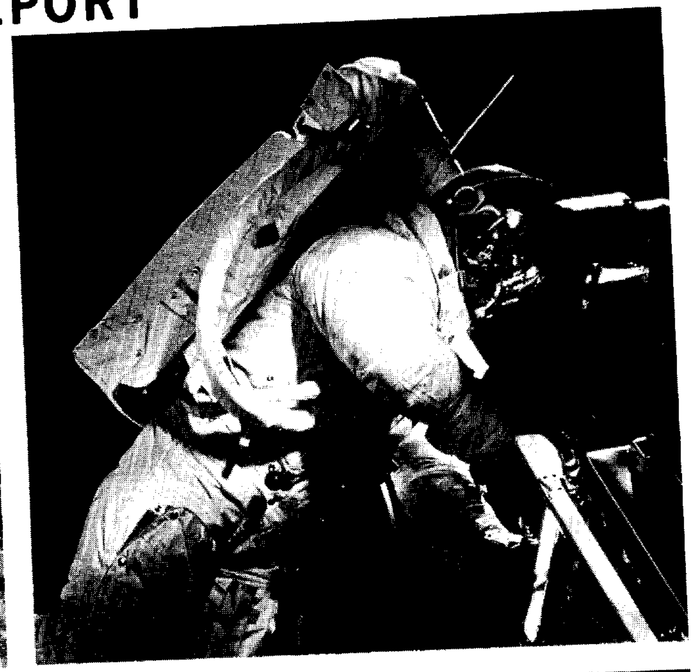
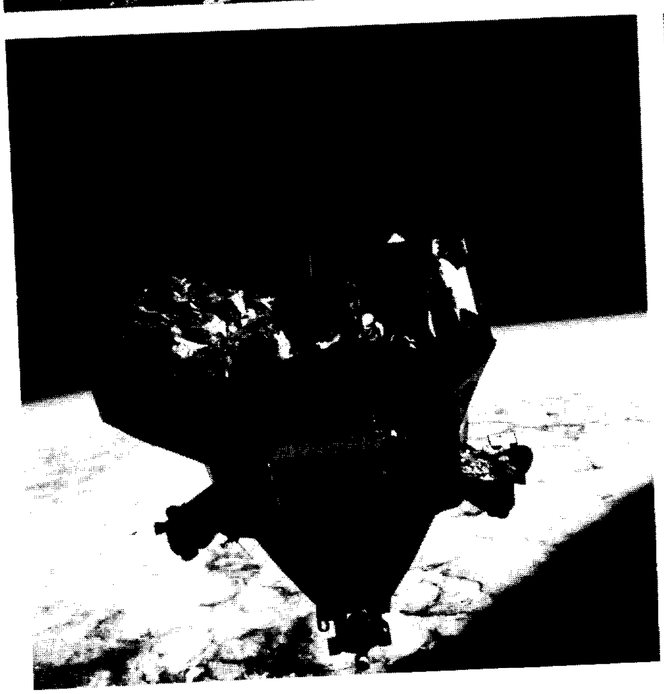
Report No. M-932-69-09

POST LAUNCH MISSION OPERATION REPORT





APOLLO 9 (AS-504) MISSION

OFFICE OF MANNED SPACE FLIGHT

Prepared by: Apollo Program Office-MAO

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GENERAL

The Apollo 9 (AS-504) mission was the first manned flight involving the Lunar Module. The crew were James A. McDivitt, Commander; David R. Scott, Command Module Pilot; and Russell L. Schweickart, Lunar Module Pilot. Launch had been initially scheduled for 28 February 1969, but was postponed for three days because all three crewmen had virus respiratory infections. The countdown was accomplished without any unscheduled holds and the AS-504 Space Vehicle was successfully launched from Launch Complex 39 at Kennedy Space Center, Florida, on Monday, 3 March 1969. Recovery of the flight crew and Command Module was successfully accomplished on 13 March 1969, for a flight duration of 241 hours 53 seconds.

Initial review of test data indicates that overall performance of the launch vehicle, spacecraft, and flight crew together with ground support and control facilities and personnel was satisfactory, and that all primary mission objectives were accomplished.

NASA OMSF PRIMARY MISSION OBJECTIVES FOR APOLLO 9

PRIMARY OBJECTIVES

- Demonstrate crew/space vehicle/mission support facilities performance during a manned Saturn V mission with CSM and LM.
- Demonstrate LM/crew performance.
- Demonstrate performance of nominal and selected backup Lunar Orbit Rendezvous (LOR) mission activities, including:
  - Transposition, docking, LM withdrawal
  - Intervehicular crew transfer
  - Extravehicular capability
  - SPS and DPS burns
  - LM active rendezvous and docking

CSM/LM consumables assessment.

Sam C. Phillips
Lt. General, USAF
Apollo Program Director

Date: 14 FEB 69

George E. Mueller
Associate Administrator for
Manned Space Flight

Date: 17 Feb 1969

RESULTS OF APOLLO 9 MISSION

Based upon a review of the assessed performance of Apollo 9, launched 3 March 1969 and completed 13 March 1969, this mission is adjudged a success in accordance with the objectives stated above.

Sam C. Phillips
Lt. General, USAF
Apollo Program Director

Date: 30 APRIL 1969

George E. Mueller
Associate Administrator for
Manned Space Flight

Date: MAY 5 1969

COUNTDOWN

The terminal countdown for Apollo 9 began at T-28 hours at 10:00 p.m. EST, 1 March 1969. The only holds encountered were two planned holds: one at T-16 hours for 3 hours, and one at T-9 hours for 6 hours. The count was resumed for the last time at 2:00 a.m. EST, 3 March 1969, and proceeded to launch at 11:00:00 a.m. EST.

FLIGHT SUMMARY

The Apollo 9 mission was launched from Kennedy Space Center, Florida, at 11:00:00 a.m. EST, 3 March 1969. All launch vehicle stages performed satisfactorily, but burned slightly longer than planned, inserting the S-IVB/spacecraft combination into a nominal orbit of 102.3 by 103.9 nautical miles (NM).

After post-insertion checkout was completed, the Command/Service Module (CSM) was separated from the S-IVB, transposed, and docked with the Lunar Module (LM). The docked spacecraft was separated from the S-IVB at 4:08:05 GET (Ground Elapsed Time). After separation, two unmanned S-IVB burns were performed to place the S-IVB/Instrument Unit on an earth-escape trajectory. After the third burn, the planned propellant dumps could not be performed.

After spacecraft separation from the launch vehicle, four Service Propulsion System (SPS) firings were made with the CSM/LM docked.

At approximately 43.5 hours GET, the Lunar Module Pilot (LMP) and the Commander (CDR) transferred to the LM. The first manned firing of the LM Descent Propulsion System (DPS) was initiated about 6 hours later. The two crewmen then returned to the Command Module (CM) for the fifth SPS firing.

At approximately 70 hours GET, the LMP and CDR again transferred to the LM for the LMP's 37-minute extravehicular activity (EVA). During this period, the Command Module Pilot (CMP) opened the CM hatch and retrieved thermal samples from the CSM exterior.

At about 89 hours GET, the CDR and LMP returned to the LM for the third time to perform the CSM/LM rendezvous. The LM primary guidance system was used to conduct the rendezvous with backup calculations being made by the CM computer. The phasing and insertion maneuvers were performed using the DPS to set up the rendezvous. The Ascent and Descent Stages were separated, followed by a concentric sequence initiation maneuver using the LM Reaction Control System. The LM Ascent Propulsion System (APS) was fired to establish the constant delta height. The terminal phase of the rendezvous began on time, and the spacecraft were again docked at about 99 hours GET. The Ascent Stage was jettisoned about 2.5 hours later. Shortly after, the APS was fired to propellant depletion. The firing lasted 350 seconds and resulted in an orbit of 3747 by 124.5 NM.

The sixth SPS firing, to lower apogee, was delayed because the +X translation to precede the maneuver was not programmed properly. However, the maneuver was rescheduled and successfully completed in the next revolution.

During the last three days, a seventh SPS firing was made to raise the apogee, and the SO65 Multispectral Photography Experiment and landmark tracking were accomplished.

Unfavorable weather in the planned landing area caused the deorbit maneuver (SPS 8) to be delayed for one revolution. This decision was made the day before splashdown and recovery forces were redeployed. Final parachute descent and splashdown were within sight of the prime recovery ship in the Atlantic Ocean. Splashdown was near the target point of 23 degrees 15 minutes north latitude, 68 degrees west longitude, as determined from the onboard computer solution. The crew were safely aboard the prime recovery ship, USS Guadalcanal, within 1 hour of splashdown.

Table 1 presents a summary of mission events.

TABLE 1

SUMMARY OF MISSION EVENTS

EVENT	TIME (GET) HR: MIN: SEC	
	PLANNED*	ACTUAL
First Motion	00:00:00	00:00:00
Maximum Dynamic Pressure	00:01:21	00:01:26
S-IC Center Engine Cutoff	00:02:14	00:02:14
S-IC Outboard Engine Cutoff	00:02:40	00:02:43
S-IC/S-II Separation	00:02:40	00:02:44
S-II Ignition	00:02:42	00:02:44
Jettison S-II Aft Interstage	00:03:10	00:03:14

* LV events based on MSFC LV Operational Trajectory, dated 31 January 1969.
SC events based on MSC SC Operational Trajectory, Revision 2, 20 February 1969.

Jettison Launch Escape Tower	00:03:16	00:03:18
S-II Engine Cutoff Command	00:08:51	00:08:56
S-II/S-IVB Separation	00:08:52	00:08:57
S-IVB Engine Ignition	00:08:55	00:09:01
S-IVB Engine Cutoff	00:10:49	00:11:05
Parking Orbit Insertion	00:10:59	00:11:15
Separation and Docking Maneuver Initiation	02:33:49	02:41:16
Spacecraft Docking	03:05:00 (Approx)	03:01:59
Spacecraft Final Separation	04:08:57	04:08:06
S-IVB Restart Preparation	04:36:12	04:36:17
S-IVB Reignition (2nd Burn)	04:45:50	04:45:56
S-IVB Second Cutoff Signal	04:46:52	04:46:58
S-IVB Restart Preparations	05:59:35	05:59:41
SPS Burn 1	06:01:40	05:59:01
S-IVB Reignition (3rd Burn)	06:07:13	06:07:19
S-IVB Third Cutoff Signal	06:11:14	06:11:21
Start LOX Dump	06:12:44	Not Accomplished
LOX Dump Cutoff	06:23:54	Not Accomplished
Start LH_2 Dump	06:24:04	Not Accomplished
LH_2 Dump Cutoff	06:42:19	Not Accomplished
SPS Burn 2	22:12:00	22:12:04
SPS Burn 3	25:18:30	25:17:39

SPS Burn 4	28:28:00	28:24:41
Docked DPS Burn	49:42:00	49:41:35
SPS Burn 5	54:25:19	54:26:12
Undocking	92:39:00	92:39:36
CSM/LM Separation	93:07:40	93:02:54
DPS Phasing	93:51:34	93:47:35
DPS Insertion	95:43:22	95:39:08
Concentric Sequence Initiation - LM RCS Burn	96:21:00	96:16:07
Constant Delta Height - APS Burn	97:05:27	96:58:15
Terminal Phase Initiation	98:00:10	97:57:59
CSM/LM Docking	99:13:00 (Approx)	99:02:26
APS Burn to Propellant Depletion	101:58:00	101:53:15
SPS Burn 6	121:58:48	123:25:07
SPS Burn 7	169:47:54	169:39:00
SPS Burn 8 (Deorbit)	**	240:31:15
Entry Interface (400,000 ft)	**	240:44:10
Drogue Chute Deployment (25,000 feet Approx)	**	240:55:08
Splashdown	**	241:00:54

** Premission planned deorbit was changed to permit shift in landing point due to weather and sea conditions in initial planned recovery area. One additional orbit was added.

MISSION PERFORMANCE

The significant portions of the Apollo 9 mission are discussed herein. Space vehicle systems and mission support performance are covered in succeeding sections.

TRAJECTORY

The CSM/LM/IU/S-IVB combination was inserted into earth orbit at 00:11:15 GET after a normal launch phase. The resulting orbital elements and maneuver parameters are given in Table II for all engine firings.

Four SPS maneuvers were performed prior to the first docked DPS firing. Each of the first three SPS maneuvers was made without requiring a +X translation to settle propellants. The fourth SPS maneuver was preceded by an 18-second +X translation made with the Service Module Reaction Control System (SM RCS).

The fifth docked SPS maneuver resulted in the perigee being approximately 5 NM less than planned causing the rendezvous to be initiated 4 minutes earlier. Small cutoff errors of this magnitude were expected, and real-time trajectory planning for both rendezvous and deorbit was conducted to accommodate minor adjustments in the initiation times and velocity increments. Out-of-plane components were added during the flight to certain preplanned maneuvers to provide substantial reduction in spacecraft weight without significantly changing the orbital parameters for subsequent maneuvers.

The trajectory aspects of the rendezvous exercise will be discussed in the rendezvous section.

After the Ascent Stage jettison, a separation maneuver of 3 feet per second (fps) was performed by the SM RCS. The APS engine was then fired to propellant depletion.

The sixth SPS maneuver was delayed one revolution when the accompanying ullage burn did not occur at the proper time, but was completed nominally.

The seventh SPS maneuver was restructured in real time to provide a desired higher burn time and was successfully accomplished.

The deorbit maneuver was made over Hawaii during revolution 152, and CM/SM separation was performed. The CM landed at 241:00:53 GET near 23 degrees 15 minutes north latitude and 68 degrees west longitude.

TABLE II SUMMARY OF MANEUVERS

	BURN TIME (SECONDS)			Δ V (FEET PER SECOND)			RESULTANT ORBIT		
	*Prelaunch PLANNED	Real Time PLANNED	ACTUAL	*Prelaunch PLANNED	Real Time PLANNED	ACTUAL	*Prelaunch PLANNED	Real Time PLANNED	ACTUAL
First Service Propulsion	5.0	4.96	5.2	36.8	36.8	36.6	125.2 X 108.7	128.2 X 110.2	127.6 X 111.3
Second Service Propulsion	111.3	111.2	110.3	849.6	850.6	850.5	190.2 X 109.1	189.8 X 107.7	192.5 X 110.7
Third Service Propulsion	280.0	281.9	279.9	2548.2	2570.7	2567.9	268.2 X 111.3	270.3 X 109.4	274.9 X 112.6
Fourth Service Propulsion	28.1	28.4	27.9	299.4	300.9	300.5	268.7 X 111.4	273.8 X 109.3	275.0 X 112.4
First Descent Propulsion	367.0	370.6	372.0	1734.0	1744.0	1737.5	267.6 X 111.8	269.9 X 109.1	274.6 X 112.1
Fifth Service Propulsion	41.5	43.2	43.3	552.3	575.4	572.5	130.2 X 129.7	129.8 X 129.8	131.0 X 125.9
Ascent Propulsion Firing to Depletion	389.0**	444.9**	362.4	6074.9**	7427.5**	5373.4	4673.3 X 128.9**	6932.3 X 125.9	3760.9 X 126.6
Sixth Service Propulsion	2.4	1.33	1.40	62.7	38.8	33.7	127.9 X 94.6	120.2 X 104.8	123.1 X 108.5
Seventh Service Propulsion	9.9	25.0	24.9	252.8	653.3	650.1	238.7 X 93.9	250.4 X 97.9	253.2 X 100.7
Eighth Service Propulsion	11.7	11.6	11.7	323.3	325.0	322.7	241.8 X -15.1	238.5 X ---	240.0 X -4.7

NOTES: * Prelaunch planned refers to Apollo 9 Spacecraft Operational Trajectory, Revision 2, 20 February 1969.
** APS burn to depletion planned for unattainable apogee value to insure propellant depletion cutoff.

EXTRAVEHICULAR ACTIVITY

Extravehicular activity (EVA), planned for the third day, was reduced from 2 hours 15 minutes to about 1 hour of depressurized LM activity. This change was made because the LMP experienced a minor in-flight illness during the first two days of the mission.

Preparation for EVA began at approximately 71 hours GET. The CDR and the LMP were in the LM and the CMP in the CM. At approximately 73 hours GET, after donning the Portable Life Support System (PLSS) and the Oxygen Purge System (OPS), the LMP egressed through the forward hatch and moved to the external foot restraints on the platform. During this time the CM was depressurized and the side hatch was opened. Thermal sample retrieval was photographically recorded with the sequence cameras. The LMP used the handrails to evaluate body control and transfer techniques. Ingress was completed at about 74 hours GET. Both hatches were then secured and the vehicles repressurized. The PLSS was successfully recharged with oxygen and water.

The lithium hydroxide cartridge from the system was returned to the CM for post-flight metabolic analysis.

The repressurization cycles for both vehicles were nominal, and post-EVA procedures were followed without difficulty.

RENDEZVOUS

The CDR and the LMP transferred to the LM on the fifth day for the rendezvous. The rendezvous exercise began on schedule with a 5-fps separation maneuver using the SM RCS.

A phasing maneuver of 90.5 fps was performed with the LM DPS about 2.8 NM from the CSM. Approximately 12 NM above and 27 NM behind the CSM, the DPS was used to impart a 43.1-fps insertion velocity to the LM. At a range of 75 NM from the CSM, the Ascent and Descent Stages of the LM were separated, and a concentric sequence initiation maneuver of 40.0 fps was made with the LM RCS.

Approximately 10 NM below and 78 NM behind the CSM, the constant delta height maneuver was performed with the APS imparting a velocity change of 41.5 fps. The terminal phase began on time with a 22.3-fps LM RCS maneuver.

Braking maneuvers were conducted on schedule, and stationkeeping was maintained at a distance of approximately 100 feet so that photographs could be taken from both vehicles. Docking was successfully completed at about 99 hours GET. Problems were experienced in using the Crewman Optical Alignment Sight (COAS) in both vehicles during docking. The combination of a bright CM, a dimly lighted CM target, and a relatively dim reticle in the alignment sight made LM docking a difficult task.

LM rendezvous navigation and maneuver targeting using both the primary and the backup guidance systems were satisfactory. Radar data were successfully used, both automatically by the primary system and through manual insertion in the About Guidance System, to correct rendezvous state vectors. Maneuver solutions from both onboard systems and from ground computations appeared to correlate closely. The crew selected the primary system solutions for all maneuvers through the first midcourse correction performed after terminal phase initiation.

Rendezvous navigation and mirror-image targeting in the CM were performed satisfactorily; however, loss of the LM tracking light prevented sextant measurements from the CM when both vehicles were in darkness. Preliminary data indicate that CM maneuver calculations for terminal phase initiation were satisfactory.

FLIGHT CREW PERFORMANCE

Crew performance was excellent throughout the mission, and the flight was conducted essentially in accordance with the nominal plan.

Preparation for transfer to the LM required longer than anticipated, primarily because of the time required for the crewmen to don the space suits. The suit supply hoses were a source of interference and also contributed to the longer preparation time. As a result, about 1 hour was added to the preparation time for subsequent transfers.

Visual and photographic inspection of the entire spacecraft was accomplished after rendezvous and before docking.

FLIGHT CREW BIOMEDICAL EVALUATION

The launch was postponed for 72 hours because of symptoms of upper respiratory infections in all three crewmen. Physical examinations 3 hours before launch revealed no infection.

The planned medical operations were conducted as scheduled except that the LMP experienced some nausea and vomiting prior to and following the initial transfer to the LM.

Plans for EVA were modified because of the LMP's illness. The physiological parameters were essentially normal throughout the mission. The LMP's work rate during EVA was on the order of 500 Btu/hr.

FLIGHT CONTROL

Flight control performance was satisfactory in providing operational support for the Apollo 9 mission. Minor spacecraft problems were encountered, but none was such that either the mission operations or the flight plan was significantly altered.

Early in the mission, a caution and warning light on Hydrogen Tank 1 was observed just prior to an automatic cycle of the heaters. This condition persisted and the crew had to be disturbed during a rest period at 81 hours GET to increase the hydrogen tank pressure.

On the third day, the crew were about 1 hour behind the timeline, resulting in cancelling all the planned communications tests except the LM secondary S-band test and the LM two-way relay with television.

On the fourth day, the EVA was abbreviated and the external transfer from the LM to the CM was not performed. The activity was restricted to the LM forward platform because of concern about the LMP's earlier illness and proper readiness for the rendezvous on the following day.

At approximately 78 hours GET, after the tunnel hardware had been installed, a crewman made an unplanned return to the LM to open a circuit breaker. This change shortened the rest period about 30 minutes.

On the fifth day, LM activation was performed approximately 40 minutes early to insure an on-time rendezvous initiation.

The LM VHF telemetry and S-band power amplifier were lost for 6 and 12 hours, respectively, after the APS firing to depletion. These failures were expected because of the lack of cooling. The electrical system capability for this spacecraft was several hours longer than predicted. LM support terminated at 113:42:00 GET.

On the sixth day, the sixth SPS maneuver was delayed by one revolution. The crew reported that the +X translation did not occur. A procedural error was made in loading the CM computer, since the proper SM RCS quads were not selected. The computer was reloaded, and one revolution later, the maneuver was made satisfactorily.

On the eighth day, the seventh SPS maneuver was increased to 25 seconds in duration to permit a test of the Propellant Utilization and Gaging System (PUGS).

RECOVERY

Recovery of the Apollo 9 Command Module and crew was completed in the West Atlantic by the prime recovery ship, USS Guadalcanal. The following table is a list of significant recovery events on 13 March 1969:

EVENT	EST
First VHF contact	11:51 a.m.
First beacon and voice contact	11:57 a.m.
First visual contact	11:59 a.m.
Landing	12:01 p.m.
Swimmers deployed	12:07 p.m.
Flotation collar installed	12:14 p.m.
CM hatch open	12:27 p.m.
First astronaut aboard helicopter	12:39 p.m.
All astronauts in helicopter	12:46 p.m.
Astronauts on deck	12:50 p.m.
CM aboard recovery ship	2:13 p.m.

The CM remained in the stable I flotation attitude. Sea-state conditions were very moderate at the recovery site.

SYSTEMS PERFORMANCE

Engineering data reviewed to date indicate that all mission objectives were attained. Further detailed analysis of all data is continuing and appropriate refined results of systems performance will be reported in MSFC and MSC technical reports. Summaries of the significant anomalies and discrepancies are presented in Tables III, IV, and V.

TABLE III

LAUNCH VEHICLE DISCREPANCY SUMMARY

DESCRIPTION	REMARKS
Oscillations occurred in the S-II center engine chamber pressure and the S-II structure late in the burn. Oscillations have occurred on four flights and five static firings, but only after 320 seconds of S-II burn.	Apparently caused by coupling between the center engine and the stage structure. Fix will be early center engine cutoff at 299 seconds on Apollo 10.
S-IVB APS Module No. 2 helium supply pressure decayed slowly.	Leak in teflon seals upstream of the regulator. Change of seal material to rubber has been approved. Closed.
S-IVB helium regulator lock-up pressure exceeded the redline during countdown, and the helium pneumatic pressure was high throughout the mission.	Internal leakage in regulator caused by wear on poppet. Modified regulator has been tested and installed on S-IVB-505. Redline has been raised from 585 to 630 psi.
S-IVB third burn anomaly: Gas generator pressure spike at start, engine chamber pressure oscillations, loss of engine control pneumatic pressure, abnormal attitude control system oscillations, decrease in engine performance during burn, and inability to dump residual propellants after burn.	Caused by extreme out-of-spec engine start conditions which resulted in excessive engine chamber pressure oscillations and possible gas generator damage, followed by loss of pneumatic system. There is no evidence that the causes of this anomaly are applicable to an in-spec engine start. The flight mission rules allowing restart with recirculation systems inoperative are being revised for Apollo 10.

