

The RCS thrusters are radiation-cooled, pressure-fed, bipropellant rocket engines that operate in a pulse mode to generate short thrust impulses for fine attitude corrections (navigation alignment maneuvers) or in a steady-state mode to produce continuous thrust for major attitude or translation changes. In the pulse mode, the thrusters are fired intermittently in bursts of less than 1 second duration-the minimum pulse may be as short as 14 milliseconds -however, the thrust level does not build up to the full 100 pounds that each thruster can produce. In the steady- state mode, the thrusters are fired continuously (longer than 1 second) to produce a stabilized 100 pounds of thrust until the shutoff command is received.

Two electric heaters, which encircle the thruster injector, control propellant temperature by conducting heat to the combustion chamber and the propellant solenoid valves. The heaters maintain the cluster at approximately +140 degrees F. ensuring that the combustion chambers are properly preheated for instantaneous thruster starts. The astronauts can determine, by use of a temperature indicator and a related selector switch, if a cluster temperature is below the minimum operational temperature of 119 degrees F and take corrective action to restore the cluster temperature.

Propellants are prevented from entering the thrusters by dual-coil, solenoid-operated shutoff valves at the fuel and oxidizer inlet ports. These valves are normally closed; they open when an automatic or a manual command energizes the primary or secondary coil, respectively. Seven milliseconds after receiving the thruster-on command, the valves are fully opened and the pressurized propellants flow through the injector into the combustion chamber where ignition occurs. By design, the fuel valve opens 2 milliseconds before the oxidizer valve, to provide proper ignition characteristics. Orifice at the valve inlets meter the propellant flow so that an oxidizer to fuel mixture ratio of 2 to 1 is obtained at the injector.

As the propellants mix and burn, the hot combustion gases increase the chamber pressure. accelerating the gas particles through the chamber exit. The gases are expanded through the divergent section of the nozzle at supersonic velocity, eventually building up to reach a reactive force of 100 pounds of thrust in the vacuum of space. The gas temperature within the combustion chamber stabilizes at approximately 5,200 degrees F. The temperature at the non-ablative chamber wall is maintained at a nominal 2,200 degrees F by a combined method of film cooling (a fuel stream sprayed against the wall) and radiation cooling (dissipation of heat from the wall surface into space).

When the thruster-off command is received, the coils in the propellant valves re-energize, and spring pressure closes the valves. Propellant trapped in the injector is ejected and burned for a short time, while thrust decays to zero pounds.

When a thruster-on signal commands a very short duration pulse, engine thrust may be just beginning to rim when the pulse is ended and the propellant valves close Under these conditions, the thrusters do not develop the full- capacity thrust of 100 pounds.

A failure detection system informs the astronauts should a thruster fail on (fires without an on command) or off (does not fire despite an on command). Either type of failure produces the same indication: a warning light goes on and the talkback related to the failed thruster pair changes effects of a thruster-on failure, opposing thrusters will automatically receive fire commands and keep firing until the failed-on thruster has been disabled. A thruster-off condition is detected by a pressure switch, which senses combustion chamber pressure. When a fire command is received, the solenoid valves of the thruster open, resulting in ignition and subsequently in pressure buildup in the combustion chamber. When the pressure reaches 10.5 psia, the switch closes, indicating that proper firing is in process. When a very short duration fire command is received (a pulse of less than 80 milliseconds). the combustion chamber pressure may not build up enough for a proper firing. Short pulse skipping does not result in a failure indication, unless six consecutive pulses to the same thruster have not produced a response In this case, the warning light and the talkback inform the astronauts that they have a nonfiring thruster, which must be isolated.

EQUIPMENT

EXPLOSIVE VALVES

The explosive valves are single-cartridge-actuated, normally closed valves. The cartridge is fired by applying power to the initiator bridgework. The resultant heat fires the initiator, generating gases in the valve explosion chamber at an extremely high rate. The gases drive the valve piston into the housing, aligning the piston port permanently with the helium pressurization line.

PROPELLANT QUANTITY MEASURING DEVICE

The propellant quantity measuring device, consisting of a helium pressure/temperature probe and an analog computer for each system, measures the total quantity of propellants (sum of fuel and oxidizer) in the fuel and oxidizer tanks. The output voltage of the analog computer is fed to an indicator and is displayed to the astronauts on two scales (one for each RCS system) as percentage of propellant remaining in the tanks.

The propellant quantity measuring device uses a probe to sense the pressure/temperature ratio of the gas in the helium tank. This ratio, directly proportional to the mass of the gas, is fed to an analog computer that subtracts the mass in the helium tank from the total mass in the system, thereby deriving the helium mass in the propellant tanks. Finally, propellant tank ullage volume is subtracted from total tank volume to obtain the quantity of propellant remaining. Before firing the helium isolation explosive valves, the quantity displayed exceeds 100%, so that, after the valves are opened and the gas in the helium tank becomes less dense, the indicated quantity will be 100%.

