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VEHICLE TEST PLAN
Apollo Mission A-202
(Spacecraft 011)
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NAS9-150

15 December 1964

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FOREWORD

This document provides the overall requirements for the testing of Apollo Spacecraft 011 during ground checkout and flight. The test flight, designated as Mission A-202, will be conducted from the Eastern Test Range of the Air Force Missile Test Center. It will be the second in a series of Block I Apollo spacecraft tests.

The test vehicle will consist of an unmanned command module, a service module, a spacecraft adapter, and a Saturn IB launch vehicle. The primary purpose of this test flight will be to evaluate the performance of the command module heat shield ablator during a high-heat-load entry and to demonstrate the compatibility and structural performance of the space vehicle.

Comments on this document should be directed to the Test Requirements Group of Apollo Development Analysis, S&ID, Downey, Extension 3622 or 3.

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ABSTRACT

This vehicle test plan contains the present objective and evaluation criteria, test vehicle configuration and mass properties data, flight constraints, flight plans, and the schedule for Mission A-202 (Spacecraft 011). In addition, the spacecraft ground checkout plans and GSE requirements are specified for Downey and Field Operations. Apollo data requirements, test reporting requirements, and general spacecraft requirements also appear in this document.

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1.0 INTRODUCTION

1.1 PURPOSE OF DOCUMENT

This document presents the current effort and plans of S&ID in support of Apollo Mission A-202 (Spacecraft 011). This vehicle test plan details the engineering requirements, objectives, evaluation criteria, vehicle checkout plans, mission tasks, data requirements, and other appropriate requirements (from assembly through data evaluation after the mission). This information is used by Apollo Engineering and Apollo Test and Operations (ATO) for mission planning and by NASA in preparation of the mission directive.

1.2 PRECEDENCE OF DOCUMENTS

The Vehicle Test Plan for Mission A-202 takes precedence over all other contractor reports and documents concerned with Spacecraft 011 testing; and in turn, it is superseded by the NASA Project Apollo Flight Mission Directive for Mission A-202.

1.3 REVISIONS

This vehicle test plan incorporates the present requirements and plans of NAA. It will be revised as required to reflect major changes in requirements and plans in support of this mission. A revision is presently planned to include mission rules, operational time lines, contingencies, and a more comprehensive flight plan.

1.4 DEFINITIONS

The definitions of terms used within this document are presented in Appendix A.

2.0 FLIGHT TEST OBJECTIVES

2.1 MISSION SUMMARY

The Saturn-Apollo trajectory profile during S-IB powered flight will approximate a normal insertion-into-orbit profile. The S-IVB powered boost profile and subsequent service propulsion subsystem (SPS) firings will be tailored to provide optimum heat-load entry conditions for the command module. After SM-CM separation, the command module will be oriented for a G&N controlled heat-load entry. Conditions at 400,000 feet altitude will be nearly 28,700 fps inertial velocity and -3.5 inertial flight path angle. Touchdown will occur in the ocean near Midway Island approximately 100 minutes after launch.

2.2 FIRST-ORDER TEST OBJECTIVES

The first-order test objectives for Mission A-202 are as follows:

1. Evaluate the thermal performance of the command module heat shield ablator during a high heat-load (30,000 to 40,000 Btu/ft^2), long duration entry

2. Demonstrate the compatibility and structural performance of CSM-Saturn IB

2.3 SECOND-ORDER TEST OBJECTIVES

The following list indicates the second-order objectives.

1. Demonstrate command module adequacy for manned reentry from low earth orbit

2. Evaluate performance of emergency detection system (EDS). (EDS will be closed-loop if Mission 201 is successful and open-loop if Mission 201 data does not verify EDS design.)

3. Demonstrate nominal mode separation of LES and boost protective cover (BPC) from the CSM, CSM from the S-IVB, and command module from the service module

4. Determine performance of guidance and navigation (G&N) subsystem, stabilization and control subsystem (SCS), environmental control subsystem (ECS) (pressure and temperature control), electrical power subsystem (EPS), command module reaction control subsystem (RCS), service module RCS, and telecommunications.

5. Evaluate G&N performance during boost and a closed-loop entry

2.4 THIRD-ORDER TEST OBJECTIVES

The third-order objectives for Mission A-202 are as follows:

1. Demonstrate multiple SPS restart (at least three burns of at least three seconds duration at ten seconds or greater intervals)

2. Evaluate SPS propellant retention device after maximum practical zero-g coast by demonstrating no-ullage SPS start and operation (at least 20 seconds burn time)

3. Demonstrate operation of the parachute recovery subsystem and recovery aids following entry

4. Determine launch and exit environment to verify ground test environmental criteria

5. Evaluate countdown operations

6. Demonstrate satisfactory operation of the HF and VHF subsystems after entry

3.0 TEST OBJECTIVE EVALUATION CRITERIA

The evaluation criteria presented in this document consists of detailed, specific requirements and data required to establish method and degree of compliance with the test objectives specified as (1) demonstrate, (2) determine, or (3) evaluate. Tables 3-1 and 3-2 present a summary of the number and types of spacecraft measurements to be made to satisfy the test objectives. The evaluation criteria for Spacecraft 011 test objectives are presented in the following paragraphs.

3.1 FIRST-ORDER OBJECTIVES

3.1.1 Command Module Heat Shield

The test objective to evaluate the command module heat shield is as follows:

> Evaluate the thermal performance of the command module heat shield ablator during a high heat load (30,000 to 40,000 Btu/ft^2), long duration entry.

3.1.1.1 Vehicle Dynamics

Aerodynamic trim and lift-to-drag ratio changes will be correlated with changes in the ablative surface. The pressure measurements will be correlated with shape changes as the ablation material erodes. The trim lift and drag of the vehicle will be determined by the relationship of the on-board accelerometers with these forces.

3.1.1.2 Temperature Control

Temperature measurements will be used to evaluate the effectiveness of the heat shield to maintain temperatures within design requirements.

3.1.1.3 Structural Performance

The data obtained from the strain gauge measurements on the aft heat shield will be used to determine the stresses experienced during entry. Recovery of the command module is essential for heat shield evaluation, since all heat shield measurements are recorded by the on-board recorders. Post-flight inspection of the heat shield will provide additional necessary information on performance.

Table 3-1. Summary of Test Objective Supporting Measurements

Test Objectives	Type of Measurement	No. of Measurements
FIRST ORDER TEST OBJECTIVES		
1. Evaluate CM heat shield ablator performance during high heat load	Calorimeter Pressure Thermocouple Char Ablation Strain	38 32 88 3 3 16
2. Demonstrate compatibility and structural performance of CSM-Saturn IB	Ground observation Ground RF reception	
SECOND ORDER TEST OBJECTIVES		
1. Demonstrate CM adequacy for manned entry from low earth orbit	See measurement summary, Table 3-2	
2. Evaluate EDS performance	Events Analog	19 4
3. Demonstration of nominal mode separation of LES and BPC from CSM, CSM from S-IVB, CM from SM	Ground cameras Tracking Events Analog	 20 4
4. Determine performance of SCS, ECS, EPS, CM-RCS, SM-RCS, telecommunication	Subsystem measurements - See measurement summary, Table 3-2	
5. Evaluate G&N performance during boost and closed-loop entry	Temperature Pressure Vibration Events Analog	3 1 3 23 30

Table 3-1. Summary of Test Objective Supporting Measurements (Cont)

Test Objectives	Type of Measurement	No. of Measurements
THIRD ORDER TEST OBJECTIVES		
1. Demonstrate multiple SPS restart	SPS chamber pressure CSM acceleration Subsystem pressure Subsystem temperature	1 2 5 16
2. Demonstrate propellant retention device by no-ullage start and operation of SPS	SPS chamber pressure CSM acceleration Subsystem pressure Subsystem temperature	1 2 5 16
3. Demonstrate operation of parachute recovery and recovery aids following entry	Events Analog	8 1
4. Determine launch and exit environment to verify ground test environmental criteria	See measurement summary, Table 3-2	
5. Evaluate countdown operations	Checkout instrumentation	
6. Demonstrate operation of HF and VHF subsystems after entry	See measurement summary, Table 3-2	

Table 3-2. Measurement Summary

	Strain S	Temperature T	Pressure P	Event X	Acceleration A	Vibration D	Rate R	Voltage V	Frequency F	Current C	Quantity Q	Acoustical Y	Position H	PH-Acidity Z	Radiation K	Total
Command Module																
Structure	12	3	6													21
Heat shield	16	88	32	3			38									177
Electrical		8	1					15	2	4						30
ELS			1	8												9
ECS		5	8				1				3					17
G&N		3	1	23		3		28		2						60
SCS		1		27				15					8			51
Flight technology					5	4						2				11
RCS		18	10													28
EDS (crew safety)				2												2
C&I		6		5				23	1						1	36
Service Module																
Structure	23	30	1	1												55
Electrical		9	9	9			6			3				3		39
ECS		4	4								4					12
Flight technology					3	6										9
SPS		16	6								4		4			30
RCS		12	16								2					30
EDS (crew safety)				1												1
C&I		1														1
LES				32				8								40
Spacecraft LEM Adapter																
Structure	24	12														36
Flight technology						3										3
BOOSTER																
EDS (crew safety)				12												12
TOTALS	75	216	95	123	8	16	45	89	3	9	13	2	12	3	1	710

3.1.1.4 Data Requirements

Radar, optical tracking, and photographic coverage (sequential and documentary), in addition to the telemetry and on-board recorder data, are required. The accumulation of radiometer data during entry is desirable. Heat shield temperature, flux, ablation, and char measurements will be made. Pressure measurements on the heat shield will also be made. Temperature measurements will be provided for other critical areas (command module vent duct heat sink and aft heat shield tension ties). Ablation and pressure measurements will be provided on the scimitar antennas.

