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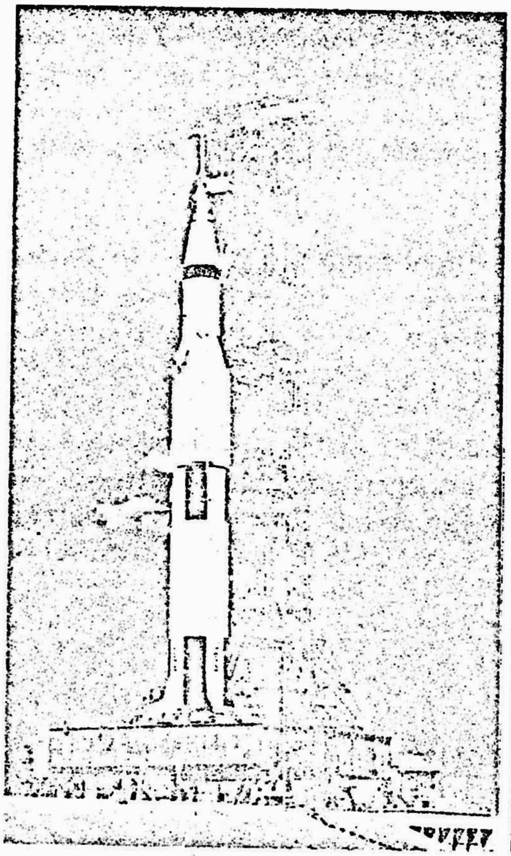
MISSION OPERATION REPORT



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APOLLO 8 (AS-503) MISSION

OFFICE OF MANNED SPACE FLIGHT
Prepared by: Apollo Program Office - MAO

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FOREWORD

MISSION OPERATION REPORTS are published expressly for the use of NASA Senior Management, as required by the Administrator in NASA Instruction 6-2-10, dated August 15, 1963. The purpose of these reports is to provide NASA Senior Management with timely, complete, and definitive information on flight mission plans, and to establish official mission objectives which provide the basis for assessment of mission accomplishment.

Initial reports are prepared and issued for each flight project just prior to launch. Following launch, updating reports for each mission are issued to keep General Management currently informed of definitive mission results as provided in NASA Instruction 6-2-10.

Because of their sometimes highly technical orientation, distribution of these reports is limited to personnel having program-project management responsibilities. The Office of Public Affairs publishes a comprehensive series of pre-launch and post-launch reports on NASA flight missions, which are available for general distribution.

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APOLLO 8 MISSION OPERATION REPORT

The Apollo 8 Mission Operation Report is published in two volumes - I, The Mission Operation Report (MOR) and II, The Mission Operation Report Supplement.

This format was designed to provide a mission oriented document in the MOR with only a very brief description of the space vehicle and support facilities. The MOR Supplement is a reference document with a more comprehensive description of the space vehicle and associated systems.

GENERAL

The goal of the Apollo Program is to enhance the manned space flight capability of the United States by developing, through logical and orderly evolution, the ability to land men on the moon and return them safely to earth.

To accomplish the goal of lunar landing and return in this decade, the Apollo Program has focused on the development of a highly reliable launch vehicle and spacecraft system. This has been done through a logical sequence of Apollo missions designed to qualify the flight hardware, ground support systems and operational personnel in the most effective manner.

The Apollo 8 mission is the second manned flight of the Apollo spacecraft and the first manned Saturn V mission. The mission is designed to test the space vehicle, mission support facilities, and crew on a lunar orbital mission.

PROGRAM DEVELOPMENT

The first Saturn vehicle was successfully flown on 27 October 1961 to initiate operations of the Saturn I Program. A total of 10 Saturn I vehicles (SA-1 to SA-10) were successfully flight tested to provide information on the integration of launch vehicle and spacecraft and to provide operational experience with large multi-engined booster stages (S-I, S-IV).

The next generation of vehicles, developed under the Saturn IB Program, featured an uprated first stage (S-IB) and a more powerful new second stage (S-IVB). The first Saturn IB was launched on 26 February 1966. The first three Saturn IB missions (AS-201, AS-203, and AS-202) successfully tested the performance of the launch vehicle and spacecraft combination, separation of the stages, behavior of liquid hydrogen in a weightless environment, performance of the Command Module heat shield at low earth orbital entry conditions, and recovery operations.

The planned fourth Saturn IB mission (AS 204) scheduled for early 1967 was intended to be the first manned Apollo flight. This mission was not flown because of a spacecraft fire, during a manned pre-launch test, that took the lives of the prime flight crew and severely damaged the spacecraft. The SA-204 launch vehicle was later assigned to the Apollo 5 mission.

The Apollo 4 mission was successfully executed on 9 November 1967. This mission initiated the use of the Saturn V launch vehicle (SA-501) and required an orbital restart of the S-IVB third stage. The spacecraft for this mission consisted of an unmanned Command and Service Module (CSM) and a Lunar Module Test Article (LTA). The CSM Service Propulsion System (SPS) was exercised, including restart, and the Command Module Block II heat shield was subjected to the combination of high heat load, high heat rate, and aerodynamic loads representative of lunar return entry.

The Apollo 5 mission was successfully launched and completed on 22 January 1968. This was the fourth mission utilizing Saturn IB vehicles (AS-204). This flight provided for unmanned orbital testing of the Lunar Module (LM-1). The LM structure, staging, proper operation of the Lunar Module Ascent Propulsion System (APS) and Descent Propulsion System (DPS), including restart, was verified. Satisfactory performance of the S-IVB/Instrument Unit (IU) in orbit was also demonstrated.

The Apollo 6 mission (second unmanned Saturn V) was successfully launched on 4 April 1968. Some flight anomalies encountered included oscillation related to propulsion-structural longitudinal coupling, Spacecraft Lunar Adapter (SLA) area structural integrity and certain malfunctions of the J-2 engines in the S-II and S-IVB stages. The spacecraft flew the planned trajectory, but preplanned high velocity reentry conditions were not achieved. A majority of the mission objectives for Apollo 6 were accomplished.

The Apollo 7 mission (first manned Saturn IB) was successfully launched on 11 October 1968. This was the fifth and last planned Apollo mission utilizing Saturn IB vehicles (AS-205). The mission provided for the first manned orbital tests of the Block II Command and Service Module. All primary mission objectives were successfully accomplished. In addition, all planned Detailed Test Objectives, plus three that were not originally scheduled, were satisfactorily accomplished.

The Apollo 8 mission will provide the first manned flight of the Saturn V and the first Apollo mission in the lunar environment.

THE APOLLO 8 MISSION

The Apollo 8 mission is the third Saturn V mission and is intended to verify crew, space vehicle, and mission support facilities on a lunar orbit flight plan. The mission is planned for slightly more than six days (147 hours).

The nominal flight plan will closely resemble the manned lunar landing mission planned for later in the Apollo program. Apollo 8 will demonstrate several nominal lunar landing mission activities including translunar injection (TLI), cislunar navigation and communications, lunar orbit insertion (LOI), passive thermal control (PTC), and trans-earth injection (TEI).

The Apollo 8 launch will be on a selected launch azimuth between 72° and 107°. Launch azimuth selection will be made just prior to launch to effect the desired moon/spacecraft rendezvous. The ascent-to-earth orbit will include the S-IC and S-II boost phase with the S-IVB orbit insertion burn. The spacecraft will remain attached to the S-IVB in a 100 nautical mile circular parking orbit for approximately two revolutions.

On the second parking orbit, while over the Pacific Ocean, the S-IVB engine will be restarted to insert the CSM and S-IVB into a translunar trajectory. The nominal TLI will provide a "free return" to earth if the deboost into a lunar orbit is not initiated. A second opportunity for TLI will be available on the third revolution if necessary.

Shortly after TLI, the CSM will separate from the S-IVB, transpose, and then move to a safe distance before the S-IVB propellant dump. The S-IVB then executes a retrograde dump of residual propellants and Auxiliary Propulsion System (APS) burn to completion to achieve a "slingshot" effect which reduces the probability that the S-IVB will impact with the CSM, the moon's surface, or return to earth.

The spacecraft will remain in translunar coast (TLC) for approximately three days. During this time navigational sightings, midcourse corrections (MCC), and passive thermal control will be exercised.

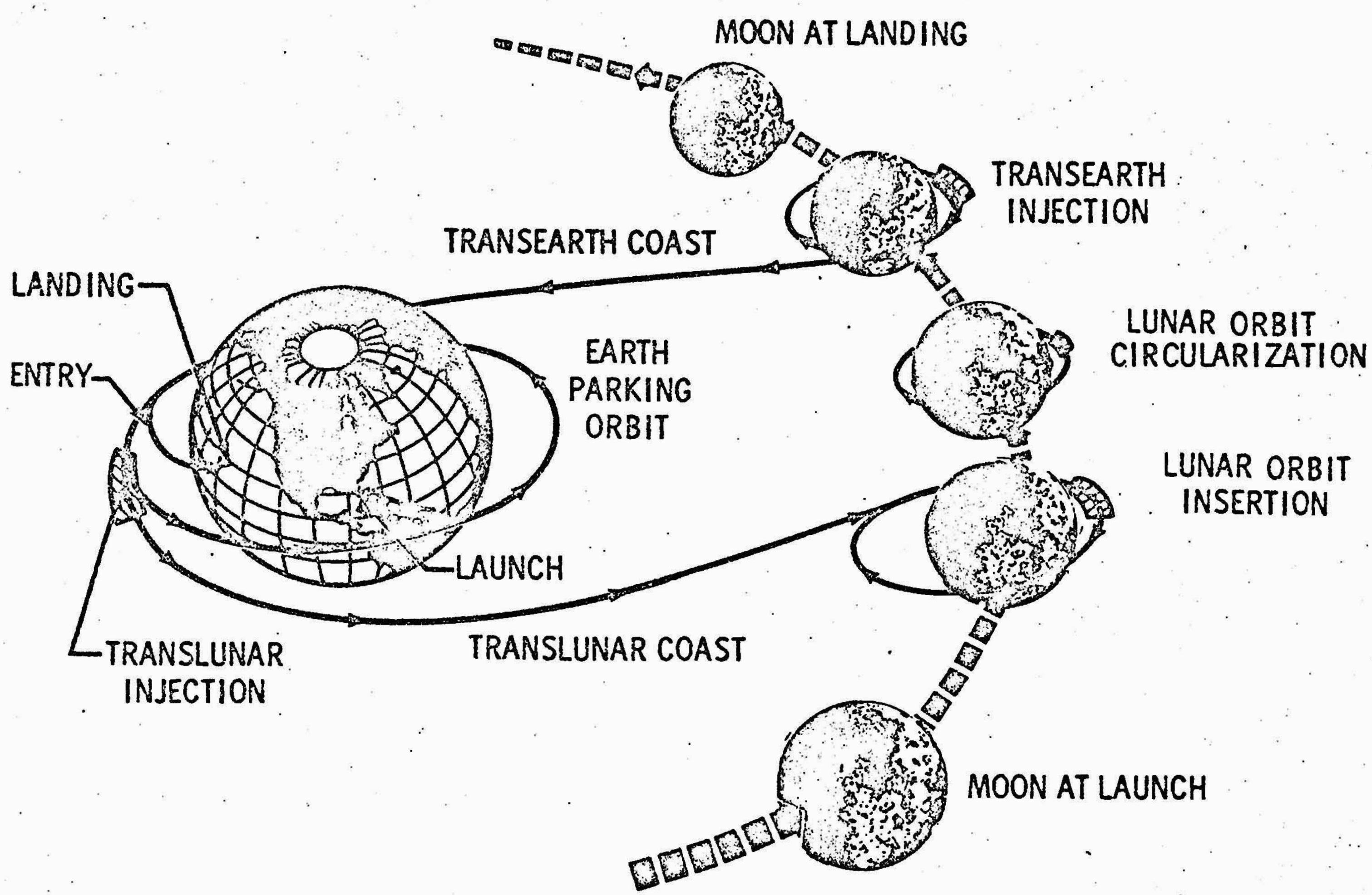
Nominally, at approximately 69 hours into the mission, the Service Propulsion System (SPS) will perform the first lunar orbit insertion (LOI_1) maneuver and will place the CSM into a 60 x 170 nautical mile lunar orbit. Following insertion and system checks, the orbit will be circularized at 60 nautical miles by a second SPS maneuver (LOI_2).

The spacecraft will remain in lunar orbit for a total of 10 revolutions (approximately 20 hours) during which time extensive photography and landmark sightings will be made. TV coverage of the lunar surface is planned.

Return to earth will begin when the SPS performs the TEI at approximately 89 hours into the mission. The transearth coast (TEC) will have a duration of approximately 58 hours. A total of three MCC's are allotted during this time for return corridor control. This period will also provide for evaluation of crew activities, navigation sightings, subsystem performance, and mission support facilities.

Approximately two hours prior to entry interface (EI) the Guidance, Navigation and Control System (GNCS) is prepared for an automatic entry. Entry velocity will be approximately 36,220 fps with splashdown planned for the Pacific recovery area. Figure 1 shows the general mission profile.

MISSION PROFILE



NASA MISSION OBJECTIVES

FOR APOLLO 8

PRIMARY OBJECTIVES

- Demonstrate crew/space vehicle/mission support facilities performance during a manned Saturn V mission with CSM
- Demonstrate performance of nominal and selected backup Lunar Orbit Rendezvous (LOR) mission activities, including:
  - Translunar injection
  - CSM navigation, communications, and midcourse corrections
  - CSM consumables assessment and passive thermal control

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

Date: 12 Dec 1968

George E. Mueller
Associate Administrator for
Manned Space Flight

Date: 13 Dec 1968

DETAILED TEST OBJECTIVES

The following detail amplifies and defines more explicitly basic tests, measurements, and evaluations which are planned to achieve the primary objectives given previously.

Launch Vehicle

- Verify launch vehicle capability for free return translunar injection
- Demonstrate S-IVB restart capability
- Verify J-2 engine modification
- Confirm J-2 engine environment in S-II and S-IVB stages
- Confirm launch vehicle longitudinal oscillation environment during S-IC stage burn period
- Verify modifications incorporated in the S-IC stage to suppress low frequency longitudinal oscillations
- Demonstrate helium heater repressurization system operation
- Verify capability to inject S-IVB/IU/LTA-B into a lunar "slingshot" trajectory
- Demonstrate capability to safe S-IVB stage

Spacecraft

- Perform lunar landmark tracking
- Demonstrate ground operational support for a CSM lunar orbit mission
- Obtain data on passive thermal control system
- Perform manual and automatic acquisition, tracking and communications with MSFN using the high gain CSM S-band antenna during a lunar mission
- Perform translunar and transearth midcourse corrections
- Perform lunar orbit insertion GNCS controlled SPS burn

- Prepare for translunar injection and monitor the GNCS and LV tank pressure displays during the TLI burn
- Demonstrate SLA panel jettison in a zero-g environment
- Obtain data on the spacecraft dynamic response
- Perform star-earth landmark sightings navigation during translunar and transearth phases
- Perform star-lunar horizon sightings during the translunar and transearth phases
- Perform star-earth horizon sightings during translunar and transearth phases
- Demonstrate CSM passive thermal control modes during the translunar and transearth mission phases
- Perform a GNCS controlled entry from a lunar return
- Perform SPS LOI and TEI burns and monitor the primary and auxiliary gauging systems
- Monitor the GNCS and displays during launch
- Obtain data on the CM crew procedures and timeline for lunar orbit mission activities
- Obtain data on CSM consumables for a CSM lunar orbit mission
- Perform a transearth insertion GNCS controlled SPS burn

SECONDARY OBJECTIVES

Apollo secondary objectives are established by the development centers to provide additional engineering or scientific data. For Apollo 8 the secondary test objectives are as follows:

Launch Vehicle

- Verify the on-board CCS and ground system interface and operation in a deep space environment

Spacecraft

- Communicate with MSFN using the CSM S-band omni antennas at lunar distance
- Demonstrate the performance of the Block II thermal protection system during a manned lunar return entry
- Obtain data to determine the effect of the tower jettison motor, S-II retro and SM RCS exhausts and other sources of contamination on the CSM windows
- Obtain data on the Block II ECS performance during manned lunar return entry conditions
- Obtain IMU performance data in the flight environment
- Perform an IMU alignment and a star pattern visibility check in daylight
- Perform a CSM/S-IVB separation and a CSM transposition on a lunar mission timeline
- Obtain photographs during the transearth, translunar and lunar orbit phases

SAFETY PLANNING

GENERAL

As a result of the normal developmental cycle, a large number of changes and additions have been made to the Apollo safety systems and procedures. A detailed discussion of changes is not within the scope of this report though some items are covered in later descriptive sections. This section generalizes the procedural or program aspects of safety related improvements.