TABLE IV

COMMAND/SERVICE MODULE DISCREPANCY SUMMARY

DESCRIPTION	REMARKS
Unable to translate the CSM to the left. Propellant isolation valves in two SM RCS quads were found to be closed.	Apparently caused by mechanical shock at CSM/S-IVB separation. The crew will check the valve positions after separation on Apollo 10 and subsequent missions.
Master alarm occurred coincident with hard docking without any accompanying annunciator.	Caused by a sensor transient or a momentary short circuit due to mechanical shock. Also occurred during the CSM 106 docking test.
During the third SPS burn, eight master alarms occurred because of indications of propellant unbalance.	Caused by erroneous readings from the primary probe in the SPS oxidizer tank. The master alarm and warning functions from the PUGS have been deleted on CSM 106 and subsequent spacecraft. Closed.
The scanning telescope mechanism jammed frequently when driven manually, but worked normally in automatic mode.	A pin from a counter drum was found wedged in a split gear. Units on Apollo 10 and subs will be replaced with units that have been inspected. Closed.
Fuel Cell No. 2 condenser outlet temperature exceeded the normal range several times.	The bypass valve that controls coolant temperature operated improperly because of contamination in the glycol. For subsequent missions, Block I valves which are less susceptible to contaminants will be installed and the radiators will be vibrated and flushed 30 to 45 days before launch. Closed.
Automatic control of the pressure in the cryogenic hydrogen tanks was lost and pressure was controlled manually.	Probably caused by an intermittent open circuit in the motor switch control circiut. No hardware change will be made. Closed.
The first two attempts to undock were unsuccessful because the release switch was not held long enough. Before the 2nd docking, the "flag" check showed the capture latches on the probe were not cocked; recycling the switch produced a cocked indication.	The Apollo Operations Handbook has been revised to clarify the procedure for extending the probe. Closed.

TABLE IV (CONTINUED)

DESCRIPTION	REMARKS
CSM would not respond to multiple uplink realtime commands for about 10 hours; only the first command was accepted. The problem was cleared by cycling the up-telemetry command reset switch.	Caused by flight hardware associated with the message-acceptance pulse.
The CM computer failed twice to respond properly to programs entered by DSKY. The ground verified correct loading except for the last entry, which is not monitored.	Probably caused by procedural error in making the last entry on the DSKY. Closed.
The entry monitor system scribe did not continuously cut through the emulsion on the scroll during entry.	Caused by a leak in the scroll assembly which caused hardening of the emulsion. On Apollo 10, the scroll assembly will be leak tested and a sharper stylus will be used. Closed.
After recovery, one docking ring separation charge holder was out of its channel far enough to possibly foul or cut the parachute riser lines during deployment.	A spring has been incorporated to retain the charge holders on CM 106 and subsequent spacecraft. Closed.

TABLE V

LUNAR MODULE DISCREPANCY SUMMARY

DESCRIPTION	REMARKS
During the first 30 seconds of the 1st DPS burn, the supercritical helium regulator manifold pressure dropped to 180 psia and then recovered to a normal 240 psia. An anomalous pressure rise also occurred during prelaunch servicing.	Flow was probably blocked momentarily by freezing of air or other contaminants in the supercritical helium tank heat exchanger. Servicing equipment and procedures have been revised. Closed.
The DPS supercritical helium tank pressure began decaying at the end of the 1st DPS burn at a rate indicating a 0.1 lb/hr leak.	Possible leak upstream of the solenoid latch valve. The LM-4 flight configuration will be checked to assure adequate strength margins for thermal, vibration, and squib valve firing shock. The squib valve braze joints will also be tested.
The oxygen purge system light did not come on during a self-test prior to rendezvous, after being erratic earlier.	Failure of the main power switch actuator mechanism, which has been redesigned for Apollo 10 and subs. Closed.
The LMP's push-to-talk switches on the umbilical and on the attitude controller were inoperative after about 89 hours GET. LMP used the VOX mode for remainder of LM operations.	Probably caused by a discontinuity (broken wire) in the common wire to the push-to-talk switches which are in parallel. Closed.
The abort guidance system (AGS) warning light remained on continuously in standby and operating modes during period five. The AGS operated nominally throughout the mission.	Probably a malfunction of the caution and warning circuitry, but the failure mode cannot be identified because the AGS parameters are not displayed or telemetered. Closed.
The DPS engine was rough for a few seconds at 27% throttle during the second DPS burn.	Caused by helium trapped in the propellant lines during the previous SPS burns, which has no detrimental effect on the system. Closed.
The tracking light failed during ascent/descent staging.	Probably caused by a failure in the pulse forming network. Mission simulations are being run on the LM-4 light.

TABLE V (CONTINUED

DESCRIPTION	REMARKS
The Crewman Optical Alignment Sight (COAS) reticle was difficult to see during rendezvous.	Background light washed out the reticle image. On LM-4 and subsequent LM's, the light filter will be replaced with a diffuser lens and a detachable filter assembly will be provided. Closed.
At the start of the APS burn to depletion, the helium pressure to the propellant tanks regulated at 177 psia instead of the expected 185 psia. At 290 seconds, the pressure increased from 176 to 180 psia.	Possible failure modes will be simulated on a regulator and the behavior of the regulated pressure will be determined. The presently identified types of failure that can cause a downward shift in regulation pressure produce no detrimental effects in DPS operation.
The Data Entry and Display Assembly operator error light remained on, and multiple depression of the "Clear" button was required to extinguish the light.	Probably caused by failure of contacts to close on one of the two switches in the "Clear" pushbutton. Closed.
When the forward hatch was opened for EVA, it tended to bind at the top and it also would not stay open.	A thermal blanket which interfered with the hatch will be retained with tape on LM-4. The door stop is being studied for possible improvement.

MISSION SUPPORT

LAUNCH COMPLEX

No major problems occurred during the terminal countdown. Launch damage to the pad was minimal and ground system performance was as expected.

NETWORK

Overall mission support by the Mission Control Center and the Manned Space Flight Network was considered satisfactory throughout the mission. Mission Control Center hardware, communications, and computer systems experienced very few problems with no major data losses. Network telemetry, tracking, and command support were satisfactory. The few failures which were experienced had minimal impact on Mission Control Center operations. Carnarvon was the only site which had persistent support problems in that the command and telemetry computers experienced outages.

HF communications reception during some periods was marginal at several sites; however, the requirement for HF communications was kept at a minimum by using satellite communications systems when possible. Although several minor communications outages were experienced, no significant data losses were experienced. A number of significant problems were experienced with air-to-ground communications primarily because of ground procedural errors.

The most significant anomalies and discrepancies are presented in Table VI.

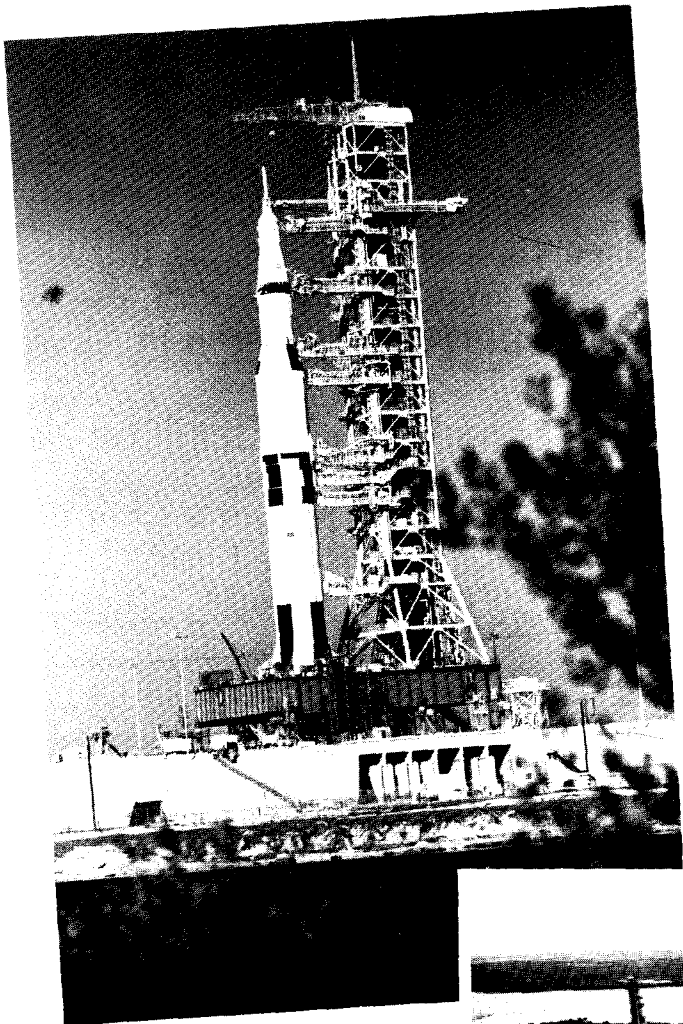
TABLE VI

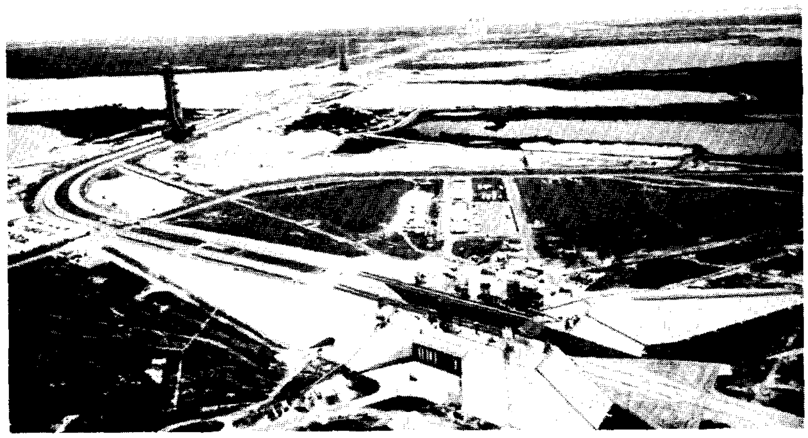
MISSION SUPPORT DISCREPANCY SUMMARY

DESCRIPTION	REMARKS
During the fourth revolution, over Guaymas, air-to-ground voice was lost for approximately 6 minutes.	Caused by a procedural error at the Mission Control Center, which had been improperly configured for the transmissions.
During extravehicular activity, air-to-ground transmissions to the spacecraft were lost from Guaymas, Texas, Merrit Island, Bermuda, and USNS Vanguard stations. Downlink voice was remoted to the Mission Control Center nominally during the same period.	The loss of uplink capability was caused by a combination of the stations being configured to uplink S-band only (rather than S-band and VHF simultaneously) and the spacecraft crew having the S-band volume fully decreased as planned. The problem was further complicated by the inability to transmit VHF voice from Bermuda because of a simultaneous transmission on that frequency from the LM and a suppression of the VHF uplink by the continuously keyed Portable Life Support System.
Air-to-ground communications were lost for approximately 3 minutes over Texas during revolution 119.	Caused by a patching error at Texas.

MISSION OPERATION REPORT (APOLLO)

SUPPLEMENT





25 FEBRUARY 1969

OFFICE OF MANNED SPACE FLIGHT
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SPACE VEHICLE

The primary flight hardware of the Apollo program consists of a Saturn V Launch Vehicle and an Apollo Spacecraft. Collectively, they are designated the Apollo-Saturn V Space Vehicle (SV) (Figure 1).

SATURN V LAUNCH VEHICLE

The Saturn V Launch Vehicle (LV) is designed to boost up to 285,000 pounds into a 105 nautical mile earth orbit and to provide for lunar payloads of 100,000 pounds. The Saturn V LV consists of three propulsive stages (S-IC, S-II, S-IVB), two interstages, and an Instrument Unit (IU).

S-IC Stage

General

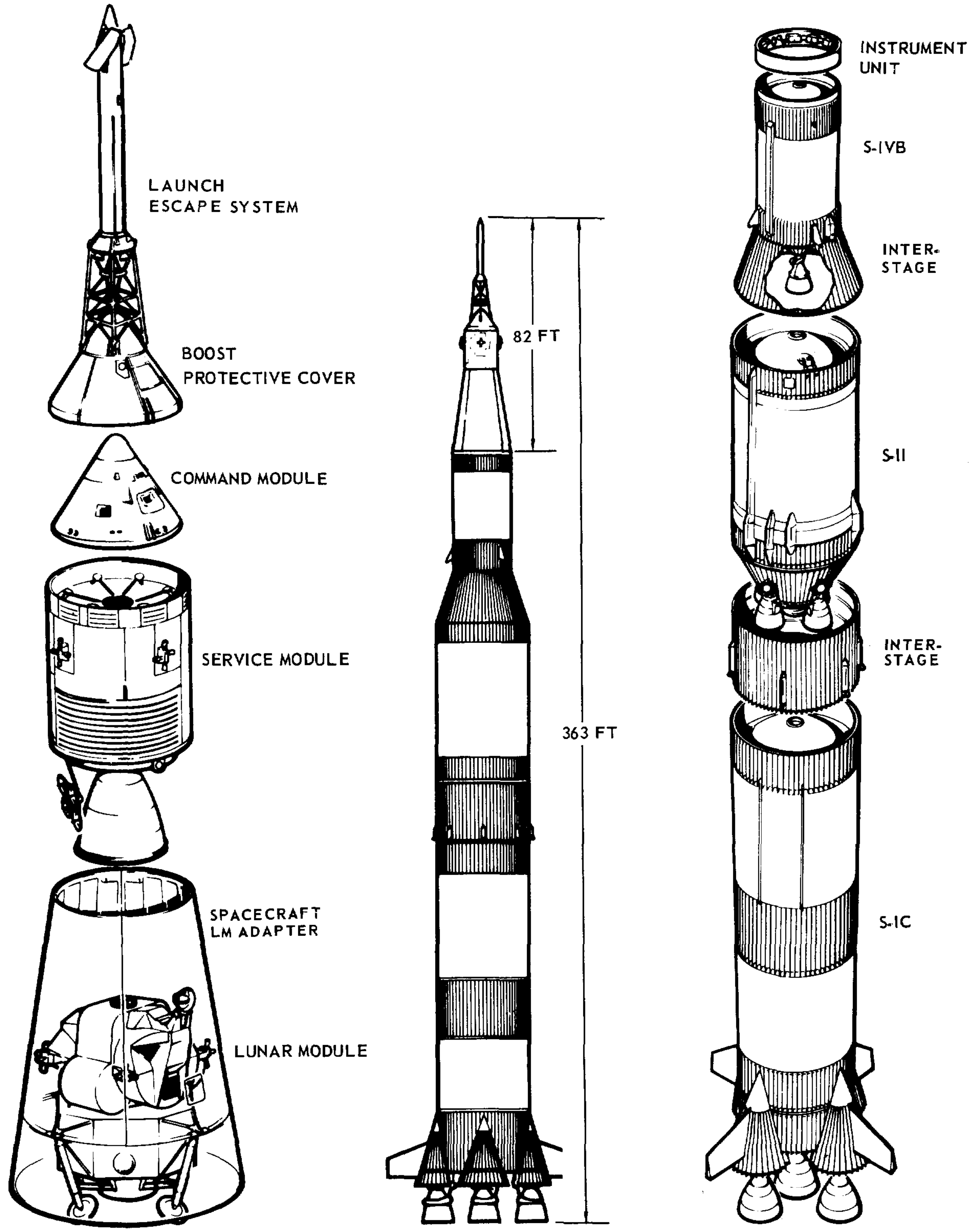
The S-IC stage (Figure 2) is a large cylindrical booster, 138 feet long and 33 feet in diameter, powered by five liquid propellant F-1 rocket engines. These engines develop a nominal sea level thrust total of approximately 7,650,000 pounds and have an operational burn time of 159 seconds. The stage dry weight is approximately 295,300 pounds and the total loaded stage weight is approximately 5,031,500 pounds. The S-IC stage interfaces structurally and electrically with the S-II stage. It also interfaces structurally, electrically, and pneumatically with Ground Support Equipment (GSE) through two umbilical service arms, three tail service masts, and certain electronic systems by antennas. The S-IC stage is instrumented for operational measurements or signals which are transmitted by its independent telemetry system.

Structure

The S-IC structural design reflects the requirements of F-1 engines, propellants, control, instrumentation, and interfacing systems. Aluminum alloy is the primary structural material. The major structural components are the forward skirt, oxidizer tank, intertank section, fuel tank, and thrust structure. The forward skirt interfaces structurally with the S-IC/S-II interstage. The skirt also mounts vents, antennas, electrical and electronic equipment.

The 47,298-cubic foot oxidizer tank is the structural link between the forward skirt and the intertank structure which provides structural continuity between the oxidizer and fuel tanks. The 29,215-cubic foot fuel tank provides the load carrying structural link between the thrust and intertank structures. Five oxidizer ducts run from the oxidizer tank, through the fuel tank, to the F-1 engines.

APOLLO-SATURN V SPACE VEHICLE



SPACECRAFT SPACE VEHICLE LAUNCH VEHICLE

Fig. 1

S-IC STAGE

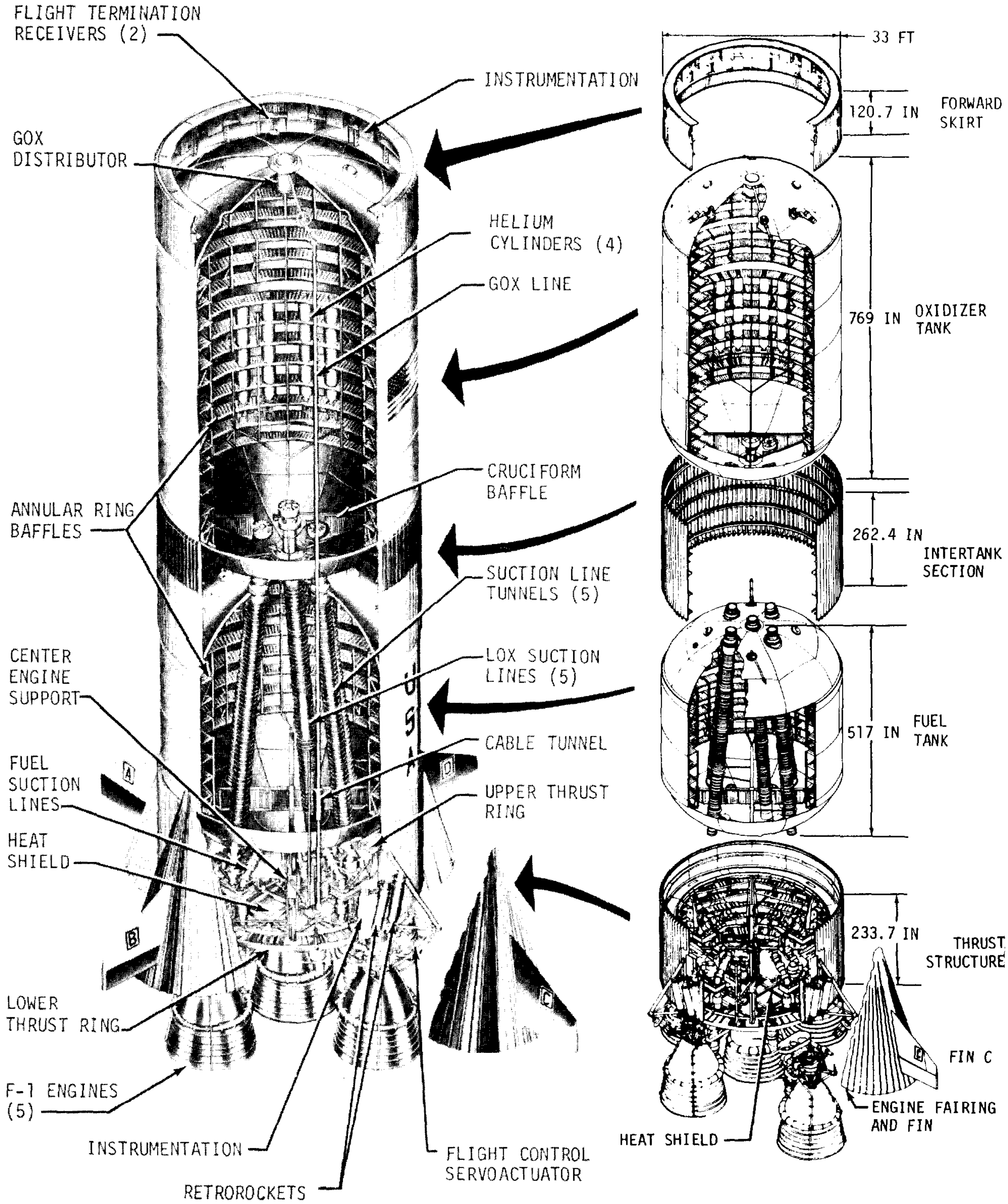


Fig. 2

The thrust structure assembly redistributes the applied loads of the five F-1 engines into nearly uniform loading about the periphery of the fuel tank. Also, it provides support for the five F-1 engines, engine accessories, base heat shield, engine fairings and fins, propellant lines, retrorockets, and environmental control ducts. The lower thrust ring has four holddown points which support the fully loaded Saturn V Space Vehicle (approximately 6,483,000 pounds) and also, as necessary, restrain the vehicle during controlled release.

Propulsion

The F-1 engine is a single-start, 1,530,000-pound fixed-thrust, calibrated, bi-propellant engine which uses Liquid Oxygen (LOX) as the oxidizer and Rocket Propellant-1 (RP-1) as the fuel. The thrust chamber is cooled regeneratively by fuel, and the nozzle extension is cooled by gas generator exhaust gases. Oxidizer and fuel are supplied to the thrust chamber by a single turbopump powered by a gas generator which uses the same propellant combination. RP-1 is also used as the turbopump lubricant and as the working fluid for the engine hydraulic control system. The four outboard engines are capable of gimbaling and have provisions for supply and return of RP-1 as the working fluid for a thrust vector control system. The engine contains a heat exchanger system to condition engine-supplied LOX and externally supplied helium for stage propellant tank pressurization. An instrumentation system monitors engine performance and operation. External thermal insulation provides an allowable engine environment during flight operation.

The normal in-flight engine cutoff sequence is center engine first, followed by the four outboard engines. Engine optical-type depletion sensors in either the oxidizer or fuel tank initiate the engine cutoff sequence. In an emergency, the engine can be cut off by any of the following methods: Ground Support Equipment (GSE) Command Cutoff, Emergency Detection System, or Outboard Cutoff System.

Propellant Systems

The propellant systems include hardware for fill and drain, propellant conditioning, and tank pressurization prior to and during flight, and for delivery to the engines. Fuel tank pressurization is required during engine starting and flight to establish and maintain a Net Positive Suction Head (NPSH) at the fuel inlet to the engine turbopumps. During flight, the source of fuel tank pressurization is helium from storage bottles mounted inside the oxidizer tank. Fuel feed is accomplished through two 12-inch ducts which connect the fuel tank to each F-1 engine. The ducts are equipped with flex and sliding joints to compensate for motions from engine gimbaling and stage stresses.