PROPELLANT STORAGE TANKS

The four propellant tanks, one fuel and one oxidizer tank for each system, are cylindrical with hemispherical ends; they are made of titanium alloy. In each tank, the propellant is stored in a Teflon bladder, which is chemically inert and resistant to the corrosive action of the propellants. The bladder is supported by a standpipe running lengthwise in the tank. The propellant is fed into the tank from a fill point accessible from the exterior of the LM. A bleed line that extends up through the standpipe draws off gases trapped in the bladder. Helium flows between the bladder and the tank wall and acts upon the bladder to provide positive propellant expulsion.

The efficiency of a rocket engine is expressed in terms of specific impulse, which is the impulse-producing capacity per unit weight of propellant The nominal specific impulse of the RCS thrusters at steady-state firing is 281 seconds. The thrusters have a favorable high-thrust to minimum-impulse ratio, meaning that they produce a comparatively high thrust for their size, as well as a very low thrust impulse. In addition, the thrusters have a fast response time. Response time is the elapsed time between a thruster-on command and stable firing at rated thrust, and between a thruster-off command and thrust decay to an insignificant value. Finally, the thrusters have a long cycle life, denoting that the thrusters can be restarted many times.

Each thruster consists of a fuel valve, an oxidizer valve, an injector head assembly, a combustion chamber, an extension nozzle, and thruster instrumentation.

The fuel and oxidizer valves are normally closed, two coil, solenoid valves that control propellant flow to the injector. Each valve has an inlet filter, an inlet orifice, a spool assembly, a spring, an armature, and a valve seat. The primary and the secondary coils are wound on a magnetic core in the spool assembly. These coils receive the thruster on and off commands. The fuel and oxidizer valves are identical except for the inlet orifice, the valve seat, and the spool assembly. Because the ratio of oxidizer to fuel at the combustion chamber must be approximately 2 to 1, the diameters of the inlet orifices and the valve seat exits differ in the two valves. The spool assembly in the fuel valve produces a faster armature response to open the fuel valve 2 milliseconds before the oxidizer valve. Permitting fuel to enter the combustion chamber first reduces the possibility of ignition delay, which could cause temporary over pressurization (spiking) in the combustion chamber. Spiking is also held to a minimum by preheating and pre--pressurizing the combustion chamber.

When the thruster-off command is given, the coils re-energize, releasing the armature poppets. Spring and propellant pressure return the armature poppet of each valve to its seat, shutting off propellant flow into the injector.

The injector head assembly supports the fuel and oxidizer valves and the mounting flange for the combustion chamber. The propellant impingement and chamber cooling arrangement in the injector consists of four concentric orifice rings and a pre-igniter cup. initial combustion occurs in the pre-igniter cup (a precombustion chamber) where a single fuel spray and oxidizer stream impinge. This provides a smoother start transient because it raises the main combustion chamber pressure for satisfactory ignition. The main fuel flow is routed through holes in a tube to a chamber that channels the fuel to an annulus. The annulus routes fuel to three concentric fuel rings. The outermost ring sprays fuel onto the combustion chamber wall, where it forms a boundary layer for cooling. The middle ring has eight orifices that spray fuel onto the outer wall of the pre-igniter cup to cool the cup. Eight primary orifices of the middle ring eject fuel to mix with the oxidizer. The main oxidizer flow is routed through holes in the oxidizer pre-igniter tube, to a chamber that supplies the eight primary oxidizer orifices of the innermost ring. The primary oxidizer and fuel orifices are arranged in doublets, at angles to each other, so that the emerging propellant streams impinge. Due to the hydraulic delay built into the injector, ignition at these eight doublets occurs approximately 4 milliseconds later than ignition inside the pre-igniter cup.

The combustion chamber is made of machined molybdenum, coated with silicon to prevent oxidation of the base metal. The chamber is cooled by radiation and by a film of fuel vapor. The extension nozzle is fabricated from L605 cobalt base alloy; eight stiffening rings are machined around its outer surface to maintain nozzle shape at high temperatures. The combustion chamber and extension nozzle are joined together by a large coupling nut and lockring.

HEATERS

Two redundant, independently operating heating systems are used simultaneously to heat the RCS clusters. Two electric heaters, one from each system, encircle the injector area of each thruster. The heaters normally operate in an automatic mode; redundant thermal switches (two connected in parallel for each thruster) sense injector temperature and turn the heaters on and off to maintain the temperature close to +140 degrees F. The heaters of the primary heating system are powered directly from their circuit breakers. Power to the redundant system is routed through switches that permit the astronauts to operate this system for each cluster individually, either under automatic thermal switch control or with heaters continuously on, or off.

Electrical Power

The Electrical Power Subsystem (EPS) is the principal source of electrical power necessary for the operation of the LM. The electrical power is supplied by seven silver-zinc batteries: five in the descent stage and two in the ascent stage. The batteries provide dc for the EPS d-c section; two solid-state inverters supply the a-c section. Both sections supply operating power to respective electrical buses, which supply all LM subsystems through circuit breakers. Other batteries supply power to trigger explosive devices, to operate the portable life support system, and to operate scientific equipment.