PRE-LAUNCH PROCEDURES

Procedures for maximum launch complex area safety are provided by "Apollo/Saturn V Ground Safety Plan" K-V-053. Overall safety conditions at the launch pad area are monitored by the Systems Safety Supervisor who has the authority to stop any operation when a condition exists which, in his opinion, is imminently hazardous.

All test procedures are required to contain applicable emergency procedures. Prior to implementation, these test procedures are reviewed by the KSC Safety Office which makes the final determination as to the hazardous classification of all tests. The procedures that are evaluated as hazardous are then compared against a set of checklists to ensure compliance with safety practices.

To ensure maximum readiness of personnel to cope with emergencies, personnel emergency procedure training and practice are given on a regular basis. In addition, training proficiency is reviewed before conducting all hazardous operations. The training program that has been initiated includes the use of fire-fighting equipment, toxic propellant safety, safety locker qualification, first aid and the handling of personnel requiring aid in the event of an emergency. It also includes the development of procedures and training in emergency crew egress. Certification of course completion is mandatory for Launch Complex (LC) 39 personnel.

RANGE SAFETY

The range safety system provides essentially real time space vehicle position, trajectory, and impact prediction information from launch through orbital insertion. The Range Safety Officer (RSO) uses consoles, plotting boards, visual displays, and information from range observers for making his decision on safe or unsafe space vehicle trajectories.

LAUNCH COUNTDOWN AND TURNAROUND CAPABILITY AS-503

Countdown for the Apollo 8 mission will begin with a pre-count period during which launch vehicle and spacecraft preparations will take place independently until coordinated space vehicle countdown activities begin. Table 1 shows the significant launch countdown events.

TABLE 1

LAUNCH COUNTDOWN SEQUENCE OF EVENTS

COUNTDOWN HRS: MIN: SEC:	EVENT
28: 00: 00	Start LV and SC countdown activities
24: 30: 00	S-II power-up
24: 00: 00	S-IVB power-up
09: 00: 00	Six hour built-in hold
09: 00: 00	End of built-in hold; close CM and BPC hatch
08: 59: 00	Clear pad for LV cryo loading
07: 28: 00	Start S-IVB LO_2 loading
07: 04: 00	S-IVB LO_2 loading complete; start S-II LO_2 loading
06: 27: 00	S-II LO_2 loading complete; start S-IC LO_2 loading
04: 57: 00	S-IC LO_2 loading complete
04: 54: 00	Start S-II LH_2 loading
04: 11: 00	S-II LH_2 loading complete; start S-IVB LH_2 loading
03: 30: 00	1 hour built-in hold
03: 30: 00	Flight crew departs MSO
03: 28: 00	S-IVB LH_2 loading complete
03: 13: 00	Closeout crew on station; start ingress preps
02: 40: 00	Start flight crew ingress
02: 10: 00	Flight crew ingress complete
01: 40: 00	Close SC hatch
01: 00: 00	Start RP-1 level adjust
00: 42: 00	Arm LES pyro buses
00: 35: 00	RP-1 level adjust complete
00: 15: 00	SC on internal power
00: 05: 30	Arm S and A Devices
00: 03: 07	Terminal Count Sequencer (TCS) start
00: 00: 17.0	Guidance reference release command
00: 00: 08.9	S-IC ignition command
00: 00: 00	Lift-off

SCRUB/TURNAROUND

Scrub/turnaround times are based upon the amount of work required to return the space vehicle to a safe condition and to complete the recycle activities necessary to resume launch countdown preparation for a subsequent launch attempt. Planning guidelines for the various scrub/turnaround plans are based upon no serial time for repairs or holds, or for systems retesting resulting from repairs; performing tasks necessary to attain launch with the same degree of confidence as for the first launch attempt; and, not requiring unloading of hypergolic propellants and RP-1 from the space vehicle.

TURNAROUND CONDITIONS VS. TIME

Should a hold occur from T-22 minutes to T-16.7 seconds and a recycle to T-22 minutes is required under the conditions stated in the Launch Mission Rules, the applicable recycle operations in the Launch Vehicle and Spacecraft procedures are followed. The decision is then made to hold at this point with the intention of resuming the count, or to scrub and initiate the turnaround procedures. A cutoff after T-16.7 seconds results in a scrub. For a hold prior to T-22 minutes, which results in a scrub, the turnaround procedures are initiated from the point of hold.

Post LV Cryo Load (with Fuel Cell Cryo Reservicing)

Turnaround time is 67 hours, 30 minutes, consisting of 30 hours, 30 minutes for recycle time and 37 hours for countdown time. Time is based upon scrub occurring between 16.7 seconds and 8.9 seconds during original countdown.

Post LV Cryo Load (No Fuel Cell Cryo Reservicing)

Turnaround time is 22 hours, 30 minutes, consisting of 13 hours, 30 minutes for recycle time and 9 hours for countdown time.

Pre LV Cryo Load (with Fuel Cell Cryo Reservicing)

Turnaround time is 61 hours, 15 minutes, consisting of 24 hours 15 minutes for recycle time and 37 hours for countdown time.

Pre LV Cryo Load (No Fuel Cell Cryo Reservicing)

The capability for approximately a one day hold exists at T-9 hours of the countdown. This capability provides for a launch attempt at the opening of the next launch window.

DETAILED FLIGHT MISSION DESCRIPTION

NOMINAL MISSION

Pre-Launch

The AS-503 space vehicle for the Apollo 8 mission is planned to be launched from Launch Complex 39, pad A, at Kennedy Space Center, Florida on 21 December 1968. The launch window opens at 0751 EST and closes at 1232 EST on this date. Should holds in the launch countdown or weather require a scrub, there are six days remaining in December during which the mission could be launched. Table 2 shows these days and launch window durations.

TABLE 2
DECEMBER LAUNCH WINDOWS

December Launch Days for Apollo 8	WINDOW (EST) Open	Close
21	0751	1232
22	0926	1405
23	1058	1535
24	1221	1658
25	1352	1820
26	1516	1820
27	1645	1820

A variable launch azimuth of 72° to 107° capability will be available to assure a launch on time. This is the first Apollo mission which has employed the variable launch azimuth concept. The concept is necessary to compensate for the relative relationship of the earth and moon at launch time.

Launch and Earth Parking Orbit

The Apollo 8 mission will begin with the boost to orbit by a full burn of the S-IC and S-II stages and a partial burn of the S-IVB stage of the Saturn V launch vehicle. Shortly after S-IC/S-II staging, four camera capsules will be jettisoned from the S-IC. Table 3 shows the mission sequence of events.

Insertion into the 100-nautical mile earth parking orbit will occur approximately 11 minutes, 32 seconds ground elapsed time (GET) from lift-off. The spacecraft and the S-IVB will remain in this orbit while all systems are checked and readied for the second burn of the S-IVB, the translunar injection (TLI) burn.

TABLE 3

APOLLO 8

SEQUENCE OF EVENTS

Time from Lift-off HR: MIN: SEC:				Event
00:	00:	00		Lift-off
00:	00:	11		Roll and Pitch Program Initiate
00:	00:	28		Roll Complete
00:	01:	17		Maximum Dynamic Pressure
00:	02:	05		S-IC Center Engine Cutoff
00:	02:	31		S-IC Outboard Engine Cutoff
00:	02:	32		S-IC/S-II Separation
00:	02:	33		S-II Ignition
00:	02:	55		Camera Capsule Ejection
00:	03:	00		Second Plane Separation
00:	03:	07		Launch Escape Tower Jettison
				Mode I/Mode II Abort Changeover
00:	08:	40		S-II Cutoff
00:	08:	41		S-II/S-IVB Separation
00:	08:	45		S-IVB Ignition
00:	09:	50		Mode IV Capability Begins
00:	10:	18		Mode II/Mode III Abort Changeover
00:	11:	31		Insertion into Earth Parking Orbit
02:	50:	31		Translunar Injection Ignition
02:	55:	43		Translunar Injection Cutoff
				Translunar Coast Begins
03:	09:	14		S-IVB/CSM Separation
04:	44:	54		Begin Maneuver to Slingshot Attitude
05:	07:	54		LOX Dump Begins
05:	12:	54		LOX Dump Ends
			TLI + 6 Hrs	Midcourse Correction 1
			TLI +25 Hrs	Midcourse Correction 2
			LOI -22 Hrs	Midcourse Correction 3
			LOI - 8 Hrs	Midcourse Correction 4
69:	07:	30		Lunar Orbit Insertion (LOI_1) Initiation
69:	11:	36		Lunar Orbit Insertion (LOI_1) Termination
73:	30:	54		Lunar Orbit Insertion (LOI_2) Initiation
73:	31:	04		Lunar Orbit Insertion (LOI_2) Termination

Time from Lift-off HR: MIN: SEC:				Event
89:	04:	02		Transearth Injection Initiate
89:	18:	33		Transearth Injection Terminate
			TEI +15 Hrs	Midcourse Correction 5
			TEI +33 Hrs	Midcourse Correction 6
			EI - 2 Hrs	Midcourse Correction 7
146:	35:	00		CM/SM Separation
146:	50:	00		Entry Interface
147:	10:	00		SPLASHDOWN

Translunar Injection and Translunar Coast

TLI will occur during revolution 2 (first opportunity) or 3 (second opportunity) over the Pacific and will last approximately five minutes. A nominal TLI burn will place the spacecraft on a circumlunar, earth-intersecting trajectory. This is called a free return trajectory which means that with no further SPS burns the spacecraft will fly around the moon and safely return to earth.

Within 20 minutes after TLI, the spacecraft will separate from the S-IVB, transpose, and perform the spacecraft/S-IVB evasive maneuver. The S-IVB will then execute the dump of residual propellants plus an APS burn to completion which will impart a differential velocity (ΔV) to put the S-IVB on a trajectory which is planned to reduce the probability of a S-IVB recontact with the CSM, impacting the moon's surface, or returning to earth.

Periodically during the approximately 66 hours of translunar coast, the spacecraft trajectory will be assessed to determine if a midcourse correction is required. Up to four MCC opportunities have been identified to maintain the free return trajectory and will ensure that the spacecraft will be at least 60 nautical miles above the moon's surface at its point of closest approach.

Lunar Orbit Insertion and Lunar Orbit

When the spacecraft reaches pericynthion (point nearest surface) behind the moon, it performs the lunar orbit insertion (LOI_1) burn. This is a retrograde Service Propulsion System (SPS) burn of approximately four minutes duration that inserts the spacecraft into a 60 x 170 nautical mile lunar orbit. The spacecraft will remain in this orbit for approximately two revolutions. At third pericynthion, a second SPS burn (LOI_2), lasting approximately 10 seconds, will be performed which will circularize the spacecraft orbit at 60 nautical miles above the lunar surface.

The spacecraft will orbit the moon eight more times for a total of 10 lunar orbits. A 60-nautical mile lunar orbit has a period of approximately two hours; therefore the spacecraft will be in the lunar environment for approximately 20 hours. A schedule of activities while in lunar orbit is shown on Table 4.

Transearth Injection, Transearth Coast, and Entry

While passing behind the moon on the 10th revolution, the spacecraft will perform the transearth injection (TEI) burn. This SPS burn will have a duration of approximately 3.4 minutes and place the spacecraft on a return-to-earth trajectory. During the approximate 58 hours of transearth coast, the spacecraft will perform up to three midcourse corrections, if required, to ensure that the spacecraft will enter the atmosphere with the proper combination of velocity and flight path angle for safe entry. The

TABLE 4

LUNAR ACTIVITIES

Revolution	Activity
1 & 2	LOI, camera preparation, eat period, COAS ground track determination, control point and pseudo landing site observations, photographs of targets of opportunity and TV transmission.
3 & 4	LOI_2, two-hour, CMP rest period, landmark training photography, vertical stereo photography and landmark lighting evaluation.
5 & 6	Three-hour CDR rest period, one control point landmark tracking, and a pseudo landing site tracking during each daylight period. Each tracking consists of four marks.
7 & 8	Two-hour LMP rest period, three control point landmark trackings, and a pseudo landing site tracking during each daylight period.
9 & 10	Two-hour CMP rest period, convergent stereo photography an eat period and the TEI maneuver.

NOTE: The IMU is realigned once during each dark period in lunar orbit.

spacecraft will enter the atmosphere at 400,000 feet with a velocity of approximately 36,220 feet per second and will land approximately 1350 nautical miles down range of the entry point. Figure 2 shows entry conditions for Apollo 8. Table 5 lists the entry and landing points for the different days applicable to the December launch windows. Splashdown for the nominal Apollo 8 mission, with a 21 December launch, is planned for approximately six days and three hours after lift-off.

REENTRY

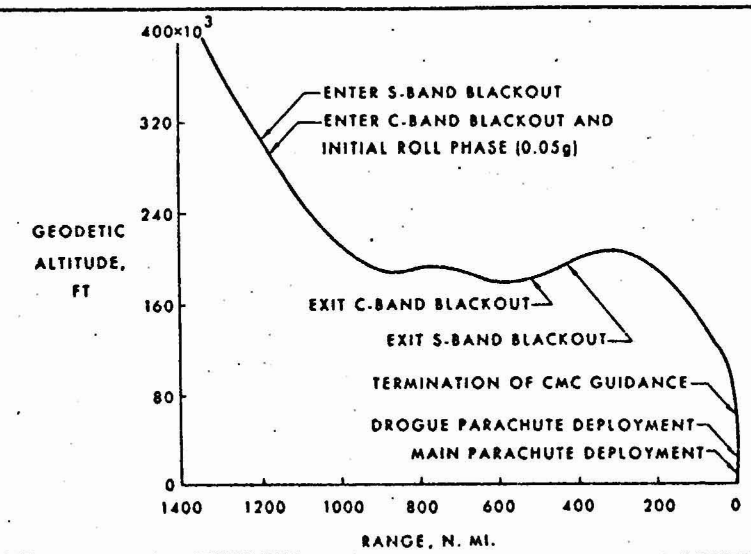


 Fig. 2

TABLE 5

CM END-OF-MISSION ENTRY AND LANDING POINTS*

Day of Launch	Entry Point		Landing Point	
	Latitude	Longitude	Latitude	Longitude
21 Dec	14°42'N	174°30'E	4°55'N	165°00'W
22 Dec	5°35'N	173°50'E	1°00'S	165°00'W
23 Dec	1°20'N	174°35'E	8°10'S	165°00'W
24 Dec	10°15'S	172°15'E	12°50'S	165°00'W
25 Dec	18°55'S	171°25'E	18°00'S	165°00'W
26 Dec	25°00'S	170°45'E	22°10'S	165°00'W
27 Dec	22°25'S	170°55'E	25°25'S	165°00'W

*These points are for a 72° launch azimuth. Other launch azimuths will change the data slightly.

TV transmissions from the CM are planned for various phases of the mission (Table 6).

TABLE 6

NOMINAL MISSION TV OPERATION TIMELINE

MSFN Station	Station Time of Acquisition (GET - Hr:Min)	Δ Time (Min)	Remarks
Goldstone	31:15	15	1. Exterior Views (Earth-Moon) - 100 mm Lens, 9° FOV
Goldstone	55:15	15	2. Interior Views - Wide Angle Lens; 160° FOV
Madrid	71:35	15	
Goldstone	85:37	15	
Goldstone	104:15	15	
Goldstone	128:00	15	

CONTINGENCY OPERATIONS

If an anomaly occurs after lift-off that would prevent the AS-503 space vehicle from following its nominal flight plan, an abort or an alternate mission will be initiated. An abort would provide only for an acceptable CM/crew recovery while an alternate mission would attempt to achieve some of the mission objectives before providing for an acceptable CM/crew recovery.