Gaseous Oxygen (GOX) is used for oxidizer tank pressurization during flight. A portion of the LOX supplied to each engine is diverted into the engine heat exchangers where it is transformed into GOX and routed back to the tanks. LOX is delivered to the engines through five suction lines which are supplied with flex and sliding joints.

Flight Control System

The S-IC thrust vector control consists of four outboard F-1 engines, gimbal blocks to attach these engines to the thrust ring, engine hydraulic servoactuators (two per engine), and an engine hydraulic power supply. Engine thrust is transmitted to the thrust structure through the engine gimbal block. There are two servo-actuator attach points per engine, located 90 degrees from each other, through which the gimbaling force is applied. The gimbaling of the four outboard engines changes the direction of thrust and as a result corrects the attitude of the vehicle to achieve the desired trajectory. Each outboard engine may be gimbaled $\pm 5°$ within a square pattern at a rate of 5° per second.

Electrical

The electrical power system of the S-IC stage consists of two basic subsystems: the operational power subsystem and the measurements power subsystem. Onboard power is supplied by two 28-volt batteries. Battery number 1 is identified as the operational power system battery. It supplies power to operational loads such as valve controls, purge and venting systems, pressurization systems, and sequencing and flight control. Battery number 2 is identified as the measurement power system. Batteries supply power to their loads through a common main power distributor, but each system is completely isolated from the other. The S-IC stage switch selector is the interface between the Launch Vehicle Digital Computer (LVDC) in the IU and the S-IC stage electrical circuits. Its function is to sequence and control various flight activities such as telemetry calibration, retrofire initiation, and pressurization.

Ordnance

The S-IC ordnance systems include the propellant dispersion (flight termination) and the retrorocket systems. The S-IC Propellant Dispersion System (PDS) provides the means of terminating the flight of the Saturn V if it varies beyond the prescribed limits of its flight path or if it becomes a safety hazard during the S-IC boost phase. A transmitted ground command shuts down all engines and a second command detonates explosives which longitudinally open the fuel and oxidizer tanks. The fuel opening is 180° (opposite) to the oxidizer opening to minimize propellant mixing.

Eight retrorockets provide thrust after S-IC burnout to separate it from the S-II stage. The S-IC retrorockets are mounted in pairs external to the thrust structure in the fairings of the four outboard F-1 engines. The firing command originates in the IU and activates redundant firing systems. At retrorocket ignition the forward end of the fairing is burned and blown through by the exhausting gases. The thrust level developed by seven retrorockets (one retrorocket out) is adequate to separate the S-IC stage a minimum of six feet from the vehicle in less than one second.

S-II Stage

General

The S-II stage (Figure 3) is a large cylindrical booster, 81.5 feet long and 33 feet in diameter, powered by five liquid propellant J-2 rocket engines which develop a nominal vacuum thrust of 230,000 pounds each for a total of 1,150,000 pounds. Dry weight of the S-II stage is approximately 84,600 pounds. The stage approximate loaded gross weight is 1,064,600 pounds. The S-IC/S-II interstage weighs 11,665 pounds. The S-II stage is instrumented for operational and R&D measurements which are transmitted by its independent telemetry system. The S-II stage has structural and electrical interfaces with the S-IC and S-IV stages, and electric, pneumatic, and fluid interfaces with GSE through its umbilicals and antennas.

Structure

Major S-II structural components are the forward skirt, the 37,737-cubic foot fuel tank, the 12,745-cubic foot oxidizer tank (with the common bulkhead), the aft skirt/thrust structure, and the S-IC/S-II interstage. Aluminum alloy is the major structural material. The forward and aft skirts distribute and transmit structural loads and interface structurally with the interstages. The aft skirt also distributes the loads imposed on the thrust structure by the J-2 engines. The S-IC/S-II interstage is comparable to the aft skirt in capability and construction. The propellant tank walls constitute the cylindrical structure between the skirts. The aft bulkhead of the fuel tank is also the forward bulkhead of the oxidizer tank. This common bulkhead is fabricated of aluminum with a fiberglass/phenolic honeycomb core. The insulating characteristics of the common bulkhead minimize the heating effect of the relatively hot LOX (-297°F) on the LH_2 (-423°F).

Propulsion

The S-II stage engine system consists of five single-start, high-performance, high-altitude J-2 rocket engines of 230,000 pounds of nominal vacuum thrust each. Fuel is liquid hydrogen (LH_2) and the oxidizer is liquid oxygen (LOX). The four outer J-2 engines are equally spaced on a 17.5-foot diameter circle and are

S-II STAGE

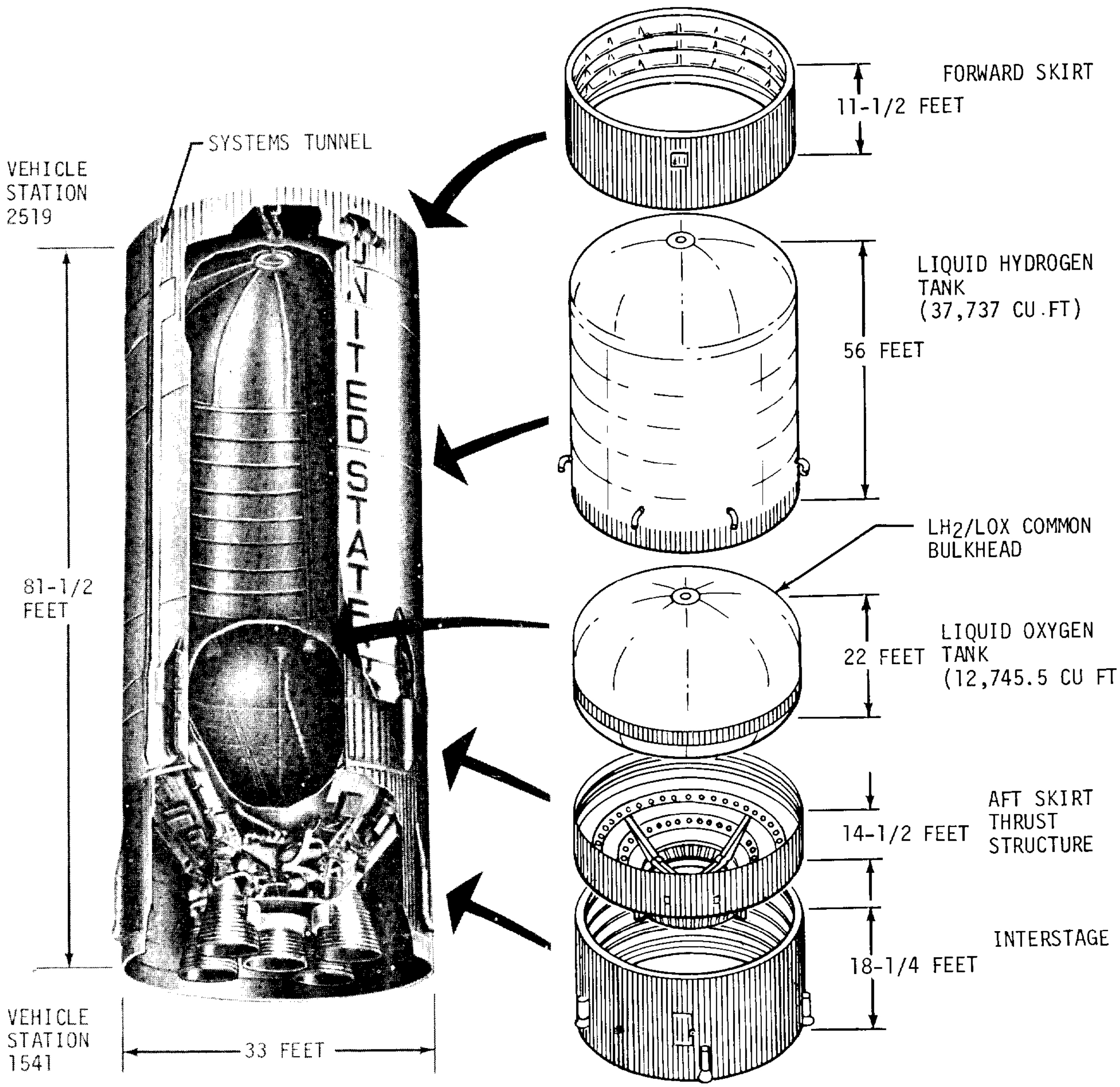


Fig. 3

capable of being gimbaled through a ±7 degree square pattern to allow thrust vector control. The fifth engine is fixed and is mounted on the centerline of the stage. The engine valves are controlled by a pneumatic system powered by gaseous helium which is stored in a sphere inside the start tank. An electrical control system that uses solid state logic elements is used to sequence the start and shutdown operations of the engine. Electrical power is stage-supplied.

The J-2 engine may receive cutoff signals from several different sources. These sources include engine interlock deviations, EDS automatic or manual abort cutoffs, and propellant depletion cutoff. Each of these sources signals the LVDC in the IU. The LVDC sends the engine cutoff signal to the S-II switch selector, which in turn signals the electrical control package, which controls all local signals necessary for the cutoff sequence. Five discrete liquid level sensors per propellant tank provide initiation of engine cutoff upon detection of propellant depletion. The cutoff sensors will initiate a signal to shut down the engines when two out of five engine cutoff signals from the same tank are received.

Propellant Systems

The propellant systems supply fuel and oxidizer to the five engines. This is accomplished by the propellant management components and the servicing, conditioning, and engine delivery subsystems. The propellant tanks are insulated with foam-filled honeycomb which contains passages through which helium is forced for purging and leak detection. The LH_2 feed system includes five 8-inch vacuum-jacketed feed ducts and five prevalves.

During powered flight, prior to S-II ignition, Gaseous Hydrogen (GH_2) for LH_2 tank pressurization is bled from the thrust chamber hydrogen injector manifold of each of the four outboard engines. After S-II engine ignition, LH_2 is preheated in the regenerative cooling tubes of the engine and tapped off from the thrust chamber injector manifold in the form of GH_2 to serve as a pressurizing medium. The LOX feed system includes four 8-inch, vacuum-jacketed feed ducts, one uninsulated feed duct, and five prevalves. LOX tank pressurization is accomplished with GOX obtained by heating LOX bled from the LOX turbopump outlet.

The propellant management system monitors propellant mass for control of propellant loading, utilization, and depletion. Components of the system include continuous capacitance probes, propellant utilization valves, liquid level sensors, and electronic equipment. During flight, the signals from the tank continuous capacitance probes are monitored and compared to provide an error signal to the propellant utilization valve on each LOX pump. Based on this error signal, the propellant utilization valves are positioned to minimize residual propellants and assure a fuel-rich cutoff by varying the amount of LOX delivered to the engines.

Flight Control System

Each outboard engine is equipped with a separate, independent, closed-loop, hydraulic control system that includes two servoactuators mounted in perpendicular planes to provide vehicle control in pitch, roll, and yaw. The servoactuators are capable of deflecting the engine ± 7 degrees in the pitch and yaw planes (± 10 degrees diagonally) at the rate of 8 degrees per second.

Electrical

The electrical system is comprised of the electrical power and electrical control subsystems. The electrical power system provides the S-II stage with the electrical power source and distribution. The electrical control system interfaces with the IU to accomplish the mission requirements of the stage. The LVDC in the IU controls in-flight sequencing of stage functions through the stage switch selector. The stage switch selector outputs are routed through the stage electrical sequence controller or the separation controller to accomplish the directed operation. These units are basically a network of low-power transistorized switches that can be controlled individually and, upon command from the switch selector, provide properly sequenced electrical signals to control the stage functions.

Ordnance

The S-II ordnance systems include the separation, ullage rocket, retrorocket, and propellant dispersion (flight termination) systems. For S-IC/S-II separation, a dual-plane separation technique is used wherein the structure between the two stages is severed at two different planes. The second-plane separation jettisons the interstage after S-II engine ignition. The S-II/S-IVB separation occurs at a single plane located near the aft skirt of the S-IVB stage. The S-IVB interstage remains as an integral part of the S-II stage. To separate and retard the S-II stage, a deceleration is provided by the four retrorockets located in the S-II/S-IVB interstage. Each rocket develops a nominal thrust of 34,810 pounds and fires for 1.52 seconds. All separations are initiated by the LVDC located in the IU.

To ensure stable flow of propellants into the J-2 engines, a small forward acceleration is required to settle the propellants in their tanks. This acceleration is provided by four ullage rockets mounted on the S-IC interstage. Each rocket develops a nominal thrust of 23,000 pounds and fires for 3.75 seconds. The ullage function occurs prior to second-plane separation.

The S-II Propellant Dispersion System (PDS) provides for termination of vehicle flight during the S-II boost phase if the vehicle flight path varies beyond its prescribed limits or if continuation of vehicle flight creates a safety hazard. The S-II PDS may be safed after the launch escape tower is jettisoned. The fuel tank linear shaped charge, when detonated, cuts a 30-foot vertical opening in the tank. The oxidizer tank destruct charges simultaneously cut 13-foot lateral openings in the oxidizer tank and the S-II aft skirt.

S-IVB Stage

General

The S-IVB stage (Figure 4) is a large cylindrical booster 59 feet long and 21.6 feet in diameter, powered by one J-2 engine. The S-IVB stage is capable of multiple engine starts. Engine thrust is 232,000 pounds for the first burn and 206,000 pounds for subsequent burns. This stage is also unique in that it has an attitude control capability independent of its main engine. Dry weight of the stage is 25,300 pounds. The launch weight of the stage is 259,160 pounds. The interstage weight of 8,080 pounds is not included in the stated weights. The stage is instrumented for functional measurements or signals which are transmitted by its independent telemetry system.

Structure

The major structural components of the S-IVB stage are the forward skirt, propellant tanks, aft skirt, thrust structure and aft interstage. The forward skirt provides structural continuity between the fuel tank walls and the IU. The propellant tank walls transmit and distribute structural loads from the aft skirt and the thrust structure. The aft skirt is subjected to imposed loads from the S-IVB aft interstage. The thrust structure mounts the J-2 engine and distributes its structural loads to the circumference of the oxidizer tank. A common, insulated bulkhead separates the 2830-cubic foot oxidizer tank and the 10,418-cubic foot fuel tank and is similar to the common bulkhead discussed in the S-II description. The predominant structural material of the stage is aluminum alloy. The stage interfaces structurally with the S-II stage and the IU.

Main Propulsion

The high-performance 232,000-pound thrust J-2 engine as installed in the S-IVB stage has a multiple restart capability. The engine valves are controlled by a pneumatic system powered by gaseous helium which is stored in a sphere inside a start bottle. An electrical control system that uses solid state logic elements is used to sequence the start and shutdown operations of the engine. Electrical power is supplied from aft battery No. 1.

S-IVB STAGE

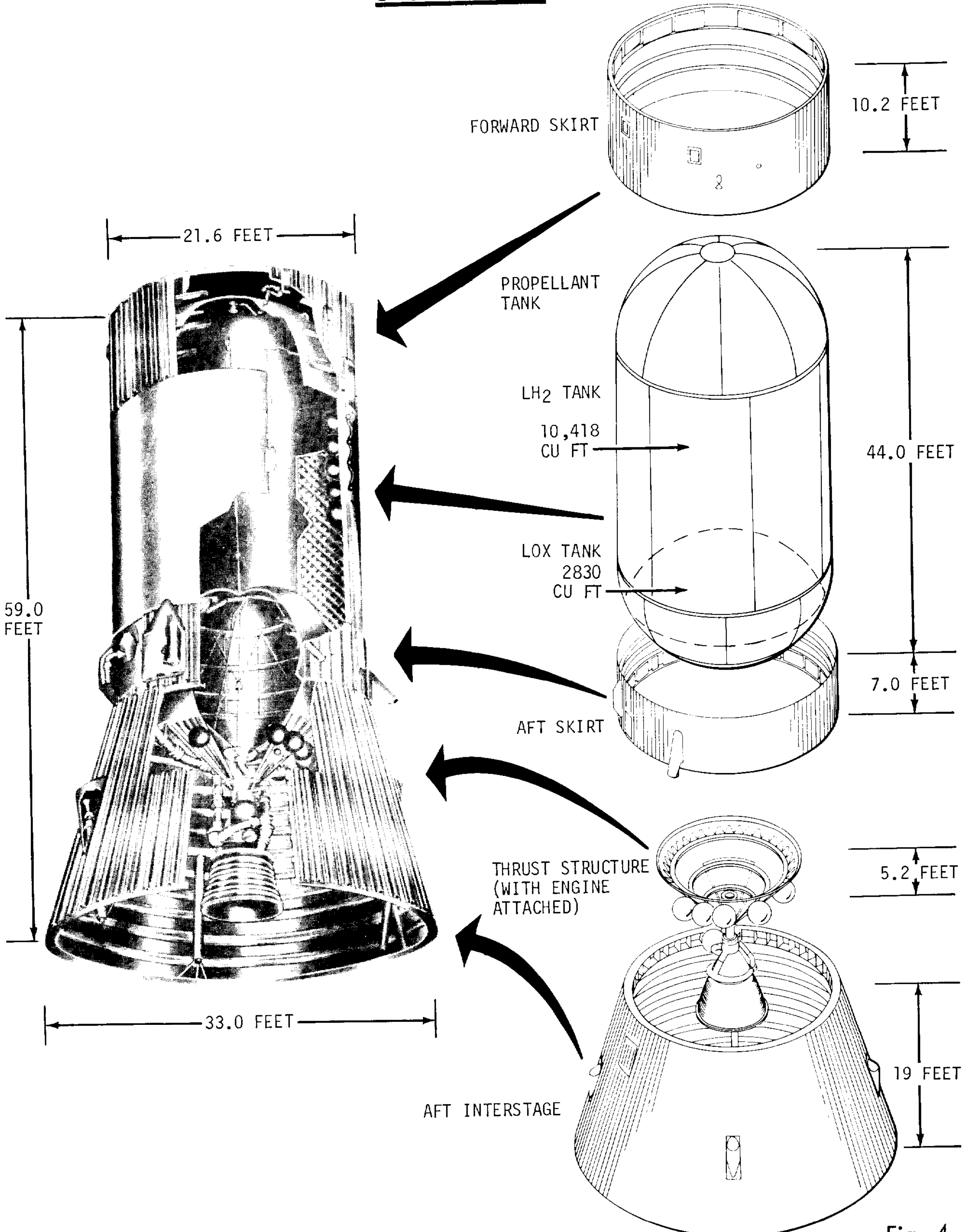


Fig. 4

During engine operation, the oxidizer tank is pressurized by flowing cold helium (from helium spheres mounted inside the fuel tank) through the heat exchanger in the oxidizer turbine exhaust duct. The heat exchanger heats the cold helium, causing it to expand. The fuel tank is pressurized during engine operation by GH_2 from the thrust chamber fuel manifold. Thrust vector control in the pitch and yaw planes during burn periods is achieved by gimbaling the entire engine.

The J-2 engine may receive cutoff signals from the following sources; EDS, range safety systems, "Thrust OK" pressure switches, propellant depletion sensors, and an IU programmed command (velocity or timed) via the switch selector.

The restart of the J-2 engine is identical to the initial start except for the fill procedure of the start tank. The start tank is filled with LH_2 and GH_2 during the first burn period by bleeding GH_2 from the thrust chamber fuel injection manifold and LH_2 from the Augmented Spark Igniter (ASI) fuel line to refill the start tank for engine restart. (Approximately 50 seconds of mainstage engine operation is required to recharge the start tank.)

To insure that sufficient energy will be available for spinning the fuel and oxidizer pump turbines, a waiting period of between approximately 80 minutes to 6 hours is required. The minimum time is required to build sufficient pressure by warming the start tank through natural means and to allow the hot gas turbine exhaust system to cool. Prolonged heating will cause a loss of energy in the start tank. This loss occurs when the LH_2 and GH_2 warm and raise the gas pressure to the relief valve setting. If this venting continues over a prolonged period the total stored energy will be depleted. This limits the waiting period prior to a restart attempt to six hours.

Propellant Systems

LOX is stored in the aft tank of the propellant tank structure at a temperature of -297°F. A six-inch, low-pressure supply duct supplies LOX from the tank to the engine. During engine burn, LOX is supplied at a nominal flow rate of 392 pounds per second, and at a transfer pressure above 25 psia. The supply duct is equipped with bellows to provide compensating flexibility for engine gimbaling, manufacturing tolerances, and thermal movement of structural connections. The tank is prepressurized to between 38 and 41 psia and is maintained at that pressure during boost and engine operation. Gaseous helium is used as the pressurizing agent.

The LH_2 is stored in an insulated tank at less than -423°F. LH_2 from the tank is supplied to the J-2 engine turbopump by a vacuum-jacketed, low-pressure, 10-inch duct. This duct is capable of flowing 80 pounds per second at -423°F and at a transfer pressure of 28 psia. The duct is located in the aft tank side wall above the common bulkhead joint. Bellows in this duct compensate for engine gimbaling, manufacturing tolerances, and thermal motion. The fuel tank is prepressurized to 28 psia minimum and 31 psia maximum.

The PU subsystem provides a means of controlling the propellant mass ratio. It consists of oxidizer and fuel tank mass probes, a PU valve, and an electronic assembly. These components monitor the propellant and maintain command control. Propellant utilization is provided by bypassing oxidizer from the oxidizer turbo-pump outlet back to the inlet. The PU valve is controlled by signals from the PU system. The engine oxidizer/fuel mixture mass ratio varies from 4.5:1 to 5.5:1.