3.1.2 CSM-Saturn IB Structural Performance

The test objective for the demonstration of the structural performance of the CSM-Saturn IB is as follows:

> Demonstrate the compatibility and structural performance of CSM-Saturn IB.

This objective will be satisfied if no rupturing or structure separation occurs in the CSM-Saturn IB combination during boost. In addition, the lack of structure degradation of the following items will be demonstrated: ECS (including radiators), service module RCS, cryogenic tanks and plumbing, EPS radiators, SPS, G&N, SCS, and CSM. Photographic coverage and radar tracking will be used to observe and detect structural failure.

3.2 SECOND-ORDER OBJECTIVES

3.2.1 Command Module Adequacy

The test objective for the demonstration of command module adequacy is as follows:

> Demonstrate command module adequacy for manned reentry from low earth orbit

The flight test of Spacecraft 011 is the first with complete spacecraft subsystems on board. Testing of the subsystems will support the first manned mission; therefore, a demonstration of system adequacy is required. Data will be required that will indicate whether the spacecraft subsystems are functioning or not. Emphasis will be placed on spacecraft subsystems that are necessary for crew survival during manned missions.

3.2.2 Emergency Detection Subsystem

The test objective to evaluate the emergency detection subsystem (EDS) is as follows:

> Evaluate performance of EDS. (EDS will be closed-loop if Mission 201 is successful, and open-loop if Mission 201 data does not verify EDS design.)

3.2.2.1 Open-loop

With the EDS in the open-loop mode all EDS automatic aborts are inhibited. The open-loop mode is implemented by placing the EDS auto abort enable switch in its OFF position. The open-loop performance will be evaluated by an analysis of the telemetry data for the absence of inadvertent signals on the booster-EDS interface and an analysis of the telemetry data for those switching functions and displays that are operable in the open-loop circuitry.

3.2.2.2 Closed-loop

With the EDS in the closed-loop mode, the EDS auto abort enable switch is in its AUTO or ON position, and all necessary switching functions are performed automatically within the launch vehicle EDS. The requirements during an unmanned mission are as follows:

1. All switching functions that should occur automatically within the EDS shall perform satisfactorily.

2. All display circuits and displays shall operate when required to do so.

3. If a critical situation arises, which produces a necessity within the EDS for an automatic abort command to be generated, this command shall reach the launch vehicle EDS/LES interface and operate the engine cutoff circuits in the spacecraft.

Verification of these requirements will be accomplished by analysis of the data obtained by telemetering the switching and display signals and photographing the display panels.

3.2.3 Separation Modes

The test objective for demonstrating the separation modes is as follows:

> Demonstrate nominal mode separation of the LES and BPC from the CSM, the CSM from the S-IVB, and the command module from the service module.

The satisfactory operation of the launch escape tower-command module separation will be verified by accomplishing the separation without collision or damage to the command module. Dual mission sequencer signals will be monitored and verified. Optical tracking will be provided to verify adequate separation.

Satisfactory operation of the CSM-Saturn IVB separation mechanism will be verified by launch vehicle and spacecraft separation without collision or damage to the CSM complex. Dual mission sequencer signals will be monitored and verified. The SCS pitch, yaw, and roll rates will be monitored by the PCM ground station. During SLA-CSM separation, the ability to provide SPS engine gimbal retention will be verified, as well as the clearance of the engine nozzle by the adapter. The separation shall not cause the adapter panels to fold back over the SIV-B antennas, interfering with further information transmission.

Proper operation of the CM-SM separation mechanism will be demonstrated by accomplishing separation without bumping or damage to the command module structure and ablator. The mission sequencer signals will be monitored, and their timing verified. Accelerometers in the command module will provide acceleration data during separation by means of the PCM link. The performance of the service module jettison controller, which controls the service module RCS so as to impart a predetermined ΔV to the service module for a CM-SM separation, will be verified by the successful separation.

3.2.4 Spacecraft Subsystem Performance

The test objective for determining the performance of spacecraft subsystems is as follows:

> Determine the performance of G&N subsystem, SCS, ECS (pressure and temperature control), EPS, command module RCS, service module RCS, and telecommunications.

3.2.4.1 Guidance and Navigation Subsystem

The criteria for verifying G&N performance is covered in Section 3.2.5.

3.2.4.2 Stabilization and Control Subsystem

Emphasis will be placed on the ability of the SCS (in conjunction with the G&N, service module RCS and command module RCS) to control the CSM in the various modes of operation. The performance of the SCS will be verified during ΔV mode, attitude control mode, and entry mode. In addition, the ability of the SCS to stabilize the command module at separation (CM-SM) and recovery required attitude will be verified.

3.2.4.3 Environmental Control Subsystem

The operation of the ECS under zero-g and other flight conditions will be determined. Adequate operation of the coolant loop in the heat transport medium between the ECS and the electronic cooling subsystem, and between these subsystems and the heat rejection (radiator and water evaporation) subsystem will be verified. The ability of the ECS to control the command module atmosphere and temperature will be verified. During entry, the ability of the ECS to maintain cabin temperature and pressure will be verified.

The operation of the ECS will be determined to the extent allowed by the available instrumentation. The instrumentation available for the Spacecraft 011 ECS will supply the temperature data for the inverter, battery, MIT IMU, guidance computer, radiator outlet, cabin air, suit circuit, and evaporator outlet.

3.2.4.4 Electrical Power Subsystem

The fuel cell (FC) performance will be determined under various environmental conditions. Fuel cell/spacecraft interface effects will be determined. The effect of launch and zero gravity environments on the performance of fuel cell subsystems will be determined.

The limited performance characteristics of the heater, heater controls, quantity sensors, etc., will be established. The demands of the environmental control and FC subsystems on the cryogenic storage subsystem during actual flight will be determined. The effects of launch and ascent conditions on the heat rejection characteristics of the fuel cell radiator will be determined. Determination of the radiator capability for the mission will help establish the supplemental cooling water requirements.

The ability of the entry and post-landing batteries to provide adequate power for entry and recovery will be verified.

3.2.4.5 Command Module Reaction Control Subsystem

The possibility of system overpressurization as the result of boost heating effects on the command module RCS propellant tanks will be investigated. The ability of the command module RCS to control the command module entry (in conjunction with the G&N and SCS) will be verified. Gross performance of the command module RCS will be determined. Areas of interest will include engine response and specific impulse. The capability for dumping fuel, purging the system, and propellant tank depressurization of the command module RCS will be determined.

The following data will be obtained with the available instrumentation.

Propellant tank pressure, which will allow monitoring of regulator operation and ullage pressure prior to command module RCS activation

Propellant line temperatures to determine whether propellant temperatures are within operating limits

Fuel and oxidizer line (manifold) pressure to demonstrate proper operation of the propellant feed subsystem and indicate proper engine inlet pressures

Engine outer wall and valve outlet temperatures to determine the amount of heat soak back to the engine valves from entry heating and engine operation

Helium tank temperature, which will be used to correlate with tank pressure during filling for proper loading and during flight for determining pressure fluctuations due to temperature changes or to leaks

Helium tank pressure to monitor source decay

3.2.4.6 Service Module Reaction Control Subsystem

The possibility of subsystem overpressurization as the result of boost heating effects on the service module RCS propellant tanks will be investigated. Determination of the G&N, service module RCS, and SCS attitude orientation and correction capability will be made. The aerothermodynamic effects on the service module RCS will be determined from the flight data. Compatibility of the service module RCS with the boost environment will be determined, as well as the gross performance of the service module RCS during pulsing and steady-state operations. Also to be investigated will be the response and specific impulse of the engines.

The following data will be obtained with the available instrumentation.