ABORTS

Launch Aborts

During launch, the velocity, altitude, atmosphere, and launch configuration change rapidly; therefore, several abort modes, each adapted to a portion of the launch trajectory, are required.

Mode I abort procedure is designed for safe recovery of the CM following aborts occurring between Launch Escape System (LES) activation (approximately T minus 30 minutes) and Launch Escape Tower (LET) jettison, approximately 3 minutes GET. The procedure consists of the LET pulling the Command Module (CM) away from the space vehicle and propelling it a safe distance down range. The resulting landing point lies between the launch site and approximately 490 nautical miles down range. The Mode II abort would be performed from the time the LET is jettisoned until the full-lift CM landing point is 3200 nautical miles down range, approximately 10 minutes GET. The procedure consists of separating the command and service module (CSM) combination from the launch vehicle, separating the CM from the SM, and then letting the CM free fall to entry. The entry would be a full-lift, or maximum range trajectory, with a landing 400 to 3200 nautical miles downrange on the ground track. Mode III abort can be performed from the time the full-lift landing point range reaches 3200 nautical miles until orbital insertion. The procedure would consist of separating the CSM from the launch vehicle and then, if necessary, performing a retrograde burn with the SPS so that the half-lift landing point is no farther than 3350 nautical miles down range. A half-lift entry would be flown which causes the landing point to be approximately 70 nautical miles south of the nominal ground track between 3000 and 3350 nautical miles down range. The Mode IV abort procedure is an abort to earth parking orbit and could be performed anytime after the SPS has the capability to insert the CSM into orbit. This capability begins at approximately 10 minutes GET. The procedure would consist of separating the CSM from the launch vehicle and, two minutes later, performing a posigrade SPS burn to insert the CSM into earth parking orbit with a perigee of at least 75 nautical miles. The CSM could then remain in earth orbit for an earth orbital alternate mission, or, if necessary, return to earth in the West Atlantic or Central Pacific Ocean after one revolution. This mode of abort is preferred over the Mode III abort and would be used unless an immediate return to earth is necessary during the launch phase. The last phase abort procedure is an Apogee Kick (AK) Mode.

This mode is a variation of the Mode IV wherein the SPS burn to orbit occurs at apogee altitude to raise the perigee to 75 nautical miles. The maneuver is executed whenever the orbital apogee at S-IVB cutoff is favorably situated and the corresponding Mode IV ΔV requirement is greater than 100 feet per second. Like the Mode IV contingency orbit insertion (COI), this maneuver is prime when the capability exists, except for those modified situations where an immediate return to earth is required.

Earth Parking Orbit Aborts

Once the CSM/S-IVB is safely inserted into earth parking orbit, a return-to-earth abort would be performed by separating the CSM from the S-IVB and then utilizing the SPS for a retrograde burn to place the CM on an atmosphere-intersecting traject After entry the crew would fly a guided flight path to a preselected target point if possible.

Aborts Associated with TLI

Ten Minute Abort

If an abort is necessary during TLI the S-IVB would be cut off early and the crew would initiate an onboard calculated retrograde SPS abort burn. The SPS burn would be performed approximately 10 minutes after TLI cutoff and would ensure a safe CM entry. The elapsed time from abort initiation to landing would vary from approximately 25 minutes to 4 hours, depending on the length of the TLI maneuver performed prior to S-IVB cutoff. For aborts initiated during the latter portion of TLI, a second SPS burn could be necessary to correct for dispersed entry conditions. Since this abort would be used only in extreme emergencies with respect to crew survival, the landing point would not be considered in executing the abort. The SM RCS will be used for land avoidance. No meaningful landing point predictions can be made since they will vary with the launch azimuth, the location of the TLI, the duration of the TLI burn prior to cutoff, and execution errors of the abort maneuvers.

Ninety Minute Abort

The ninety minute abort would be applicable after the TLI maneuver has been completed. If any malfunction occured during the TLI burn, and after careful check, it became apparent that it was necessary to return to earth, an abort procedure specifying an SPS burn at a certain CSM attitude would be transmitted to the crew. The abort would be initiated at approximately TLI cutoff plus 90 minutes. Unlike the previous procedure this abort would be targeted to a preselected landing location called a recovery line. The location of these lines is shown in Figure 8. If possible, the abort would be targeted to the Mid-Pacific recovery line but for some time-critical situations, the abort would be targeted to the Atlantic Ocean line or the East Pacific line. The abort maneuver would be a retrograde SPS burn followed by a midcourse correction, if necessary, performed near apogee to provide the proper CM entry conditions. The elapsed time between abort initiation and landing would normally vary between 11 and 18 hours.

Aborts During Translunar Coast

The abort procedure for this phase of the mission would be similar to the 90 minute abort. If conditions warrant an abort, abort information specifying a combination of SPS burn time and CSM attitude would be sent to the crew. Deep space aborts will be targeted to, in order of priority, (1) the Mid-Pacific line, (2) the Atlantic Ocean line, (3) the East Pacific or West Pacific lines, and (4) the Indian Ocean line. Regardless of the recovery line selected, the landing latitude should remain nearly the same. The minimum elapsed time between abort initiation and CM landing increases with translunar coast flight time. About the time the CSM enters the moon's sphere of gravitational influence, it becomes faster to perform a circumlunar abort rather than returning directly to earth.

Aborts During Lunar Orbit Insertion

Aborts following an early shutdown of the SPS during the lunar orbit insertion (LOI) maneuver are divided into two categories, Mode I and Mode III. There is no Mode II procedure on the Apollo 8 mission because this mode is necessary only for a mission which carries a Lunar Module.

Mode I

The Mode I procedure would be used for aborts following LOI shutdowns from ignition to approximately two minutes into the burn. This procedure would consist of performing a posigrade SPS burn as soon as possible after cutoff to put the CSM back on a return-to-earth trajectory.

Mode III

The Mode III procedure would be used for aborts following LOI shutdowns from approximately two minutes into the burn until nominal cutoff. After two minutes of LOI burn, the CSM will have been inserted into an acceptable lunar orbit. Therefore, the abort procedure would be to let the spacecraft go through one or two lunar revolutions prior to doing a posigrade SPS burn at pericynthion. This would place the CSM on a return-to-earth trajectory targeted to the Mid-Pacific recovery line.

Aborts During Lunar Orbit

Aborts from the lunar orbit would be accomplished by performing the transearth injection burn (TEI) early. The abort would be targeted to the Mid-Pacific recovery line.

Aborts During Transearth Injection

The abort procedures for early cutoff of TEI are the inverse of the LOI abort procedures. That is, for early cutoffs between TEI ignition and approximately two minutes, a Mode III abort would be performed. After this time a Mode I abort would be used. All TEI aborts should result in landings on the Mid-Pacific recovery line.

Aborts During Transearth Coast

From TEI until entry minus 24 hours, the only abort procedure that could be performed is to use the SPS or the SM/reaction control system for a posigrade burn that will decrease the transearth flight time and change the longitude of landing. After entry minus 20 hours, no further abort burns will be performed. This is to ensure that the CM maintains the desired entry velocity and flight path angle combination that will allow a safe entry.

ALTERNATE MISSIONS

Alternate mission plans for Apollo 8 have been prepared for the following contingencies:

1. Failure of the S-IVB late in its first burn which results in an early CSM separation and a subsequent SPS burn to a COI.

2. S-IVB failure prior to TLI or insufficient propellant for restart.

3. Premature or non-nominal TLI termination resulting in an ellipse with associated energy such that a ΔV greater than $\sim$3000 fps is required at TLI+3 hours for an SPS MCC.

4. Dispersed trajectories resulting from a malfunction of the S-IVB during TLI.

The alternate missions resulting from the above contingencies are described in the following paragraphs. The descriptions are essentially qualitative since the associated maneuvers will be computed in real time using the real-time auxiliary computing facility (RTACF).

Alternate Mission 1

If the S-IVB malfunctions late in its first burn, the CSM has the capability to achieve a low orbit (contingency orbit) using the SPS. And, if the COI requires less than 900 fps, the CSM has the capability to later inject into a 4000 nautical mile apogee ellipse.

If the above conditions exist, the plan is to remain in low earth orbit for approximately 68-72 hours, making MCC's consistent with the lunar timeline. These maneuvers could be used for perigee preservation. In the time period of 68-72 hours GET, the SPS is ignited over US or ETR MSFN stations for a 4200 fps burn to inject the CSM into a 4000 nautical mile apogee ellipse. The injection point is a function of the lighting conditions desired under the apogee for star/landmark navigation sightings.

The CSM remains in the high apogee ellipse for approximately 24 hours. While in the high apogee ellipse, the crew work and rest cycles would be quite similar to that of the lunar mission. The SPS retrograde burn occurs over the US-ETR; the resulting ellipse is approximately 100 x 400 nautical miles. The CSM remains in the low earth orbit for the remainder of the 10-day mission. Remaining SPS propellant can be used for orbit trimming and shaping and deorbit.

In the event the COI ΔV requires more than 900 fps the CSM would not inject into the high ellipse, but remain in low earth orbit. End of mission recovery area lighting conditions are a problem for earth orbit alternates from missions which lift-off early in the first two days of the December launch window. Nodal regression during the 10 days of earth orbit together with the early lift-off results in early morning (pre-sunrise) landings. This condition can be improved in some cases by nodal plane-change maneuvers; this method, however, is somewhat restricted by ΔV capability.

A typical alternate mission 1 timeline is shown in Table 7 for the 21 December, 72° launch azimuth opportunity.

Alternate Mission 2

Alternate mission 2 is planned for a failure of the S-IVB to restart for the TLI.

If, after achieving earth parking orbit (EPO), the S-IVB for any reason cannot be reignited with the CSM attached, the spacecraft can be separated, and the same sequence outlined for the alternate 1 mission can be followed, except that there is no SPS ΔV limitation as a result of COI. Therefore, the SPS injection to a 4000 nautical mile apogee ellipse would always be performed (assuming the CSM is in a "GO" condition). The same lighting problems for navigation and lighting exists here as in alternate 1.

Shown in Table 8 is a typical timeline for alternate mission 2.

TABLE 7 - TYPICAL SPS MANEUVER TIMELINES FOR THE EARTH ORBITAL ALTERNATE MISSIONS OF THE APOLLO 8 ALTERNATE 1 MISSION[a]

Mission Time, hr:min	Event	Duration, min:sec	ΔV, fps	Resulting h_a/h_p, n.mi.	MSFN Coverage
0:11	COI	00:56	600	182/100	BDA, insertion ship
9:07	MCC	00:10	110	110/110	HAW
61:00	MCC2	--	--	107/107	--
70:10	SPS injection	05:07	4180	4000/105	TEX, MLA, GBI
93:51	Deboost	03:02	3668	400/105	MLA, GBI
100:03	MCC3	00:15	25	400/90	HAW
165:01	MCC4	00:02	363	200/90	CRO
236:51	Deorbit	00:11	282	190/-10	HAW

[a] All timelines assume 21 Dec launch, 72° launch azimuth, injection on first opportunity

TABLE 8 - TYPICAL SPS MANEUVER TIMELINES FOR THE EARTH ORBITAL ALTERNATE MISSIONS OF THE APOLLO 8 ALTERNATE 2 MISSION[a]

Mission Time, hr:min	Event	Duration, min:sec	ΔV, fps	Resulting h_a/h_p, n.mi.	MSFN Coverage
09:11	MCC	00:04	45	125/102	HAW
61:00	MCC2	--	--	115/102	--
70:08	SPS injection	05:27	4160	4000/103	TEX, MLA, GBI
93:48	Deboost	03:15	3680	400/103	MLA, GBI
100:01	MCC3	00:15	27	400/90	HAW
165:40	MCC4	00:02	359	200/90	CRO
236:46	Deorbit	00:12	286	190/-10	HAW

[a] All timelines assume 21 Dec launch, 72° launch azimuth, injection on first opportunity

Alternate Mission 3

Alternate mission 3 contains four subalternates, each depending upon the apogee that can be achieved after early termination of the S-IVB TLI burn.

Alternate 3A

In the event the S-IVB reignites but malfunctions necessitating engine shutdown, and the resulting apogee is less than 4000 nautical miles, the SPS is used to raise the apogee to 4000 nautical miles. The maneuver to achieve the 4000 nautical mile apogee would be performed at the second perigee. The CSM remains in this orbit for approximately one day (or six orbits), maneuvers to a low earth orbit, and continues a low earth orbit mission. This SPS retrograde maneuver to low earth orbit is performed over the injection ships, roughly 24 hours after TLI. A typical alternate mission 3A timeline is shown in Table 9 for the 21 December, 72° launch azimuth opportunity.

Alternate 3B

If the premature S-IVB shutdown during TLI results in an apogee of 4000 to 25,000 nautical miles, the plan is to perform an SPS phasing maneuver at the first perigee passage to change the orbital period such that at TLI-plus-24-hours the CSM passes through perigee, and such that perigee is located over one or both injection ships, or Hawaii, Canberra, or Carnarvon MSFN stations (depending upon the day of launch, since perigee moves southerly during the launch window). This is done to cover the large SPS burn used to lower the CSM into a 100 x 400 nautical mile low earth orbit. The phasing maneuver could take place anywhere from 6 hours to 16 hours GET, depending upon apogee altitude.

This alternate mission sequence is to perform a phasing maneuver at first perigee pass (3 to 14 hours after TLI), coast in the phasing ellipse until approximately TLI-plus-24-hours, maneuver into the 100 x 400 nautical mile orbit, and proceed with a low earth orbit mission (10 days). There is ample time (20 to 25 hours) spent in the high ellipse to perform navigation exercises; however, in some regions of the launch window the landmark lighting conditions underneath the apogee resulting from the TLI premature cutoff are rather poor.

A typical timeline for alternate mission 3B is shown in Table 9.

Alternate 3C

If the TLI burn results in an apogee of 25,000 to 60,000 nautical miles the procedure is to perform a retrograde phasing maneuver at the first perigee to alter the orbit period such that a later perigee occurs over a selected Pacific recovery area.

TABLE 9 - TYPICAL SPS MANEUVER TIMELINES FOR THE EARTH ORBITAL ALTERNATE MISSIONS OF THE APOLLO 8 ALTERNATE 3 MISSION[a]

Mission Time, hr:min	Event	Duration, min:sec	ΔV, fps	Resulting h_a/h_p, n.mi.	MSFN Coverage
Alternate 3A					
02:51	TLI c/o	--	--	2000/104	Injection ship, HAW
05:02	Boost to high apogee	02:24	1620	4000/106	Injection ship, GWM
19:56	Deboost	04:13	3679	400/106	TEX, MLA, GBI
63:02	MCC	00:19	29	400/90	TEX, MLA
100:15	MCC2	00:02	337	200/90	CRO
167:00	MCC3	--	--	200/90	--
236:35	Deorbit	00:15	283	188/-10	HAW
Alternate 3B					
02:52	TLI c/o	--	--	23,000/110	Injection ship, HAW
15:37	Phasing	00:07	69	22,100/110	None
27:48	Deboost	08:41	7810	400/110	Injection ship
63:21	MCC1	00:15	35	400/90	MLA, GBI
97:43	MCC2	00:02	336	200/90	CRO
167:00	MCC3	--	--	200/90	--
236:09	Deorbit	00:12	292	196/-11	HAW

[a]All timelines assume 21 Dec launch, 72° launch azimuth, injection on first opportunity

TABLE 9 - (Continued)

Mission Time, hr:min	Event	Duration, min:sec	ΔV, fps	Resulting h_a/h_p, n.mi.	MSFN Coverage
Alternate 3C					
02:52	TLI c/o	--	--	50,000/112	Injection ship, HAW
35:28	Phasing	01:00	638	29,900/111	None (Perigee)
70:29	Deboost to semisynchronous orbit	00:46	518	21,800/110	Injection ship
186:15	MCC1	--	--	--	--
236:11	Deorbit	00:04	50	21,800/25	CRO, TAN, CNB

All timelines assume a 21 Dec launch, 72° launch azimuth, injection on first opportunity.