The descent stage batteries power the LM from T-30 minutes until the docked phases of the mission, at which time the LM receives electrical power from the CSM. After separation from the CSM, during the powered descent phase of the mission, the descent stage batteries am paralleled with the ascent stage batteries. Paralleling the batteries ensures the minimum required voltage for all possible LM operations Before lift-off from the lunar surface, ascent stage battery power is introduced, descent battery power is terminated, and descent battery feeder lines are dead faced and severed. Ascent stage battery power is then used until after final docking and astronaut transfer to the CM. The batteries are controlled and protected by electrical control assemblies, a relay junction box, and a dead face relay box, in conjunction with the control and display panel.

In addition to being the primary source of electrical power for the LM during the mission, the EPS is the distribution point for externally generated power during prelaunch and docked operations. Prelaunch d-c and a-c power is initially supplied from external ground power supplies until approximately T-7 hours. At this time, the vehicle ground power supply unit is removed and d-c power from the launch umbilical tower is connected. From launch until LM-CSM transition and docking, the EPS distributes internally generated d-c power. After docking, LM power is shut down and the CSM supplies d-c power to the LM. Before LM-CSM separation, all LM internally supplied electrical power is restored.

FUNCTIONAL DESCRIPTION

The outputs of the five descent stage batteries and two ascent stage batteries are applied to four electrical control assemblies. The two descent stage electrical control assemblies provide an independent control circuit for each descent battery. The two ascent stage electrical control assemblies provide four independent battery control circuits, two control circuits for each ascent battery. The electrical control assembly monitors reverse-current, overcurrent, and overtemperature within each battery. Each battery control circuit can detect a bus or feeder short. If an overcurrent condition occurs in a descent or ascent battery, the control circuit operates a main feed contactor associated with the malfunctioning battery to remove the battery from the distribution system.

Ascent and descent battery main power feeders are routed through circuit breakers to the d c buses. From them buses, power is distributed through circuit breakers to all LM subsystems. The two inverters, which make up the a-c section power source, are connected to either of two a-c buses. Either inverter, when selected. can supply the LM a- c requirements.

Throughout the mission, the astronauts monitor the primary a-c and d-c voltage levels, d-c current levels, and the status of all main power feeders. The electrical power control and indicator panel in the cabin has talkbacks that indicate main power feeder status, indicators that display battery and bus voltages and currents, and component caution lights. The component caution lights are used to detect low bus voltages, out-of-limit, a-c bus frequencies, and battery malfunctions. Backup a-c and d-c power permits the astronauts to disconnect, substitute, or reconnect batteries, feeder lines, buses, or inverters to assure a continuous electrical supply.

EQUIPMENT

DESCENT STAGE BATTERIES

Each battery is composed of silver-zinc plates, with a potassium hydroxide electrolyte. Each battery has 20 cells, weighs 135 pounds. and has a 415 ampere hour capacity (approximately 25 amperes at 28 volts dc for 16 hours, at +80 degrees F). Normally, the descent stage batteries are paralleled so that they discharge equally. The batteries can operate in a vacuum while cooled by an Environmental Control Subsystem (ECS) cold rail assembly to which the battery heat sink surface is mounted. Five thermal sensors monitor cell temperature limits (+145 degrees +/-5 degrees F) within each battery; they cause a caution light to go on to alert the astronaut to a battery over-temperature condition. The batteries initially have high- voltage characteristics; a low-voltage tap is provided (at the 17th cell) for use from T-30 minutes through transposition and docking. The high-voltage tap is used for all other normal LM operations. If one descent stage battery fails, the remaining descent stage batteries can provide sufficient power.

ASCENT STAGE BATTERIES

The two ascent stage batteries are identical. Each battery is composed of silver-zinc plates, with a potassium hydroxide electrolyte. Each battery weighs 125 pounds, and has a 296-ampere-hour capacity (50 amperes at 28 volts for 5.9 hours, at +80 degrees F). To provide independent battery systems, the batteries are normally not paralleled during the ascent phase of the mission. The batteries can operate in a vacuum while cooled by ECS cold rails to which the battery heat sink surface is mounted. The nominal operating temperature of the batteries is approximately +80 degrees F. Battery temperature in excess of +145 degrees +/-5 degrees F closes a thermal sensor. causing a caution light to go on. The astronaut then takes corrective action to disconnect the faulty battery. The batteries ordinarily supply the d-c power requirements. from normal staging to final docking of the ascent stage with the orbiting CSM or during any malfunction that requires separation of the ascent and descent stages. If one ascent stage battery fails, the remaining battery provides sufficient power to accomplish safe rendezvous and docking with the CSM during any part of the mission.

DESCENT STAGE ELECTRICAL CONTROL ASSEMBLIES

The two descent stage electrical control assemblies control and protect the descent stage batteries. Each assembly has a set of control circuits for each battery accommodated. A failure in one set of battery control circuits does not affect the other set. The protective circuits of the assembly automatically disconnect a descent stage battery if an overcurrent condition occurs and cause a caution light to go on if a battery over-current, reverse-current, or over- temperature condition is detected.