Propellant line temperature to determine whether propellant temperatures are within operating limits

Fuel and oxidizer line (manifolded) pressure to demonstrate the proper operation of the propellant feed system and indicate proper engine inlet pressures

Engine valve and injector head temperatures to determine the amount of heat soak-back from boost heating and engine operation

Engine package temperature to determine whether engines are within operational limits

Helium manifold pressure to monitor regulator operation

Helium tank pressure to monitor source decay

3.2.4.7 Telecommunication

3.2.4.7.1 Communications Subsystem. The performance of that portion of the communications subsystem (CI) required to support the first manned mission will be determined. Excluded are the unified S-band equipment, S-band power amplifier equipment, 2-kmc antenna switch, 2-kmc high-gain antenna equipment, HF orbital antenna, and television equipment. Performance determination will be accomplished by an analysis of the communications equipment signals, as received by MSFN or recovery forces equipment, and of the telemetered or tape recorded communications equipment parameters. Any apparent substandard performance of communications equipment may be further evaluated by post-flight operation of such equipment. The performance of the following functions will be determined.

1. VHF/FM PCM transmission to the MSFN from the CSM

2. Reception of UHF up-data from the MSFN

3. Transmission of VHF/AM simulated voice communications (400-cps tone) to the MSFN from the CSM

4. Providing a time reference for the CSM communications subsystems and for use by other CSM subsystems (This time reference is to be synchronized with the CSM G&N subsystem.)

5. Providing data processing for conditioning and encoding CSM analog and digital data into a PCM format for transmission

6. Recording of PCM, PDM, and analog data

7. Transmission of VHF/AM simulated voice communication (400-cps tone) to the recovery forces until touchdown of the command module

8. Providing for VHF and HF direction finding by recovery forces

9. Providing for pulsed C-band radar tracking by the MSFN

3.2.4.7.2 Instrumentation. The adequacy of the instrumentation will be determined with special attention on the heat shield transducers and tape recorders, and the proper range coverage of the transducers. The operation of the signal conditioner and transducers will be verified for correct frequency response for the telemetry channel used.

3.2.5 Guidance and Navigation Subsystem

The test objective for the evaluation of the G&N is as follows:

Evaluate G&N performance during boost and a closed-loop entry.

The G&N subsystem performance will be evaluated for the following modes of operation.

1. ΔV control. The G&N subsystem shall be capable of controlling the spacecraft during a thrusting phase. This operation will verify the G&N-SCS-SPS tie-in.

2. Spacecraft control. The G&N subsystem shall be capable of maintaining and controlling various spacecraft attitudes. This operation will verify the G&N-SCS-service module RCS tie-in.

3. Entry mode. The G&N subsystem shall be capable of switching to the desired entry program and monitoring the entry trajectory. This will verify the ability of the G&N to control the spacecraft during entry and the G&N-SCS-command module RCS tie-in.

4. Up-link/Down-link. The G&N subsystem shall be capable of sending information to the communications subsystem for transmission to the MSFN and receiving MSFN transmitted information from the communications subsystem. These operations will verify the G&N-telemetry-updata link tie-in.

This G&N performance will be evaluated by analyses of the data on the stability and accuracy of the spacecraft during the various modes of operation. The compatibility of the G&N with the boost environment will be evaluated. A comparison will be made of the output data of the ST 124 in the S-IVB and the G&N subsystem in the spacecraft during boost.

3.3 THIRD-ORDER OBJECTIVES

3.3.1 Service Propulsion Subsystem Start and Shutdown Performance

The test objective is as follows:

Demonstrate multiple SPS restart (at least three burns of at least three seconds duration of ten second or greater intervals).

A total of four starts and restarts of the SPS engine will be demonstrated, one restart with no ullage maneuver. Burn times will be 250 seconds, 95 seconds after no ullage restart, and two three-second burns. Data will be obtained to verify start and shutdown performance.

The data on the effectiveness of the baffles to control slosh, geysering, vortexing, and pull-through of ullage (gas) will be limited to that obtained from the propellant location sensors. Structural performance of the SPS during the boost acoustic and vibration environment will be demonstrated. The structural integrity of the SPS engine nozzle extension will not be impaired during CSM-SLA-Saturn IVB separation.

3.3.2 Service Propulsion Subsystem No-Ullage Start

The test objective is as follows:

Evaluate SPS propellant retention device after maximum practical zero-g coast by demonstrating no-ullage SPS start and operation (at least 20 seconds burn time).

SPS restart will be demonstrated, without RCS ullage, after approximately 45 minutes of zero-g coast. Burn time will be approximately 95 seconds. Investigation of zero-g effects on SPS restarts and on RCS firing requirements will be made.

3.3.3 Earth Landing Subsystem

The test objective is as follows:

Demonstrate operation of the parachute recovery subsystem and recovery aids following entry.

The gross performance of the earth landing subsystem (ELS) will be demonstrated. The adequacy of the timing for events initiation by the ELS sequencer will be verified. Entry environment effects, such as high-temperature boundary layer gases leaking into the parachute compartment, will be shown to be negligible upon full blossoming of the parachutes.

3.3.4 Launch and Exit Environments

The test objective is as follows:

Determine launch and exit environment to verify ground test environmental criteria.

The launch and exit environment will be determined from the vibration, temperature, acoustic, strain gauge, and acceleration data to be obtained throughout the launch vehicle boost phase and during SPS firing. Emphasis will be placed on determining the boost environment for the command module.

3.3.5 Countdown Operations

The test objective is as follows:

Evaluate countdown operations.

After the satisfactory completion of the spacecraft mission objectives, data will indicate the adequacy of the spacecraft countdown operations. Countdown operations will be evaluated in the areas of spacecraft subsystem checkout adequacy.

3.3.6 HF/VHF Subsystems

The test objective is as follows:

Demonstrate satisfactory operation of the HF and VHF subsystems after entry.

After the entry RF blackout period, the satisfactory operation of the HF subsystem, the VHF/AM transmitter-receiver, and the VHF/FM transmitter will be demonstrated, as permitted by line-of-sight conditions to properly equipped MSFN or recovery forces equipment. This will be accomplished by analysis of signals from this equipment as received by the MSFN or recovery forces, and of the telemetered or tape recorded equipment parameters. Any apparent substandard performance of the equipment may be further evaluated by post-flight operation of the equipment.

4.0 TEST VEHICLE CONFIGURATION

4.1 VEHICLE SUBSYSTEM PRIORITIES

The following lists indicate the spacecraft subsystems. Each subsystem is assigned a priority (P ~ Primary, S ~ Secondary) in accordance with the definitions of priorities specified in Appendix A. Each subsystem listed is a complete configuration unless specifically denoted otherwise.

4.1.1 Command Module Subsystems On-Board (Spacecraft Construction)

1.	Mission control programmer	P
2.	Guidance and navigation	P
3.	Heat shield	P
4.	Stabilization and control	P
5.	Reaction control	P
6.	Electrical power	P
7.	Mission sequencer	P
8.	Environmental control	P
9.	Displays and controls	S
10.	Emergency detection	P
11.	Recovery	P
12.	Earth landing	P
13.	Instrumentation (refer to measurement list, SID 63-511)	P
14.	Communications	P

4.1.2 Command Module Significant Subsystem Deletions

1. Crew couches, support and restraints
2. Crew provisions
3. Spare LIOH packs
4. Scientific equipment
5. Television equipment

4.1.3 Service Module Subsystems On-Board (Spacecraft Construction)

1.	Service propulsion	P
2.	Reaction control	P
3.	Environmental control	P
4.	Electrical power	P
5.	CM-SM separation	P
6.	Service module jettison controller	P

4.1.4 Service Module Significant Subsystem Deletions

1. 2-kmc high-gain directional antenna
2. HF orbital antenna

4.1.5 Spacecraft LEM Adapter P

The adapter to be used between the spacecraft and booster will be of spacecraft construction and will utilize a hinge adapter separation subsystem.

4.1.6 Launch Escape Subsystem P

1. Launch escape motor
2. Pitch control motor
3. Jettison motor
4. Separation mechanism
5. Canard
6. Ballast
7. Nose cone (Q-ball)
8. Boost protection cover

4.2 VEHICLE DESCRIPTION SUMMARY

Detailed descriptions of the spacecraft and its various subsystems are available in SID 64-1237, Vehicle Model Specification—Basic, Block I. Deletions, additions, and deviations to the basic Block I configuration are described in SID 63-703, Vehicle Contract End Item Specification, Spacecraft 011.

4.3 MASS PROPERTIES DATA

An estimated weight summary of the spacecraft is presented in Table 4-1. Other data, such as center of gravity, moments of inertia, mass data summaries, and time histories are available in SID 64-1474, Preliminary Weight and Balance Report, Spacecraft 011.

Table 4-1. Weight Summary

Component	Estimated Weight (pounds)	
Command module	11,000	
Service module	10,200	21,200
*SPS propellant		16,800
Total CSM, including propellant		38,000
Spacecraft-LEM adapter		3,900
Total injected spacecraft		41,900
Launch escape system (including boost cover)		8,200
Total launch spacecraft		50,100

*Final propellant loading data will be determined at a later date.

5.0 FLIGHT CONSTRAINTS

Flight constraints are defined as the minimum testing that must be completed satisfactorily prior to the flight of Spacecraft 011. Failure to complete these tests satisfactorily could result in a flight delay.