At this perigee another SPS maneuver lowers apogee to approximately 22,000 nautical miles. The resulting ellipse is a semisynchronous (12-hour period) orbit whose perigees occur over the same two points in the Pacific and Atlantic, once each per day. The spacecraft would then remain in this ellipse for 10 days and deorbit from the semisynchronous orbit into the Pacific prime recovery area. Contingency deorbit can be performed from this ellipse at all true anomalies except a relatively small band about perigee (about 25°). This procedure provides two deorbit opportunities per day to stationary locations, one each in the Atlantic and Pacific oceans.

It may be advantageous to make the final orbit slightly less than 12 hours in period to allow the perigee to advance slowly eastward toward the prime area throughout the mission. The recovery ship could move westward, and therefore reduce the amount of required perigee progression. The disadvantage in allowing the Pacific perigee point to progress toward the prime recovery area is that the Atlantic perigee moves onto land. The remainder of the 10 days is spent in this semisynchronous orbit, allowing ample opportunity for navigation exercises. However, for some portions of the launch window, the landmark lighting conditions under the TLI-established apogee position are somewhat marginal.

A typical timeline for alternate mission 3C is shown in Table 9.

Alternate 3D

If the resulting apogee altitude is greater than 60,000 nautical miles, the SPS ΔV required at TLI-plus-3-hours to complete the injection is approximately 3000 fps. At this point, the decision would be made whether or not to perform this maneuver and continue the lunar mission. The TLI-plus-3-hours MCC will always be computed by the RTCC, and if the required ΔV is less than approximately 3000 fps, the MCC would be performed. This alternate could result in a circumlunar flyby or lunar orbit depending upon ΔV available.

Alternate Mission 4

This alternate mission was planned for the contingencies resulting from dispersed trajectories resulting from a S-IVB malfunction during TLI. Planning was based upon an attained apogee of greater than 80,000 nautical miles. In this situation, either a nominal lunar orbit mission of a lunar flyby may be flown. The alternative is dependent upon the ΔV requirements.

There are many variables which could affect this alternate mission; therefore, the RTCC has been programmed for a number of possibilities which can be computed in real time. In general, the realtime decisions will be associated with evaluation and execution of MCC's.

GO/NO-GO RULES

Go/no-go rules have been established for various plateaus or major phases of the Apollo 8 mission. These rules are presented on Figure 3.

GO/NO-GO RULE FOR CRITICAL PHASES OF THE APOLLO 8 MISSION

PRINCIPAL GO/NO-GO RULES FOR LAUNCH PHASE

- LAUNCH PHASE CONTINUED TO INSERTION IF AT ALL POSSIBLE SINCE IT IS SAFER TO REENTER FROM EARTH ORBIT THAN ATTEMPT ABORT DURING LAUNCH
- LAUNCH ABORTED FOR:
  - VIOLATION OF AUTO/MANUAL EDS LIMITS
  - TWO ENGINE OUT ON S-II (TIME DEPENDENT)
  - FAILURE OF SECOND PLANE SEPARATION
  - S-IVB LOSS OF THRUST (TIME DEPENDENT)
  - VIOLATION OF TRAJECTORY LIMIT LINES
  - LOSS OF CABIN PRESSURE AND O_2 MANIFOLD LEAK
  - LOSS OF THREE FUEL CELLS AND ONE BATTERY
  - UNCONTROLLABLE SHORTED MAIN BUS
  - LOSS OF BOTH AC BUSES DURING MODE I OR MODE II
  - SUSTAINED LEAK OR LOSS OF N_2 PRESSURE IN BOTH CM-RCS SYSTEMS (MODE I ONLY)

PRINCIPAL GO/NO-GO RULES FOR TLI PHASE

- TLI MANEUVER ATTEMPTED IF S-IVB CONSUMABLES ARE SUFFICIENT FOR GUIDED CUTOFF, IF NO S-IVB PROBLEMS ARE PRESENT WHICH WOULD RESULT IN UNSAFE RESTART, AND IF CSM HAS TOTAL SYSTEMS CAPABILITY WITH REDUNDANCY
- TLI INHIBITED FOR:
  - FAILURES IN S-IVB GUIDANCE AND CONTROL SYSTEM
  - FAILURES IN S-IVB PROPULSION SYSTEM OR INSUFFICIENT CONSUMABLES.
  - LOSS OF CABIN INTEGRITY, LEAK, FIRE, OR SMOKE IN CABIN
  - LOSS OF SURGE TANK AND REPRESS PACK OR ONE CRYO TANK
  - LOSS OF ONE MAIN O_2 REGULATOR OR LEAK IN O_2 MANIFOLD
  - TOTAL OR PARTIAL LOSS OF PRIMARY OR SECONDARY COOLANT LOOP
  - FAILURE OF BOTH H_2O ACCUMULATORS OR LOSS OF POTABLE OR WASTE H_2O TANK
  - LOSS OF ONE FUEL CELL, ONE ENTRY BATTERY, OR TWO INVERTERS
  - LOSS OF ONE BATTERY, MAIN, AC, OR BATTERY RELAY BUS
  - LOSS OF TWO-WAY COMMUNICATIONS OR GO/NO-GO INSTRUMENTATION
  - PREMATURE ACTIVATION OF SMJC OR DROGUE CHUTE DEPLOY
  - LOSS OF BOTH BMAGS IN EITHER ROLL, PITCH, OR YAW OR BOTH FDAI'S
  - LOSS OF G & N (CMC, NAV, DSKY, ISS, OR OSS)
  - VIOLATION OF SPS PRESSURE LIMITS
  - LOSS OF BOTH GN_2 TANK PRESSURES
  - ARMING OF CM-RCS SYSTEMS OR LOSS OF ONE SM-RCS QUAD OR CM-RCS SYSTEM

PRINCIPAL GO/NO-GO RULES FOR LOI PHASE

- LOI INHIBITED WITH ENTRY INTO NEXT BEST PTP OR CIRCUMLUNAR FLIGHT ATTEMPTED FOR FAILURES RESULTING IN LOSS OF CAPABILITY OR REDUNDANCY IN THOSE SYSTEMS REQUIRED FOR SAFE RETURN TO EARTH. ABORT METHOD DEPENDENT ON EFFECT OF FAILURE AND PREVAILING CONDITIONS.
  - LOSS OF CABIN INTEGRITY, GLYCOL LEAK, FIRE, OR SMOKE IN CABIN
  - LEAK IN O_2 MANIFOLD OR FAILURE OF MAIN O_2 REGULATOR
  - TOTAL OR PARTIAL LOSS OF PRIMARY OR SECONDARY COOLANT LOOP
  - FAILURE OF BOTH H_2O ACCUMULATORS OR LOSS OF POTABLE OR WASTE H_2O TANK
  - LOSS OF ONE FUEL CELL, ONE ENTRY BATTERY, OR TWO INVERTERS
  - LOSS OF ANY CRYO TANK, OR SURGE TANK AND REPRESS PACK
  - LOSS OF ONE BATTERY, MAIN, AC, OR BATTERY RELAY BUS
  - LOSS OF TWO-WAY COMMUNICATIONS OR GO/NO-GO INSTRUMENTATION
  - PREMATURE ACTIVATION OF SMJC OR DROGUE CHUTE DEPLOY
  - LOSS OF BMAGS IN EITHER ROLL, PITCH, OR YAW OR BOTH FDAI'S
  - LOSS OF EITHER TVC LOOP IN EITHER PITCH OR YAW
  - LOSS OF AUTO ATTITUDE CONTROL IN PITCH AND YAW
  - LOSS OF G & N (CMC, NAV, DSKY, ISS, OR OSS)
  - VIOLATION OF SPS ENGINE PRESSURE OR TEMPERATURE LIMITS
  - FAILURE OF ONE BANK OF BALL VALVES OR GROUND AT ONE SPS SOL DRIVER OUTPUT
  - LOSS OF ONE QUAD, ONE PITCH, ONE YAW, OR TWO ROLL THRUSTERS
  - ARMING OF CM-RCS SYSTEMS OR LOSS OF ONE SM-RCS QUAD OR CM-RCS SYSTEM

PRINCIPAL GO/NO-GO RULES FOR LUNAR ORBIT PHASE

- TRANSEARTH INJECTION PERFORMED AT NEXT BEST OPPORTUNITY FOR FAILURES RESULTING IN DEGRADED SYSTEMS CAPABILITY THAT WOULD AFFECT ABILITY TO PERFORM TEI MANEUVER OR FAILURES IN ECS AFFECTING LIFE SUPPORT
  - LOSS OF CABIN INTEGRITY, GLYCOL LEAK, FIRE, OR SMOKE IN CABIN
  - LEAK IN O_2 MANIFOLD
  - LOSS OF PRIMARY COOLANT LOOP
  - LOSS OF POTABLE OR WASTE H_2O TANK
  - LOSS OF ONE FUEL CELL, TWO BATTERIES, OR TWO INVERTERS
  - LOSS OF ONE MAIN, AC, OR BATTERY RELAY BUS
  - PREMATURE ACTIVATION OF SMJC OR DROGUE CHUTE DEPLOY
  - LOSS OF BOTH BMAGS IN EITHER ROLL, PITCH OR YAW OR BOTH FDAI'S
  - LOSS OF DIRECT RCS CONTROL FROM BOTH RHC'S
  - LOSS OF AUTO ATTITUDE CONTROL IN PITCH AND YAW
  - LOSS OF ONE QUAD, ONE PITCH, ONE YAW, OR TWO ROLL THRUSTERS
  - ARMING OF CM-RCS SYSTEMS OR LOSS OF ONE SM-RCS QUAD OR CM-RCS SYSTEM
  - LOSS OF G & N (CMC, NAV, DSKY, ISS, OR OSS)
  - DECAY IN SPS SOURCE, OXID, OR FUEL TANK PRESS, OR EXCESSIVE ΔP
  - VIOLATION OF SPS ENGINE PRESSURE OR TEMPERATURE LIMITS

SPACE VEHICLE DESCRIPTION

The Apollo 8 Mission will be performed by an Apollo Saturn V space vehicle designated AS-503, which consists of a three stage Saturn V Launch Vehicle (SA-503) and an Apollo Block II Spacecraft (CSM 103). A complete description of the space vehicle and its subsystems is included in the Mission Operation Report Supplement. The following is a brief description of the various stages.

The Saturn V Launch Vehicle consists of three propulsion stages and an Instrument Unit (IU).

The Apollo Spacecraft payload for Apollo 8 consists of a Launch Escape Assembly (LEA), Block II Command and Service Module, a Spacecraft LM Adapter, and a Lunar Module Test Article.

A list of current weights for the space vehicle is contained in Table 10.

TABLE 10

APOLLO 8 WEIGHT SUMMARY
(All Weights in Pounds)

STAGE/MODULE	INERT WEIGHT	TOTAL EXPENDABLES	TOTAL WEIGHT	FINAL SEPARATION WEIGHT
S-IC Stage	305,650	4,490,140	4,795,790	380,440
S-IC/S-II Interstage	12,610	--	12,610	--
S-II Stage	88,600	949,220	1,037,820	104,280
S-II/S-IVB Interstage	8,760	--	8,760	--
S-IVB Stage	26,000	237,640	263,640	29,340
Instrument Unit	4,880	--	4,880	--
Launch Vehicle At Ignition			6,123,500	
SC/LM Adapter	4,060	--	4,060	--
LM Test Article B	19,900	--	19,900	--
Service Module	10,670	40,580	51,250	21,440
Command Module	12,160	--	12,160	11,030
Launch Escape System	8,890	--	8,890	(Splashdown)
Spacecraft At Ignition			96,260	
Space Vehicle at Ignition			6,219,760	
S-IC Thrust Buildup			-85,880	
Space Vehicle at Lift-off			6,133,880	
Space Vehicle at Orbit Insertion			284,390	
Space Vehicle at Translunar Injection			121,720	
Spacecraft at Post-TLI Separation			63,410	

LAUNCH VEHICLE DESCRIPTION

First Stage (S-IC)

The S-IC is powered by five F-1 rocket engines each developing approximately 1,522,000 pounds of thrust at sea level and building up to 1.7 million pounds before cutoff. One engine, mounted on the vehicle longitudinal centerline, is fixed; the remaining four engines, mounted in a square pattern about the center line, are gimballed for thrust vector control by signals from the control system housed in the IU. The F-1 engines utilize LOX (liquid oxygen) and RP-1 (kerosene) as propellants.

Four cameras are mounted in jettisonable capsules on the S-IC forward skirt. Two are arranged to provide coverage of S-IC/S-II separation and the remaining two look into the LOX tank through fiber optics to record behavior during tank depletion. All four capsules are jettisoned after S-IC/S-II separation. Two TV cameras are mounted to provide real time visual monitoring of F-1 engine operation during flight.

Second Stage (S-II)

The S-II is powered by five J-2 rocket engines each developing approximately 200,000 pounds of thrust in a vacuum. One engine, mounted on the vehicle longitudinal centerline, is fixed; the remaining four engines, mounted in square pattern about the center line, are gimballed for thrust vector control by signals from the control system housed in the IU. The J-2 engines utilize LOX and LH_2 (liquid hydrogen) as propellants.

Third Stage (S-IVB)

The S-IVB consists of a cylindrical mainstage powered by a single J-2 engine which is a high performance, multiple-start engine developing approximately 200,000 pounds of thrust in a vacuum. The engine is gimballed for thrust vector control in pitch and yaw. Roll control is provided by the APS modules containing motors to provide roll control during mainstage operations and yaw and roll control during non-propulsive orbital flight.

Instrument Unit (IU)

The IU contains the following: Electrical system, self-contained and battery powered; Environmental Control, provides thermal conditioning for the electrical components and guidance systems contained in the assembly; Guidance and Control, used in solving guidance equations and controlling the attitude of the vehicle; Measuring and Telemetry, monitors and transmits flight parameters and vehicle operation information to ground stations; Radio Frequency, provides for tracking and command signals.

SPACECRAFT DESCRIPTION

The Apollo 8 spacecraft consists of a Command Module (CM 103), Service Module (SM 103), a Spacecraft Lunar Module Adapter (SLA-11), Lunar Module Test Article-B (LTA-B), and a Launch Escape Assembly (LEA).

Command Module (CM)

CM 103 is a Block II Command Module which contains automatic and manual equipment to control and monitor the spacecraft systems as well as communications equipment and systems to provide for safety and comfort of the flight crew. The primary structure is encompassed by three heat shields forming a truncated, conic structure. The CM consists of a forward compartment, a crew compartment, and an aft compartment (Figure 4).

Service Module (SM)

The Service Module is a cylindrical structure which contains systems to supplement those in the CM. The SM also contains the Service Propulsion System (SPS) which can provide 20,500 pounds of thrust in a vacuum.

Common Command and Service Module Systems

There are a number of systems which are common to the CM and SM.