The major elements of each assembly are high and low-voltage main feed contactors, current monitors, overcurrent relays, reverse-current relays, and power supplies. An auxiliary relay supplies system logic contact closures to other control assemblies in the LM power distribution system.

The reverse current relay causes a caution light to go on when current flow in the direction opposite to normal current flow exceeds 10 amperes for at least 4 seconds. Unlike the overcurrent relay, the reverse-current relay does not open the related main feed contactor and is self-resetting when the current monitor ceases to detect a reverse- current condition. During reverse-current conditions, the related contactor must be manually switched open. The control assembly power supplies provide ac for current-monitor excitation and regulated dc for the other circuits.

ASCENT STAGE ELECTRICAL CONTROL ASSEMBLIES

The two ascent stage electrical control assemblies individually control and protect the two ascent stage batteries in nearly the same manner as the descent stage control assemblies. Each assembly contains electrical power feed contactors, an over-current relay, a reverse-current relay, and a current monitor. Each ascent stage battery can be connected to its normal or backup main feeder line via the normal or the backup main feed contactor in its respective assembly. Both batteries are thereby connected to the primary d-c power buses. The normal feeder line has overcurrent protection; the backup feeder line does not.

RELAY JUNCTION BOX

The relay junction box provides the following:

Control logic and junction points for connecting external prelaunch power (via the launch umbilical tower) to the LM Pilot's d-c bus

Control and power junction points for connecting descent stage and ascent stage electrical control assemblies to the LM Pilot's d-c bus

Dead-facing (electrical isolation) of half of the power feeders between the descent and ascent stages.

The relay junction box controls the low-voltage contactors of batteries 1 and 4 (on and off) from the launch umbilical tower and CSM, and all low and high-voltage descent power contactors (off) on receipt of an abort stage command. The junction box includes abort logic relays, which, when energized by an abort stage command, close the ascent stage battery main feed contactors and open the dead-face main feed contactors and dead-face relays. The dead-face relay is manually opened and closed or automatically opened when the abort logic relays close. The dead-face relay in the junction box dead-faces half of the main power feeders between the descent and ascent stages; the other half of the power feeders is dead-faced by the dead-face relay in the dead-face relay box. The ascent stage then provides primary d-c power to the LM.

DEAD-FACE RELAY BOX

The dead-face relay box dead-faces those power feeders that are not controlled by the relay junction box, in the same manner as the relay junction box. Two individual dead-facing facilities (28 volts for each circuit breaker panel) are provided.

INVERTERS

Two identical redundant, 400 Hz inverters individually supply the primary a-c power required in the LM. Inverter output is derived from a 28-volt d-c input. The output of the inverter stage is controlled by 400-Hz pulp drives developed from a 6.4-kilopulse-per-second (kps) oscillator, which is, in turn, synchronized by timing pulses from the Instrumentation Subsystem. An electronic tap changer sequentially selects the output of the tapped transformer in the inserter stage, convening the 400-Hz square wave to an approximate sine wave of the same frequency. A voltage regulator maintains the inverter output at 115 volts ac during normal load conditions by controlling the amplitude of a dc-to-dc converter output. The voltage regulator also compensates for variations in the d-c input and a-c output load. When the voltage at a bus is less than 112 volts ac, or the frequency is less than 398 Hz or more than 402 Hz, a caution light goes on. The light goes off when the malfunction is remedied.

CIRCUIT BREAKER AND EPS CONTROL PANELS

All primary a-c and d-c power feed circuits are protected by circuit breakers on the Commander's and LM Pilot's buses. The two d c buses are electrically connected by the main power feeder network. Functionally redundant LM equipment is placed on both d-c buses (one on each bus), so that each bus can individually perform a mission abort.

SENSOR POWER FUSE ASSEMBLIES

Two sensor power fuse assemblies, in the aft equipment bay, provide a secondary d-c bus system that supplies excitation to transducers in other subsystems that develop display and telemetry data. During prelaunch procedures, primary power is supplied to the assemblies from the Commander's 28 volt d-c bus Before launch, power from the launch umbilical tower is disconnected, and power is subsequently available to the sensor power fuse assemblies from the LM Pilot's 28-volt d-c bus. Each assembly comprises a positive d-c bus, negative return bus, and 40 fuses. All sensor return lines are routed to a common ground bus.

Communications

The Communications Subsystem (CS) provides in-flight and lunar surface communications links between the LM and CSM, the LM and MSFN, and the LM and the extravehicular astronaut (EVA). When both astronauts are outside the LM, the LM relays communications between the astronauts and MSFN. When the astronauts are in the Lunar Roving Vehicle (LRV), the Lunar Communications Relay Unit (LCRU), mounted on the LRV, is the communications relay. The CS consists of S-band and VHF equipment.