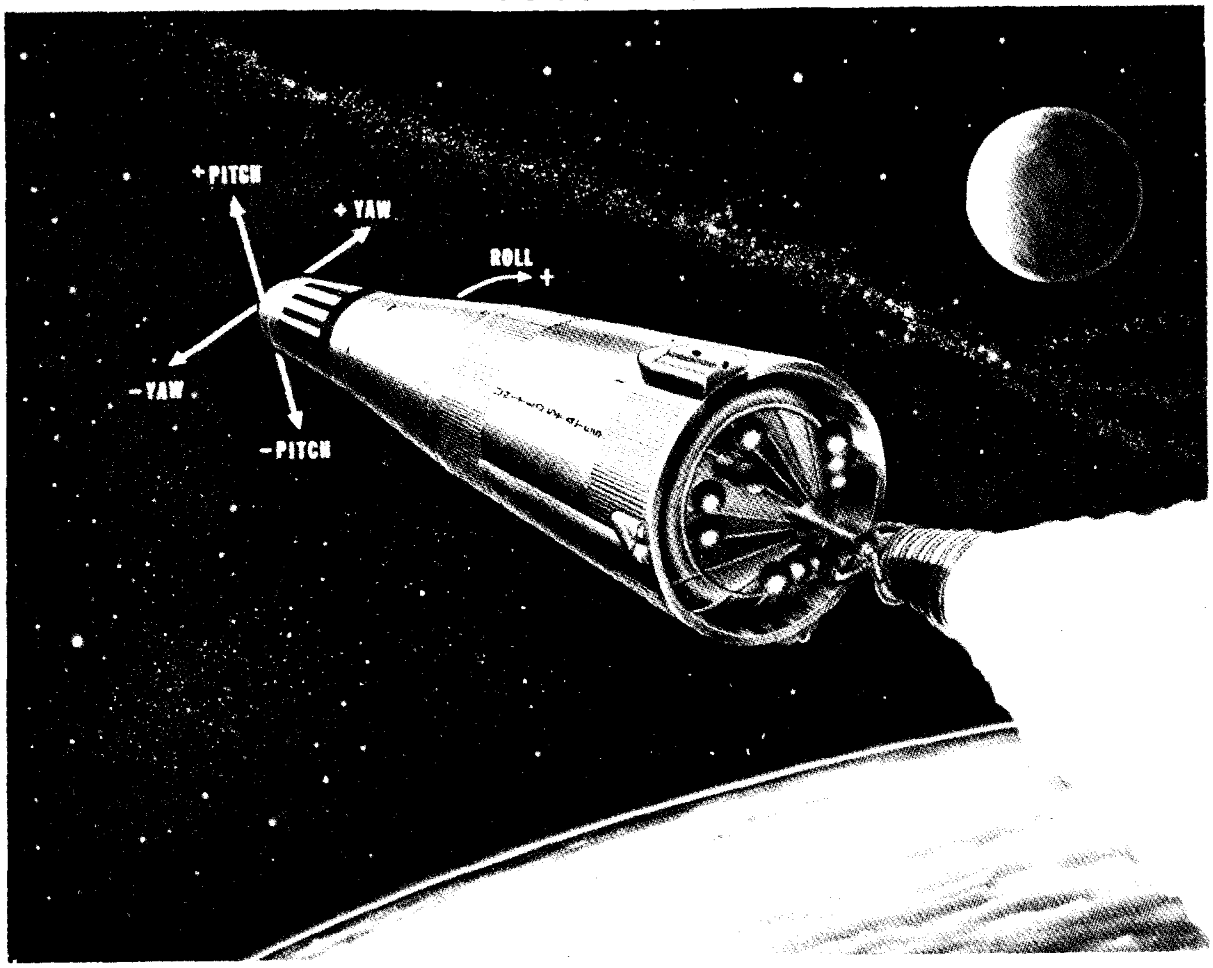
Flight Control System

The flight control system incorporates two systems for flight and attitude control. During powered flight, thrust vector steering is accomplished by gimbaling the J-2 engine for pitch and yaw control and by operating the Auxiliary Propulsion System (APS) engines for roll control. The engine is gimbaled in a ± 7.5 degree square pattern by a closed-loop hydraulic system. Mechanical feedback from the actuator to the servovalve provides the closed engine position loop. Two actuators are used to translate the steering signals into vector forces to position the engine. The deflection rates are proportional to the pitch and yaw steering signals from the flight control computer. Steering during coast flight is by use of the APS engine alone.

Auxiliary Propulsion System

The S-IVB APS provides three-axis stage attitude control (Figure 5) and main stage propellant control during coast flight. The APS engines are located in two modules 180° apart on the aft skirt of the S-IVB stage (Figure 6). Each module contains four engines; three 150-pound thrust control engines, and one 70-pound thrust ullage engine. Each module contains its own oxidizer, fuel, and pressurization system. A positive expulsion propellant feed subsystem is used to assure that hypergolic propellants are supplied to the engines under "zero g" or random gravity conditions. Nitrogen tetroxide (N_2O_4), is the oxidizer and monomethyl hydrazine (MMH), is the fuel for these engines.

APS FUNCTIONS



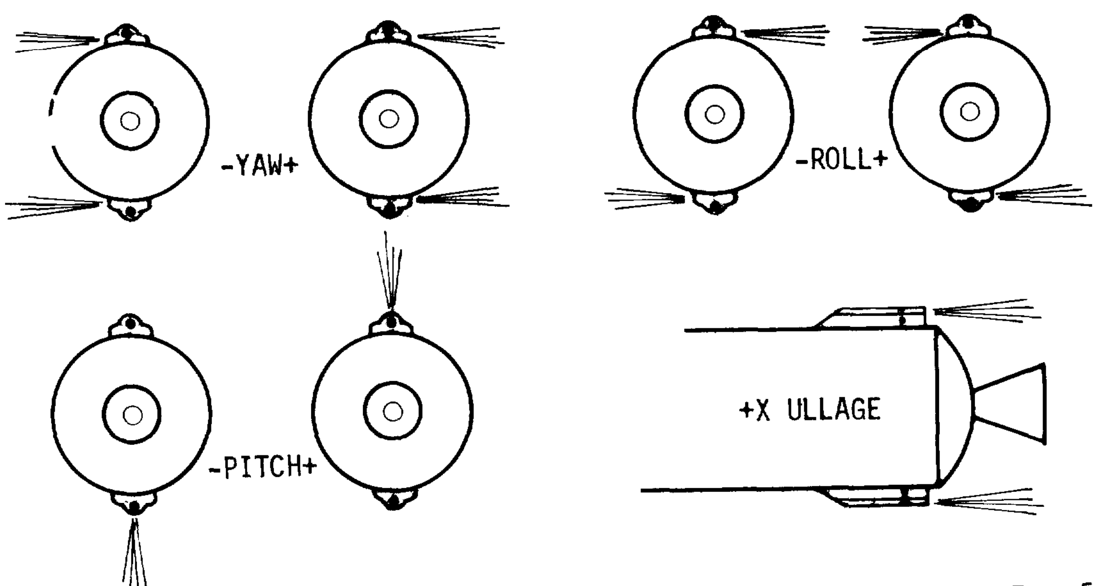


Fig. 5

APS CONTROL MODULE

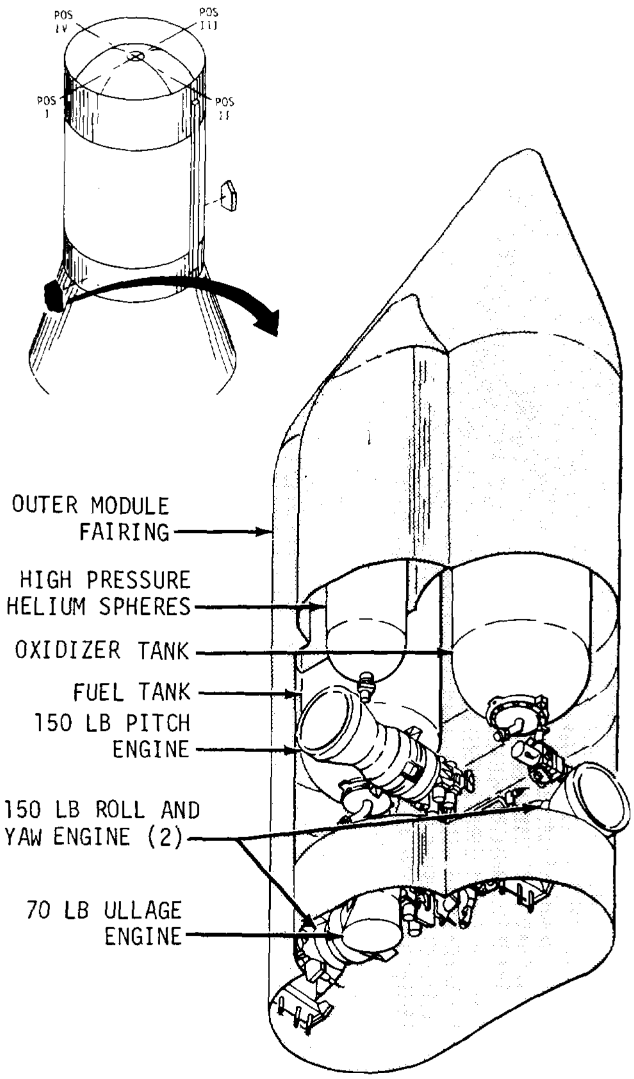


Fig. 6

Electrical

The electrical system of the S-IVB stage is comprised of two major subsystems: the electrical power subsystem which consists of all the power sources on the stage; and the electrical control subsystem which distributes power and control signals to various loads throughout the stage. Onboard electrical power is supplied by four silver-zinc batteries. Two are located in the forward equipment area and two in the aft equipment area. These batteries are activated and installed in the stage during the final pre-launch preparations. Heaters and instrumentation probes are an integral part of each battery.

Ordnance

The S-IVB ordnance systems include the separation, ullage rocket, and PDS systems. The separation plane for S-II/S-IVB staging is located at the top of the S-II/S-IVB interstage. At separation four retrorocket motors mounted on the interstage structure below the separation plane fire to decelerate the S-II stage with the interstage attached.

To provide propellant settling and thus ensure stable flow of fuel and oxidizer during J-2 engine start, the S-IVB stage requires a small acceleration. This acceleration is provided by two jettisonable ullage rockets for the first burn. The APS provides ullage for subsequent burns.

The S-IVB PDS provides for termination of vehicle flight by cutting two parallel 20-foot openings in the fuel tank and a 47-inch diameter hole in the LOX tank. The S-IVB PDS may be safed after the launch escape tower is jettisoned. Following S-IVB engine cutoff at orbit insertion, the PDS is electrically safed by ground command.

Instrument Unit

General

The Instrument Unit (IU) (Figure 7), is a cylindrical structure 21.6 feet in diameter and 3 feet high installed on top of the S-IVB stage. The IU contains the guidance, navigation, and control equipment for the Launch Vehicle. In addition, it contains measurements and telemetry, command communications, tracking, and emergency detection system components along with supporting electrical power and environmental control systems.

Structure

The basic IU structure is a short cylinder fabricated of an aluminum alloy honeycomb sandwich material. Attached to the inner surface of the cylinder are "cold plates" which serve both as mounting structure and thermal conditioning units for the electrical/electronic equipment.

Navigation, Guidance, and Control

The Saturn V Launch Vehicle is guided from its launch pad into earth orbit by navigation, guidance, and control equipment located in the IU. An all-inertial system utilizes a space-stabilized platform for acceleration and attitude measurements. A Launch Vehicle Digital Computer (LVDC) is used to solve guidance equations and a Flight Control Computer (FCC) (analog) is used for the flight control functions.

SATURN INSTRUMENT UNIT

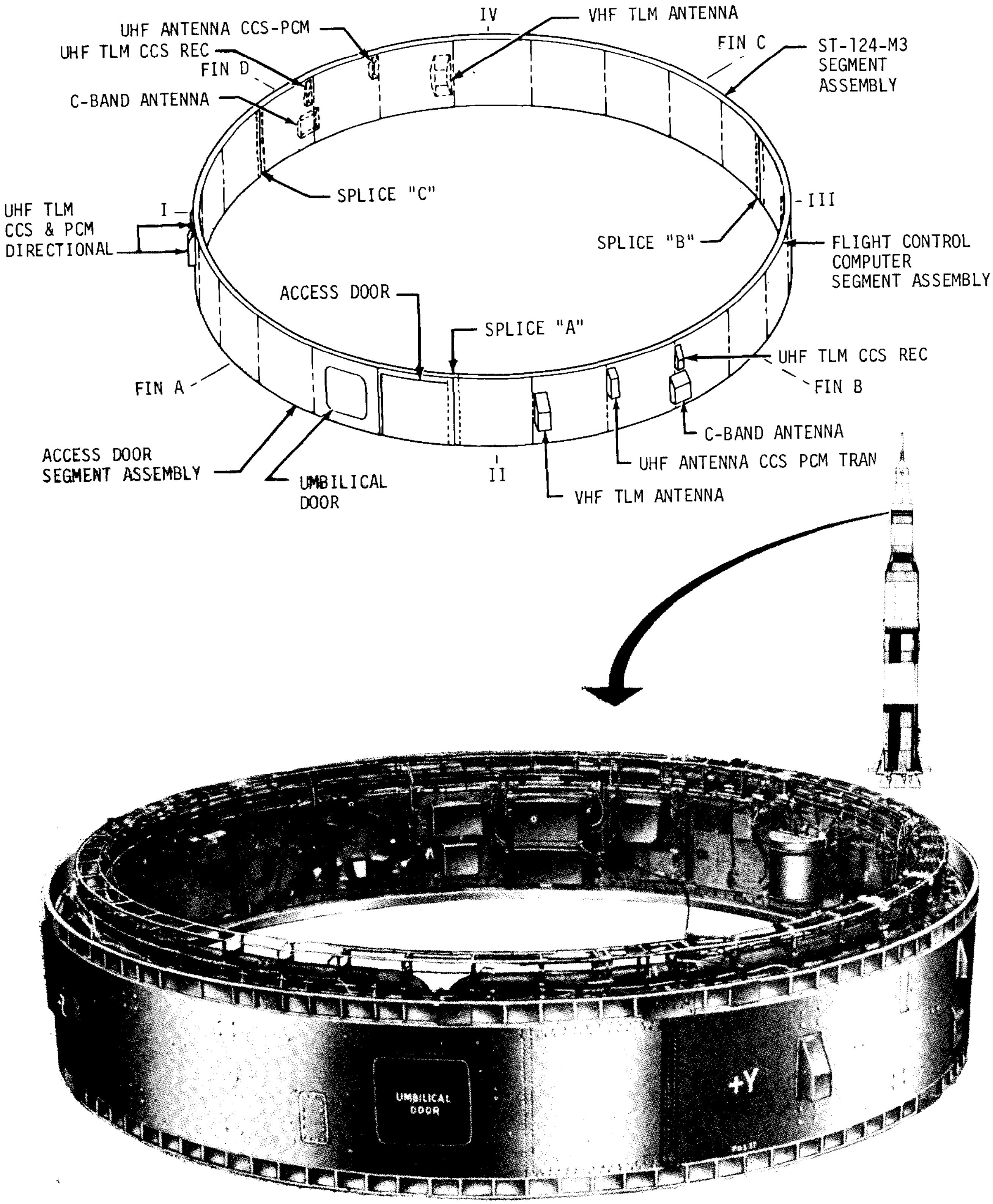


Fig. 7

The three-gimbal stabilized platform (ST-124-M3) provides a space-fixed coordinate reference frame for attitude control and for navigation (acceleration) measurements. Three integrating accelerometers, mounted on the gyro-stabilized inner gimbal of the platform, measure the three components of velocity resulting from vehicle propulsion. The accelerometer measurements are sent through the Launch Vehicle Data Adapter (LVDA) to the LVDC. In the LVDC, the accelerometer measurements are combined with the computed gravitational acceleration to obtain velocity and position of the vehicle. During orbital flight, the navigational program continually computes the vehicle position, velocity, and acceleration. Guidance information stored in the LVDC (e.g., position, velocity) can be updated through the IU command system by data transmission from ground stations. The IU command system provides the general capability of changing or inserting information into the LVDC.

The control subsystem is designed to maintain and control vehicle attitude by forming the steering commands to be used by the controlling engines of the active stage. The control system accepts guidance computations from the LVDC/LVDA Guidance System. These guidance commands, which are actually attitude error signals, are then combined with measured data from the various control sensors. The resultant output is the command signal to the various engine actuators and APS nozzles. The final computations (analog) are performed within the FCC. The FCC is also the central switching point for command signals. From this point, the signals are routed to their associated active stages and to the appropriate attitude control devices.

Measurements and Telemetry

The instrumentation within the IU consists of a measuring subsystem, a telemetry subsystem and an antenna subsystem. This instrumentation is for the purpose of monitoring certain conditions and events which take place within the IU and for transmitting monitored signals to ground receiving stations.

Command Communications System

The Command Communications System (CCS) provides for digital data transmission from ground stations to the LVDC. This communications link is used to update guidance information or command certain other functions through the LVDC. Command data originates in the Mission Control Center (MCC) and is sent to remote stations of the Manned Space Flight Network (MSFN) for transmission to to the Launch Vehicle.

Saturn Tracking Instrumentation

The Saturn V IU carries two C-band radar transponders and an Azusa/GLOTRAC tracking transponder. A combination of tracking data from different tracking systems provides the best possible trajectory information and increased reliability through redundant data. The tracking of the Saturn Launch Vehicle may be divided into four phases: powered flight into earth orbit; orbital flight; injection into mission trajectory; and coast flight after injection. Continuous tracking is required during powered flight into earth orbit. During orbital flight, tracking is accomplished by S-band stations of the MSFN and by C-band radar stations.

IU Emergency Detection System Components

The Emergency Detection System (EDS) is one element of several crew safety systems. There are nine EDS rate gyros installed in the IU. Three gyros monitor each of the three axes (pitch, roll, and yaw) thus providing triple redundancy. The control signal processor provides power to and receives inputs from the nine EDS rate gyros. These inputs are processed and sent on to the EDS distributor and to the flight control computer. The EDS distributor serves as a junction box and switching device to furnish the spacecraft display panels with emergency signals if emergency conditions exist. It also contains relay and diode logic for the automatic abort sequence.

An electronic timer in the IU allows multiple engine shutdowns without automatic abort after 30 seconds of flight. Inhibiting of automatic abort circuitry is also provided by the vehicle flight sequencing circuits through the IU switch selector. This inhibiting is required prior to normal S-IC engine cutoff and other normal vehicle sequencing. While the automatic abort is inhibited, the flight crew must initiate a manual abort if an angular-overrate or two engine-out condition arises.

Electrical Power Systems

Primary flight power for the IU equipment is supplied by silver-zinc oxide batteries at a nominal voltage level of 28 $\pm$2 vdc. Where ac power is required within the IU it is developed by solid state dc to ac inverters. Power distribution within the IU is accomplished through power distributors which are essentially junction boxes and switching circuits.

Environmental Control System

The Environmental Control System (ECS) maintains an acceptable operating environment for the IU equipment during pre-flight and flight operations. The ECS is composed of the following:

1. The Thermal Conditioning System (TCS) which maintains a circulating coolant temperature to the electronic equipment of 59° $\pm$1°F.
2. Pre-flight purging system which maintains a supply of temperature and pressure regulated air/gaseous nitrogen in the IU/S-IVB equipment area.
3. Gas bearing supply system which furnishes gaseous nitrogen to the ST-124-M3 inertial platform gas bearings.
4. Hazardous gas detection sampling equipment which monitors the IU/S-IVB forward interstage area for the presence of hazardous vapors.

APOLLO SPACECRAFT

The Apollo Spacecraft (S/C) is designed to support three men in space for periods up to two weeks, docking in space, landing on and returning from the lunar surface, and safely reentering the earth's atmosphere. The Apollo S/C consists of the Spacecraft LM Adapter (SLA), the Service Module (SM), the Command Module (CM), Launch Escape System (LES), and the Lunar Module (LM).

Spacecraft LM Adapter

General

The SLA (Figure 8) is a conical structure which provides a structural load path between the LV and SM and also supports the LM. Aerodynamically, the SLA smoothly encloses the irregularly shaped LM and transitions the space vehicle diameter from that of the upper stage of the LV to that of the SM. The SLA also encloses the nozzle of the SM engine and the High Gain Antenna.

Structure

The SLA is constructed of 1.7-inch thick aluminum honeycomb panels. The four upper jettisonable, or forward, panels are about 21 feet long, and the fixed lower, or aft, panels about 7 feet long. The exterior surface of the SLA is covered completely by a layer of cork. The cork helps insulate the LM from aerodynamic heating during boost. The LM is attached to the SLA at four locations around the lower panels.

SPACECRAFT LM ADAPTER

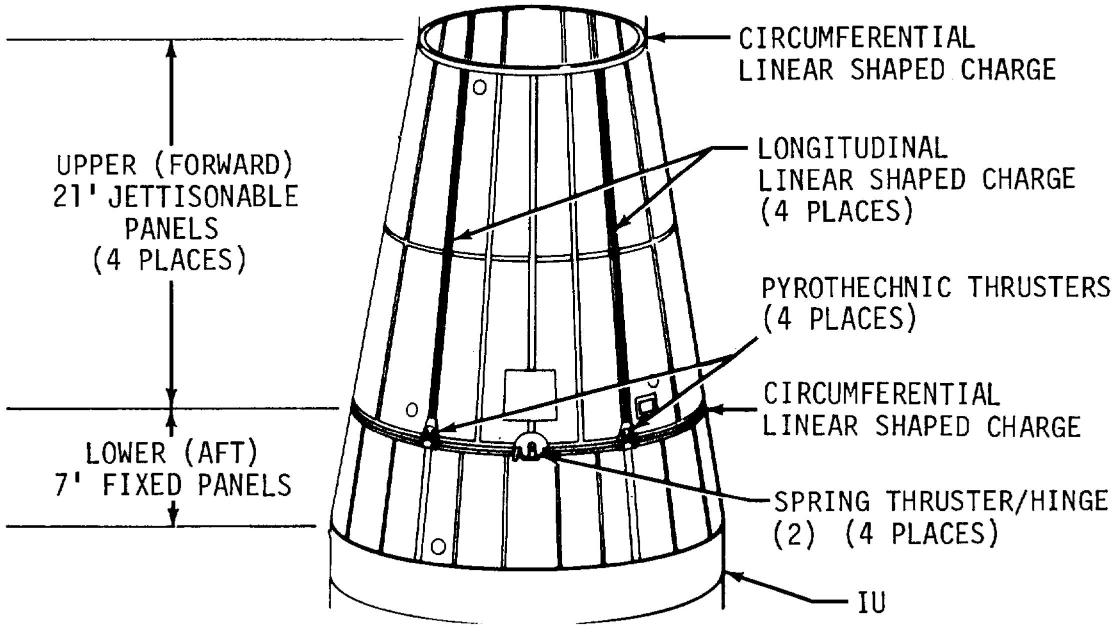


Fig. 8

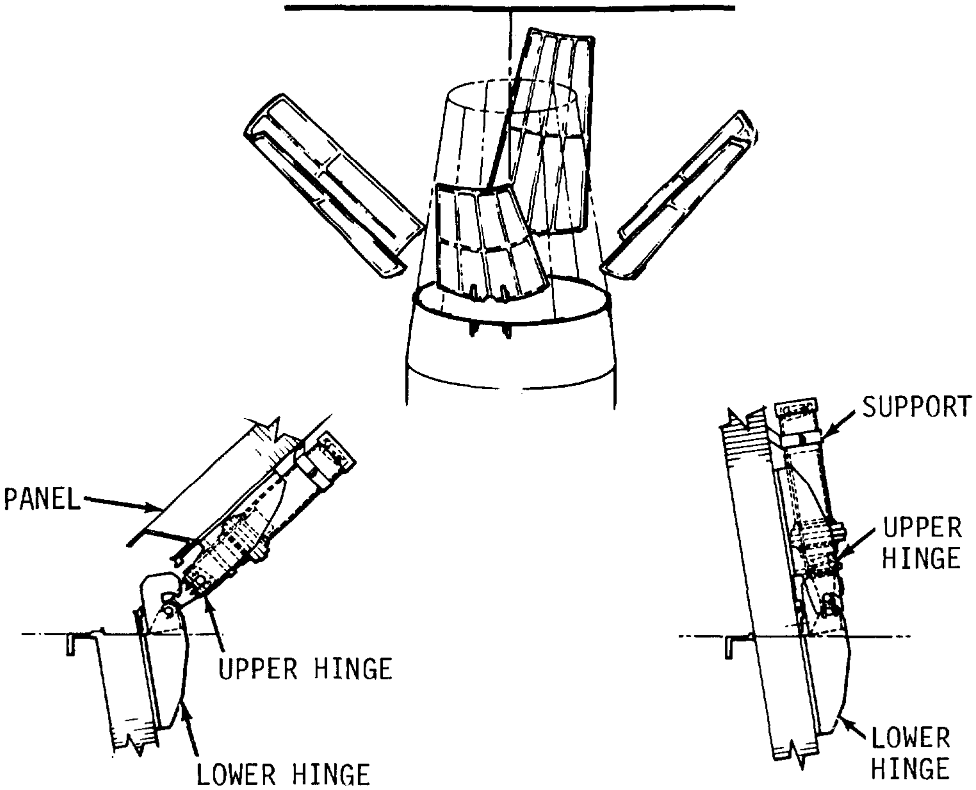
SLA-SM Separation

The SLA and SM are bolted together through flanges on each of the two structures. Explosive trains are used to separate the SLA and SM as well as for separating the four upper jettisonable SLA panels. Redundancy is provided in three areas to assure separation; redundant initiating signals, redundant detonators and cord trains, and "sympathetic" detonation of nearby charges.