Guidance and Navigation (G&N) System

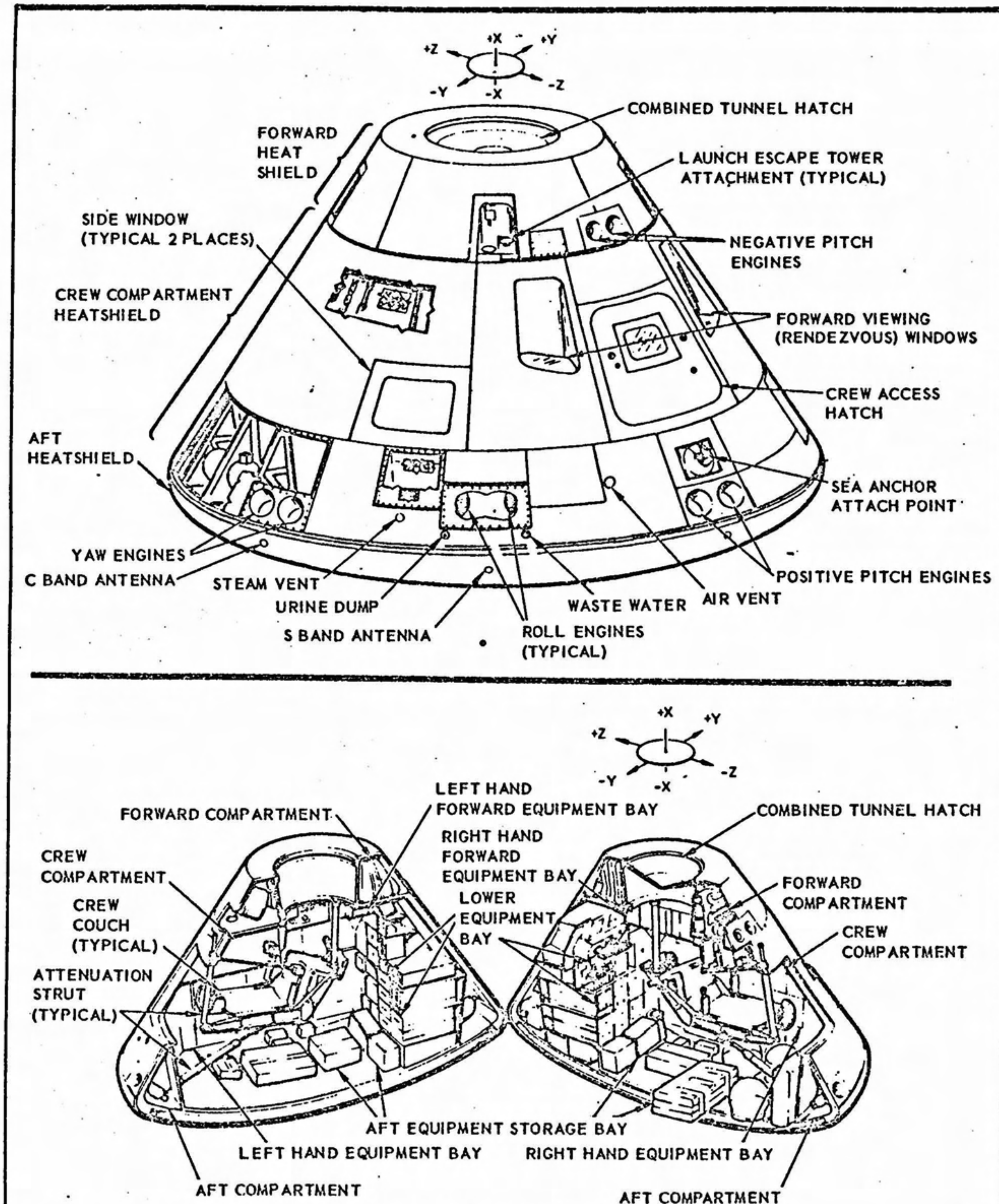
Measures spacecraft attitude and velocity, determines trajectory, controls spacecraft attitude, controls the thrust vector of the SPS engine, and provides abort information and display data.

Stabilization and Control System (SCS)

Provides control and monitoring of the spacecraft attitude, backup control of the thrust vector of the SPS engine and a backup inertial reference.

Reaction Control System (RCS)

Provides thrust for attitude maneuvers of the spacecraft in response to automatic control signals from the SCS in conjunction with the G&N system.



BLOCK II COMMAND MODULE

Electrical Power System (EPS)

Supplies all electrical power required by the CSM. The primary power source consists of three fuel cells which are the prime spacecraft power from lift-off through CM/SM separation. Five batteries - three for entry and post-landing and two for pyrotechnic uses - are located in the CM.

Environmental Control System (ECS)

Provides a controlled cabin environment and dispersion of CM equipment heat loads.

Telecommunications (T/C) System

Provides for the acquisition, processing, storage, transmission and reception of telemetry, tracking, and ranging data between the spacecraft and ground stations.

Sequential (SEQ) Systems

Major subsystems are the sequential events control system (SECS), emergency detection system (EDS), launch escape system (LES), and earth landing system (ELS). The systems interface with the RCS or SPS during an abort.

Spacecraft LM Adapter (SLA)

SLA-11 connects the CSM and the IU and houses the LTA-B. It is a truncated conic structure with a cone element length of 29 feet. The upper section is made up of four 21-foot high panels which swing open at the top and are jettisoned away from the spacecraft by springs attached to the lower fixed panels.

Lunar Module Test Article (LTA-B)

LTA-B (Figure 5) is a steel cylindrical test article instrumented to provide "g" loading values. The "g" loadings are measured by six accelerometers attached to this structure. LTA-B is attached to the SLA by four aluminum struts and remains attached throughout the mission.

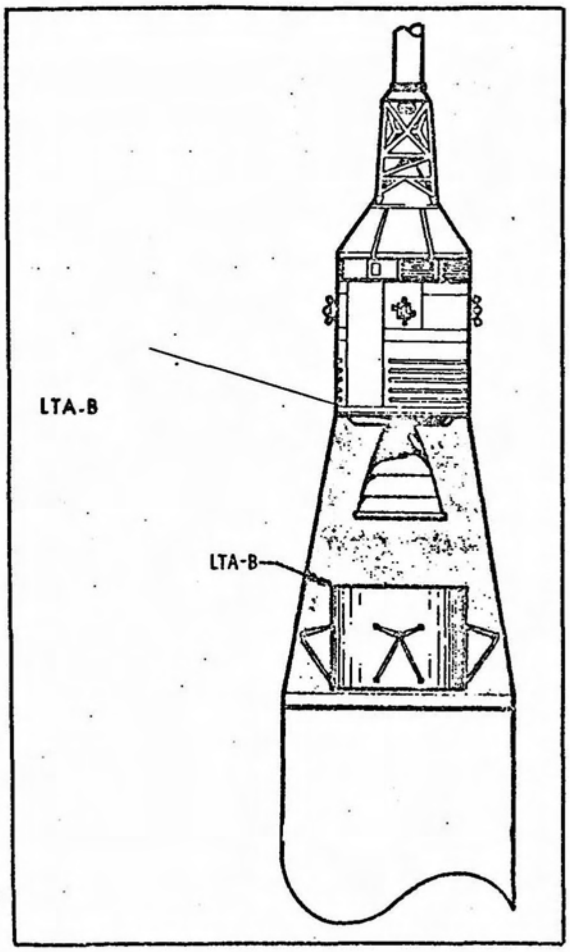


Fig 5

Launch Escape Assembly (LEA)

The LEA provides the means for separating the CM from the launch vehicle during pad or suborbital aborts. This assembly consists, in main, of the launch escape tower, launch escape motor, and tower jettison motor. Both motors utilize solid propellants. Redundant analysis and tests have been performed to maximize confidence in this emergency system.

Configuration Differences

The space vehicle for Apollo 8 varies in its configuration from that flown on Apollo 7 and those to be flown on subsequent Apollo missions. These differences are the result of the normal growth, planned changes, and experience gained on previous missions. Figure 6 shows the major configuration differences between AS-205 (CSM 101) and AS-503 (CSM 103).

HUMAN SYSTEM PROVISIONS

The major human system provisions included for the Apollo 8 mission are: Space Suits, Bioinstrumentation System, Medical Provisions, Crew Personal Hygiene, Crew Meals, Sleeping Accommodations, Oxygen Masks, and Survival Equipment. These systems provisions are described in detail in the Mission Operation Report Supplement.

As a result of difficulties with the biomedical harnesses used in Apollo 7, sturdier biomedical harnesses will be worn by the Apollo 8 crew members and a complete spare harness will be carried in stowage. In addition, medication carried on board has been increased as a result of Apollo 7 experience. Other configuration differences, as they apply to human systems, are noted on Figure 6.

MAJOR CONFIGURATION DIFFERENCES

STRUCTURES

	101	103
• Modify Foward Hatch to a Combined Forward Crew Hatch		X
• Structural Mod to SM Aft Bulkhead to Assure a 1.4 Factor of Safety for Saturn V		X
• Increase CM-SM Tension Tie Thickness		X
• Redesign SM/SLA Interface to Install Bolts from Outside		X
• Reduce Couch Strut Load/Stroke Criteria and Add Lockouts		X
• Addition of 0.03 inch of Cork on SLA		X
• Addition of LTA-B		X
• Unitized Outboard Couches	X	
• Foldable Couches (All Positions)		X

COMMUNICATIONS SUBSYSTEM

	101	103
• Conversion of Spacecraft Ground Intercom from Two-Wire to Four-Wire System		X
• Add S-band High-Gain Antenna		X
• High-Gain Antenna Automatic Reacquisition		X

ENVIRONMENTAL CONTROL SUBSYSTEM

	101	103
• Redesign ECS Radiator Flow Proportioning valve		X
• Change Material of CO_2 Absorber Elements		
- Stainless	X	
- Aluminum		X
• Demand Regulator Capability to		
- Connect Suit-to-cabin ΔP Transducer to DC Interface connector		X
- Relocation of Suit-to-Cabin ΔP Transducer		X
- Signal Amplifier Internal Circuit Change		X
• Cabin Pressure Relief Valve-Change Stroke to Reduce Variation in Relief Pressure		X

SERVICE PROPULSION SUBSYSTEM (SPS)

	101	103
• ID Ball valve to -IE Ball valve		X
• Propellant Gaging Bias Correction		X

SPACE SUIT

	101	103
• Provide Hard Ring Neck Dam	X	
• Incorporate IVCL on EV PGA's		X
• Provide Larger Helmet (Borman Only)		X
• Relocate Gas Connectors Inward		X
• Redesign Knee Convolute Cable Ending		X
• IVCL Cross Section Change		X

GUIDANCE AND NAVIGATION

	101	103
• Onboard Software -Sundisk to Colossus		X

STABILIZATION CONTROL SUBSYSTEM (SCS)

	101	103
• Single Jet Isolation to Provide Separate SCS Power Control		
- Powered From Bank Switches	X	
- Powered From Separate Switches		X

ORDNANCE

	101	103
• SLA Panels (Jettisonable)		X
• LM Ejection Thruster		X

INSTRUMENTATION

	101	103
• Van Allen Belt Dosimeter		X
• Add POGO Instrumentation		X
• Nuclear Particle Detection System		X
• Delete Two SPS Transfer Line Temperature Measurements and add Two Temperature Measurements to Propellant Utilization Valve (High and Low Bit Rate)		X

DISPLAYS AND CONTROLS

	101	103
• IU-Up TLM Inhibit Switch on Panel No. 2		X
• Redundant Launch Vehicle Attitude Error Display (Hardware change for MSFC only)		X

CREW EQUIPMENT

	101	103
• Deletion of Right-Hand Crewman's Right-hand Arm Rest		X

CM - RCS

	101	103
• Add Holes to RCS Tank Abrasion PADS to Allow Depressurization of He		X

ENTRY MONITOR SUBSYSTEM

	101	103
• Entry Monitor Scroll Patterns Changed From Earth Orbital To Lunar Mission		X

Fig. 6

LAUNCH COMPLEX

The AS-503 space vehicle (SV) will be launched from Launch Complex 39 at Kenneu Space Center, Florida. LC 39 was designed and built to the mobile concept wherein the space vehicle is checked out in an enclosed building before being moved to the pad for final preparations and launch.

The major components of LC 39 include the Vehicle Assembly Building (VAB), the Launch Control Center (LCC), the Mobile Launcher (ML), the Crawler Transporter (C/T), the Mobile Service Structure (MSS), and the Launch Pad.

The LCC is a permanent four story structure located adjacent to the VAB and serves as the focal point for monitoring and controlling vehicle checkout and launch activities for all Saturn V launches.

The ground floor of the structure is devoted to service and support functions. Telemetry equipment occupies the second floor and the third floor is divided into firing rooms, computer rooms, and offices. Firing room 1 will be used for Apollo 8.-

The AS-503 space vehicle was received at KSC and assembly and initial overall checkout was performed in the VAB on the mobile launcher. Rollout occurred on 9 October 1968. Transportation, to the pad, of the assembled space vehicle and ML is provided by the crawler transporter (C/T) which also moves the MSS to the pad after the ML and SV have been secured. The MSS provides 360-degree access to the space vehicle at the launch pad by means of five vertically adjustable, elevator serviced, enclosed platforms. The MSS is removed to its park position prior to launch.

The emergency egress route system at LC 39 is made up of three major components: the high speed elevators, slide tube, and slide wire. The primary route for egress from the CM is via the elevators and, if necessary, through the slide tube which exits into an underground blast room. Apollo 8 is the first mission to employ the slide wire on LC 39. This system is attached to the ML 409 feet above ground and extends approximately 2500 feet west of the ML where it is attached to a 30-foot tower.

Refer to the Mission Operation Report Supplement for a more thorough description of LC 39.

MISSION SUPPORT

Mission support is provided by the Launch Control Center (LCC), the Mission Control Center (MCC), the Manned Space Flight Network (MSFN), and the recovery forces. The LCC is essentially concerned with pre-launch checkout, countdown and with launching the space vehicle, while MCC located at Houston, Texas, provides centralized mission control from lift-off through recovery. MCC functions within the framework of a Communications, Command and Telemetry System (CCATS); Real Time Computer Complex (RTCC); Voice Communications System; Display/Control System; and, a Mission Operations Control Room (MOCR). These systems allow the flight control personnel to remain in contact with the spacecraft, receive telemetry and operational data which can be processed by the CCATS and RTCC for verification of a safe mission or compute alternatives. The MOCR is staffed with specialists in all aspects of the mission who provide the flight director with real time evaluations of mission progress.

The MSFN is a worldwide communications network which is controlled by the MCC during Apollo missions. The network is composed of fixed stations (Figure 7) and supplemented by mobile stations (Table 11) which are optimally located within a global band extending from approximately 40° South latitude to 40° North latitude. Station capabilities are summarized in Table 12.

The functions of these stations are to provide tracking, telemetry, command and communications (voice, television) both on an updata link to the spacecraft and on a down data link to the MCC. Connection between these many MSFN stations and the MCC is provided by NASA Communications (NASCOM). Refer to the Mission Operation Report Supplement for more detail on Mission Support.

The Apollo 8 mission will be the first opportunity to acquire mission experience for the Deep Space Instrumentation Facilities (DSIF). There are three stations in this network - Madrid (MAD), Goldstone (GDS), and Honeysuckle Creek (HSK). These stations, each having 85-foot antennas, will be tested with the onboard spacecraft high-gain and omni-antennas to evaluate long range transmissions.

Real-time TV signals from the spacecraft will be available through Madrid and Goldstone to the MCC.

In addition, the 210-foot antenna at Goldstone will be in use for Apollo 8. This antenna will perform a backup function and will be used in a passive mode recording telemetry and voice transmissions.

M-932-68-08

ORBITAL EVENTS GROUND TRACK

1st EDITION, 26 NOVEMBER 1968

PREPARED UNDER THE DIRECTION OF THE DEPARTMENT OF DEFENSE BY THE AERONAUTICAL CHART AND INFORMATION CENTER, UNITED STATES AIR FORCE FOR THE NATIONAL AERONAUTICS AND SPACE ADMINISTRATION.

Lithographed by ACIC 11-68

12/9/68

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NOTE: The representation of international boundaries on this chart is not necessarily authoritative.

Fig 7

MERCATOR PROJECTION
SCALE 1:40,000,000 AT THE EQUATOR

LEGEND

International boundary
Capital cities
Other cities
Airfields
HC 130 home base airfields

TRACKING STATION SYMBOLS

Location symbol and call letters .... MIL/MLA/CNV
Station capabilities .... (SCVT)
S. Unified S Band (Radar tracking, Telemetry, A-G Voice and Command. NOTE: Ship HTV does not have command capability.)
C. C Band Radar tracking
V. VHF A-G Voice
T. VHF Telemetry

TRACKING LIMIT SYMBOLS

USB command capability
No command capability
Tracking limits are based upon 100 NM spacecraft altitude for a 0° antenna elevation. USB station tracking limits reflect obstruction and keyhole limitations per antenna coverage report X-820-67-581.
NOTE: SHIP POSITIONS FOR RED, MER AND HTV ARE NOMINAL FOR 21 DECEMBER 1968 LAUNCH DATE ONLY AND CURRENT THROUGH FSR 43 DATED 26 NOVEMBER 1968.