5.1 FLIGHT TESTS

1. Satisfactory completion of the following objective of Spacecraft 002: Demonstrate the structural integrity of the launch escape vehicle airframe structure for an abort in the power-on tumbling boundary region.

2. Satisfactory completion of the following objective of Boilerplate 13: Demonstrate the primary mode of LET jettison using the tower jettison motor.

5.2 GROUND TESTS

1. Satisfactory completion of one service module RCS and one SPS (propulsion static firing) for Spacecraft 011 mission profile (Airframe 001)

2. Satisfactory completion of acoustic tests (ATR-603B) on a Block I CSM (Airframe 007)

3. Satisfactory completion of tower water drop tests (ATR-201) (Airframe 007)

4. Satisfactory completion of one 12-hour open-sea recovery communications test (Airframe 007)

5. Satisfactory completion of static structural limit load tests on the service module (ATR 300) and CSM (ATR 600) (Airframe 004)

6. Satisfactory completion of structural limit load tests on the spacecraft-LEM adapter (ATR-404)

7. Satisfactory completion of spacecraft-LEM adapter separation tests (ATR-101)

8. Satisfactory completion of flotation tests (Boilerplate 29)

[redacted]

9. Satisfactory completion of service module panel modes and CM-SM structural modal tests (Boilerplate 27)

10. Satisfactory completion of launch vehicle structural modal testing (Boilerplate 27)

11. Satisfactory completion of landing capability and recovery operations testing (Boilerplate 25)

12. Satisfactory completion of hybrid mission evaluator test and guidance and control (G&C) ICD verification tests in G&C laboratory

13. Satisfactory completion of qualification level "0" on the following critical subsystems:

    Service propulsion
    Reaction control
    Guidance and navigation
    Stabilization and control
    Environmental control
    Electrical power
    Launch escape
    Earth landing
    Heat shield
    Mechanical separation
    Mission sequencer
    Mission programmer
    Service module jettison controller
    Communications
    Instrumentation
    Recovery aids

6.0 MISSION RULES

The Mission rules present the limitations and conditions which must be satisfied prior to launch. The launch limitations define the required operational readiness of the vehicle and range support system in support of the mission objectives. The range support systems are assigned an operational readiness requirement, and each vehicle subsystem and measurement requirement is keyed to the test objectives and assigned an operational readiness requirement in accordance with NASA nomenclature (i.e., mandatory, highly desirable, and desirable). Launch conditions define the allowable wind velocity and direction, visibility, and launch angles.

The Mission rules for Mission A-202, Spacecraft 011, will be supplied at a later date as Appendix B to this document.

7.0 SCHEDULE

The test schedule for Spacecraft 011, as established by Master Development Schedule 8, is presented in Figure 7-1.

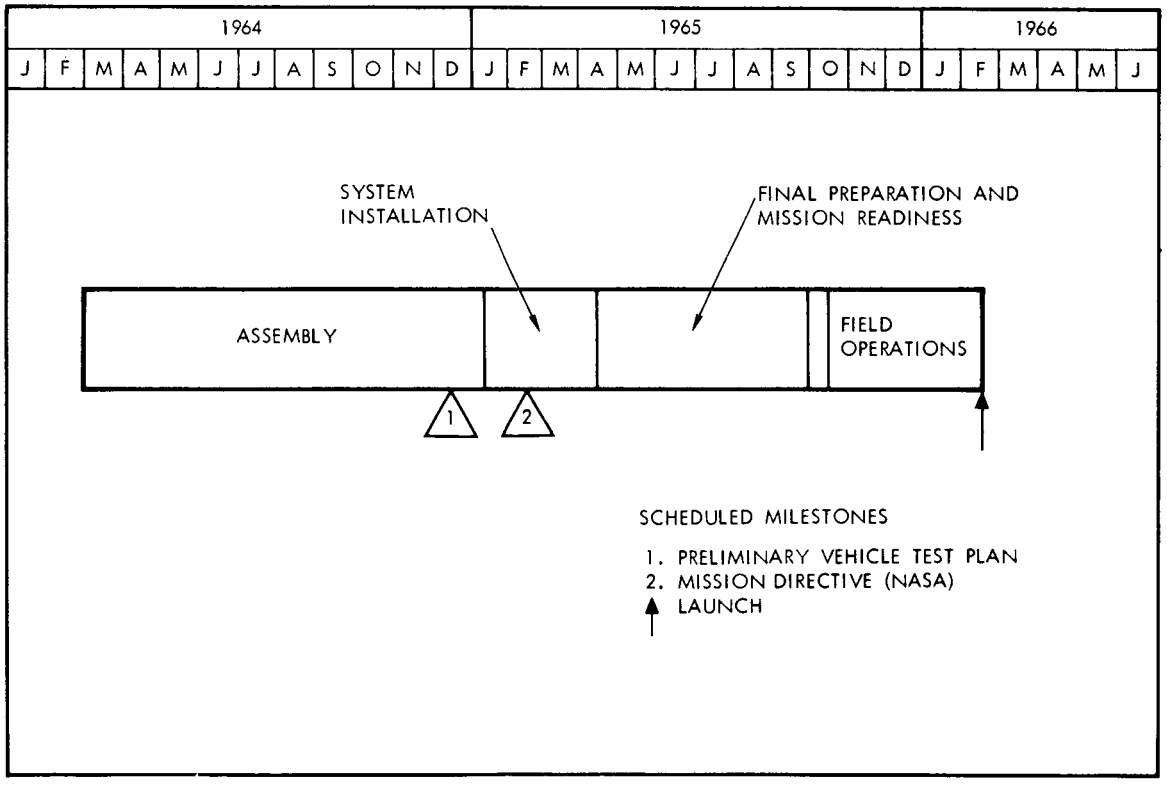


Figure 7-1. Spacecraft 011 Test Schedule

8.0 VEHICLE CHECKOUT PLAN

The basic philosophy adopted for Spacecraft 011 checkout is to perform as much as possible of the checkout at Downey in order to minimize field checkout flow times. Individual, combined, and integrated subsystem checkouts will be conducted with as many of the subsystems as can be operated safely at Downey. Field checkout will include only those tests that cannot be performed at Downey because of the hazards involved, or that must be repeated because of invalidation due to demating, mating, or shipping operations. All tests on the vehicle will be performed with acceptance checkout equipment (ACE). ACE carry-on installation and removal operations will be minimized to reduce the total checkout time span.

8.1 VEHICLE PREPARATION AND CHECKOUT REQUIREMENTS, DOWNEY

Downey vehicle preparation and checkout requirements are presented in SID 64-330-1, Vehicle Ground Operations Requirements Plan, Spacecraft 011. The referenced plan presents the required ground operations scheduled for Spacecraft 011 at Downey, including spacecraft flow, test and checkout sequence, GSE requirements, GSE configuration arrangements, and handling equipment diagrams.

Test data sheets will be prepared for each test, and they will present the following information: test location, time requirements, subsystems, prerequisities, objectives, and vehicle configuration, as well as test description, sequence of test and checkout procedures, required GSE, measurement parameters, associated process specifications, and special considerations. Facilities descriptions for Buildings 290 (including checkout stations), 260, 6, and 1 will be provided, in addition to special descriptions of ACE facilities and operations.

The referenced document (SID 64-330-1) outlines assembly, installation and continuity checks, individual and combined system checkouts, modifications and closeout, command module heat shield installation, CSM cleaning, weight and balance, mission readiness testing that consists of integrated subsystems checkout, and preparation for transportation and delivery of the spacecraft modules to Florida. The final checkout plan will be presented at a later date.

8.2 CHECKOUT OPERATIONS, DOWNEY

The spacecraft checkout operations at Downey will be conducted in the test preparation area of Building 290. A plan view of Building 290 showing

equipment locations and the flow of the spacecraft through the facility is presented in Figure 8-1. Individual subsystem, combined subsystem, and quality vibration verification testing will be performed at Station 7. After the downstream installations and modifications have been completed at Station 7, the spacecraft modules and GSE will be moved to Station 2 for integrated subsystem testing. A Downey checkout flow plan for Spacecraft 011 is shown in Figure 8-2.

8.2.1 Station 7 Activation and Checkout

Activation and checkout of Station 7 will be completed before installation of the spacecraft in the station. During this period, all GSE and equipment required to perform the subsystem, and quality vibration verification testing will be installed and checked out with the facility and ACE. The pre-test activation also includes checkout of the ground telemetry station.