Pryotechnic type and spring type thrusters (Figure 9) are used in deploying and jettisoning the SLA upper panels. The four double-piston pyrotechnic thrusters are located inside the SLA and start the panels swinging outward on their hinges. The two pistons of the thruster push on the ends of adjacent panels thus providing two separate thrusters operating each panel. The explosive train which separates the panels is routed through two pressure cartridges in each thruster assembly. The pyrotechnic thrusters rotate the panels 2 degrees establishing a constant angular velocity of 33 to 60 degrees per second. When the panels have rotated about 45 degrees, the partial hinges disengage and free the panels from the aft section of the SLA, subjecting them to the force of the spring thrusters.

The spring thrusters are mounted on the outside of the upper panels. When the panel hinges disengage, the springs in the thruster push against the fixed lower panels to propel the panels away from the vehicle at an angle of 110 degrees to the centerline at a speed of about 5-1/2 miles per hour. The panels will then depart the area of the spacecraft.

SLA PANEL JETTISONING



SPRING THRUSTER AFTER PANEL DEPLOYMENT, AT START OF JETTISON

SPRING THRUSTER BEFORE PANEL DEPLOYMENT

Fig. 9

SLA-LM Separation

Spring thrusters also are used to separate the LM from the SLA. After the command/service module has docked with the LM, mild charges are fired to release the four adapters which secure the LM in the SLA. Simultaneously, four spring thrusters mounted on the lower (fixed) SLA panels push against the LM Landing Gear Truss Assembly to separate the spacecraft from the launch vehicle.

The separation is controlled by two LM Separation Sequence Controllers located inside the SLA near the attachment point to the Instrument Unit (IU). The redundant controllers send signals which fire the charges that sever the connections and also fire a detonator to cut the LM-IU Umbilical. The detonator impels a guillotine blade which severs the umbilical wires.

Service Module

General

The Service Module (SM) (Figure 10) provides the main spacecraft propulsion and maneuvering capability during a mission. The SM provides most of the spacecraft consumables (oxygen, water, propellant, hydrogen) and supplements environmental, electrical power, and propulsion requirements of the CM. The SM remains attached to the CM until it is jettisoned just before CM reentry.

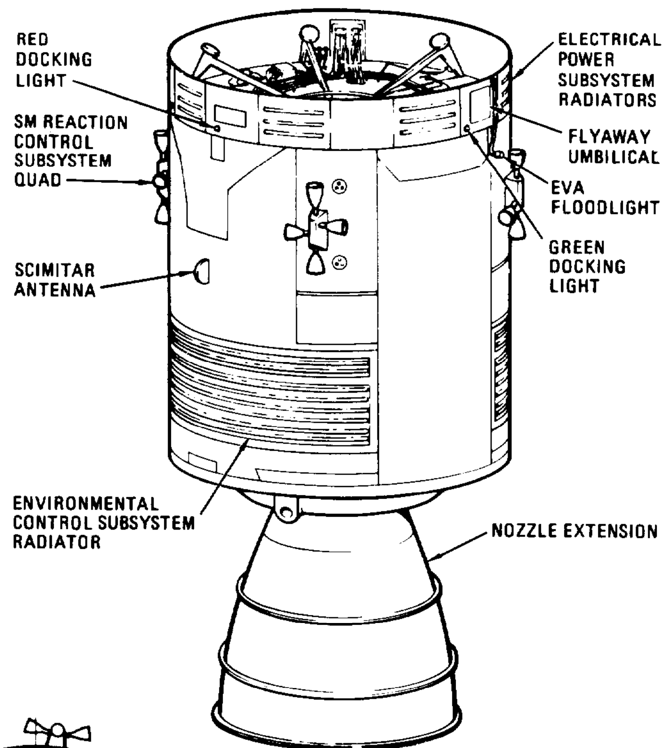
Structure

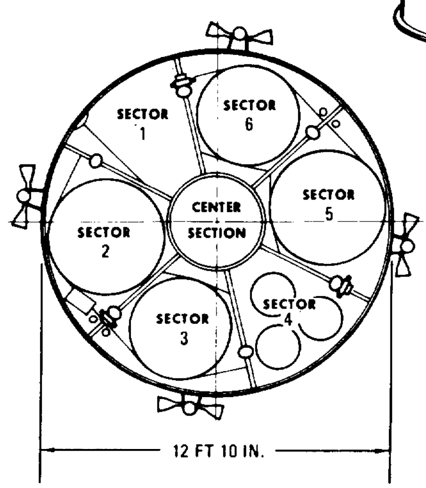
The basic structural components are forward and aft (upper and lower) bulkheads, six radial beams, four sector honeycomb panels, four reaction control system honeycomb panels, aft heat shield, and a fairing. The forward and aft bulkheads cover the top and botton of the SM. Radial beam trusses extending above the forward bulkhead support and secure the CM. The radial beams are made of solid aluminum alloy which has been machined and chem-milled to thicknesses varying between 2 inches and 0.018 inch. Three of these beams have compression pads and the other three have shear-compression pads and tension ties. Explosive charges in the center sections of these tension ties are used to separate the CM from the SM.

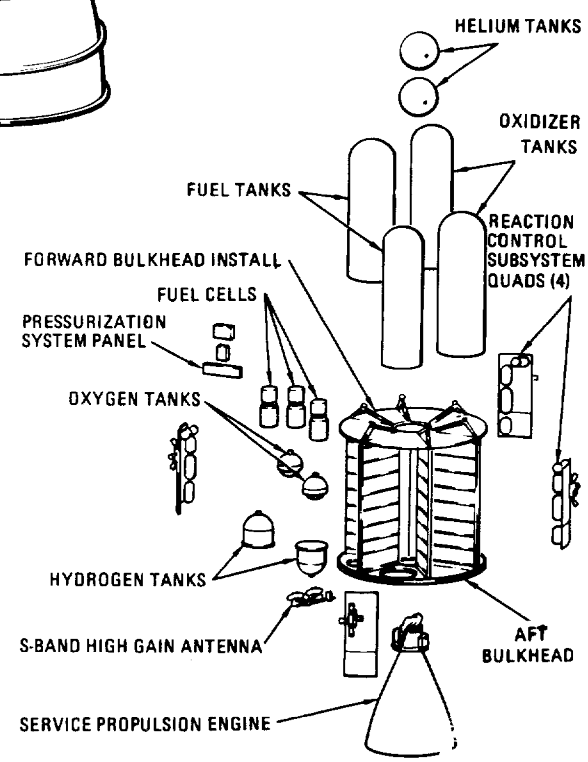
An aft heat shield surrounds the service propulsion engine to protect the SM from the engine's heat during thrusting. The gap between the CM and the forward bulkhead of the SM is closed off with a fairing which is composed of eight electrical power system radiators alternated with eight aluminum honeycomb panels. The sector and reaction control system panels are one inch thick and are made of aluminum honeycomb core between two aluminum face sheets. The sector panels are bolted to the radial beams. Radiators used to dissipate heat from the environmental control subsystem are bonded to the sector panels on opposite sides of the SM. These radiators are each about 30 square feet in area.

The SM interior is divided into six sectors and a center section. Sector one is currently void. It is available for installation of scientific or additional equipment should the need arise. Sector two has part of a space radiator and an RCS engine quad (module) on its exterior panel and contains the SPS oxidizer sump tank. This tank is the larger of the two tanks that hold the oxidizer for the SPS engine. Sector three has the rest of the space radiator and another RCS engine quad on its exterior panel and contains the oxidizer storage tank. This tank is the second of two SPS oxidizer tanks and is fed from the oxidizer sump tank in sector two. Sector four contains most of the electrical power generating equipment. It contains three fuel cells, two cryogenic oxygen and two cryogenic hydrogen tanks and a power control relay box. The cryogenic tanks supply oxygen to the environmental control subsystem and oxygen and hydrogen to the fuel cells. Sector five has part of an environmental control radiator and an RCS engine quad on the exterior panel and contains the SPS engine fuel sump tank. This tank feeds the engine and is also connected by feed lines to the fuel storage tank in sector six.

SERVICE MODULE







SECTOR 2 }	SERVICE PROPULSION SUBSYSTEM
SECTOR 3 }	OXIDIZER TANKS
SECTOR 4	OXYGEN TANKS, HYDROGEN TANKS, FUEL CELLS
SECTOR 5 }	SERVICE PROPULSION SUBSYSTEM
SECTOR 6 }	FUEL TANKS

CENTER SECTION - SERVICE PROPULSION ENGINE AND HELIUM TANKS

Fig. 10

Sector six has the rest of the environmental control radiator and an RCS engine quad on its exterior and contains the SPS engine fuel storage tank which feeds the fuel sump tank in sector five. The center section contains two helium tanks and the SPS engine. The tanks are used to provide helium pressurant for the SPS propellant tanks.

Propulsion

Main spacecraft propulsion is provided by the 20,500-pound thrust Service Propulsion System (SPS). The SPS engine is a restartable, non-throttleable engine which uses nitrogen tetroxide as an oxidizer and a 50-50 mixture of hydrazine and unsymmetrical dimethylhydrazine as fuel. This engine is used for major velocity changes during the mission such as midcourse corrections, lunar orbit insertion, transearth injection, and CSM aborts. The SPS engine responds to automatic firing commands from the guidance and navigation system or to commands from manual controls. The engine assembly is gimbal-mounted to allow engine thrust-vector alignment with the spacecraft center of mass to preclude tumbling. Thrust vector alignment control is maintained automatically by the stabilization and control system or manually by the crew. The Service Module Reaction Control System (SM RCS) provides for maneuvering about and along three axes. (See page 40 for more comprehensive description.)

Additional SM Systems

In addition to the systems already described the SM has communication antennas, umbilical connections, and several exterior mounted lights. The four antennas on the outside of the SM are the steerable S-band high-gain antenna, mounted on the aft bulkhead; two VHF omnidirectional antennas, mounted on opposite sides of the module near the top; and the rendezvous radar transponder antenna, mounted in the SM fairing.

The umbilicals consist of the main plumbing and wiring connections between the CM and SM enclosed in a fairing (aluminum covering), and a "flyaway" umbilical which is connected to the launch tower. The latter supplies oxygen and nitrogen for cabin pressurization, water-gylcol, electrical power from ground equipment, and purge gas.

Seven lights are mounted in the aluminum panels of the fairing. Four (one red, one green, and two amber) are used to aid the astronauts in docking, one is a floodlight which can be turned on to give astronauts visibility during extravehicular activities, one is a flashing beacon used to aid in rendezvous, and one is a spotlight used in rendezvous from 500 feet to docking with the LM.

SM/CM Separation

Separation of the SM from the CM occurs shortly before reentry. The sequence of events during separation is controlled automatically by two redundant Service Module Jettison Controllers (SMJC) located on the forward bulkhead of the SM. Physical separation requires severing of all the connections between the modules, transfer of electrical control, and firing of the SM RCS to increase the distance between the CM and SM. A tenth of a second after electrical connections are deadfaced, the SMJC's send signals which fire ordnance devices to sever the three tension ties and the umbilical. The tension ties are straps which hold the CM on three of the compression pads on the SM. Linear-shaped charges in each tension tie assembly sever the tension ties to separate the CM from the SM. At the same time, explosive charges drive guillotines through the wiring and tubing in the umbilical. Simultaneously with the firing of the ordnance devices, the SMJC's send signals which fire the SM RCS. Roll engines are fired for five seconds to alter the SM's course from that of the CM, and the translation (thrust) engines are fired continuously until the propellant is depleted or fuel cell power is expended. These maneuvers carry the SM well away from the entry path of the CM.

Command Module

General

The Command Module (CM) (Figure 11) serves as the command, control, and communications center for most of the mission. Supplemented by the SM, it provides all life support elements for three crewmen in the mission environments and for their safe return to earth's surface. It is capable of attitude control about three axes and some lateral lift translation at high velocities in earth atmosphere. It also permits LM attachment, CM/LM ingress and egress, and serves as a buoyant vessel in open ocean.

Structure

The CM consists of two basic structures joined together: the inner structure (pressure shell) and the outer structure (heat shield). The inner structure, the pressurized crew compartment, is made of aluminum sandwich construction consisting of a welded aluminum inner skin, bonded aluminum honeycomb core and outer face sheet. The outer structure is basically a heat shield and is made of stainless steel brazed honeycomb brazed between steel alloy face sheets. Parts of the area between the inner and outer sheets is filled with a layer of fibrous insulation as additional heat protection.

COMMAND MODULE

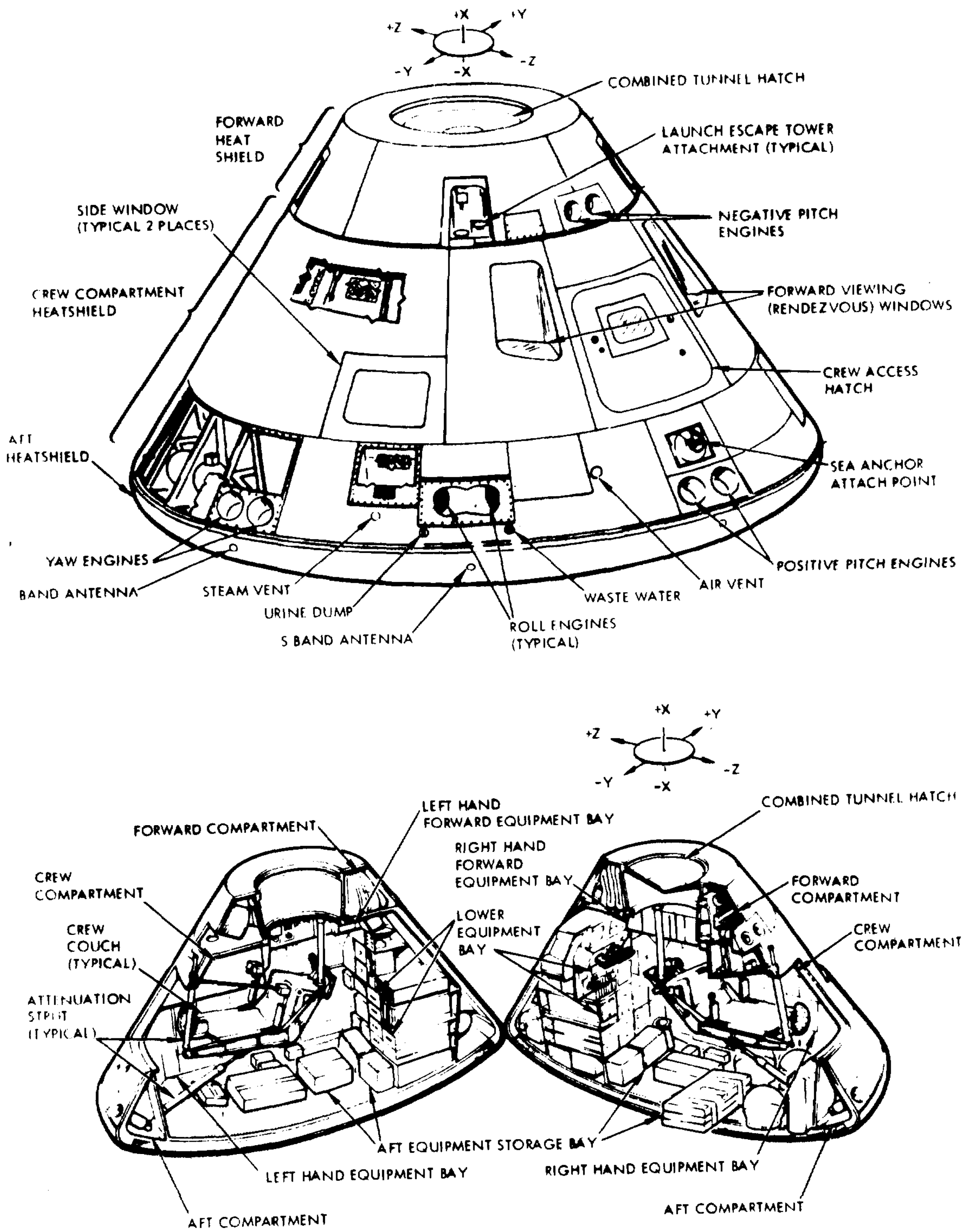


Fig. 11

Thermal Protection (Heat Shields)

The interior of the CM must be protected from the extremes of environment that will be encountered during a mission. The heat of launch is absorbed principally through the Boost Protective Cover (BPC), a fiberglass structure covered with cork which encloses the CM. The cork is covered with a white reflective coating. The BPC is permanently attached to the launch escape tower and is jettisoned with it.

The insulation between the inner and outer shells, plus temperature control provided by the environmental control subsystem, protects the crew and sensitive equipment in space. The principal task of the heat shield that forms the outer structure is to protect the crew during reentry. This protection is provided by ablative heat shields of varying thicknesses covering the CM. The ablative material is a phenolic epoxy resin. This material turns white hot, chars, and then melts away, conducting relatively little heat to the inner structure. The heat shield has several outer coverings: a pore seal, a moisture barrier (white reflective coating), and a silver Mylar thermal coating.

Forward Compartment

The forward compartment is the area around the forward (docking) tunnel. It is separated from the crew compartment by a bulkhead and covered by the forward heat shield. The compartment is divided into four 90-degree segments which contain earth landing equipment (all the parachutes, recovery antennas and beacon light, and sea recovery sling, etc.), two RCS engines, and the forward heat shield release mechanism.

The forward heat shield contains four recessed fittings into which the legs of the launch escape tower are attached. The tower legs are connected to the CM structure by frangible nuts containing small explosive charges, which separate the tower from the CM when the Launch Escape System is jettisoned. The forward heat shield is jettisoned at about 25,000 feet during return to permit deployment of the parachutes.

Aft Compartment

The aft compartment is located around the periphery of the CM at its widest part, near the aft heat shield. The aft compartment bays contain 10 RCS engines; the fuel, oxidizer, and helium tanks for the CM RCS; water tanks; the crushable ribs of the impact attenuation system; and a number of instruments. The CM-SM umbilical is also located in the aft compartment.

Crew Compartment

The crew compartment has a habitable volume of 210 cubic feet. Pressurization and temperature are maintained by the ECS. The crew compartment contains the controls and displays for operation of the spacecraft, crew couches, and all the other equipment needed by the crew. It contains two hatches, five windows, and a number of equipment bays.

Equipment Bays

The equipment bays contain items needed by the crew for up to 14 days, as well as much of the electronics and other equipment needed for operation of the spacecraft. The bays are named according to their position with reference to the couches. The lower equipment bay is the largest and contains most of the guidance and navigation electronics, as well as the sextant and telescope, the Command Module Computer (CMC), and a computer keyboard. Most of the telecommunications subsystem electronics are in this bay, including the five batteries, inverters, and battery charger of the electrical power subsystem. Stowage areas in the bay contain food supplies, scientific instruments, and other astronaut equipment.

The left-hand equipment bay contains key elements of the ECS. Space is provided in this bay for stowing the forward hatch when the CM and LM are docked and the tunnel between the modules is open. The left-hand forward equipment bay also contains ECS equipment, as well as the water delivery unit and clothing storage.

The right-hand equipment bay contains waste management system controls and equipment, electrical power equipment, and a variety of electronics, including sequence controllers and signal conditioners. Food also is stored in a compartment in this bay. The right-hand forward equipment bay is used principally for stowage and contains such items as survival kits, medical supplies, optical equipment, the LM docking target, and bioinstrumentation harness equipment.

The aft equipment bay is used for storing space suits and helmets, life vests, the fecal canister, portable life support systems (backpacks), and other equipment, and includes space for stowing the probe and drogue assembly.

Hatches

The two CM hatches are the side hatch, used for getting in and out of the CM, and the forward hatch, used to transfer to and from the LM when the CM and LM are docked. The side hatch is a single integrated assembly which opens outward and has primary and secondary thermal seals. The hatch normally contains a small window, but has provisions for installation of an airlock. The latches for the side hatch are so designed that pressure exerted against the hatch serves only to increase the locking pressure of the latches. The hatch handle mechanism also operates a mechanism which opens the access hatch in the BPC. A counterbalance assembly which consists of two nitrogen bottles and a piston assembly enables the hatch and BPC hatch to be opened easily. In space, the crew can operate the hatch easily without the counter balance, and the piston cylinder and nitrogen bottle can be vented after launch. A second nitrogen bottle can be used to open the hatch after landing. The side hatch can readily be opened from the outside. In case some deformation or other malfunction prevented the latches from engaging, three jack-screws are provided in the crew's tool set to hold the door closed.

The forward (docking) hatch is a combined pressure and ablative hatch mounted at the top of the docking tunnel. The exterior or upper side of the hatch is covered with a half-inch of insulation and a layer of aluminum foil. This hatch has a six-point latching arrangement operated by a pump handle similar to that on the side hatch and can also be opened from the outside. It has a pressure equalization valve so that the pressure in the tunnel and that in the LM can be equalized before the hatch is removed. There are also provisions for opening the latches manually if the handle gear mechanism should fail.

Windows

The CM has five windows: two side (numbers 1 and 5), two rendezvous (numbers 2 and 4), and a hatch window (number 3 or center). The hatch window is over the center couch. The windows each consist of inner and outer panes. The inner windows are made of tempered silica glass with quarter-inch thick double panes, separated by a tenth of an inch. The outer windows are made of amorphous-fused silicon with a single pane seven tenths of an inch thick. Each pane has an anti-reflecting coating on the external surface and a blue-red reflective coating on the inner surface to filter out most infrared and all ultraviolet rays. The outer window glass has a softening temperature of 2800°F and a melting point of 3110°F. The inner window glass has a softening temperature of 2000°F. Aluminum shades are provided for all windows.

Impact Attenuation

During a water impact the CM deceleration force will vary considerably depending on the shape of the waves and the dynamics of the CM's descent. A major portion of the energy (75 to 90 percent) is absorbed by the water and by deformation of the CM structure. The impact attenuation system reduces the forces acting on the crew to a tolerable level. The impact attenuation system is part internal and part external. The external part consists of four crushable ribs (each about four inches thick and a foot in length) installed in the aft compartment. The ribs are made of bonded laminations of corrugated aluminum which absorb energy by collapsing upon impact. The main parachutes suspend the CM at such an angle that the ribs are the first point of the module that hits the water. The internal portion of the system consists of eight struts which connect the crew couches to the CM structure. These struts absorb energy by deforming steel wire rings between an inner and an outer piston.