GROUND TRACK SYMBOLS

Apollo Spacecraft 72
Elapsed time from launch, in hours and minutes, is contained in boxes centered on the ground track at 10 minute intervals for each of launch azimuths 72°, 90° and 108°.
Ground track segments are shown for specific intermediate launch azimuths. Launch azimuths are labeled in degrees on the ground track and segments.

TRANSLUNAR INJECTION SYMBOLS

Translunar Injection Burn Initiation .... TLI (21 DEC)
Translunar Injection Burn Termination .... TLT (21 DEC)
Each TLI and TLT line is designated by appropriate launch date.

APOLLO EARTH ORBIT CHART (AEO)

APOLLO MISSION 8

FOR DECEMBER 1968 LAUNCH DATES

TABLE 11

MSFN MOBILE FACILITIES

APOLLO SHIPS (4 required)

FUNCTION	SUPPORT	LOCATION	NAME
Apollo Insertion Ship	Insertion, abort contingencies	25°N, 49°W	USNS VANGUARD
Apollo Injection Ship	Orbital event support	2.5°N, 155.5°E	USNS REDSTONE
Apollo Injection Ship	Also reentry area support ship	7.5°N, 178°W - I 18°N, 159°E - R	USNS MERCURY
Apollo Reentry Ship	Also supports injection	25°N, 175°W - I 10°N, 177°E - R	USNS HUNTSVILLE

APOLLO AIRCRAFT (5 required)

A/RIA will support the mission on specified revolutions from assigned Test Support Positions (TSP). In addition, A/RIA will cover reentry (400,000 ft) thru crew recovery. A/RIA #1, #2 and #3 will operate in the Pacific Sector and A/RIA #4 and #5 in the Indian Ocean.

I - Injection position
R - Reentry position

TABLE 12

NETWORK CONFIGURATION FOR THE AS-503 MISSION

Systems	Tracking				USB				TLM					CMD			Data Processing			Comm					Other	
Facilities	C-band (High Speed)	C-band (Low Speed)	ODOP	Optical	USB	Voice (A/G)	Command	Telemetry	VHF Links	FM Remoting	Mag Tape Recording	Decoms	Displays		CMD Destruct		642B TLM	642B CMD	1218	High Speed Data	Wideband Data	TTY	Voice (SCAMA)	VHF A/G Voice	Video (TV)	SPAN
CIF											X	X								X	X		X		X	
TEL 4									X	X	X															
CNV	X		X	X											X									X		
PAT*	X	X																								
MLA	X	X																								
MIL					X	X	X	X	X	X	X	X	X				X	X	X	X		X	X	X	X	
GBI	X*	X*							X		X				X											
GBM					X	X	X	X	X	X	X	X					X	X	X	X		X	X	X	X	
GTK	X	X													X											
ANG					X	X	X	X	X	X	X	X					X	X	X	X		X	X	X	X	
ANT	X	X							X		X															
BDA	X	X			X	X	X	X	X	X	X	X	X		X		X	X	X	X		X	X	X	X	
ACN					X	X	X	X	X	X	X	X					X	X	X	X		X	X	X	X	
ASC	X	X																								
MAD					X	X	X	X		X	X	X					X	X	X	X		X	X		X	
MADX					X	X	X	X											X			X	X			
CYI		X			X	X	X	X	X	X	X	X					X	X	X	X		X	X	X	X	X
PRE		X																								
TAN		X							X		X											X	X	X		
CRO	X	X			X	X	X	X	X	X	X	X					X	X	X	X		X	X	X	X	X
HSK					X	X	X	X		X	X	X					X	X	X	X		X	X		X	
HSKX					X	X	X	X											X			X	X			
GWM					X	X	X	X	X	X	X	X					X	X	X	X		X	X	X	X	
HAW		X			X	X	X	X	X	X	X	X	X				X	X	X	X		X	X	X	X	
CAL	X	X																				X	X	X		
WHS	X	X																				X	X			
GDS					X	X	X	X		X	X	X					X	X	X	X		X	X		X	
GDSX					X	X	X	X											X			X	X			
GYM					X	X	X	X	X	X	X	X					X	X	X	X		X	X	X	X	
TEX					X	X	X	X	X	X	X	X					X	X	X	X		X	X	X	X	
HTV		X			X	X		X			X	X										X	X	X		
RED	X	X			X	X	X	X	X	X	X	X					X	X		X	X	X	X	X	X	
VAN	X	X			X	X	X	X	X	X	X	X					X	X		X		X	X	X	X	
MER	X	X			X	X	X	X	X	X	X	X					X	X		X		X	X	X	X	
ARIA (6)					X	X		X	X		X												X	X		

*Subject to availability.

RECOVERY

Recovery operations begin with touchdown of the Command Module and are terminated with the retrieval of the Command Module and astronauts. The recovery forces are made up of ships and aircraft deployed in the planned recovery areas (TABLE 13). These areas include the primary landing areas and abort and contingency landing areas.

TABLE 13

RECOVERY SHIP DEPLOYMENT

Ship Station (Figs. 8,9,10,11,12, and 13)	Ship
SRS 1	USS Guadalcanal (LPH-7)
SRS 2	USS Rankin (AKA-103)
SRS 3	USNS Vanguard (AIS)
SRS 4	USS Chukawan (AO-100)
SRS 5	USS Sandoval (LPA-194)
SRS 6	USS Francis Marion (LPA-249)
PRS	USS Yorktown (CVS-10)
SRS 7	USS Rupertus (DD-851)
SRS 8	USS Cochrane (DDG-21)
SRS 9	USS Nicholas (DD-449)

Various safety features are included on the CM to aid the astronauts and recovery team. Among these are a recovery light, sea marker dye, VHF communications, and up-righting bags which are inflated should the CM land in an inverted position.

The recovery team is trained to provide a fast and safe recovery of the CM. Upon location of the CM, helicopters with swim teams onboard are launched to deliver a flotation collar to the CM which will provided flotation capability for a minimum of 48 hours. The crew may be recovered by the helicopters or remain in the CM and be picked up by ship.

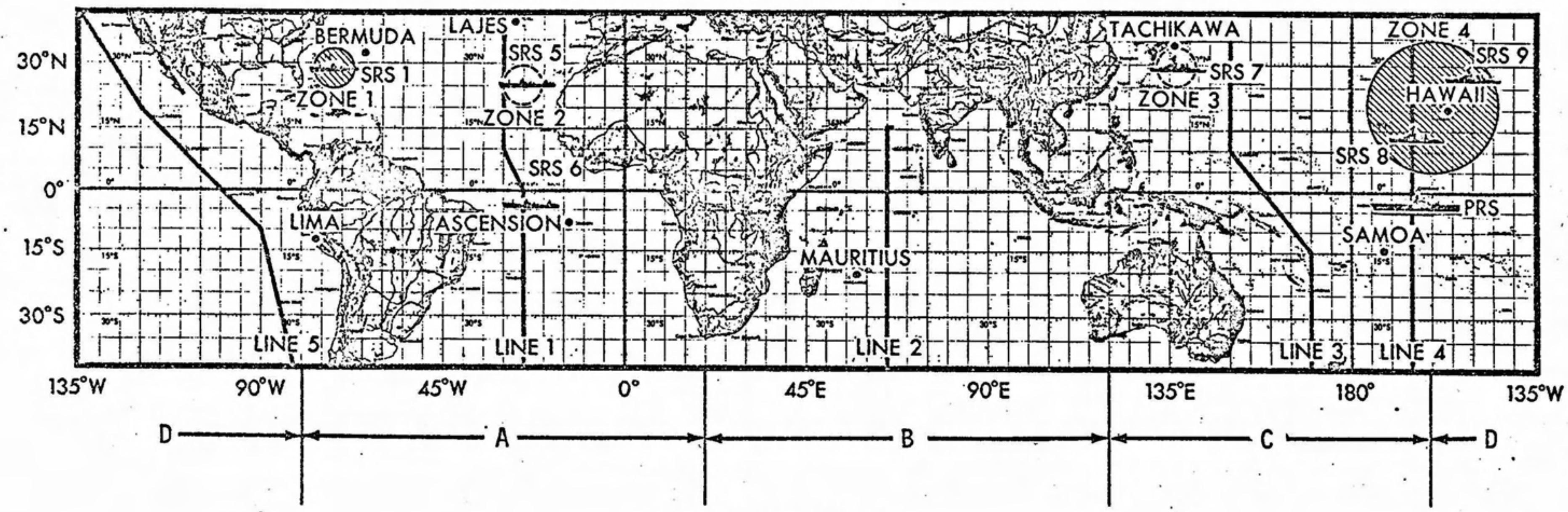
RECOVERY CONTROL

Recovery forces for the Apollo 8 mission will be deployed in both the Atlantic and Pacific Ocean (Figure 8 and Table 13). The recovery will be directed from the Recovery Control Room of the MCC and will be supported by two satellite recovery control centers: the Atlantic Recovery Control Center located at Norfolk, Virginia, and the Pacific Recovery Control Center located at Kunia in the Hawaiian Islands. In addition to the recovery control centers, there will be NASA representatives deployed with recovery forces throughout the worldwide DOD recovery network.

APOLLO 8 RECOVERY FORCE DEPLOYMENT

RECOVERY LINES AND CONTINGENCY AREAS

2-HC-130'S WILL BE STATIONED AT EACH STAGING BASE



CONTINGENCY AREAS

Fig 8

M-932-68-08

There are five recovery areas which contain the spacecraft landing points following aborts, alternate missions, and the nominal Apollo 8 mission. The areas are: Launch Site Area, Launch Abort Area, Primary Landing Area, Secondary Landing Area and Contingency Landing Areas.

LAUNCH SITE AREA

The launch site area is that area in which the CM will land following aborts which could occur between launch escape system activation and approximately 90 seconds ground elapsed time (GET).

For a specific wind direction and velocity at the launch site, the locus of possible CM landing points will lie within a relatively narrow corridor in the launch site area. This corridor, based on winds and launch azimuth, will be defined and passed to the launch site recovery forces just prior to launch. The Launch Site Area is illustrated in Figure 9.

LAUNCH ABORT AREA

The launch abort area is that area in which the CM will land following an abort initiated during the launch phase of flight. Since the launch azimuth may vary from 72° to 107°, the launch abort area is designed to include all possible CM landing points following a launch abort from any launch azimuth. Since a landing following a launch abort will be on or near the ground track, the locus of possible CM landing points is a relatively narrow corridor within the launch abort area once the launch azimuth is determined.

The launch abort area (Figure 10) is divided into two sectors, A and B. These sectors are used to differentiate between the recovery force support required in the eastern and western portions of the area. Sector A is all the area in the launch abort area that is within 1200 nautical miles of the launch site. This sector includes a majority of the landing points that occur with a "high g" entry trajectory. Sector B is all the area in the launch abort area that is between 1200 and 3400 nautical miles of the launch site.

Located in the western portion of sector A is the camera capsule landing area (Figure 11). This area, bounded by 335 and 365 downrange lines, contains the predicted camera capsule landing points and the high probability dispersion areas (30 by 15 nautical mile ellipses) around the landing points for any launch azimuth.

PRIMARY LANDING AREA

A primary landing area is an area in which a landing could occur after launch and the probability of such a landing is sufficiently high to warrant a requirement for primary recovery ship (PRS) support. The area is designed to encompass the spacecraft target