8.2.2 Subsystems Tests

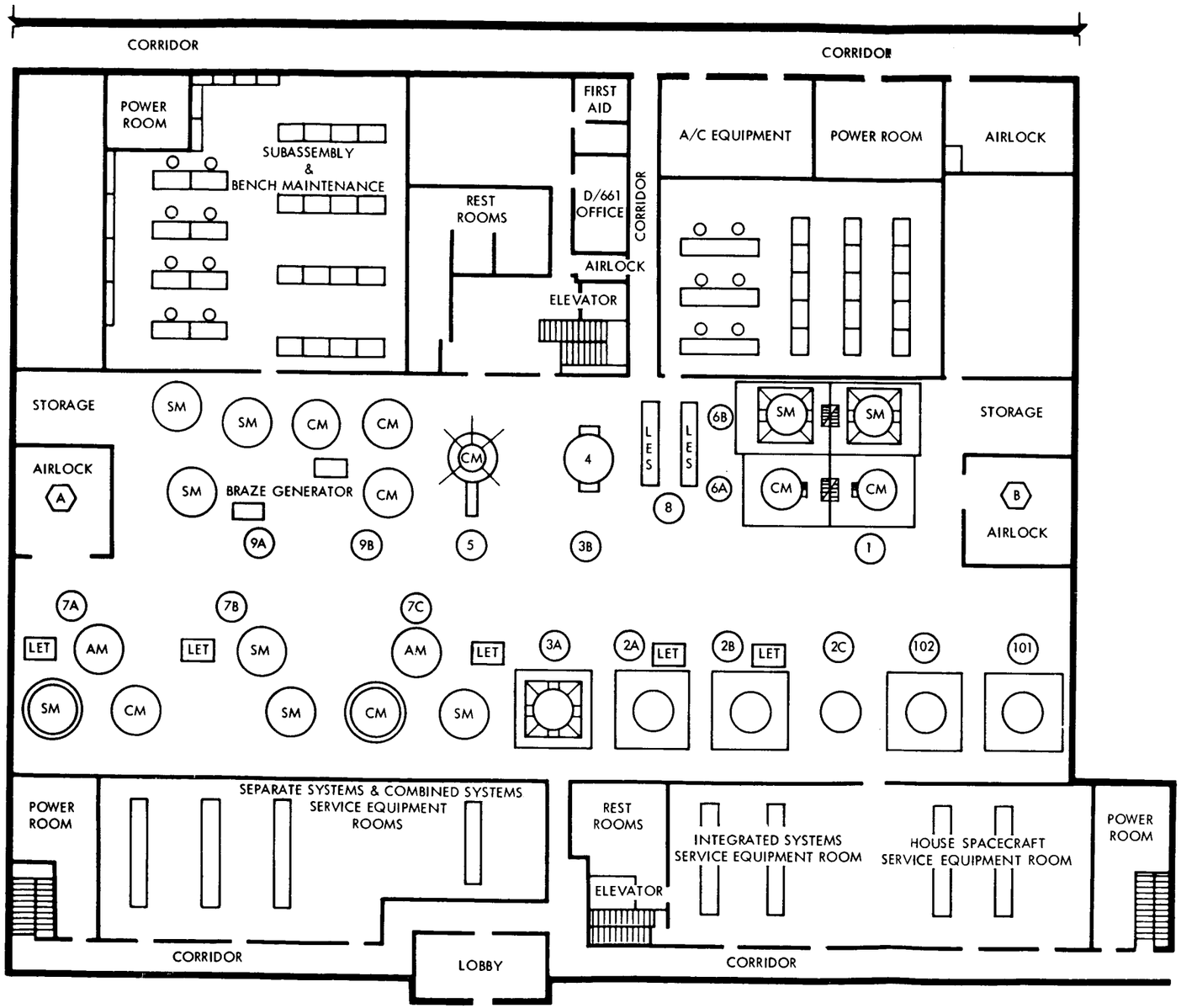
Subsystem tests will be conducted with the command module, service module, and launch escape tower installed separately in their respective test stands. Spacecraft components will be electrically mated and interconnected by means of GSE and plumbing support equipment, so that the checkout may be accomplished with complete subsystems. Before subsystem checkout, ACE-spacecraft compatibility checks will be performed. Then each spacecraft subsystem will be checked starting with the electrical power, environmental control, instrumentation and communications subsystems.

8.2.3 Quality Vibration Verification Tests

Quality vibration verification tests (QVVT) will be performed by installing thrusters on each spacecraft support stand and vibrating the vehicle while all subsystems are operating. The purpose of these tests is to establish the operational integrity of the vehicle under random vibration conditions as experienced during the launch phase. In addition, the tests will establish the electrical integrity, including the integrity of cold-solder joints that may become intermittent under these conditions. These tests are not conducted to determine airframe structural capability.

8.2.4 Downstream Installation and Modification

Certain hardware installations required for the design engineering inspection (DEI) and the weight and balance operation could create access



STATION LEGEND

- (A) NORTH AIRLOCK
- (B) SOUTH AIRLOCK
- (1) SHIPPING PREPARATION
- (2A) INTEGRATED CHECKOUT
- (2B) INTEGRATED CHECKOUT
- (2C) POLARITY CHECKER
- (3A) WEIGHT AND BALANCE, VERTICAL
- (3B) WEIGHT AND BALANCE, HORIZONTAL
- (4) CLEANING POSITIONER
- (5) HEAT SHIELD INSTALLATION
- (6A) CM MODIFICATION
- (6B) SM MODIFICATION
- (7A) INDIVIDUAL AND COMBINED SUBSYSTEMS CHECKOUT
- (7B) INDIVIDUAL AND COMBINED SUBSYSTEMS CHECKOUT
- (7C) INDIVIDUAL AND COMBINED SUBSYSTEMS CHECKOUT
- (8) LES ASSEMBLY (BUILD-UP) ALIGNMENT AND DISASSEMBLY
- (9A) SM INSTALLATION
- (9B) CM INSTALLATION
- (101) HOUSE SPACECRAFT 1
- (102) HOUSE SPACECRAFT 2

Figure 8-1. Building 290 Layout

[redacted]

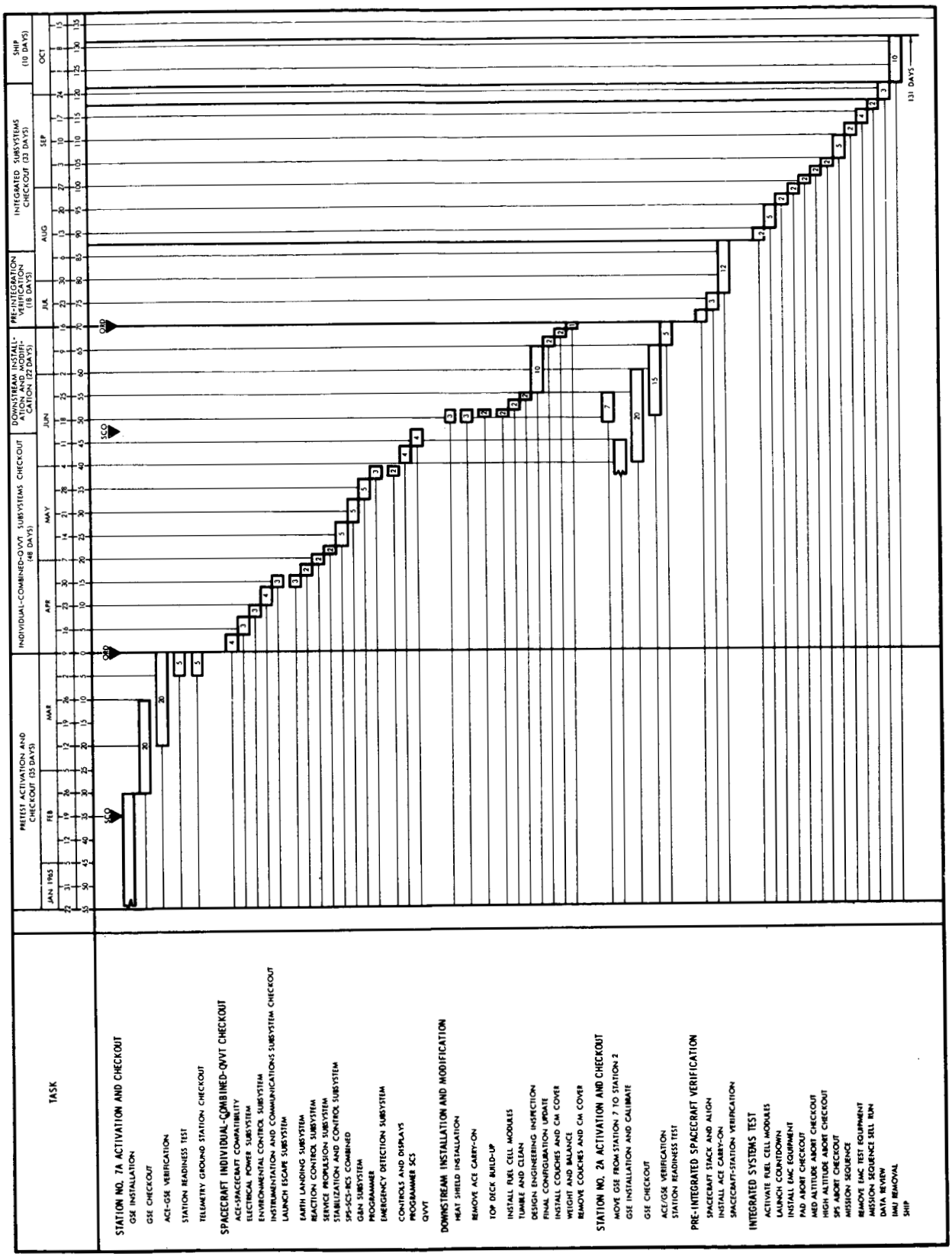


Figure 8-2. Spacecraft 011 Checkout Flow Plan, Downey

[redacted]

problems during the individual subsystem, combined subsystem, and quality vibration verification tests. Consequently, installation of the heat shield, boost protective cover, crew couches, parachutes, and mortars will be made after the quality vibration verification tests have been completed. In addition, this period is used to accomplish modifications determined to be necessary during the preceding testing and for final updating of the configuration. ACE carry-on equipment will be removed; the fuel cells will be installed; and the command module and service module will undergo a weight and balance operation to verify engineering calculations and to provide NASA with data for pre-mission calculations. After the weight and balance operation, the crew couches will be removed, and the command module will be moved to Station 2 for integrated subsystem tests.