Displays and Controls

The Main Display Console (Figure 12) has been arranged to provide for the expected duties of crew members. These duties fall into the categories of Commander, CM Pilot, and LM Pilot, occupying the left, center, and right couches respectively. The CM Pilot also acts as the principal navigator. All controls have been designed so they can be operated by astronauts wearing gloves. The controls are predominantly of four basic types: toggle switches, rotary switches with click-stops, thumbwheels, and push buttons. Critical switches are guarded so that they cannot be thrown inadvertently. In addition, some critical controls have locks that must be released before they can be operated.

MAIN DISPLAY CONSOLE

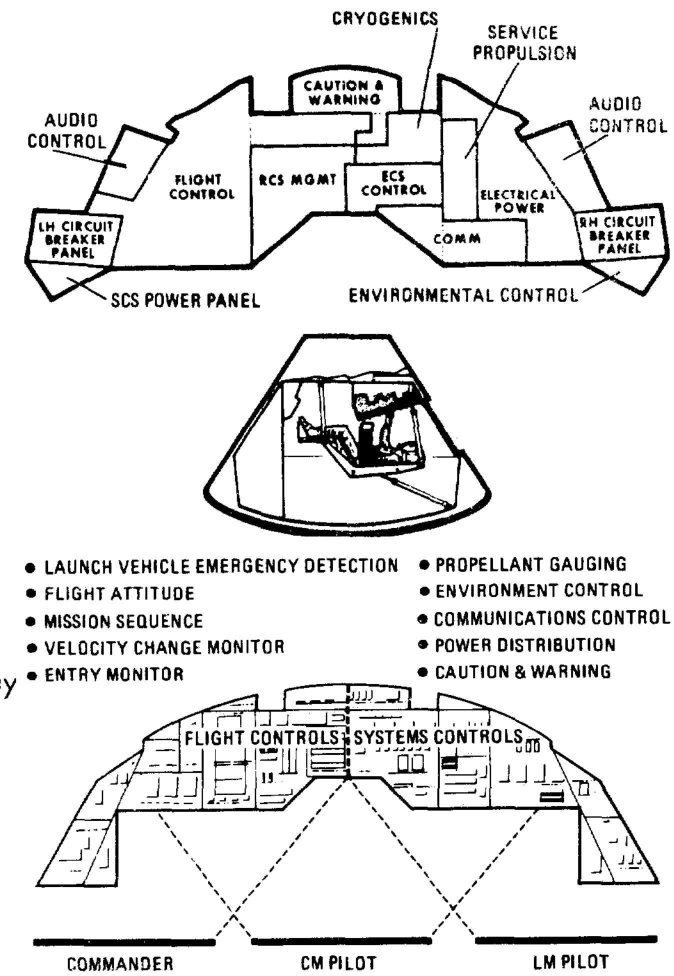


Fig. 12

Flight controls are located on the left-center and left side of the Main Display Console, opposite the Commander. These include controls for such subsystems as stabilization and control, propulsion, crew safety, earth landing, and emergency detection. One of two guidance and navigation computer panels also is located here, as are velocity, attitude, and altitude indicators.

The CM Pilot faces the center of the console and thus can reach many of the flight controls, as well as the system controls on the right side of the console. Displays and controls directly opposite him include reaction control propellant management, caution and warning, environmental control, and cryogenic storage systems.

The LM Pilot couch faces the right-center and right side of the console. Communications, electrical control, data storage, and fuel cell system components are located here, as well as service propulsion subsystem propellant management.

Other displays and controls are placed throughout the cabin in the various equipment bays and on the crew couches. Most of the guidance and navigation equipment is in the lower equipment bay, at the foot of the center couch. This equipment, including the sextant and telescope, is operated by an astronaut standing and using a simple restraint system. The non-time-critical controls of the environmental control system are located in the left-hand equipment bay, while all the controls of the waste management system are on a panel in the right-hand equipment bay. The rotation and translation controllers used for attitude, thrust vector, and translation maneuvers are located on the arms of two crew couches. In addition, a rotation controller can be mounted at the navigation position in the lower equipment bay.

Critical conditions of most spacecraft systems are monitored by a Caution And Warning System. A malfunction or out-of-tolerance condition results in illumination of a status light that identifies the abnormality. It also activates the master alarm circuit, which illuminates two master alarm lights on the Main Display Console and one in the lower equipment bay and sends an alarm tone to the astronauts' headsets. The master alarm lights and tone continue until a crewman resets the master alarm circuit. This can be done before the crewmen deal with the problem indicated. The Caution And Warning System also contains equipment to sense its own malfunctions.

Telecommunications

The telecommunications system (Figure 13) provides voice, television, telemetry, tracking, and ranging communications between the spacecraft and earth, between the CM and LM, and between the spacecraft and astronauts wearing the Portable Life Support System (PLSS). It also provides communications among the astronauts in the spacecraft and includes the central timing equipment for synchronization of other equipment and correlation of telemetry equipment. For convenience, the telecommunications subsystem can be divided into four areas: intercommunications (voice), data, radio frequency equipment, and antennas.

Intercommunications

The astronauts' headsets are used for all voice communications. Each astronaut has an audio control panel on the Main Display Console which enables him to control what comes into his headset and where he will send his voice. The three headsets and audio control panels are connected to three identical audio center modules. The audio center is the assimilation and distribution point for all spacecraft voice signals. The audio signals can be routed from the center to the

TELECOMMUNICATIONS SYSTEM

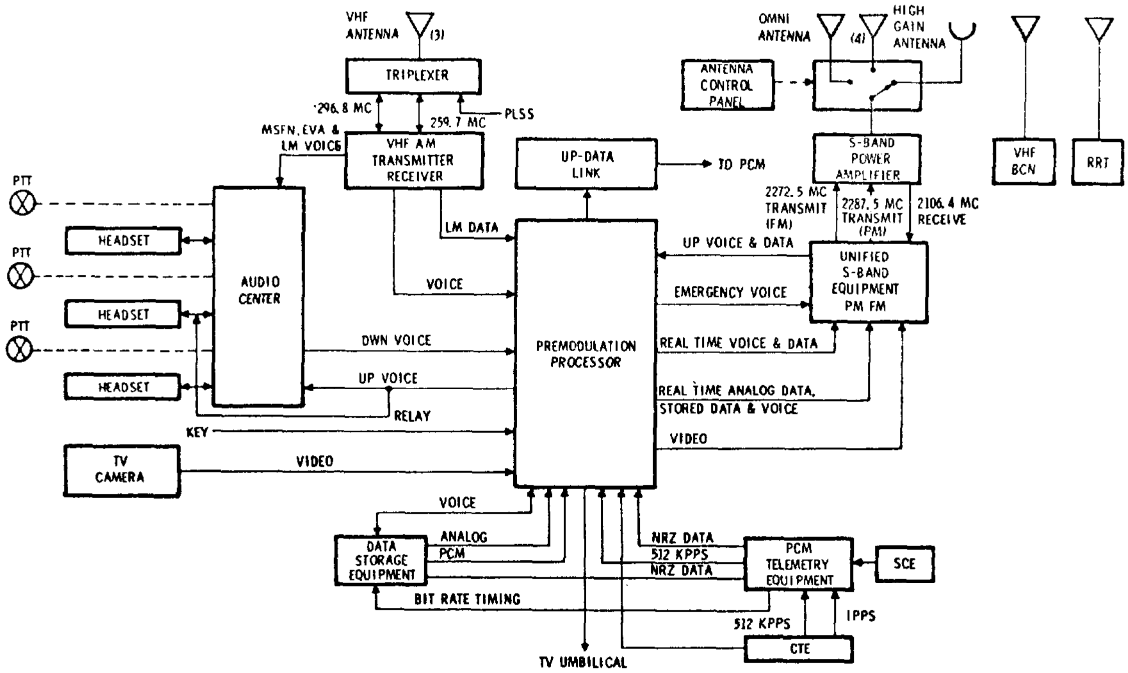


Fig. 13

appropriate transmitter or receiver, the Launch Control Center (for pre-launch checkout), the recovery forces intercom, or voice tape recorders.

Two methods of voice transmission and reception are possible: The VHF/AM transmitter-receiver and the S-band transmitter and receiver. The VHF/AM equipment is used for voice communications with the Manned Space Flight Network during launch, ascent, and near-earth phases of a mission. The S-band equipment is used during both near-earth and deep-space phases of a mission. When communications with earth are not possible, a limited number of audio signals can be stored on tape for later transmission. The CSM communication range capability is depicted in Figure 14.

Data

The spacecraft structure and subsystems contain sensors which gather data on their status and performance. Biomedical, TV, and timing data also are gathered. These various forms of data are assimilated into the data system, processed, and then transmitted to the ground. Some data from the operational systems, and some voice communications, may be stored for later transmission or for recovery after landing. Stored data can be transmitted to the ground simultaneously with voice or realtime data.

CSM COMMUNICATION RANGES

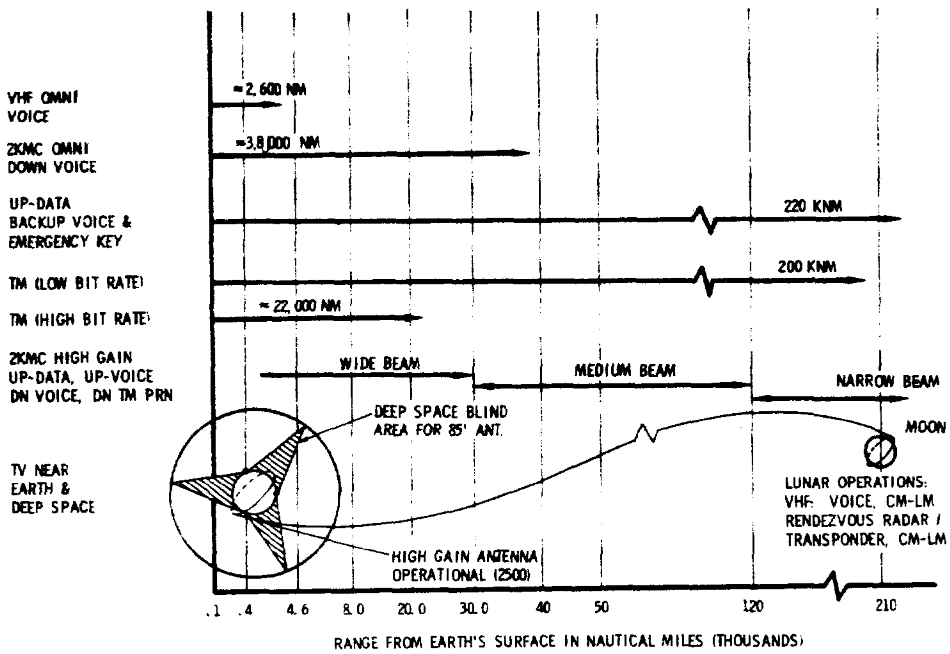


Fig. 14

Radio Frequency Equipment

The radio frequency equipment is the means by which voice information, telemetry data, and ranging and tracking information are transmitted and received. The equipment consists of two VHF/AM transceivers in one unit, the unified S-band equipment (primary and secondary transponders and an FM transmitter), primary and secondary S-band power amplifiers (in one unit), a VHF beacon, an X-band transponder (for rendezvous radar), and the premodulation processor.

The equipment provides for voice transfer between the CM and the ground, between the CM and LM, between the CM and extravehicular astronauts, and between the CM and recovery forces. Telemetry can be transferred between the CM and the ground, from the LM to the CM and then to the ground, and from extravehicular astronauts to the CM and then to the ground. Ranging information consists of pseudo-random noise and double-Doppler ranging signals from the ground to the CM and back to the ground, and of X-band radar signals from the LM to the CM and back to the LM. The VHF beacon equipment emits a 2-second signal every five seconds for line-of-sight direction finding to aid recovery forces in locating the CM after landing.

Antennas

LOCATION OF ANTENNAS

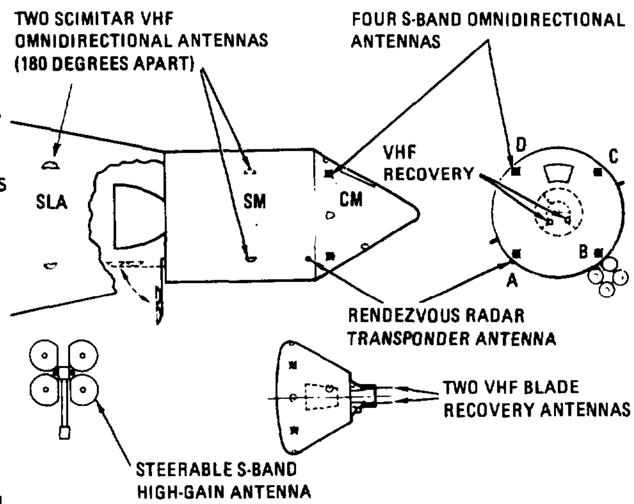


Fig. 15

There are nine antennas (Figure 15) on the CSM, not counting the rendezvous radar antenna which is an integral part of the rendezvous radar transponder. These antennas can be divided into four groups: VHF, S-band, recovery, and beacon. The two VHF antennas (called scimitars because of their shape) are omnidirectional and are mounted 180 degrees apart on the SM. There are five S-band antennas, one mounted at the bottom of the SM and four located 90 degrees apart around the CM. The S-band high-gain antennas, used for deep space communications, is composed of four 31-inch diameter reflectors surrounding an 11-inch square reflector. At launch it is folded down parallel to the SPS engine nozzle so that it fits within the spacecraft LM adapter. After the CSM separates from the SLA the antenna is deployed at a right angle to the SM center line. The four smaller surface-mounted S-band antennas are used at near-earth ranges and deep-space backup. The high-gain antenna is deployable after CSM/SLA separation. It can be steered through a gimbal system and is the principal antenna for deep-space communications. The four S-band antennas on the CM are mounted flush with the surface of the CM and are used for S-band communications during near-earth phases of the mission, as well as for a backup in deep space. The two VHF recovery antennas are located in the forward compartment of the CM, and are deployed automatically shortly after the main parachutes deploy. One of these antennas also is connected to the VHF recovery beacon.

Environmental Control System

The Environmental Control System (ECS) provides a controlled environment for three astronauts for up to 14 days. For normal conditions, this environment includes a pressurized cabin (five pounds per square inch), a 100-percent oxygen atmosphere, and a cabin temperature of 70 to 75 degrees fahrenheit. The system provides a pressurized suit circuit for use during critical mission phases and for emergencies.

The ECS provides oxygen and hot and cold water, removes carbon dioxide and odors from the CM cabin, provides for venting of waste, and dissipates excessive heat from the cabin and from operating electronic equipment. It is designed so that a minimum amount of crew time is needed for its normal operation. The main unit contains the coolant control panel, water chiller, two water-glycol evaporators, carbon dioxide odor-absorber canisters, suit heat exchanger, water separator, and compressors. The oxygen surge tank, water glycol pump package and reservoir, and control panels for oxygen and water are adjacent to the unit.

The system is concerned with three major elements: oxygen, water, and coolant (water-glycol). All three are interrelated and intermingled with other systems. These three elements provide the major functions of spacecraft atmosphere, thermal control, and water management through four major subsystems: oxygen, pressure suit circuit, water, and water-glycol. A fifth subsystem, post-landing ventilation, also is part of the environmental control system, providing outside air for breathing and cooling prior to hatch opening.

The CM cabin atmosphere is 60 percent oxygen and 40 percent nitrogen on the launch pad to reduce fire hazard. The mixed atmosphere supplied by ground equipment will gradually be changed to pure oxygen after launch as the environmental control system maintains pressure and replenishes the cabin atmosphere.

During pre-launch and initial orbital operation, the suit circuit supplies pure oxygen at a flow rate slightly more than is needed for breathing and suit leakage. This results in the suit being pressurized slightly above cabin pressure, which prevents cabin gases from entering and contaminating the suit circuit. The excess oxygen in the suit circuit is vented into the cabin.

Spacecraft heating and cooling is performed through two water-glycol coolant loops. The water-glycol, initially cooled through ground equipment, is pumped through the primary loop to cool operating electric and electronic equipment, the space suits, and the cabin heat exchangers. The water-glycol also is circulated through a reservoir in the CM to provide a heat sink during ascent.

Earth Landing System

The Earth Landing System (ELS) (Figure 16) provides a safe landing for the astronauts and the CM. Several recovery aids which are activated after splashdown are part of the system. Operation normally is automatic, timed, and activated by the sequential control system. All automatic functions can be backed up manually.

For normal entry, about 1.5 seconds after forward heat shield jettison, the two drogue parachutes are deployed to orient the CM properly and to provide initial deceleration. At about 10,000 feet, the drogue parachutes are released

ELS MAJOR COMPONENT STOWAGE

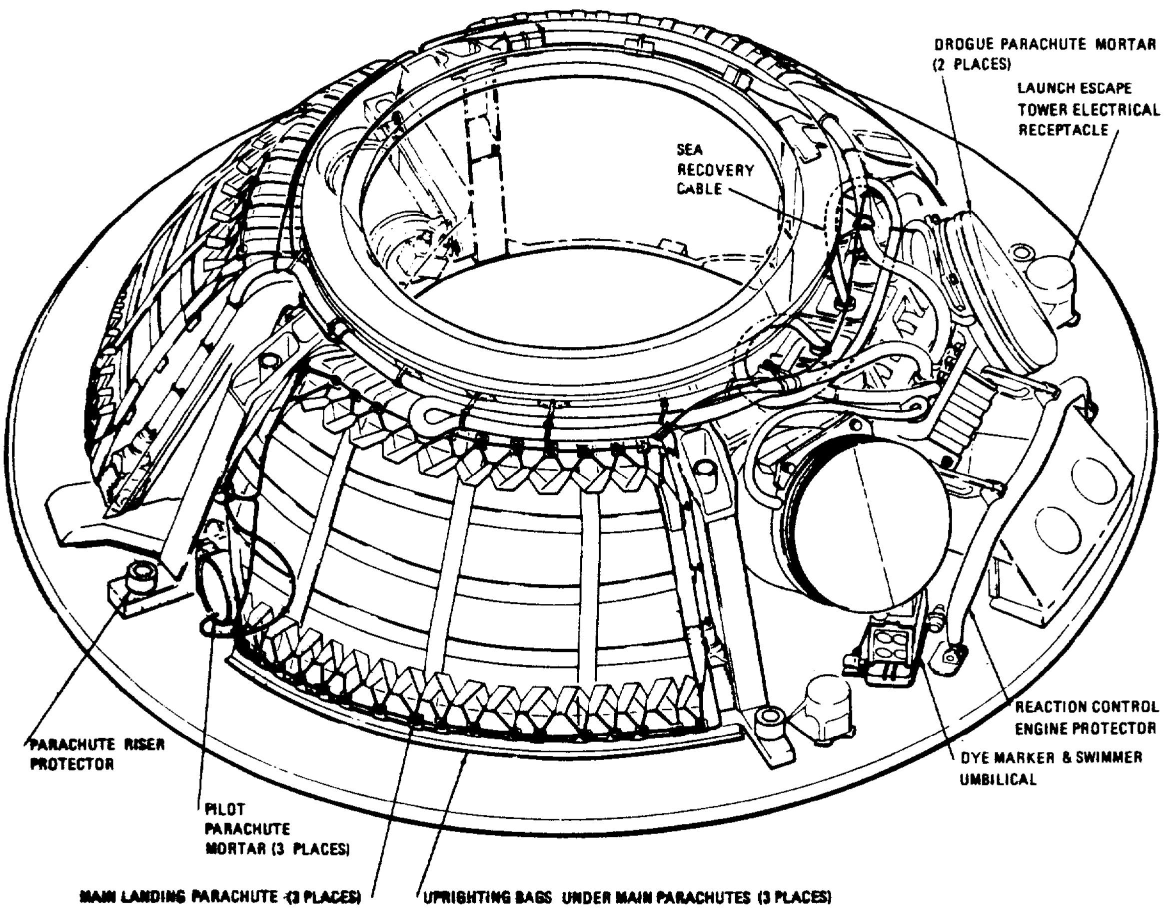


Fig. 16

and the three pilot parachutes are deployed; these pull the main parachutes from the forward section of the CM. The main parachutes initially open partially (reefed) for ten seconds to limit deceleration prior to full-diameter deployment. The main parachutes hang the CM at an angle of 27.5 degrees to decrease impact loads at touchdown.

After splashdown the crew releases the main parachutes and sets the recovery aid subsystem in operation. The subsystem consists of an uprighting system, swimmer's umbilical cable, a sea dye marker, a flashing beacon, and a VHF beacon transmitter. A sea recovery sling of steel cable is provided to lift the CM aboard a recovery ship. Three inflatable uprighting bags, stowed under the main parachutes, are available for uprighting the CM should it stabilize in an inverted floating position after splashdown.

The two VHF recovery antennas are located in the forward compartment with the parachutes. They are deployed automatically eight seconds after the main parachutes. One of them is connected to the beacon transmitter which emits a two-second signal every five seconds to aid recovery forces in locating the CM. The other is connected to the VHF/AM transmitter and receiver to provide voice communications between the crew and recovery forces.

Common Spacecraft Systems

Guidance and Control

The Apollo spacecraft is guided and controlled by two interrelated systems (Figure 17). One is the Guidance, Navigation, and Control System (GNCS). The other is the Stabilization and Control System (SCS). The two systems provide rotational, line-of-flight, and rate-of-speed information. They integrate and interpret this information and convert it into commands for the spacecraft propulsion systems.

GUIDANCE AND CONTROL FUNCTIONAL FLOW

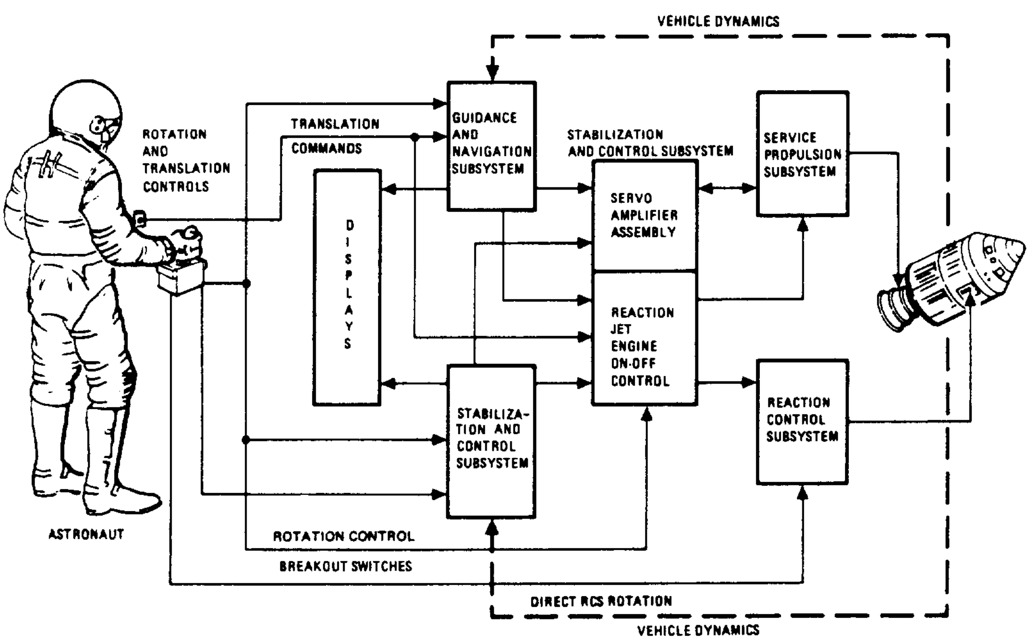


Fig. 17

Guidance, Navigation, and Control System

Guidance and navigation is accomplished through three major elements; the inertial, optical, and computer systems. The inertial subsystem senses any changes in the velocity and angle of the spacecraft and relays this information to the computer which transmits any necessary signals to the spacecraft engines. The optical subsystem is used to obtain navigation sightings of celestial bodies and landmarks on the earth and moon. It passes this information along to the computer for guidance and control purposes. The computer subsystem uses information from a number of sources to determine the spacecraft position and speed and, in automatic operation, to give commands for guidance and control.