IN-FLIGHT COMMUNICATIONS

In flight, when the LM is separated from the CSM and is on the earth side of the moon, the CS provides S-band communications with MSFN and VHF communications with the CSM. When the LM and the CSM are on the far side of the moon, VHF is used for communications between them.

EARTH SIDE (LM-MSFN)

In-flight S band communications between the LM and MSFN include voice, digital uplink signals, and ranging code signals from MSFN The LM S-band equipment transmits voice, acts as transponder to the ranging code signals, transmits biomedical and systems telemetry data, and provides a voice backup capability and an emergency key capability.

S-band voice is the primary means of communication between MSFN and the LM Backup voice communication from MSFN is possible, using the digital uplink assembly, but this unit is normally used by the MSFN to update the LM guidance computer. In response to ranging code signals sent to the LM, the S-band equipment supplies MSFN with a return ranging code signal that enables MSFN to track, and determine the range of the LM. The LM transmits biomedical data pertinent to astronaut heartbeat so that MSFN can monitor and record the physical condition of the astronauts. The LM also transmits systems telemetry data for MSFN evaluation; voice, using redundant S-band equipment; and, in case there is no LM voice capability, provides an emergency key signal so that the astronauts can transmit Morse code to MSFN.

EARTH SIDE (LM-CSM)

In-flight VHF communications between the LM and CSM include voice, backup voice, and tracking and ranging signals. Normal LM-CSM voice communications use VHF channels A and B duplex. Backup voice communication is accomplished with VHF channel B simplex or channel A simplex VHF ranging, initiated by the CSM, uses VHF channels A and B duplex.

FAR SIDE (LM-CSM)

When the LM and CSM are behind the moon, contact with MSFN is not possible VHF channels A and B are used for duplex LM-CSM voice communications. VHF channel B is used as a one-way data link to transmit system telemetry signals from the LM, to be recorded and stored by the CSM. When the CSM establishes S-band contact with MSFN, the stored data are transmitted by the CSM at 32 times the recording speed.

LUNAR SURFACE COMMUNICATIONS

When the LM is on the lunar surface, the CS provides S-band communications with MSFN and VHF communications with the EVA. The LM relays VHF signals to MSFN, using the S-band.

Communications with the CSM may be accomplished by using MSFN as a relay. LM-MSFN S-band capabilities are the same as in-flight capabilities, except that, in addition, TV may be transmitted from the lunar surface in an FM mode.

FUNCTIONAL DESCRIPTION

Each astronaut has his own audio center. The audio centers have audio amplifiers and switches that are used to route signals between the LM astronauts, and between the LM and MSFN or the CSM. The centers are redundant in that each one can be Used by either astronaut, or both astronauts can use either audio center if necessary.

In a transmission mode, the output of the audio centers goes to the VHF transceivers, or to the pre-modulation processor or to the data storage electronics assembly in the Instrumentation Subsystem (IS) If an audio center output is routed to the VHF transmitter, the transmission is through the diplexer to the selected VHF antenna. If an audio center output is routed to the pre-modulation processor (PMP) and then to the S-band transceivers, the transmitter output is applied to the diplexer, or to the S-band power amplifier, depending on power output requirements. The output from the transmitter or the power amplifier goes through the diplexer to the selected S- band antenna. If an audio center output is routed to the data storage electronics assembly, the voice transmission is recorded.

The inputs to the S-band transceivers are from the pre-modulation processor or the television camera. The outputs from the pre-modulation processor (to be transmitted by S-band transmitters) are processed voice, and PCM, EMU and biomed data. For television transmission, the 5 band power amplifier is used In normal flight, and on the lunar surface, the steerable antenna is used. When the LRV is in use, transmission is through the S-band antenna mounted on it. The S-band omni antennas are used in any one of a number of backup modes.

External RF inputs to the S-band equipment are MSFN voice, either uplink data or an uplink backup voice signal, and ranging. Received MSFN voice is routed through the pre-modulation processor to the audio centers. Received uplink data signals are routed to the digital uplink assembly to be decoded and sent to the LM guidance computer. MSFN backup voice is routed to the digital uplink assembly where it is decoded and then sent to the Commander's microphone amplifier input

EQUIPMENT

S-BAND TRANSCEIVER

The S-band transceiver assembly provides deep-space communications between the LM and MSFN S-band communications consist of voice and pseudorandom noise ranging transmission from MSFN to the LM and voice, pseudorandom noise ranging turnaround, biomed, and subsystem data transmission from the LM to MSFN. The assembly consists of two identical phase-locked receivers, two phase modulators with driver and multiplier chains, and a frequency modulator. The receivers and phase modulators provide the ranging, voice, emergency-keying, and telemetry transmit-receive functions. The frequency modulator is primarily provided for video transmission, but accommodates pulse-code-modulation telemetry (subsystem data), biomed, and voice transmission. The frequency modulator provides limited backup for both phase modulators. The operating frequencies of the S-band equipment are 2282.5 mHz (transmit) and 2101.8 mHz (receive).