8.2.5 Station 2 Activation and Checkout

GSE required to support integrated subsystem testing will be moved from Station 7 to Station 2 where it will be installed, calibrated, and checked out. GSE and ACE connections will be made and verified, and a station readiness test will be performed.

8.2.6 Integrated Subsystems Tests

The command module and service module will be stacked and aligned in the integrated test stand; the ACE carry-on equipment will be installed; and spacecraft-station verification checks will be performed. During this period, a complete spacecraft and GSE configuration review will be conducted to identify and verify the actual vehicle-GSE test configuration.

Integrated subsystem tests simulate the various mission modes, including the abort sequences. These mission readiness runs will simulate the total planned mission.

Fuel cells will be activated, and a launch countdown will be performed. EMC equipment will be installed to monitor EMI during all mission simulations. At this point, abort sequences (including the pad, medium- and high-altitude, and SPS aborts) will be performed. After the mission simulation has been satisfactorily completed, the EMC equipment will be removed, and the mission simulation will be rerun for NASA sell-off.

8.2.7 Shipping Preparations

After the customer readiness review (CARR) meeting with NASA, the spacecraft will be demated, and all open work items will be cleared, including MCR's, EO's, and squawks. Historical records consisting of the FAIR books, CCR's, and shortage sheets will be closed out, and the acceptance data package and DD-250 will be prepared. Items of GSE and flight hardware will be packaged, weighed, and loaded on trucks for transportation to the airport.

8.3 FIELD OPERATIONS REQUIREMENTS

The engineering checkout requirements for field operations are contained in SID 64-330-1, Vehicle Ground Operations Requirements Plan, Spacecraft 011. In addition to the information discussed in Section 8.1 of this document, the referenced plan encompasses the integrated operational requirements for Spacecraft 011 in the MILA Florida Facility from prelaunch and countdown checkouts through vehicle launch.

The document defines the requirements for receiving inspection, installation and subsystem checkouts, weight and balance operations, CM-SM mating and compatibility checks, and integrated subsystem checkout.

8.4 CHECKOUT OPERATIONS, MILA/KSC

8.4.1 Shipping

The command module, service module, and launch escape subsystem components and GSE items will be transported to KSC aboard the Aerospace Lines PG-377 and USAF C-133B aircraft. The spacecraft-LEM adapter (SLA) will be transported from Tulsa to KSC by helicopter. All spacecraft modules and GSE will be off-loaded at the KSC skid strip and transported by truck to the MILA (Facilities) for final assembly and prelaunch checkout. Other spacecraft components, including the LES rocket motors and parachutes, will be shipped directly from subcontractor facilities to KSC. The spacecraft will be routed through the various MILA and KSC facilities and checked out as shown in Figure 8-3.

8.4.2 Receiving Inspection

After arrival at MILA, the spacecraft and associated GSE will be processed through receiving inspection in separate facilities. The command module will go directly to the hypergolic facility, the service module to the static firing facility, the SLA to the O&C building, and the LES components

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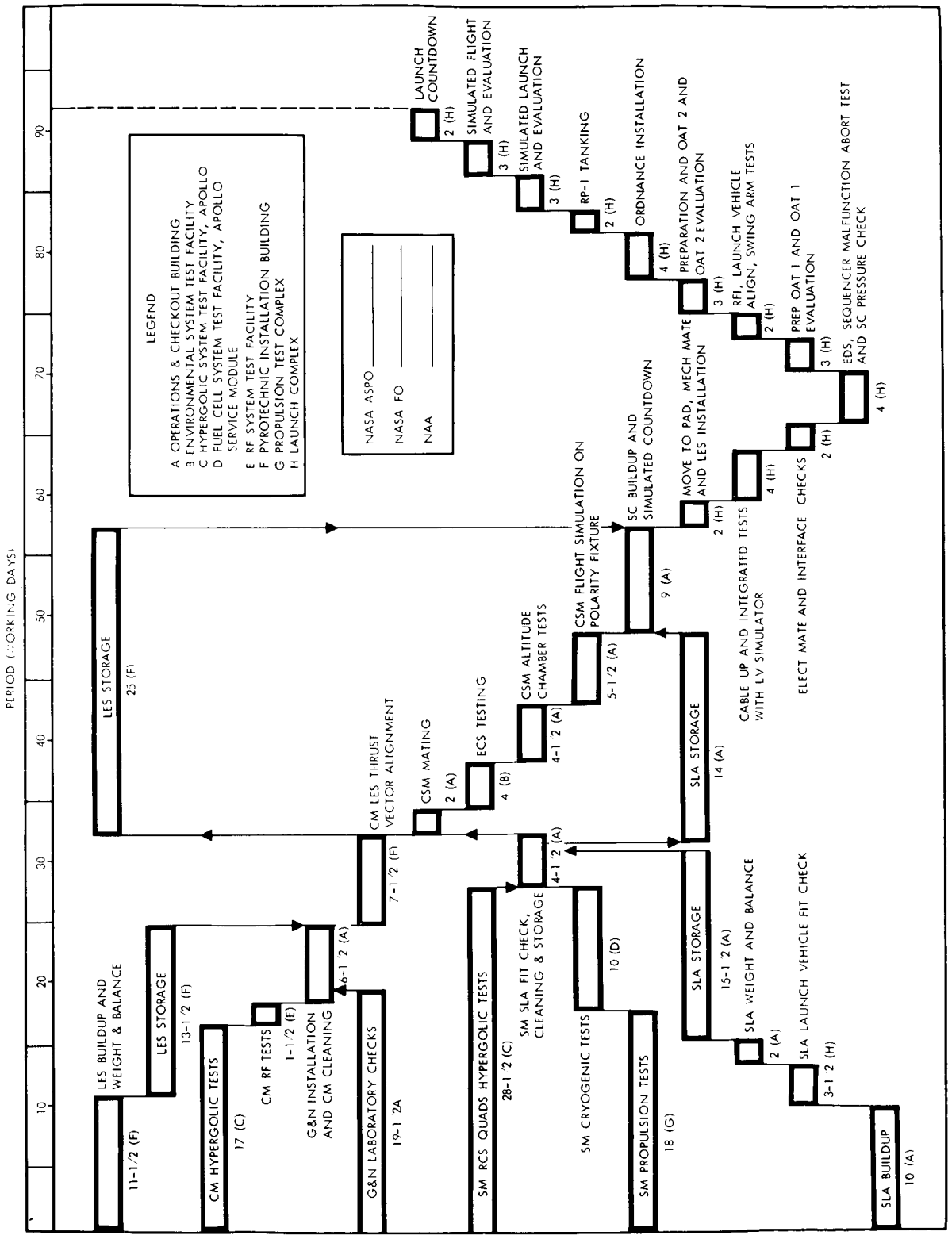

Figure 8-3. Spacecraft 011 Florida Operations Flow Plan

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to the pyrotechnic installation building. All components will be visually inspected in their respective test areas for proper configuration and for possible shipping damage. The receiving inspection will include a review of the hardware, acceptance data package, and DD-250 to verify hardware received, part shortages, and open work items, including MCR's, EO's, and squawks.

8.4.3 Command Module Checkout

8.4.3.1 Reaction Control Subsystem

The command module will be installed in the hypergolic building test cell for a functional verification of the command module RCS engines. These tests will include functional verification and leakage checks of the helium and propellant storage, distribution, and feed subsystems. Functional checks will be performed at subsystem operating pressures of valves, regulators, and other RCS components. After subsystem checkout, GSE servicing and ACE control units will be connected and validated. The RCS tests will be concluded with a cold-flow checkout of the subsystem.