Stabilization and Control System

The Stabilization and Control System (SCS) operates in three ways; it determines the spacecraft's attitude (angular position); maintains the spacecraft's attitude; and controls the direction of thrust of the service propulsion engine. Both the GNCS and SCS are used by the computer in the CM to provide automatic control of the Spacecraft. Manual control of the spacecraft attitude and thrust is provided mainly through the SCS equipment.

The Flight Director Attitude Indicators (FDAI) on the main console show the total angular position, attitude errors, and their rates of change. One of the sources of total attitude information is the stable platform of the Inertial Measurement Unit (IMU). The second source is a Gyro Display Coupler (GDC) which gives a reading of the spacecraft's actual attitudes as compared with an attitude selected by the crew. Information about attitude error also is obtained by comparison of the IMU gimbal angles with computer reference angles. Another source of this information is gyro assembly No. 1, which senses any spacecraft rotation about any of the three axes. Total attitude information goes to the CMC as well as to the FDAI's on the console. If a specific attitude or orientation is desired, attitude error signals are sent to the reaction jet engine control assembly. Then the proper reaction jet automatically fires in the direction necessary to return the spacecraft to the desired position.

The CMC provides primary control of thrust. The flight crew pre-sets thrusting and spacecraft data into the computer by means of the display keyboard. The forthcoming commands include time and duration of thrust. Accelerometers sense the amount of change in velocity obtained by the thrust. Thrust direction control is required because of center of gravity shifts caused by depletion of propellants in service propulsion tanks. This control is accomplished through electromechanical actuators which position the gimbaled SPS engine. Automatic control commands may originate in either the guidance and navigation subsystem or the SCS. There is also provision for manual controls.

Reaction Control Systems (RCS)

The Command Module and the Service Module each has its own independent system, the CM RCS and the SM RCS respectively. The SM RCS has four identical RCS "quads" mounted around the SM 90 degrees apart. Each quad has four 100-pound thrust engines, two fuel and two oxidizer tanks, and a helium pressurization sphere. The SM RCS provides redundant spacecraft attitude control through cross-coupling logic inputs from the Stabilization and Guidance Systems. Small velocity change maneuvers can also be made with the SM RCS. The CM RCS consists of two independent subsystems of six 94-pound thrust engines each. Both subsystems are activated after separation from the SM; one is used for spacecraft attitude control during entry. The other serves in standby as a backup. Propellants for both CM and SM RCS are monomethyl hydrazine fuel and nitrogen tetroxide oxidizer with helium pressurization. These propellants are hypergolic, i.e., they burn spontaneously when combined without need for an igniter.

Electrical Power System

The Electrical Power System (EPS) provides electrical energy sources, power generation and control, power conversion and conditioning, and power distribution to the spacecraft throughout the mission. The EPS also furnishes drinking water to the astronauts as a by-product of the fuel cells. The primary source of electrical power is the fuel cells mounted in the SM. Each cell consists of a hydrogen compartment, an oxygen compartment, and two electrodes. The cryogenic gas storage system, also located in the SM, supplies the hydrogen and oxygen used in the fuel cell power plants, as well as the oxygen used in the ECS.

Three silver-zinc oxide storage batteries supply power to the CM during entry and after landing, provide power for sequence controllers, and supplement the fuel cells during periods of peak power demand. These batteries are located in the CM lower equipment bay. A battery charger is located in the same bay to assure a full charge prior to entry.

Two other silver-zinc oxide batteries, independent of and completely isolated from the rest of the dc power system, are used to supply power for explosive devices for CM/SM separation, parachute deployment and separation, third-stage separation, launch escape system tower separation, and other pyrotechnic uses.

Emergency Detection System

The Emergency Detection System (EDS) monitors critical conditions of launch vehicle powered flight. Emergency conditions are displayed to the crew on the main display console to indicate a necessity for abort. The system includes provisions for a crew-initiated abort with the use of the LES or with the SPS after tower jettison. The crew can initiate an abort separation from the LV from prior to lift-off until the planned separation time. A capability also exists for commanding early staging of the S-IVB from the S-II stage when necessary. Also included in the system are provisions for an automatic abort in case of the following time-critical conditions:

1. Loss of thrust on two or more engines on the first stage of the LV.

2. Excessive vehicle angular rates in any of the pitch, yaw, or roll planes.

3. Loss of "hotwire" continuity from SM to IU.

The EDS will automatically initiate an abort signal when two or more first-stage engines are out or when LV excessive rates are sensed by gyros in the IU. The abort signals are sent to the master events sequence controller, which initiates the abort sequence. The engine lights on the Main Display Console provide the following information to the crew: ignition, cutoff, engine below pre-specified thrust level, and physical stage separation. A yellow "S-II Sep" light will illuminate at second-stage first-plane separation and will extinguish at second-plane separation. A high-intensity, red "ABORT" light is illuminated if an abort is requested by the Launch Control Center for a pad abort or an abort during lift-off via updata link. The "ABORT" light can also be illuminated after lift-off by the Range Safety Officer or by the Mission Control Center via the updata link from the Manned Space Flight Network.

Launch Escape System

General

The Launch Escape System (LES) (Figure 18) includes the LES structure, canards, rocket motors, and ordnance. The LES provides an immediate means of separating the CM from the LV during pad or suborbital aborts up through completion of second stage ignition. During an abort, the LES must provide a satisfactory earth return trajectory and CM orientation before jettisoning from the CM. The jettison or abort can be initiated manually or automatically.

Assembly

The forward or rocket section of the system is cylindrical and houses three solid-propellant rocket motors and a ballast compartment topped by a nose cone and "Q-ball" which measures attitude and flight dynamics of the space vehicle. The 500-pound tower is made of titanium tubes attached at the top to a structural skirt that covers the rocket exhaust nozzles and at the bottom to the CM by means of explosive bolts. A Boost Protective Cover (BPC) is attached to the tower and completely covers the CM. It has 12 "blowout" ports for the CM reaction control engines, vents, and an 8-inch window. This cover protects the CM from the rocket exhaust and also from the heat generated during launch vehicle boost. It remains attached to the tower and is carried away when the LES is jettisoned. Two canards mounted near the forward end of the assembly aerodynamically tumble the CM in the pitch plane during an abort so that the heat shield is forward.

LAUNCH ESCAPE SYSTEM

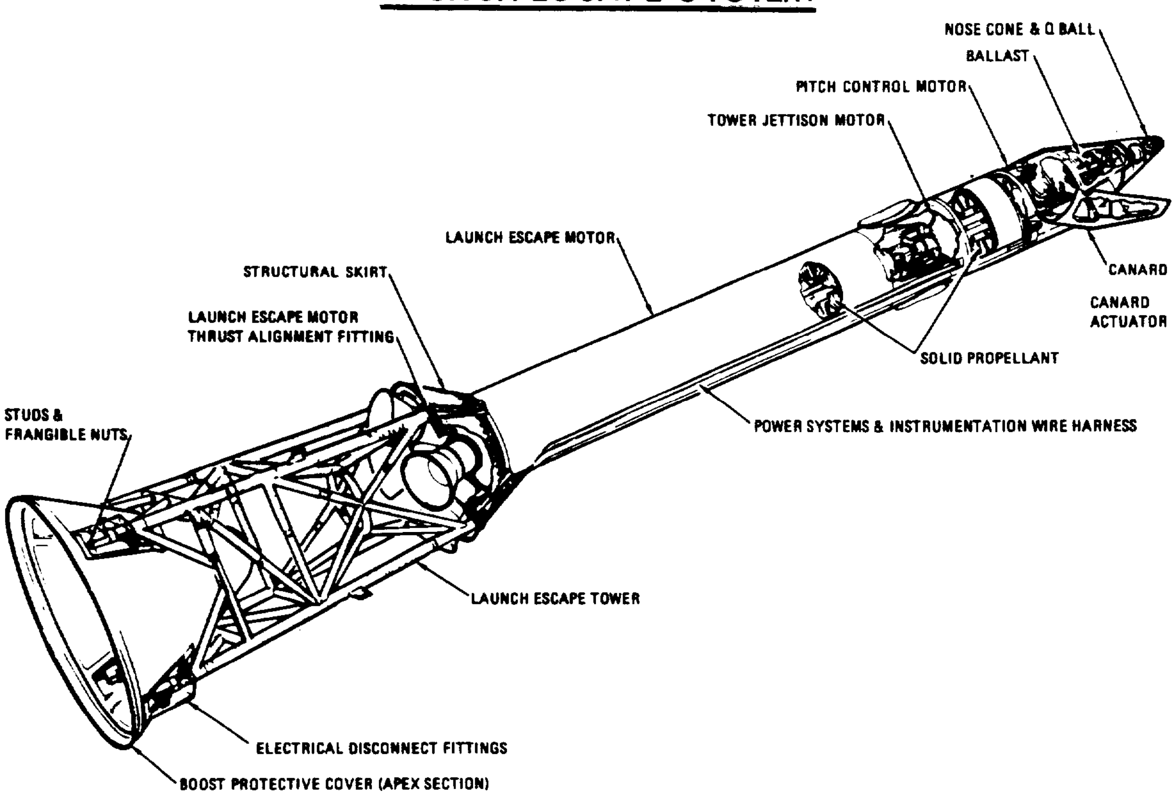


Fig. 18

Propulsion

Three solid propellant motors are used on the LES. They are:

1. The Launch Escape Motor which provides thrust for CM abort. It weighs 4700 pounds and provides 147,000 pounds of thrust at sea level for approximately eight seconds.

2. The Pitch Control Motor which provides an initial pitch maneuver toward the Atlantic Ocean during pad or low-altitude abort. It weighs 50 pounds and provides 2400 pounds of thrust for half a second.

3. The Tower Jettison Motor, which is used to jettison the LES, provides 31,500 pounds of thrust for one second.

System Operation

The system is activated automatically by the emergency detection system in the first 100 seconds or manually by the astronauts at any time from the pad to jettison altitude. The system is jettisoned at about 295,000 feet, or about 30 seconds after ignition of the second stage. After receiving an abort signal, the booster is cut off (after 30 seconds of flight), the CM-SM separation charges are fired, and the launch escape motor is ignited. The launch escape motor lifts the CM and the pitch control motor (used only at low altitudes) directs the flight path off to the side. Two canards are deployed 11 seconds after an abort is initiated. Three seconds later on extreme low-altitude aborts, or at approximately 24,000 feet on high-altitude aborts, the tower separation devices are fired and the jettison motor is started. These actions carry the LES away from the CM's landing trajectory. Four-tenths of a second after tower jettisoning, the CM's earth landing system is activated and begins its sequence of operations to bring the CM down safely. All preceeding automatic sequences can be prevented, interrupted, or replaced by crew action.

During a successful launch the LES is jettisoned by the astronauts, using the digital events timer and the "S-II Sep" light as cues. The jettisoning of the LES disables the Emergency Detection System automatic abort circuits. In the event of Tower Jettison Motor failure, the Launch Escape Motor may jettison the LES.

Lunar Module

General

The Lunar Module (LM) (Figure 19) is designed to transport two men safely from the CSM, in lunar orbit, to the lunar surface and return them to the orbiting CSM. The LM provides operational capabilities such as communications, telemetry,

LUNAR MODULE

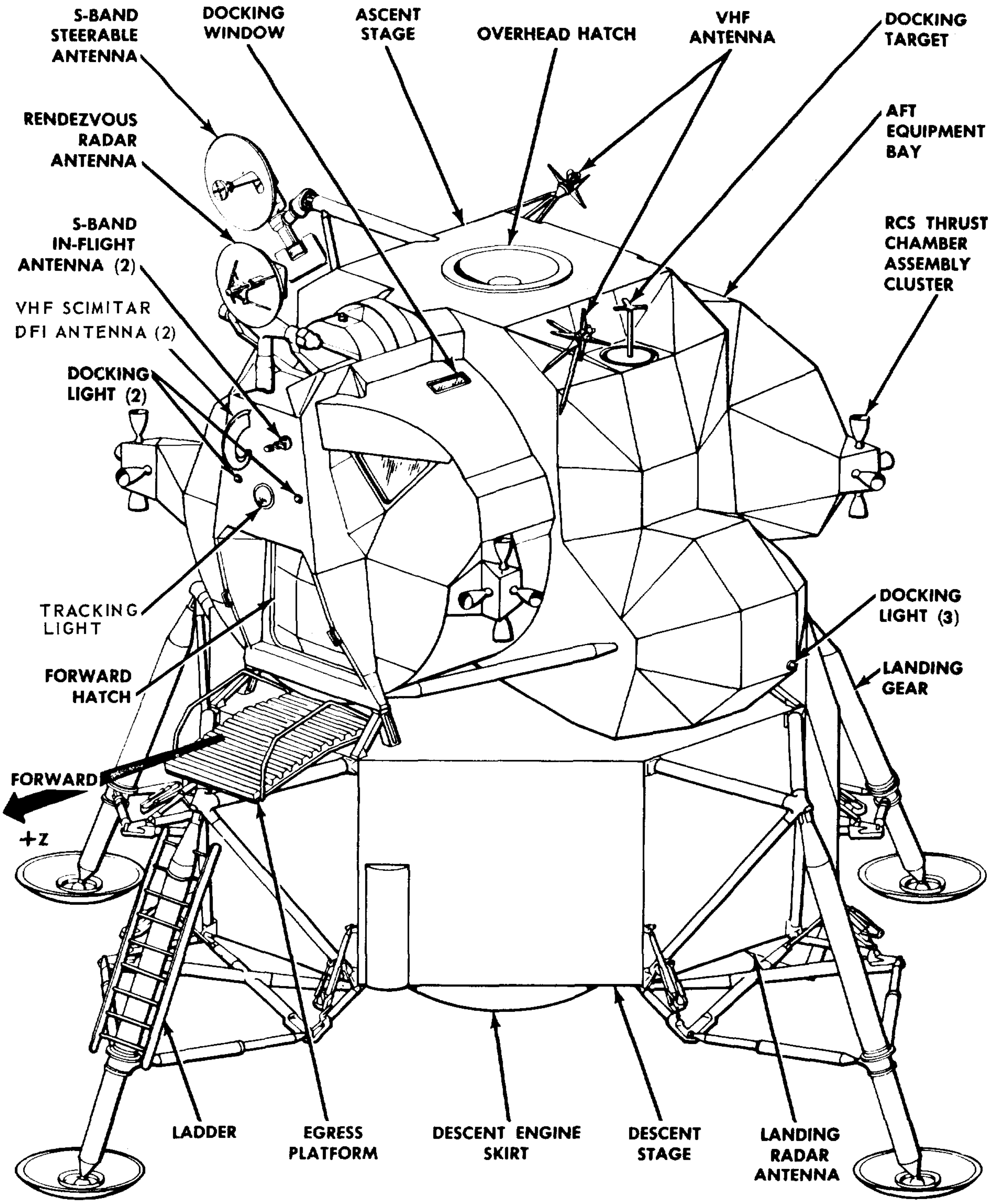


Fig. 19

environmental support, the transport of scientific equipment to the lunar surface, and the return of surface samples with the crew to the CSM. Physical characteristics are shown in Figure 20.

The Lunar Module consists of two stages; the Ascent Stage (AS), and the Descent Stage (DS). The stages are attached at four fittings by explosive bolts. Separable umbilicals and hardline connections provide subsystem continuity to operate both stages as a single unit until separate Ascent Stage operation is desired. The LM is designed to operate for 48 hours after separation from the CSM, with a maximum lunar stay time of 44 hours.

Ascent Stage

The Ascent Stage (AS) (Figure 21) accommodates two astronauts and is the control center of the LM. The stage structure provides three main sections consisting of a crew compartment and mid-section, which comprises the pressurized cabins and the unpressurized aft equipment bay. Other component parts of the structure consists of the Thrust Chamber Assembly (TCA) cluster supports, and antenna supports. The cylindrical crew compartment is of semi-monocoque, aluminum alloy construction. Large structural beams extend up the front face and across the top of the crew compartment to distribute loads applied to the cabin structure. The structural concept utilizes beams, bulkheads, and trusses to "cradle" the cabin assembly. The cabin volume is approximately 235 cubic feet.

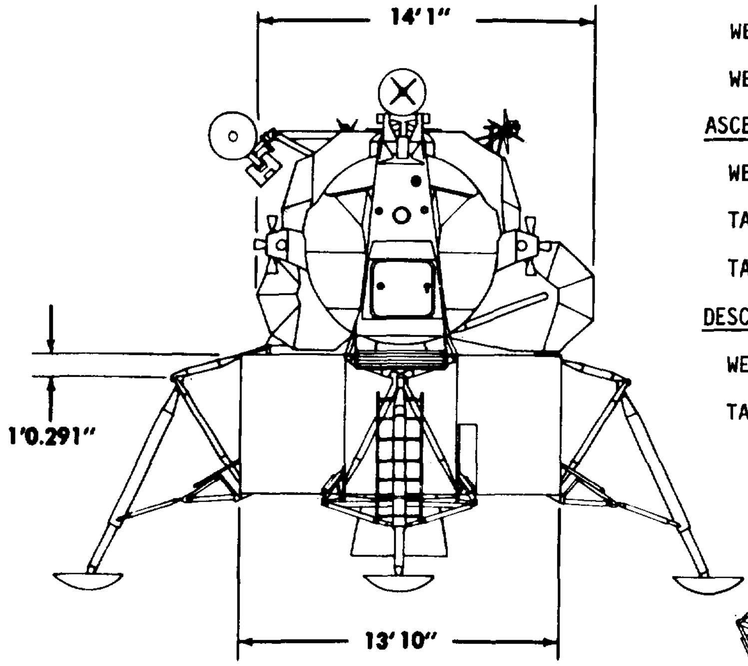
The entire Ascent Stage structure is enveloped by a vented blanket shield suspended at least two inches from the main structure. The thermal and micrometeoroid shield consists of multiple-layer aluminized mylar, nickel foil, inconel mesh, inconel sheet, and, in certain areas, H-film. The shield nominally provides thermal insulation against +350° F temperatures; with H-film, protection up to +1000° F is provided.

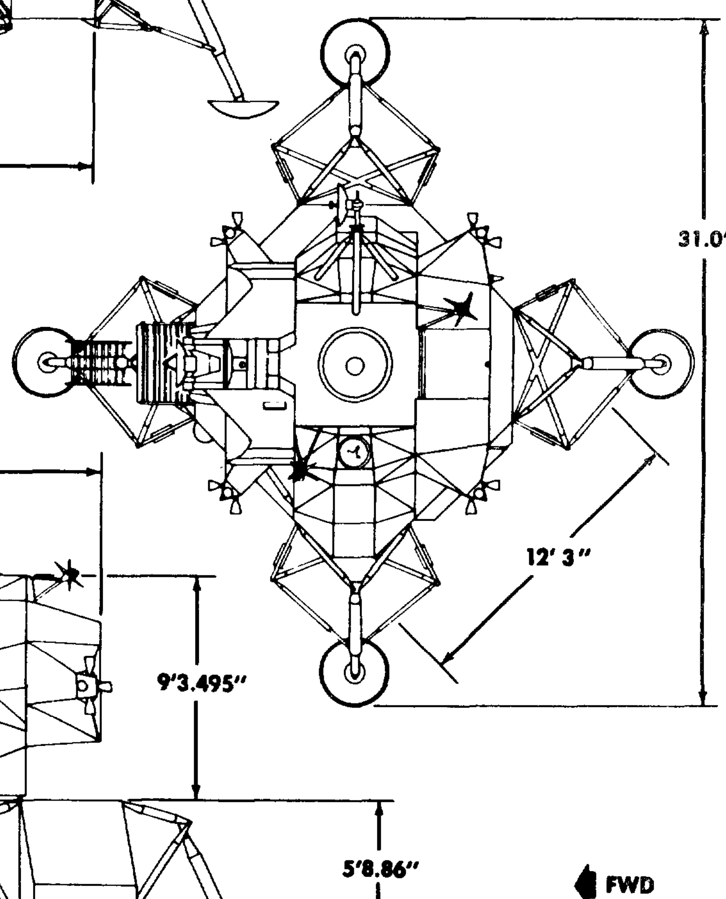
The flight station area has two front windows, a docking window, window shades, supports and restraints, an Alignment Optical Telescope (AOT), Crewman Optical Alignment Sight (COAS), data files, and control and display panels. Two hatches are provided for ingress and egress. The inward-opening forward hatch is used for EVA exit and entry. The overhead hatch seals the docking tunnel which is used for the transfer of crew and equipment internally between the docked CSM and LM.

The Ascent Stage is the nucleus of all LM systems. Two Portable Life Support Systems are stowed in the LM and provisions have been made for their replenishment: Stowage is provided for docking equipment, Extra Vehicular Visors, Extra Vehicular Gloves, Lunar Overshoes, and crew provisions in general.

LM PHYSICAL CHARACTERISTICS

TOTAL		
	WEIGHT (Propellant & Crew)	32,025 LB.
	WEIGHT (Less Prop)	9,336 LB.
ASCENT STAGE		
	WEIGHT (Less Prop)	5,070 LB.
	TANKED PROPELLANT (APS)	4,137 LB.
	TANKED PROPELLANT (RCS)	608 LB.
DESCENT STAGE		
	WEIGHT (Less Prop)	4,260 LB.
	TANKED PROPELLANT (DPS)	17,944 LB.