S-BAND POWER AMPLIFIER

The S-band power amplifier amplifies the S-band transmitter output when additional transmitted power is required. This assembly consists of two amplitrons, an input and an output isolator (ferrite circulators), and two power supplies, all mounted on a common chassis. The RF circuit is a series interconnection of the isolators and amplitrons. The amplitrons (which are characteristic of saturated, rather than linear, amplifiers) have broad bandwidth, high efficiency, high peak and average power output, but relatively low gain. The isolators protect both amplitrons and both S-band transmitter driver and multiplier chains. The isolators exhibit a minimum isolation of 20 db and a maximum insertion loss of 0.6 db. Each amplitron has its own power supply, One amplitron is designated primary; the other, secondary. Only one amplitron can be activated at a time. When neither amplitron is selected, a feedthrough path through the power amplifier exists with maximum insertion loss of 3.2 db (feedthrough mode).

VHF TRANSCEIVER

The VHF transceiver assembly provides voice communications between the LM and the CSM and, during blackout of transmission to MSFN, low-bit-rate telemetry transmission from the LM to the CSM, and ranging on the LM by the CSM. When the LM mission profile includes extravehicular activity, this equipment also provides EVA LM voice communications, and reception of EVA biomed and suit data for transmission to MSFN over the S-band The assembly consists of two solid-state superheterodyne receivers and two transmitters. One transmitter-receiver combination provides a 296.8-mHz channel (channel A); the other, a 259.7-mHz channel (channel B), for simplex or duplex voice communications. Channel B may also be used to transmit pulse code modulation data from the IS at the low bit rate and to receive biomed and suit data from the EVA during EVA-programmed missions.

SIGNAL PROCESSOR ASSEMBLY

The signal processor assembly is the common acquisition and distribution point for most CS received and transmitted data, except that low-bitrate, split-phase data are directly coupled to VHF channel B and TV signals are directly coupled to the S-band transmitter. The signal processor assembly processes voice and biomed signals and provides the interface between the RF electronics, data storage electronics assembly, and pulse code modulation and timing electronics assembly of the IS. The signal processor assembly consists of an audio center for each astronaut and a pre-modulation processor. The signal processor assembly does not handle ranging and uplink data signals. The pre-modulation processor provides signal modulation, mixing, and switching in accordance with the selected mode of operation. It also permits the LM to be used as a relay station between the CSM and MSFN, and, for EVA-programmed missions, between the EVA and MSFN. The audio centers are identical They provide individual selection, isolation, and amplification of audio signals received by the CS receivers and which are to be transmitted by the CS transmitters Each audio center contains a microphone amplifier, headset amplifier, voice operated relay (VOX) circuit, diode switches, volume control circuits, and isolation pads. The VOX circuit controls the microphone amplifier by activating it only when required for voice transmission. Audio signals are routed to and from the VHF A, VHF B. and S-band equipment’s and the intercom bus via the audio centers The intercom bus, common to both audio centers, provides hardline communications between the astronauts. Voice signals to be recorded by the data storage electronics assembly are taken from the intercom bus.

DIGITAL UPLINK ASSEMBLY

The digital uplink assembly decodes S-band uplink commands from MSFN and routes the data to the LM guidance computer. The digital uplink assembly provides a verification signal to the IS for transmission to MSFN, to indicate that the uplink messages have been received and properly decoded by the digital uplink assembly. The LM guidance computer also routes a no-go signal to the IS for transmission, to MSFN whenever the computer receives an incorrect message. The uplink commands addressed to the LM parallel those inputs available to the LM guidance computer via the display and keyboard assembly. The digital uplink assembly also provides a voice backup capability if the received S-band audio circuits in the pre-modulation processor fail.

RANGING TONE TRANSFER ASSEMBLY

The ranging tone transfer assembly operates with VHF receiver B and VHF transmitter A to provide a transponder function for CSM-LM VHF ranging. The ranging tone transfer assembly receives VHF ranging tone inputs from VHF receiver B and produces ranging tone outputs to key VHF transmitter A.

The VHF ranging tone input consists of two acquisition tone signals and one track tone signal. Accurate ranging is accomplished when the track tone signal from the CSM is received and retransmitted from the LM.

S-BAND STEERABLE ANTENNA

The S-band steerable antenna is a 26-inch diameter parabolic reflector with a point source feed that consists of a pair of cross-sleeved dipoles over a ground plane. The prime purpose of this antenna is to provide deep-space voice and telemetry communications and deep-space tracking and ranging. This antenna provides 174 degree azimuth coverage and 330 degree elevation coverage. The antenna can be operated manually or automatically. The manual mode is used for initial positioning of the antenna to orient it within +/-12 5 degrees (capture angle) of the line-of-sight signal received from the MSFN station. Once the antenna is positioned within the capture angle, it can operate in the automatic mode.

S-BAND IN-FLIGHT ANTENNAS

The two S band in-flight antennas are omnidirectional; one is forward and one is aft on the LM. The antennas are right-hand circularly polarized radiators that collectively cover 90% of the sphere at -3 db or better. They operate at 2282.5 mHz (transmit) and 2101 8 mHz (receive). These antennas are the primary S-band antennas for the LM when in flight.