8.4.3.2 RF Compatibility Tests

After completing the RCS checkout, the command module will be moved and installed in the RF systems test facility for the purpose of demonstrating the compatibility of the command module RF subsystems with each other and with the ETR ground equipment. All RF subsystems, including the telemetry transmitter, C-band radar transponder, VHF transmitter-receiver, UHF command receiver, HF transceiver, and the HF recovery beacon, will be exercised in an open-loop operating mode. The test will be conducted in two phases: The first phase will be performed with all in-flight RF subsystems in operation; the second phase will exercise only those RF subsystems involved in the recovery operation. Equipment performance will be monitored by the Apollo open-loop ground station and by the appropriate ETR ground equipment. An external GSE power source will be used to power the command module d-c electrical buses and inverters.

8.4.3.3 G&N Installation and Command Module Cleaning

The command module will be sent to the O&C building where the G&N IMU will be installed and interface compatibility checks will be conducted. After G&N installation, access panels will be removed, and the command module will be installed in the cleaning positioner for cleaning. The command module will be vacuumed in various rotational positions to remove all loose particles.

[redacted]

8.4.3.4 Command Module-LES Weight, Balance, and Thrust Vector Alignment

The command module-LES weight, balance, and thrust vector alignment will be performed to verify that weight and balance parameters are within acceptable flight limits and to perform the LE motor thrust vector alignment. The command module will be moved to the pyrotechnic installation building and installed in the weight and balance fixture. The parachutes, mortars (equipped with dummy initiators), the forward heat shield and other flight components will be installed before the horizontal and vertical weight and balance operation is performed. LES components, including the tower structure, Q-ball, electrical harnesses, and live launch escape, tower jettison, and pitch control motors, will be assembled and mated to the command module. The combined command module-LES will be weighed in the vertical position; the center of gravity will be determined; and a thrust vector alignment will be performed. This operation will be repeated if required to obtain proper tolerances. The command module-LES will be demated, and the LES will be placed in storage until it is required at the launch pad. The command module will be returned to the O&C building.

8.4.4 Service Module

8.4.4.1 SPS Firing Tests

Before the receiving inspection is started, the service module will be installed, without the RCS quads, in the static firing facility test stand in preparation for the service propulsion subsystem static firings. GSE and ACE equipment, including the command module simulator, propellant and oxidizer servicing units, and environmental equipment, will be connected, and compatibility tests will be conducted. The service module a-c and d-c buses will be energized, and instrumentation will be calibrated in preparation for SPS control verification and valve signature tests. Before SPS static firings are started, SPS low- and high-pressure leakage and functional checks, SPS nozzle gimbaling checks, and propellant-oxidizer quantity calibrations will be performed. The SPS will undergo hot firings in several operating modes. After the completion of SPS static firings, the propellant and oxidizer tanks and feed subsystems will be drained, flushed, purged, and refurbished. The static firings will be concluded with a final leak check of the entire SPS.

8.4.4.2 Service Module Quads RCS Tests

The service module RCS quads will undergo a receiving inspection before they are installed one at a time in the hypergolic facility test cell in preparation for cold-flow firings. After GSE hookup and RCS-GSE compatibility checks, RCS low- and high-pressure leakage and functional checks will be performed to verify satisfactory subsystem performance. The functional

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checks will include verification of check valves, pressure regulators, and propellant control valves. After satisfactory completion of the functional checks, the RCS will undergo a cold-flow firing. The tests will be concluded after the propellant quantity gauging subsystem has been calibrated and the RCS has been drained, flushed, purged, and functionally reverified.

8.4.4.3 Cryogenic and Fuel Cell Tests

The service module without the RCS quads will be moved to the fuel cell systems test facility and installed in the workstands in preparation for cryogenic and fuel cell tests. GSE, including servicing, cooling, and ground power units as well as the command module simulator, will be connected to the vehicle, and compatibility checks will be conducted. Electrical power and instrumentation verifications will be performed, and the cryogenic subsystem will be serviced with liquid hydrogen and liquid oxygen. After a maximum oxygen demand test has been conducted, the fuel cells will be operated with cryogenics.

8.4.4.4 SLA-Launch Vehicle Fit Check

The SLA will be received from Tulsa. After receiving inspection, it will be transported to the launch pad for a mechanical fit check with the launch vehicle instrumentation unit. This check is to provide assurance that a proper mechanical fit and electrical connector mating can be achieved without difficulty when the entire spacecraft stack is moved to the launch pad. Electrical connectors will only be mated mechanically. After the mating operation, the SLA will be removed and transported to the O&C building for mating with the service module.

8.4.4.5 Service Module-SLA Fit Check

At completion of the fuel cell tests, the service module will be moved to the O&C building; the service module RCS quads will be installed; and the service module will be lifted onto the SLA for SM-SLA mechanical fit checks. The electrical connectors will only be connected mechanically. After completion of the SM-SLA fit check, the service module will be removed from the SLA and installed on the cleaning fixture, where it will be tumbled and cleaned.

8.4.5 Spacecraft Checkout

8.4.5.1 CM-SM Mating and Interface Checks

The service module will be moved to the CSM mating stand, where the SM RCS quads and the SPS inertial simulator will be installed. Then the command module and service module will be mated.

After mating, a CM-SM alignment will be performed and verified; interface connectors will be mated; the command module airlock will be installed; a leak check at the CM-SM interface will be performed; and all GSE equipment will be connected to the vehicle. The ECS water-glycol subsystem will be serviced; power will be applied to the spacecraft buses; and all CM-SM interface electrical circuits will be verified through operation of the appropriate subsystems. The interface checks are to verify the CSM electrical and mechanical subsystems compatibility in the mated configuration. Gimbaling of the SPS engine will be performed to verify proper frequency response and step input scaling. End-to-end gain checks of the RCS and SPS thrust vector control loops will complete the interface verification.

8.4.5.2 ECS Tests

The mated CSM will be moved to the environmental system test facility and installed in the workstands. GSE and ACE, including the servicing, cooling and ground power units, will be connected, and the command module environmental hatch will be installed. After conducting command module leak checks and ECS functional control verification, the oxygen subsystem will be serviced, and an end-to-end ECS test will be conducted. At the completion of ECS testing, the cryogenic storage subsystem will be drained and purged, and the spacecraft will be returned to the O&C building.

8.4.5.3 Altitude Chamber Tests

The mated CSM will be installed in the altitude chamber, and all required GSE and associated test equipment will be connected in preparation for two altitude runs. These tests will be performed to determine cabin leakage rates and to verify satisfactory spacecraft subsystem performance under conditions approaching hard vacuum. The CSM will be in the flight configuration, except for the flight batteries, ordnance devices, the SPS bell, and the parachute subsystem. After the servicing equipment, leakage test equipment, and umbilical and pyro simulators have been connected and verified, power will be applied to the spacecraft bus, and the flight hatch will be installed. The altitude chamber pressure will be pumped down to the equivalent pressure altitude of 250,000 feet, and the cabin leakage rate will be measured by telemetry hardline. The chamber will be returned to sea level pressure, and qualitative subsystem checks will be performed on all subsystems to be operated at altitude. The altitude chamber will then be returned to a pressure altitude of 250,000 feet, and after subsystem verification a flight simulation will be performed from liftoff through touchdown. Before the final descent phase, a hard vacuum test will be conducted on the command module.

8.4.5.4 Polarity Check and Simulated Flight

The mated CSM configuration will be installed on the polarity fixture in preparation for polarity checks and a simulated flight mission. The first

test consists of an end-to-end polarity check, energizing all subsystems from the control programmer through the RCS and SPS. The polarity fixture will be operated to change the CSM attitude in pitch, yaw, and roll in order to verify proper RCS and SPS responses. Gas flow sensors using cold gas will be used for RCS verification. The second test will consist of a simulated flight with all subsystems operating, starting with umbilical ejection at T-15 seconds and continuing through touchdown.

8.4.5.5 Integrated Subsystem Tests

The CSM will be removed from the polarity fixture and installed in the integrated test stand where flight hardware, including the SPS nozzle extension, parachutes, ordnance items, initiators, and forward heat shield, are installed. Installation of the SPS nozzle will include a mechanical fit check, an alignment, and a pressure leak check. At this point, the CSM and SLA are mated by hoisting the CSM and positioning it on top of the SLA. After completion of the mating operations, the GSE, ACE, and umbilicals will be connected to the spacecraft and verified. Abort mode interface tests, including the emergency detection subsystem, will be performed before integrated subsystem tests commense. Integrated subsystem tests will consist of a simulated countdown.

8.4.5.6 Launch Pad Operation

8.4.5.6.1 Spacecraft-Launch Vehicle Mating and Interface Tests. After satisfactory completion of integrated subsystem tests in the O&C building, the spacecraft (command module, service module, and SLA) will be transported as a unit to Launch Complex 34 and mechanically mated to the launch vehicle. The LES will be installed on the spacecraft, and all LES-spacecraft electrical cables will be connected and verified. Also, all launch complex and GSE cabling will be connected to the spacecraft, and interface tests will be conducted using the launch vehicle simulator. After satisfactory completion of the facility-spacecraft interface tests, the spacecraft and launch vehicle will be electrically connected, and electrical interface checks, EDS tests, sequencer malfunction tests, and mission abort tests will be performed to verify proper pyrotechnic circuit sequencing and firing operations.