LAUNCH SITE AREA AND RECOMMENDED FORCE DEPLOYMENT

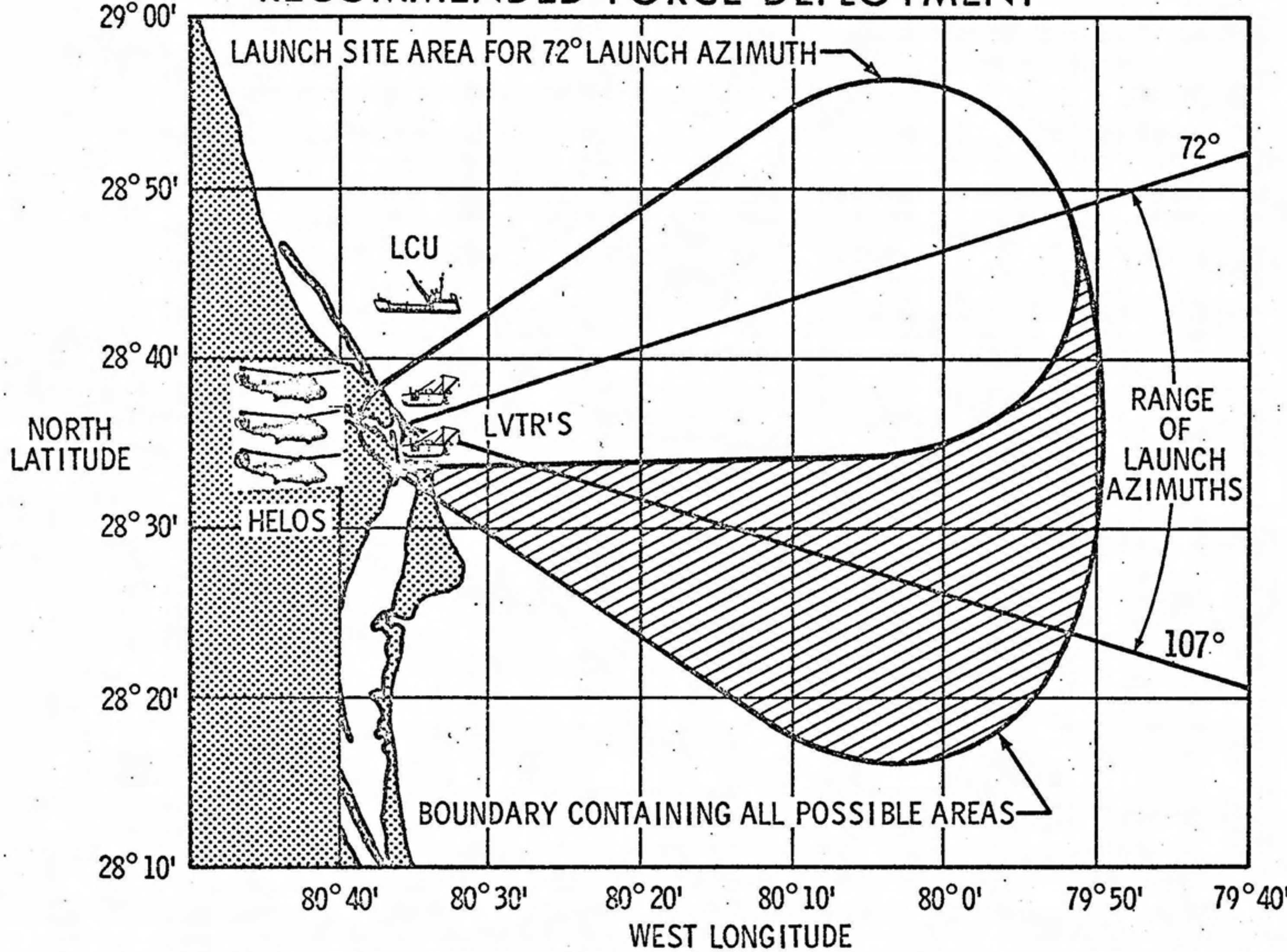


Fig 9

M-932-68-08

LAUNCH ABORT AREA AND RECOMMENDED FORCE DEPLOYMENT

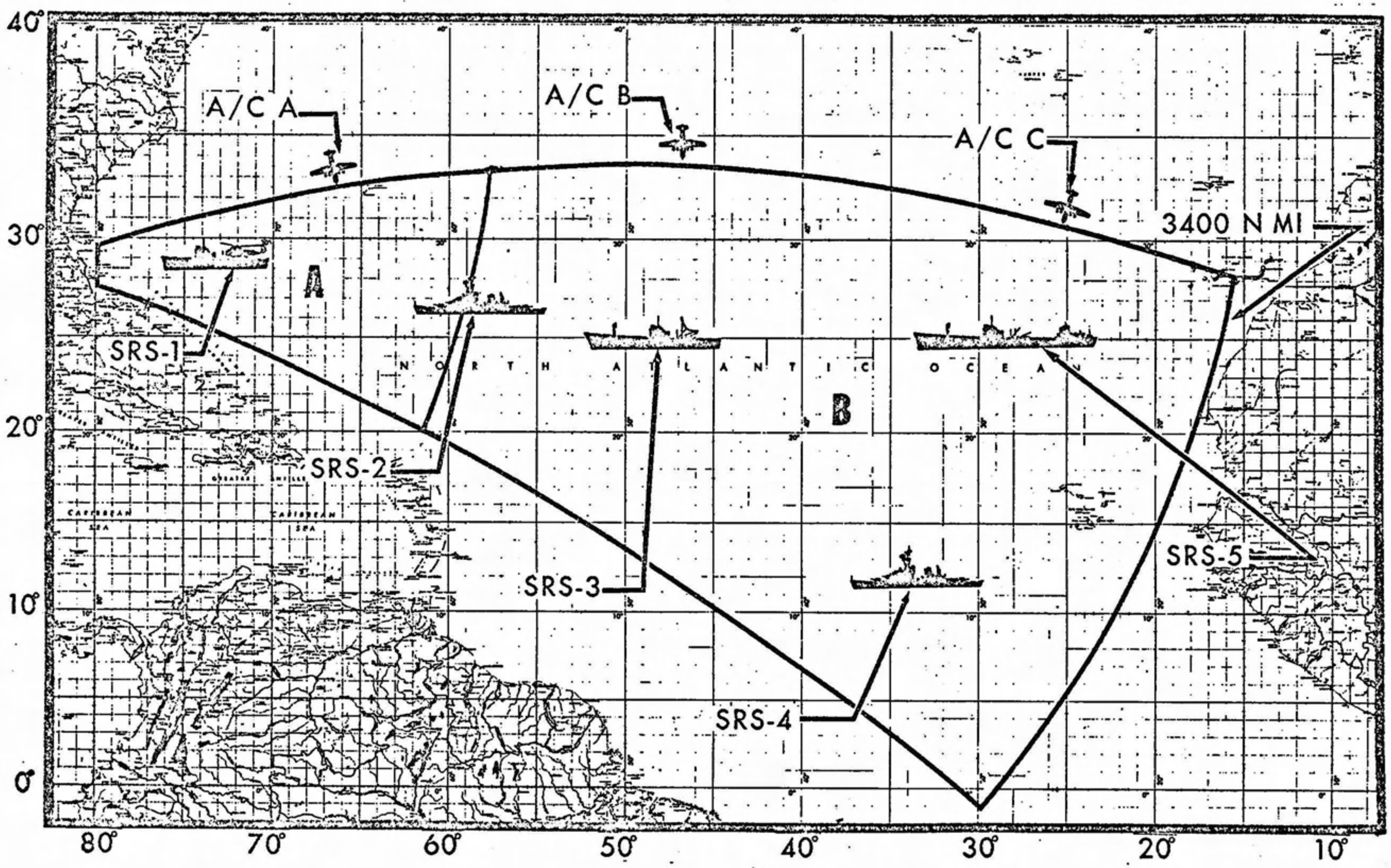


Fig 10

CAMERA CAPSULE LANDING AREA AND RECOMMENDED FORCE DEPLOYMENT

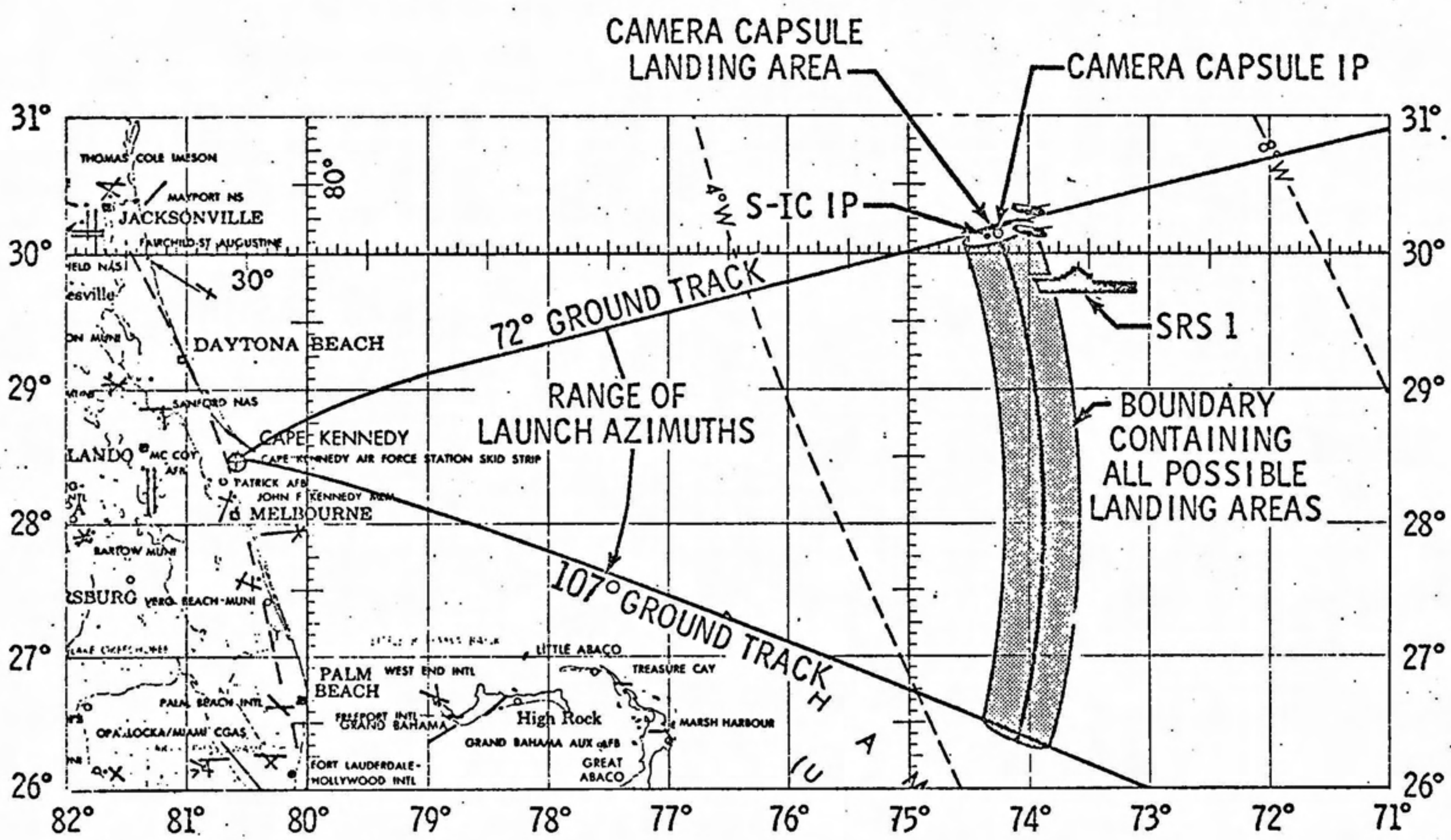


Fig 11

point and the surrounding dispersion area associated with a high-speed entry from deep space. The areas are used to define recovery support for a CM landing following the nominal mission or an abort initiated any time after the completion of TLI. For the Apollo 8 mission, the primary landing area is bounded by an ellipse 800 nautical miles long by 300 nautical miles wide. Primary landing areas will be located on or near the Mid-Pacific recovery line, one of five recovery lines designated as general locations where landing areas may be selected (Figure 12).

SECONDARY LANDING AREA

A secondary landing area is an area in which a landing could occur after launch and the probability of such a landing is sufficiently high to warrant a requirement for at least secondary recovery ship (SRS) support. The area is designed to encompass the spacecraft target point and its associated high-probability landing point dispersion.

Deep space secondary landing areas are designed to encompass the target point and dispersion area associated with a high-speed entry from deep space. These will be used to define recovery support for a CM landing from an abort following the completion of TLI. These areas are bounded by an ellipse 800 nautical miles long by 300 nautical miles wide. Deep space secondary landing areas will be selected on or near the Atlantic Ocean recovery line (Figure 13).

CONTINGENCY LANDING AREA

The contingency landing area is all the area outside the launch site, launch abort, and primary and secondary landing areas within which a landing could possibly occur. These contingency areas are shown on Figure 8.

PRIMARY LANDING AREA AND RECOMMENDED FORCE DEPLOYMENT

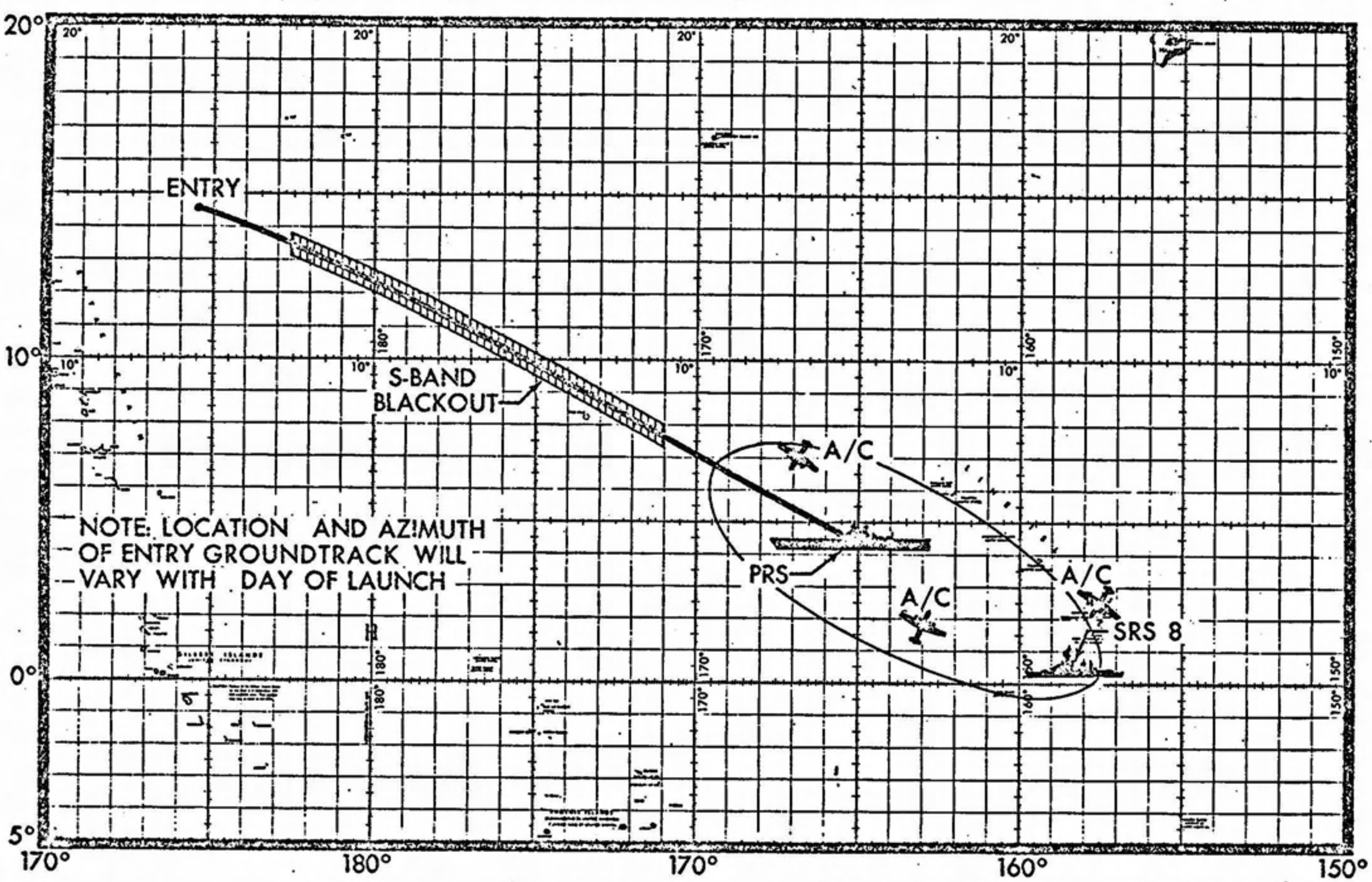


Fig 12

M-932-68-08

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DEEP SPACE SECONDARY LANDING AREA AND RECOMMENDED FORCE DEPLOYMENT

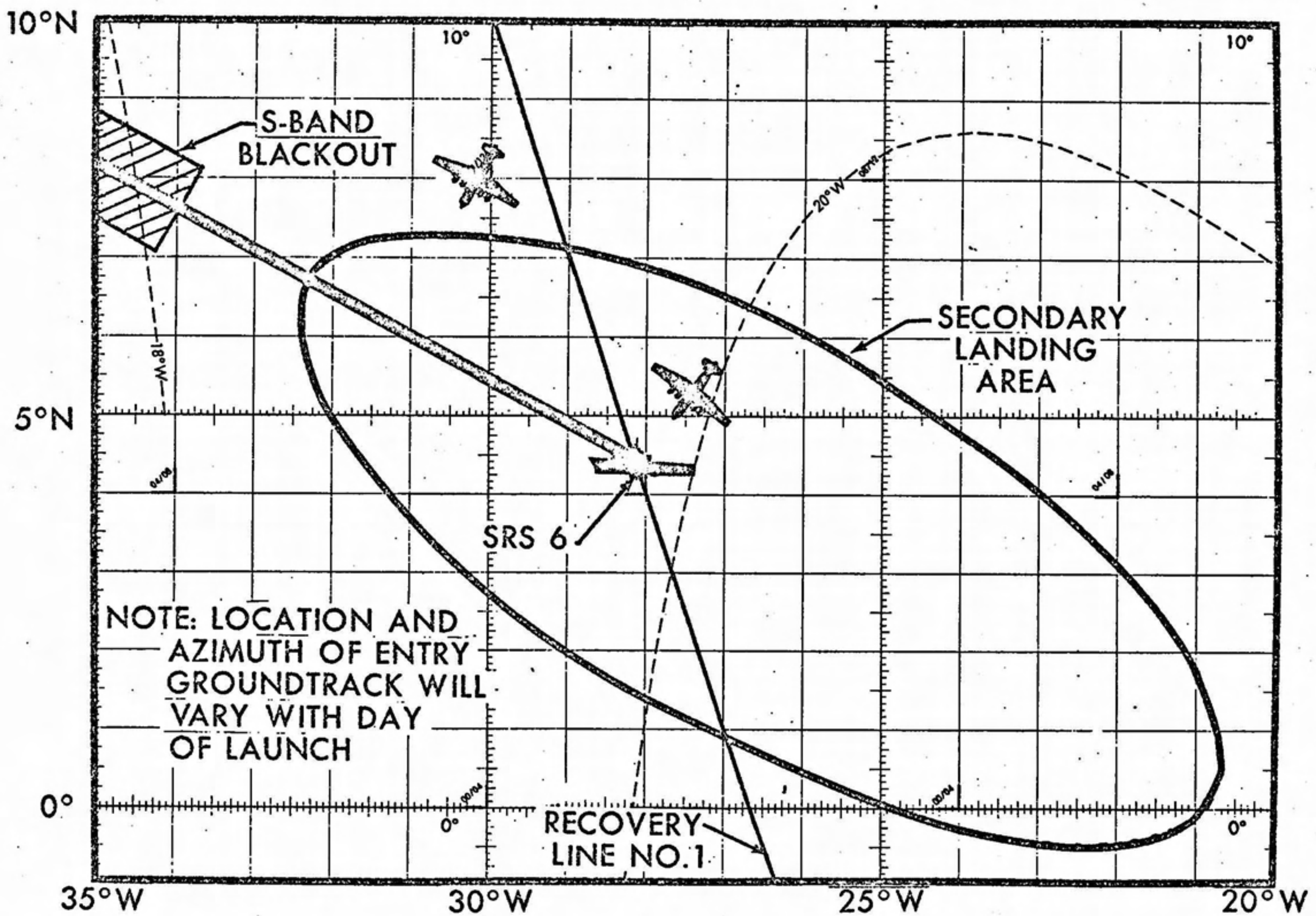


Fig 13

M-932-68-08

FLIGHT CREW

FLIGHT CREW ASSIGNMENTS

Prime Crew (Figure 14)

Command Pilot (CDR) - Frank Borman, Col., USAF
Command Module Pilot (CMP) - James A. Lovell, Jr., Capt., USN
Lunar Module Pilot (LMP) - William A. Anders, Maj., USAF

Backup Crew (Figure 15)

Command Pilot (CDR) - Neil A. Armstrong (Mr.)
Command Module Pilot (CMP) - Edwin E. Aldrin, Jr., Col., USAF
Lunar Module Pilot (LMP) - Fred W. Haise, Jr., (Mr.)