VHF IN-FLIGHT ANTENNAS

The two VHF in-flight antennas are omni- directional, right-hand, circularly polarized antennas that operate at 259.7 and 296.8-mHz.

VHF EVA ANTENNA

The VHF EVA antenna is an omnidirectional conical antenna, which is used for LM-EVA communications when the LM is on the lunar surface. It is mounted on the LM and unstowed by an astronaut in the LM after landing.

Instrumentation

The Instrumentation Subsystem (IS) monitors the LM subsystems, performs in-flight checkout, prepares LM status data for transmission, provides timing frequencies and correlated data for LM subsystems, stores voice and time- correlation data, performs lunar surface checkout, and provides scientific instrumentation for lunar experiments.

The IS monitors various parameters (status) of LM subsystems and structure and prepares the status data for telemetering via the Communications Subsystem (CS), to MSFN. In a high-bitrate mode of operation, MSFN receives 51,200 bits of information from 279 subsystem sensors every second. This, along with Guidance, Navigation, and Control Subsystem data, enables mission controllers to participate in major decisions, assist in spacecraft management during complicated astronaut activity, and maintain a detail subsystem performance history.

Caution and warning lights and two master alarm lights alert the astronauts to out-of-tolerance conditions (malfunctions) that affect the mission or their safety. In addition a 3-kHz alarm tone is routed to the astronaut headsets to advise the astronauts that a malfunction exists. The tone is especially helpful in alerting the astronauts when they are preoccupied or asleep. The master alarm lights can be turned off by pushing either illuminated lens; this also stops the tone. When a warning light (red) goes on, it indicates a malfunction that affects the mission, but could affect astronaut safety if not corrected.

FUNCTIONAL DESCRIPTION

The IS consists of subsystem sensors, a signal-conditioning electronics assembly, a pulse-code modulation and timing electronics assembly, a caution and warning electronics assembly, and a data storage electronics assembly.

The sensors continuously monitor the status of LM subsystems and provide outputs indicative of temperature, pressure, frequency, gas and liquid quantity, stage-separation distance, valve and switch positions, voltage, and current. These outputs are in analog and digital form; some are routed to the signal-conditioning electronics assembly for voltage-level conditioning. If conditioning is not required, the outputs are routed directly to the pulse- code-modulation and timing electronics assembly. The signal-conditioning electronics assembly conditions its sensor-derived inputs and routes high-level analog or digital data to the pulse code-modulation and timing electronics assembly, caution and warning electronics assembly, and crew displays.

The pulse-code-modulation and timing electronics assembly converts the conditioned and unconditioned signals to several form for telemetering. This assembly also provides subcarrier frequencies, time reference signals, and sync pulses.

The sensed subsystem data, routed in analog and digital form to the caution and warning electronics assembly, are constantly compared with internally generated references. When an out-of-tolerance condition is detected, this assembly provides a signal to light the appropriate warning or caution light and both master alarm lights and to provide the 3-kHz alarm tone to the headsets.

Basically, all caution and warning lights operate in the same manner. The following is a typical example. Signals are routed from Reaction Control Subsystem (RCS) helium tank pressure sensors to comparators in the caution and warning electronics assembly. If comparison indicates a low-pressure condition, solid-state electronic circuits are enabled, causing the RCS caution light to go on. The astronauts then monitor helium tank pressure on indicators to determine actual pressure levels.

The data storage electronics assembly is a tape recorder that records voice and time correlation data (mission elapsed time). The voice and data inputs are multiplexed and recorded. The recorder can be operated manually or semiautomatically. In the manual mode, an astronaut closes a push-to-talk switch on his attitude controller assembly or electrical umbilical and speaks into his microphone. In the semiautomatic mode, CS equipment senses voice inputs from within the cabin or from the communications receivers and activates the recorder. Voice signals from the CS intercom bus are also recorded, together with mission elapsed time.

EQUIPMENT

SUBSYSTEM SENSORS

The sensors fall into four general categories: mechanical, resistive, variable reluctance, and electrical They are located throughout the various LM subsystems and structure and are used to change physical data into electrical signals.

SIGNAL CONDITIONING ELECTRONICS ASSEMBLY

This assembly consists of two electronic replaceable assemblies, each capable of housing up to 22 plug-in subassemblies of 11 different types (converters, amplifiers, etc.). Each subassembly contains its own power supply, which is isolated from the other subassemblies. Loss of one subassembly, due to a power supply failure does not affect operation of the other subassemblies. The subassemblies perform one or more of the following seven functions: amplify d-c voltages, attenuate d-c voltages, convert ac to dc, convert frequency to dc, phase modulate ac to dc, convert resistance variations to d c voltages, and isolate signals.

Pulse-Code-Modulation and Timing Electronics Assembly

This assembly comprises two sections: timing electronics and pulse code modulation. The timing electronics section develops timing signals for the pulse code modulation section, and the LM subsystems including the mission elapsed timer. The pulse-code-modulation section converts analog and digital signals to one of two formats, normal and reduced, for telemetering: 51,200 bits per second and 1600 bits per second.