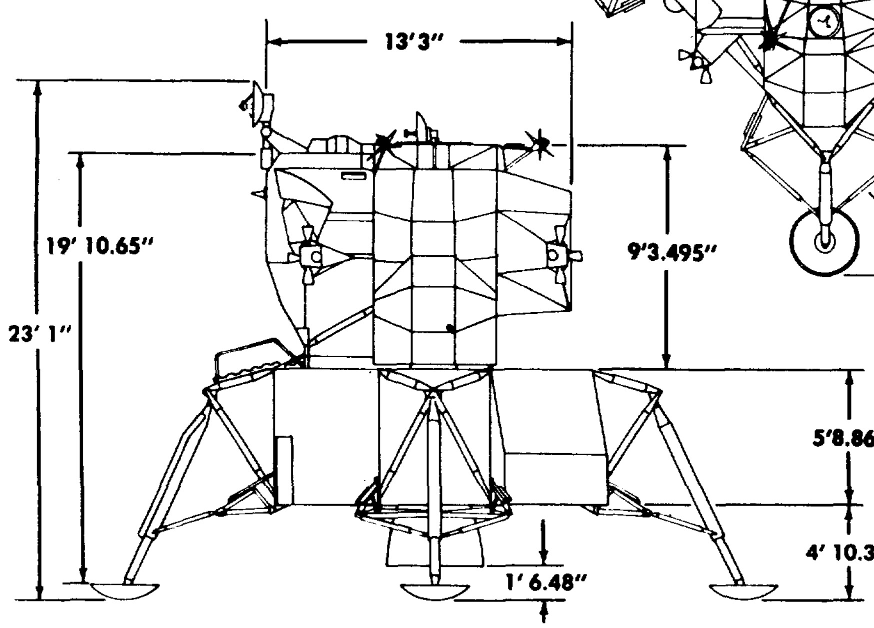
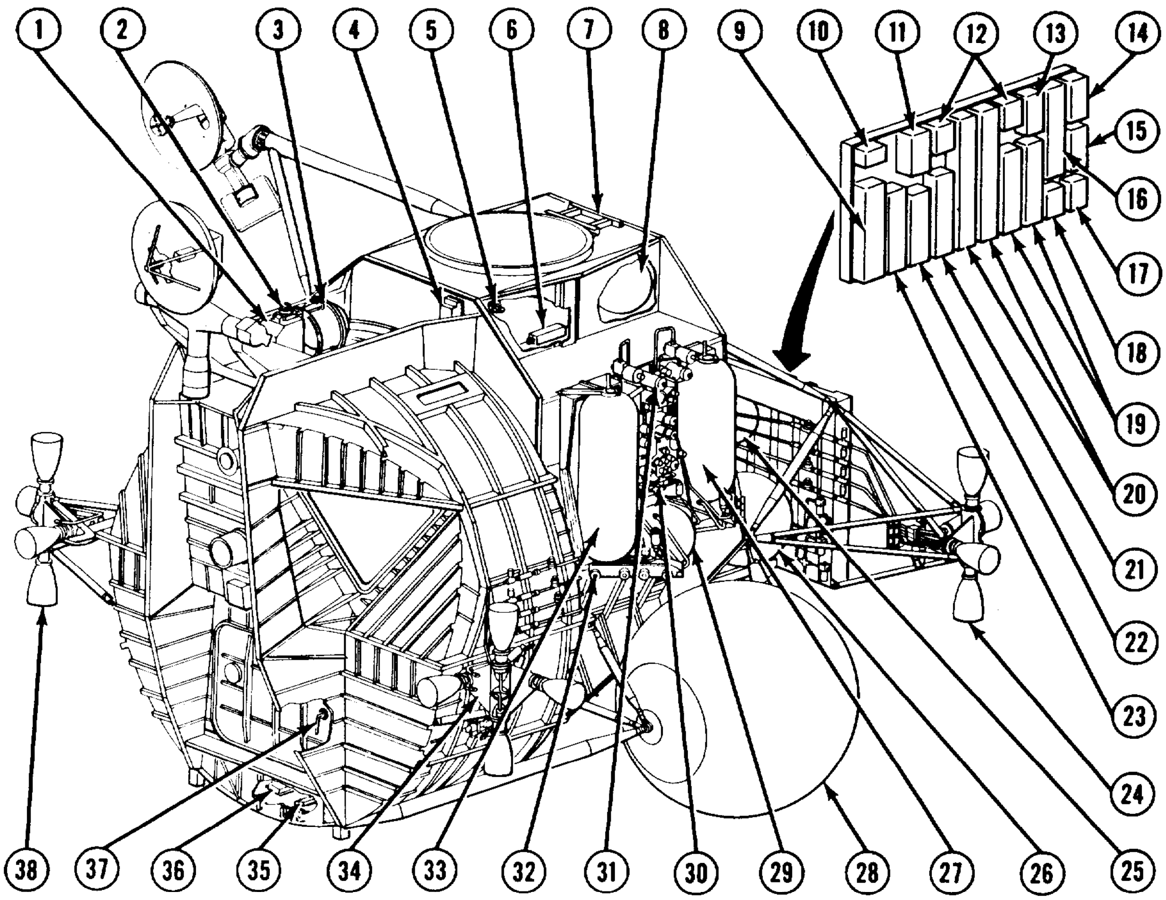


Fig. 20

LM ASCENT STAGE



KEY

1. Abort sensor assembly
2. Alignment optical telescope
3. Inertial measurement unit
4. Pulse torque assembly
5. Cabin dump and relief valve (upper hatch)
6. CSM/LM electrical umbilical fairing
7. Aft equipment bay bulkhead
8. Water tank
9. Rendezvous radar electronics assembly
10. Propellant quantity gaging system control unit
11. Caution and warning electronics assembly
12. Electrical control assembly
13. Attitude and translation control assembly
14. S-band power amplifier and diplexer
15. S-band transceiver
16. Abort electronic assembly
17. Signal processor assembly
18. VHF transceiver and diplexer
19. Inverter
20. Batteries
21. Signal-conditioning and electronic replaceable assembly No. 2
22. Pulse-code-modulation and timing equipment assembly
23. Signal-conditioning and electronic replaceable assembly No. 1
24. RCS quadrant 2
25. Gaseous oxygen tank
26. Helium tank
27. RCS fuel tank
28. APS fuel tank
29. RCS helium tank
30. RCS tank module
31. Helium pressurization module
32. Oxidizer service panel
33. RCS oxidizer tank
34. RCS quadrant 1
35. Lighting control assembly
36. Auxiliary switching relay box
37. Cabin dump and relief valve (forward hatch)
38. RCS quadrant 4

Fig. 21

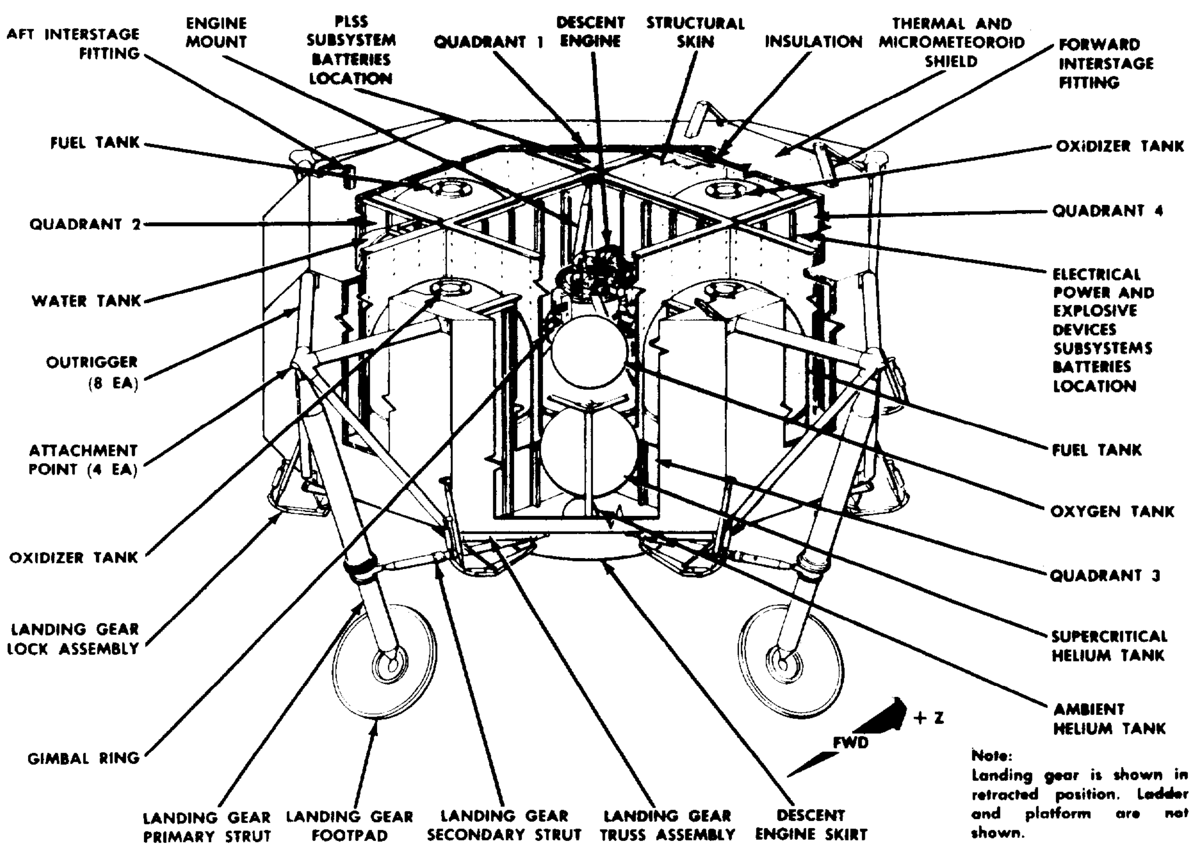
The Ascent Stage also provides external mounting for a CSM-active docking target, tracking and orientation lights, two VHF antennas, two S-band inflight antennas, an S-band steerable antenna and a rendezvous radar antenna.

The Ascent Propulsion System (APS) provides for major +X axis translations when separated from the descent Stage and a Reaction Control System (RCS) provides attitude and translational control about and along three axes.

Descent Stage

The Descent Stage (DS) (Figure 22) is the unmanned portion of the LM. It provides for major velocity changes of the LM to deorbit and land on the lunar surface. The basic structure consists of four main crossed-beams whose ends define the octagon shape of the stage. The major structural material is aluminum alloy. Thermal and micrometeoroid shielding is similar to that used on the Ascent Stage but with additional base heat shielding of nickel foil, H-film, Fibrocel, and Fiberfrax protecting the stage base from engine heat radiation.

LM DESCENT STAGE



Note:
Landing gear is shown in retracted position. Ladder and platform are not shown.

Fig. 22

The Descent Stage has four landing gear to absorb landing shock and to support the Descent Stage which must serve as a launch pad for the Ascent Stage. The Descent Stage engine nozzle extension is designed to collapse up to 28 inches and will not have any influence on LM lunar surface stability. Impact attenuation is achieved by compression of the main struts against crushable aluminum honeycomb. The landing gear trusses also provide the structural attachment points for securing the LM to the lower (fixed) portion of the Spacecraft LM Adapter (SLA). A ladder, integral to a primary landing gear start, provides access from and to the lunar surface from the ten-foot high forward hatch platform.

The Descent Stage contains the Descent Propulsion System (DPS) as well as electrical batteries, landing radar, supplements for the Environmental Control System, six batteries for the Protable Life Support Systems, a storage area for scientific equipment, an erectable S-band antenna, pyrotechnics, and generally, LM components not required for the lunar ascent phase of the mission.

Guidance, Navigation, and Control System

The Guidance, Navigation, and Control System (GN&CS) provides vehicle guidance, navigation, and control required for a manned lunar landing mission. The GN&CS utilizes a Rendezvous Radar (RR), and a Landing Radar (LR) to aid in navigation. The major subsystems of the GN&CS system are designated Primary Guidance and Navigation Subsystem (PGNS), Abort Guidance Subsystem (AGS), and the Control Electronics Subsystem (CES).

The GN&CS has a primary and alternate system path. The primary guidance path comprises the Primary Guidance and Navigation Subsystem, Control Electronics Subsystem, Landing Radar, Rendezvous Radar, and the selected propulsion system. The alternate system path comprises the Abort Guidance Subsystem, Control Electronics Subsystem, and the selected propulsion system. The term Primary Guidance, Navigation, and Control System (PGNCS) appears in certain technical mission documentation and connotes use of systems in the primary path of the LM Guidance, Navigation, and Control System.

Primary Guidance and Navigation Subsystem

The Primary Guidance and Navigation Subsystem (PGNS) establishes an inertial reference for guidance with an Inertial Measurement Unit, uses optics and radar for navigation, and a digital LM Guidance Computer (LGC) for data processing and generation of flight control signals. The inertially stabilized accelerometers sense incremental changes of velocity and attitude. Comparison of sensed instantaneous conditions against software programs generates corrections used to control the vehicle. The reference for the inertial system is aligned using the Alignment Optical Telescope, stars, horizons, and the computer. The PGNS, in conjunction with the CES, controls LM attitude, ascent or descent engine firing, descent engine thrust, and thrust vector. Control under the PGNS mode ranges from fully automatic to manual.

Abort Guidance Subsystem

The Abort Guidance Subsystem (AGS) provides an independent backup for the PGNS. The section is not utilized during aborts unless the PGNS has failed. The AGS is capable of determining trajectories required for a coelliptic rendezvous sequence to automatically place the vehicle in a safe parking/rendezvous orbit with the CSM or can display conditions to be acted upon by the astronauts to accomplish rendezvous. The activated AGS performs LM navigation, guidance, and control in conjunction with the Control Electronics Subsystem (CES). The AGS differs from the PGNS in that its inertial sensors are rigidly mounted with respect to the vehicle rather than on a stabilized platform. In this mode, the Abort Sensor Assembly (ASA) measures attitude and acceleration and supplies data to the Abort Electronics Assembly (AEA) which is a high-speed digital computer.

Control Electronics Subsystem

The Control Electronics Subsystem (CES) controls LM attitude and translation about and along three axes by processing commands from the PGNS or AGS and routing on/off commands to 16 reaction control engines, ascent engine, or descent engine. Descent engine thrust vector is also controlled by the CES.

Rendezvous Radar

The Rendezvous Radar (RR) tracks the CSM to provide relative line of sight, range and range rate data for rendezvous and docking. The transponder in the CSM augments the transmitted energy of the RR thus increasing radar capabilities and minimizing power requirements. Radar data is automatically entered into the LGC in the PGNS mode. During AGS operation, data inputs are entered into the Abort Electronics Assembly (AEA) through the Data Entry and Display Assembly (DEDA) by the crew from cabin displays. Radar data is telemetered to the Manned Space Flight Network and monitored for gross inaccuracies.

Landing Radar

The Landing Radar provides the LGC with slant range and velocity data for control of the descent to the lunar surface. Slant range data is available below lunar altitudes of approximately 25,000 feet and velocity below approximately 18,000 feet.

Main Propulsion

Main Propulsion is provided by the Descent Propulsion System (DPS) and the Ascent Propulsion System (APS). Each system is wholly independent of the other. The DPS provides the thrust to control descent to the lunar surface. The APS can provide the thrust for ascent from the lunar surface. In case of mission abort, the APS and/or DPS can place the LM into a rendezvous trajectory with the CSM from any point in the descent trajectory. The choice of engine to be used depends on the cause for abort, on how long the descent engine has been operating, and on the quantity of propellant remaining in the Descent Stage. Both propulsion systems use identical hypergolic propellants. The fuel is a 50-50 mixture of Hydrazine and Unsymetrical Di-Methyl Hydrazine and the oxidizer is Nitrogen Tetroxide. Gaseous Helium pressurizes the propellant feed systems. Helium storage in the DPS is at cryogenic temperatures in the super-critical state and in the APS it is gaseous at ambient temperatures.

Ullage for propellant settling is required prior to descent engine start and is provided by the +X axis reaction engines. The descent engine is gimbaled, throttleable and restartable. The engine can be throttled from 1050 pounds of thrust to 6300 pounds. Throttle positions above this value automatically produce full thrust to reduce combustion chamber erosion. Nominal full thrust is 9870 pounds. Gimbal trim of the engine compensates for a changing center of gravity of the vehicle and is automatically accomplished by either the PGNS or AGS. Automatic throttle and on/off control is available in the PGNS mode of operation. The AGS commands on/off operation but has no automatic throttle control capability. Manual control capability of engine firing functions has been provided. Manual thrust control override may, at any time, command more thrust than the level commanded by the LGC.

The ascent engine is a fixed, non-throttleable engine. The engine develops 3500 pounds of thrust, sufficient to abort the lunar descent or to launch the Ascent Stage from the lunar surface and place it in the desired lunar orbit. Control modes are similar to those described for the descent engine. The Ascent Propulsion System propellant is contained in two spherical titanium tanks, one for oxidizer and the other for fuel. Each tank has a volume of 36 cubic feet. Total fuel weight is 2008 pounds of which 71 pounds are unusable. Oxidizer weight is 3170 pounds of which 92 pounds are unusable. The APS has a limit of 35 starts, must have a propellant bulk temperature between 50°F and 90°F prior to start, must not exceed 460 seconds of burn time and has a system life of 24 hours after pressurization.

In general, the main propulsion systems use pyrotechnic isolation valves in pressurization and propellant lines to prevent corrosive deterioration of components. Once the APS or DPS is activated, its reliable operating time is limited but adequate for its designed use.

Reaction Control System

The Reaction Control System (RCS) stabilizes the LM, provides ullage thrust for the DPS or APS, helps to maintain the desired trajectory during descent, and controls LM attitude and translation about or along three axes during hover. Sixteen engines termed Thrust Chamber Assemblies (TCA's) of 100 pounds thrust each are mounted symmetrically around the LM Ascent Stage in clusters of four. The RCS contains two independent, parallel systems (A&B) controlling two TCA's in each cluster. Each system, operating alone, can perform all required attitude control requirements, however translational performance is slightly degraded under single system operation. The independent propellant systems have a crossfeed capability for increased operational dependability. During APS thrusting, APS propellant can supplement the RCS system. The propellant tanks utilize bladders to achieve positive expulsion feed under zero-g gravity conditions. Malfunctioning TCA pairs can be deactivated by manual switches.

The RCS TCA firing is accomplished by the Control Electronics Section of the GN&CS in response to manual commands or signals generated in the PGNS or AGS modes. RCS modes of operation are: automatic; attitude hold (semi-automatic); and manual override. The TCA's firing time ranges from a pulse of less than one second up to steady state operation.

Thirty-two heaters are used to heat the 16 TCA's. TCA temperature requirements ranging from 132°F to 154°F are important to safe and proper TCA operation. Propellant capacity of each system of the RCS is: oxidizer (N_2O_4) 207.5 pounds, 194.9 pounds usable; Fuel (50-50 N_2H_4 and UDMH) 106.5 pounds, 99.1 pounds usable.

In order to ensure reliable RCS operation, firing time for each TCA must not exceed 500 seconds with firing times exceeding one second, and 1000 seconds of pulses with firing times less than one second. RCS operation requires propellant tank temperatures between 40°F and 100°F. Firing time of vertically mounted thrusters is limited to prevent damage to descent stage insulation or the ascent stage antennas.

Electrical Power System

The Electrical Power System (EPS) contains six batteries which supply the electrical power requirements of the LM during undocked mission phases. Four batteries are located in the Descent Stage (DS) and two in the Ascent Stage (AS). Batteries for the Explosive Devices System are not included in this system description. Post-launch LM power is supplied by the DS batteries until the LM and CSM are docked. While docked, the CSM supplies electrical power to the LM up to 296 watts (peak). During the lunar descent phase, the two AS batteries are paralleled with the DS batteries for additional power assurance. The DS batteries are utilized for LM lunar surface operations and checkout. The AS batteries are brought on the line just before ascent phase staging. All batteries and busses may be individually monitored for load, voltage, and failure. Several isolation and combination modes are provided.

Two Inverters, each capable of supplying full load, convert the dc to ac for 115-volt, 400-hertz supply. Electrical power is distributed by the following buses: LM Pilot's dc bus, Commanders dc bus, and ac buses A&B.

The four Descent Stage silver-zinc oxide batteries are identical and have a 400 ampere-hour capacity at 28 volts. Because the batteries do not have a constant voltage at various states of charge/load levels, "high" and "low" voltage taps are provided for selection. The "low voltage" tap is selected to initiate use of a fully charged battery. Cross-tie circuits in the busses facilitate an even discharge of the batteries regardless of distribution combinations. The two silver-zinc oxide Ascent Stage batteries are identical to each other and have a 296 ampere-hour capacity at 28 volts. The AS batteries are normally connected in parallel for even discharge. Because of design load characteristics, the AS batteries do not have and do not require high and low voltage taps.

Nominal voltage for AS and DS batteries is 30.0 volts. Reverse current relays for battery failure are one of many components designed into the EPS to enhance EPS reliability. Cooling of the batteries is provided by the Environmental Control System cold rail heat sinks. Available ascent electrical energy is 17.8 kilowatt hours at a maximum drain of 50 amps per battery and descent energy is 46.9 kilowatt hours at a maximum drain of 25 amps per battery.

Environmental Control System

The Environmental Control System (ECS) provides a habitable environment for two astronauts for a maximum of 48 hours while the LM is separated from the CSM. Included in this capability is four cabin decompression/re-pressurization cycles. The ECS also controls the temperature of electrical and electronic equipment, stores and provides water for drinking, cooling, fire extingusihing, and food preparation. Two oxygen and two water tanks are located in the Ascent Stage. One larger oxygen tank and a larger water tank is located in the Descent Stage.

The ECS is comprised of an Atmosphere Revitalization Subsystem (ARS), an Oxygen Supply and Cabin Pressure Control Subsystem (OSCPCS), a Water Management Subsystem (WMS), a Heat Transport Subsystem (HTS), and an oxygen and water supply to the Portable Life Support System (PLSS). The ARS cools and ventilates the Pressure Garment Assemblies, controls oxygen temperature, and the level of carbon dioxide in the atmosphere, removes odors, particles, noxious gases, and excess moisture.

Oxygen Supply and Cabin Pressure Control Subsystem

The Oxygen Supply and Cabin Pressure Control Subsystem (OSCPCS) stores gaseous oxygen and maintains cabin and suit pressure by supplying oxygen to the ARS. This replenishes losses due to crew metabolic consumption and cabin or suit leakage. The oxygen tank in the Descent Stage provides oxygen during the descent and lunar-stay phases of the mission, and the two in the ascent stage are used during the ascent and rendezvous phases of the mission.

Water Management Subsystem

The Water Management Subsystem (WMS) supplies water for drinking, cooling, fire extinguishing, and food preparation; for refilling the PLSS cooling water tank; and for pressurization of the secondary coolant loop of the HTS. It also provides for delivery of water from ARS water separators to HTS sublimators and from the water tanks to ARS and HTS sublimators. The water tanks are pressurized before launch to maintain the required pumping pressure in the tanks. The Descent Stage tank supplies most of the water required until staging occurs. After staging, water is supplies by the two Ascent Stage storage tanks. A self-sealing "PLSS DRINK" valve delivers water for drinking and food preparation.

Heat Transport Subsystem

The Heat Transport Subsystem (HTS) consists of a primary coolant loop and a secondary coolant loop. The secondary loop serves as a backup loop and functions in the event the primary loop fails. A water-glycol solution circulates through each loop. The primary loop provides temperature control for batteries, electronic equipments that require active thermal control, and for the oxygen that circulates through the cabin and pressure suits. The batteries and electronic equipment are mounted on cold plates and rails through which coolant is routed to remove excess heat.

The cold plates used for equipment required for mission abort contain two separate coolant passages, one for the primary loop and one for the secondary loop. The secondary coolant loop serves only abort equipment cold plates.

In flight, excess heat rejection from both coolant loops is achieved by the primary and secondary sublimators which are vented overboard. A coolant pump recirculation assembly contains all the HTS coolant pumps and associated check and relief valves. Coolant flow from the assembly is directed through parallel circuits to the cold plates for the electronic equipment and the oxygen-to-glycol heat exchanger in the ARS.