PRIME CREW BIOGRAPHICAL DATA

Command Pilot (CDR)

NAME: Frank Borman (Colonel, USAF)

DATE OF BIRTH: 14 March 1928

PHYSICAL DESCRIPTION: Height: 5 feet, 10 inches; weight: 163 pounds

EDUCATION: Received a Bachelor of Science degree from the United States Military Academy at West Point in 1950 and a Master of Science degree in Aeronautical Engineering from the California Institute of Technology, Pasadena, California, in 1957.

ORGANIZATIONS: Member of the American Institute of Aeronautics and Astronautics and the Society of Experimental Test Pilots.

SPECIAL HONORS: Awarded the NASA Exceptional Service Medal, Air Force Astronaut Wings, and Air Force Distinguished Flying Cross: recipient of the 1966 American Astronautical Flight Achievement Award and the 1966 Air Force Association David C. Schilling Flight Trophy; co-recipient of the 1966 Harmon International Aviation Trophy; and recipient of the California Institute of Technology Distinguished Alumni Service Award for 1966.

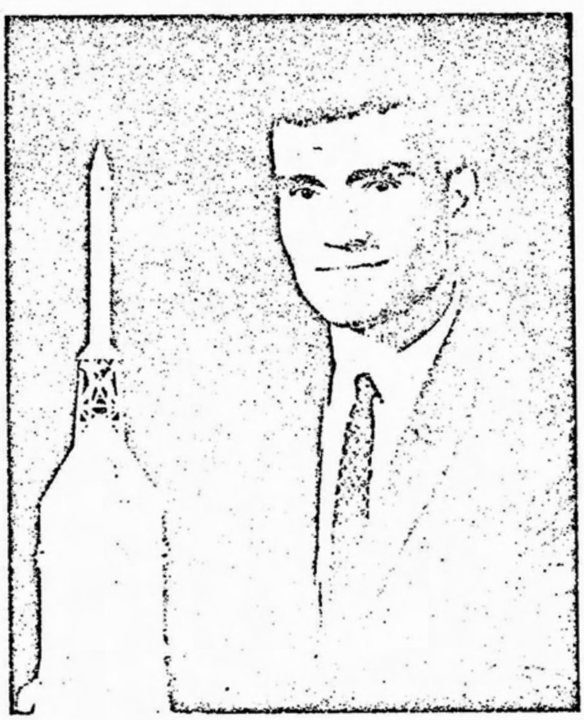
PRIME CREW

JAMES A. LOVELL, Jr. WILLIAM A. ANDERS FRANK BORMAN

BACKUP CREW

NEIL A. ARMSTRONG

EDWIN E. ALDRIN, Jr.



FRED W. HAISE, Jr.

EXPERIENCE: Borman, an Air Force Colonel, entered the Air Force after graduation from West Point and received his pilot training at Williams Air Force Base, Arizona.

From 1951 to 1956, he was assigned to various fighter squadrons in the United States and the Philippines. He became an instructor of thermodynamics and fluid mechanics at the Military Academy in 1957 and subsequently attended the USAF Aerospace Research Pilots School from which he graduated in 1960. He remained there as an instructor until 1962.

He has accumulated over 5500 hours flying time, including 4500 hours in jet aircraft.

CURRENT ASSIGNMENT: Colonel Borman was selected as an astronaut by NASA in September 1962. He has performed a variety of special duties, including an assignment as a member of the Apollo 204 Review Board.

As command pilot of the history-making Gemini 7 mission, launched on 4 December 1965, he participated in establishing a number of space "firsts" -- among which are the longest manned space flight (330 hours and 35 minutes) and the first rendezvous of two manned maneuverable spacecraft as Gemini 7 was joined in orbit by Gemini 6.

<u>Command Module Pilot (CMP)</u>

NAME: James A. Lovell, Jr. (Captain, USN)

DATE OF BIRTH: 25 March 1928

PHYSICAL DESCRIPTION: Height: 5 feet, 11 inches; weight: 170 pounds.

EDUCATION: Graduated from Juneau High School, Milwaukee, Wisconsin; attended the University of Wisconsin for two years, then received a Bachelor of Science degree from the United States Naval Academy in 1952.

ORGANIZATIONS: Member of the Society of Experimental Test Pilots and the Explorers Club.

SPECIAL HONORS: Awarded two NASA Exceptional Service Medals, the Navy Astronaut Wings, two Navy Distinguished Flying Crosses, and the 1967 FAI Gold Space Medal (Athens, Greece); and co-recipient of the 1966 American Astronautical Society Flight Achievement Award and the Harmon International Aviation Trophy in 1966 and 1967.

EXPERIENCE: Lovell, a Navy Captain, received flight training following graduation from Annapolis.

He has had numerous naval aviator assignments including a 4-year tour as a test pilot at the Naval Air Test Center, Patuxent River, Maryland. While there he served as program manager for the F4H weapon system evaluation. A graduate of the Aviation Safety School of the University of Southern California, he also served as a flight instructor and safety officer with Fighter Squadron 101 at the Naval Air Station, Oceana, Virginia.

Of the 4000 hours flying time he has accumulated, more than 3000 hours are in jet aircraft.

CURRENT ASSIGNMENT: Captain Lovell was selected as an astronaut by NASA in September 1962.

On 4 December 1965, he and command pilot Frank Borman were launched on the Gemini 7 mission. The flight lasted 330 hours and 35 minutes.

The Gemini 12 mission, with Lovell and pilot Edwin Aldrin, began on 11 November 1966. This 4-day, 59-revolution flight brought the Gemini Program to a successful close. Major accomplishments of the 94 hour, 35 minute flight included a third-revolution rendezvous with the previously launched Agena (using for the first time backup onboard computations due to a radar failure): a tethered station-keeping exercise; retrieval of a micro-meteorite experiment package from the spacecraft exterior; an evaluation of the use of body restraints specially designed for completing work tasks outside of the spacecraft; and completion of numerous photographic experiments, the highlights of which are the first pictures taken from space of an eclipse of the sun.

Lunar Module Pilot (LMP)

NAME: William A. Anders (Major, USAF)

DATE OF BIRTH: 17 October 1933

PHYSICAL DESCRIPTION: Height: 5 feet, 8 inches; weight: 145 pounds

EDUCATION: Received a Bachelor of Science degree from the United States Naval Academy in 1955 and a Master of Science degree in Nuclear Engineering from the Air Force Institute of Technology at Wright-Patterson Air Force Base, Ohio, in 1962.

ORGANIZATIONS: Member of the American Nuclear Society and Tau Beta Pi.

SPECIAL HONORS: Awarded the Air Force Commendation Medal.

EXPERIENCE: Anders, an Air Force Major, was commissioned in the Air Force upon graduation from the Naval Academy. After Air Force flight training, he served as a fighter pilot in all-weather interceptor squadrons of the Air Defense Command.

After his graduate training, he served as a nuclear engineer and instructor pilot at the Air Force Weapons Laboratory, Kirtland Air Force Base, New Mexico, where he was responsible for technical management of radiation nuclear power reactor shielding and radiation effects programs.

He has logged more than 3000 hours flying time.

CURRENT ASSIGNMENT: Major Anders was one of the third group of astronauts selected by NASA in October 1963 and has since served as backup pilot for the Gemini 11 mission.

BACKUP CREW BIOGRAPHICAL DATA

Command Pilot (CDR)

NAME: Neil A. Armstrong (Mr.)

DATE OF BIRTH: 5 August 1930

PHYSICAL DESCRIPTION: Height: 5 feet, 11 inches; weight: 165 pounds.

EDUCATION: Attended secondary school in Wapakoneta, Ohio; received a Bachelor of Science degree in Aeronautical Engineering from Purdue University in 1955. Graduate School - University of Southern California.

ORGANIZATIONS: Associate Fellow of the Society of Experimental Test Pilots; Associate Fellow of the American Institute of Aeronautics and Astronautics; and member of the Soaring Society of America.

SPECIAL HONORS: Recipient of the 1962 Institute of Aerospace Sciences Octave Chanute Award; the 1966 AIAA Astronautics Award; the NASA Exceptional Service Medal; and the 1962 John J. Montgomery Award.

EXPERIENCE: Armstrong was a naval aviator from 1949 to 1952 and flew 78 combat missions during the Korean action.

He joined NASA's Lewis Research Center in 1955 (then NACA Lewis Flight Propulsion Laboratory) and later transferred to the NASA High Speed Flight Station (now Flight Research Center) at Edwards Air Force Base, California, as an aeronautical research pilot for NACA and NASA. In this capacity, he performed as an X-15 project pilot, flying that aircraft to over 200,000 feet and approximately 4000 miles per hour.

Other flight test work included piloting the X-1 rocket airplane, the F-100, F-101, F-102, F-104, F5D B-47, the paraglider, and others.

He has logged more than 4000 hours flying time.

CURRENT ASSIGNMENT: Mr. Armstrong was selected as an astronaut by NASA in September 1962. He served as backup command pilot for the Gemini 5 and Gemini 11 flights.

As command pilot for the Gemini 8 mission, which was launched on 16 March 1966, he performed the first successful docking of two vehicles in space.

Command Module Pilot (CMP)

NAME: Edwin E. Aldrin, Jr. (Colonel, USAF)

DATE OF BIRTH: 20 January 1930

PHYSICAL DESCRIPTION: Height: 5 feet, 10 inches; weight: 165 pounds.

EDUCATION: Graduated from Montclair High School, Montclair, New Jersey; received a Bachelor of Science degree from the United States Military Academy at West Point, New York, in 1951 and a Doctor of Science degree in Astronautics from the Massachusetts Institute of Technology in 1963; recipient of an Honorary Doctorate of Science degree from Gustavus Adolphus College in 1967.

ORGANIZATIONS: Associate Fellow of the American Institute of Aeronautics and Astronautics; member of the Society of Experimental Test Pilots, Sigma Gamma Tau (aeronautical engineering society), Tau Beta Pi (national engineering society), and Sigma Xi (national science research society); and a 32 Degree Mason advanced through the Commandery and Shrine.

SPECIAL HONORS: Awarded the Distinguished Flying Cross with one oak leaf cluster, the Air Medal with two oak leaf clusters, the Air Force Commendation Medal, the NASA Exceptional Service Medal and Air Force Command Pilot Astronaut Wings, the NASA Group Achievement Award for Rendezvous Operations Planning Team, an Honorary Life Membership in the International Association of Machinists and Aerospace Workers, and an Honory Membership in the Aerospace Medical Association.

EXPERIENCE: Aldrin, an Air Force Colonel, was graduated third in a class of 475 from the United States Military Academy at West Point in 1951 and subsequently received his wings at Bryan, Texas, in 1952.

He flew 66 combat missions in F-86 aircraft while on duty in Korea with the 51st Fighter Interceptor Wing and was credited with destroying two MIG-15 aircraft. At Nellis Air Force Base, Nevada, he served as an aerial gunnery instructor and then attended the Squadron Officers' School at the Air University, Maxwell Air Force Base, Alabama.

Following his assignment as Aide to the Dean of Faculty at the United States Air Force Academy, Aldrin flew F-100 aircraft as a flight commander with the 36th Tactical Fighter Wing at Bitburg, Germany. He attended MIT, receiving a doctorate after completing his thesis concerning guidance for manned orbital rendezvous, and was then assigned to the Gemini Target Office of the Air Force Space Systems Division, Los Angeles, California. He was later transferred to the USAF Field Office at the Manned Spacecraft Center which was responsible for integrating DOD experiments into the NASA Gemini flights.

He has logged approximately 3500 hours flying time, including 2853 hours in jet aircraft and 139 hours in helicopters. He has made several flights in the lunar landing research vehicle.

CURRENT ASSIGNMENT: Colonel Aldrin was one of the third group of astronauts named by NASA in October 1963. He has since served as backup pilot for the Gemini 9 mission.

On 11 November 1966, he and command pilot James Lovell were launched in the Gemini 12 spacecraft on a 4-day, 59-revolution flight which brought the Gemini Program to a successful close. Aldrin established a new record for extravehicular activity (EVA) by accruing slightly more than 5-1/2 hours outside the spacecraft.

Lunar Module Pilot (LMP)

NAME: Fred Wallace Haise, Jr. (Mr.)

DATE OF BIRTH: 14 November 1933

PHYSICAL DESCRIPTION: Height: 5 feet, 9-1/2 inches; weight: 150 pounds.

EDUCATION: Graduated from Biloxi High School, Biloxi, Mississippi; attended Perkinston Junior College (Association of Arts); received a Bachelor of Science degree with honors in Aeronautical Engineering from the University of Oklahoma in 1959.

ORGANIZATIONS: Member of the Society of Experimental Test Pilots, Tau Beta Pi, Sigma Gamma Tau, and Phi Theta Kappa.

SPECIAL HONORS: Recipient of the A.B. Honts Trophy as the outstanding graduate of class 64A from the Aerospace Research Pilot School in 1964; awarded the American Defense Ribbon and the Society of Experimental Test Pilots Ray E. Tenhoff Award for 1966.

EXPERIENCE: Haise was a research pilot at the NASA Flight Research Center at Edwards, California, before coming to Houston and the Manned Spacecraft Center; and from September 1959 to March 1963, he was a research pilot at the NASA Lewis Research Center in Cleveland, Ohio. During this time he authored the following papers which have been published: NASA TND, entitled "An Evaluation of the Flying Qualities of Seven General-Aviation Aircraft;" NASA TND 3380, "Use of Aircraft for Zero Gravity Environment, May 1966," SAE Business Aircraft Conference Paper, entitled "An Evaluation of General-Aviation A/C Flying Qualities," 30 March-1 April 1966; and a paper delivered at the tenth symposium of the Society of Experimental Test Pilots, entitled "A Quantitative/Qualitative Handling Qualities Evaluation of Seven General-Aviation Aircraft," 1966.

He was the Aerospace Research Pilot School's outstanding graduate of class 64A and served with the US Air Force from October 1961 to August 1962 as a tactical fighter pilot and as Chief of the 164th Standardization-Evaluation Flight of the 164th Tactical Fighter Squadron at Mansfield, Ohio. From March 1957 to September 1959, he was a fighter-interceptor pilot with the 185th Fighter Interceptor Squadron in the Oklahoma Air National Guard.

He also served as a tactics and all weather flight instructor in the US Navy Advanced Training Command at NAAS Kingsville, Texas, and was assigned as a US Marine Corps fighter pilot to VMF-533 and 114 at MCAS Cherry Point, North Carolina, from March 1954 to September 1956.

His military career began in October 1952 as a Naval Aviation Cadet at the Naval Air Station in Pensacola, Florida.

He has accumulated 5800 hours flying time, including 3000 hours in jets.

CURRENT ASSIGNMENT: Mr. Haise is one of the 19 astronauts selected by NASA in April 1966.

PROGRAM MANAGEMENT

NASA HEADQUARTERS

Office of Manned Space Flight

Manned Spacecraft Center

Marshall Space Flight Center

Kennedy Space Center

LAUNCH VEHICLE	SPACECRAFT	TRACKING AND DATA ACQUISITION
Marshall Space Flight Center The Boeing Co. (S-IC) North American Rockwell (S-II) McDonnell Douglas Corp. (S-IVB) IBM Corp. (IU)	Manned Spacecraft Center North American Rockwell (LES, CSM, SLA) Manned Spacecraft Center (LTA)	Kennedy Space Center Goddard Space Flight Center Department of Defense MSFN