Data Storage Electronics Assembly

This assembly is a single-speed, four-track, magnetic tape recorder that stores voice and time correlation data. A maximum of 10 hours of recording time is provided (2.5 hours on each track) by driving the tape, at 0.6 inch per second, over the record head and, on completion of a pass, automatically switching to the next track and reversing tape direction. One tape (450 feet) is supplied in a magazine.

Caution and Warning Electronics Assembly

This assembly compares analog signals (between 0 and 5 volts dc), from the signal-conditioning electronics assembly, with preselected internally generated limits supplied by the caution and warning power supply as reference voltages. In addition to analog inputs, it receives discrete on-off and contact closure signals. All inputs are routed to detectors; the detected signals are routed through logic circuitry, enabling relay contacts that cause caution or warning lights to go on or causing talkbacks to change state. Simultaneously, the detected signal energizes a master relay driver, enabling relay contacts. These contacts route a signal to light the master alarm lights and trigger the 3-kHz tone to the headsets.

Lighting

Exterior and interior lighting aids in the performance of crew visual tasks and lessens astronaut fatigue and interior- exterior glare effects. Exterior lighting is used for LM and CSM tracking and docking maneuvers. Interior lighting illuminates the cabin and the controls and displays on the Commander's and LM Pilot's panels.

FUNCTIONAL DESCRIPTION

LM lighting is provided by exterior and interior lights and lighting control equipment. The exterior lighting enables the astronauts to guide and orient the LM visually to the CSM visually to achieve successful tracking and docking. Interior lighting is divided into seven categories: incandescent annunciators, component caution lights, floodlights, computer condition lights, integral electro-luminescent lighting, numeric electro-luminescent lighting, and incandescently illuminated pushbuttons.

EXTERIOR LIGHTING

Exterior lighting includes five docking lights, and a high-intensity tracking light.

DOCKING LIGHTS

Five docking lights mounted on the exterior of the LM provide visual orientation and permit gross attitude determination relative to a line of sight through the CSM windows during rendezvous and docking. For transposition and docking, the docking lights are turned on by a switch located at spacecraft Lunar Module adapter attachment points. This switch is automatically closed upon deployment of the adapter panels. At completion of the docking maneuver, LM power is turned off and the docking lights go off The lights are visible, and their color recognizable, at a maximum distance of 1,000 feet.

TRACKING LIGHT

The tracking light permits visual tracking of the LM by the CSM. A flash tube in the tracking light electronics assembly causes the light, which has a 60 degree beam spread, to flash at a rate of 60 flashes per minute.

INTERIOR LIGHTING

Interior lighting consists of integral panel and display lighting, backup floodlighting, and electroluminescent lighting. Electroluminescence is light emitted from a crystalline phosphor (ZNS) placed as a thin layer between two closely spaced electrodes of an electrical capacitor; one of the electrodes must be transparent. The light output varies with voltage. Advantageous characteristics are an "afterglow" of less than 1 second, low power consumption, and negligible heat dissipation.

INTEGRALLY LIGHTED COMPONENTS

There are three types of integrally lighted components: panel areas, displays, and caution and warning annunciators. The integrally lighted components use electroluminescent or incandescent devices that are controlled by on-off switches and potentiometer-type dimming controls. All panel placards are integrally lighted by white electroluminescent lamps with overlays. The displays have electroluminescent lamps within their enclosures. The numeric displays show green or white illuminated digits on a nonilluminated background; displays with pointers have a nonilluminated pointer travelling over an illuminated background. The brightness of the electroluminescent displays is varied with dimming controls which can be bypassed by a related override switch, so that full brightness will be maintained should a dimming control fail.

LUNAR CONTACT LIGHTS

Two Lunar Contact lights go on when one or more of the four lunar-surface sensing probes contact the lunar surface. A probe is mounted beneath each of the landing gear footpads.

FLOODLIGHTING

Floodlighting is used for general cabin illumination and as a secondary source of illumination for the control and display panels. Floodlighting is provided by the Commander's overhead and forward floodlights, the LM Pilot's overhead and forward floodlights, and recessed floodlights in the bottom of extending side panels. These floodlight fixtures provide an even distribution of light with minimum reflection. Every panel area has more than one lamp.

PORTABLE UTILITY LIGHTS

Two portable utility lights are used, when necessary, to supplement the cabin interior lighting. The lights, when removed from the flight data file container, connect to the overhead utility light panel. Switches provide one-step dimming for light-intensity control.

OPTICAL SIGHT RETICLE LIGHT

The crewman's optical alignment sight, used to sight the docking target on the CSM, has a reticle that is illuminated by a 28-volt d-c lamp.

ALIGNMENT OPTICAL TELESCOPE LIGHTS

A thumbwheel on the computer control and reticle dimmer assembly controls the brightness of the telescope reticle. The lamps edge-light the reticle with incandescent red light.