8.4.5.6.2 Overall Test. Two overall tests (OAT) will be conducted: The first is to demonstrate spacecraft-launch vehicle subsystems compatibility and to verify that no transients are generated in the pyrotechnic or arm-safe circuits during a launch vehicle mission simulation. The service structure will remain in place, and the umbilicals will not be pulled. The second OAT will verify spacecraft swing arm compatibility and will fire all

LES and separation subsystem test ordnance devices. The test will commence late in the launch countdown and will continue through the CSM/S-IVB separation sequence. Before simulating CSM/S-IVB separation, the LES motor firings, the separation sequences, and the command destruct system will be energized. RF compatibility tests will be performed before the second OAT in order to determine RF compatibility between the launch vehicle, spacecraft, and ETR. These tests are conducted first with the service structure in place and then repeated with the structure removed.

8.4.5.6.3 Simulated Countdown. The purpose of the simulated countdown is to provide a complete rehearsal of the actual launch procedures. The vehicle will be serviced with cryogenic and hypergolic fluids. All subsystems, including the fuel cells, will be operated in the same manner and sequence as during the actual launch countdown. The spacecraft umbilical and egress arm will be retracted, and all service connections will be broken. The test will terminate at T-O. After test completion, the fuel cells will be cooled and purged, and the spacecraft will be detanked and purged with nitrogen.

8.4.5.6.4 Simulated Flight. A simulated flight will be conducted as a final verification of spacecraft-launch vehicle performance during the flight phase. All subsystems will be in operation, except for fuel cells and some of the ordnance systems. The test will commence late in the countdown and continue through S-IVB ignition, and it will include the boost phase abort sequences. At this point, the count will be recycled to T-O, and a normal mission will be conducted through the earth landing operation. The final test will include a simulation of the alternate missions.

8.4.5.6.5 Launch Day Activities. The final launch countdown operation will include installation of the flight batteries and servicing of the hypergolics and cryogenics subsystems. The fuel cells will be started, and then a final subsystems verification and mission simulation will be performed on internal power. The ACE will be removed; the command module will be closed out; and the service structure will be rolled away. All subsystems will be monitored during the final countdown for satisfactory operation.

8.4.5.6.6 Launch Countdown. The launch countdown will proceed as follows:

1. Installation of flight batteries

2. Application of ground power

3. Servicing of cryogenics

4. Servicing of hypergolics and helium

5. Starting of fuel cells

6. Final verification of subsystems

   a. Electrical power
   b. Reaction control
   c. Guidance and navigation
   d. Stabilization and control
   e. Control programmer
   f. Radio frequency equipment
   g. Launch escape
   h. Separation
   i. Arm-safe

7. Simulation of normal subsystem operation

8. Removal of ACE carry-on equipment

9. Closeout of command module

10. Installation of final ordnance items

11. Simulation of normal mission

12. Removal of service structure

13. Place all subsystems in their launch configuration

14. Transfer to internal power

15. Monitoring of all subsystems for malfunctions

8.5 OPERATIONAL CHECKOUT PROCEDURES[1]

Operational checkout procedures (OCP) will be prepared for each checkout operation, including individual subsystem, combined subsystem, QVVT, and integrated subsystem verification. The OCP's will define the step-by-step procedure and sequence of operation for conducting each test. The OCP's will be used in the preparation of the ACE programming requirements documents, which supply information to General Electric for the preparation of the ACE tapes.

[1] The OCP titles and number assignments have not been made at this time. This information will be included when it becomes available.

9.0 FLIGHT PLAN

9.1 MISSION DESCRIPTION

A more comprehensive and updated flight plan will be provided at a later date after the revised Preliminary Reference Trajectory document has been received from NASA. The following is a general mission description based on available advanced information.

Spacecraft 011 will be launched by a Saturn IB launch vehicle from Complex 34. The Saturn-Apollo trajectory profile during S-IB powered flight will approximate a normal insertion into orbit profile. After S-IB shutdown, the first stage will be jettisoned, and the S-IVB's J-2 engine will be started. LES will be jettisoned a short time later. The S-IVB boost profile and subsequent SPS firings will be tailored to provide optimum heat load entry conditions for the command module. After J-2 shutdown, the CSM will separate from the S-IVB, and the SPS will be ignited and fired for approximately 250 seconds. The spacecraft will then be oriented for the next planned SPS firing. After the spacecraft has experienced approximately 50 minutes of zero-gravity coast and passed through an apogee of approximately 600 nautical miles, the SPS will be ignited. This second SPS firing of approximately 95 seconds will increase entry velocity by approximately 2800 feet per second. Two more SPS firings (each of 3-second duration) will be used strictly for SPS testing and will have little effect on the trajectory. After these tests, the CSM will be oriented for CM-SM separation, and command module and service module will separate. The command module will be oriented for a G&N-controlled heat-load entry. At 400,000 feet, inertial velocity will be approximately 28,700 feet per second, and the inertial flight path angle -3.5 degrees. The command module will touchdown in the Pacific Ocean near Midway Island. The total range and duration of the flight are expected to be approximately 16,000 nautical miles and 100 minutes.

9.2 TRAJECTORY DESIGN GROUND RULES

Trajectory design ground rules will be supplied at a later date.

9.3 MISSION TRAJECTORIES

Mission trajectory information will be supplied at a later date.

9.4 SEQUENCE OF EVENTS

The sequence of events will be supplied at a later date.

9.5 MSFN COVERAGE PLOTS

MSFN coverage plots will be supplied at a later date.

9.6 ENVELOPE

Envelope information will be supplied at a later date.

[redacted]

10.0 FLIGHT TASKS (COMMAND AND CONTROL)

10.1 OPERATIONAL TIME LINES

The detailed operational time lines will be supplied at a later date (in Appendix C). This information will include on-off times of subsystem operations based on the mission objectives, nominal consumables, and weight management.

10.2 CONTINGENCIES

Alternate mission tasks and modes of operation for rectifiable or tolerable abort modes, and remaining onboard supplies of fuel, oxygen, etc., will be specified in this document at a later date.

10.3 ABORT CRITERIA

The abort criteria will define the critical levels of subsystem performance in terms of ground and vehicle displayed parameters. This information will be supplied at a later date.

10.4 GROUND REAL-TIME DISPLAY AND SUBSYSTEM OPERATING LIMITS

The critical parameters to be displayed at ground tracking stations and monitored by flight controllers to determine and predict spacecraft subsystem performance will be specified in Appendix B at a later date.

11.0 POST-FLIGHT OPERATIONS

11.1 POST-FLIGHT OPERATIONS REQUIREMENTS

The engineering requirements for checkout and disposition of Spacecraft 011 subsystems during and after recovery will be defined in the Vehicle Ground Operations Requirements Plan, Spacecraft 011, SID 64-330-1. The referenced document will provide descriptive treatment of recovery operations and post-recovery testing and operations at the Florida Facility and at the contractor's facility. The information will be discussed as was the information in Sections 8.1 and 8.3.

The referenced document will define the major phases and detailed requirements for normal and emergency landings, prelaunch deployment to designated recovery areas, and recovery force capabilities, as well as the major phases of recovery operations, including location, retrieval, and post-retrieval operations. AMR subsystem checkout operation requirements will be defined. Safety requirements related to post-landing and recovery, radioactivity, and range safety will be described.

11.2 CHECKOUT OPERATIONS

The command module will be returned to Florida and installed in the O&C building following the recovery operation. A detailed visual inspection of the command module will be performed to ascertain the extent of damage, if any, and to determine the condition of the subsystems. After GSE and ACE equipment have been connected and verified, a complete command module electrical power-on check will be performed. In addition, a checkout will be conducted of those subsystems that malfunctioned or failed to perform according to specifications. After completing this testing, the command module will be shipped to Downey for detailed NAA and subcontractor testing.

12.0 APOLLO DATA REQUIREMENTS

The data requirements for the Spacecraft 011 mission are outlined in the following paragraphs. Detailed measurement requirements are specified in SID 63-511, Apollo Measurement Requirements, Spacecraft 011, and detailed tracking and support data requirements are specified in PRD-3200. A more detailed discussion of data requirements will be presented within this section at a later date.

12.1 ONBOARD DATA

Onboard data will be acquired during test preparations at Downey, and during field operations and flight testing at ETR. Measurement requirements are specified in SID 63-511 and also listed in Appendix B.

12.1.1 Test Preparation and Checkout Data

The data requirements for test preparation checkouts and launch site checkouts will be specified by the applicable operational test procedures.

12.1.2 Flight Test Data

The following data will be supplied to Engineering following completion of the flight test (refer to PRD 3200 for details):

1. Onboard Recorder Data. Onboard recorders are provided to record data for flight evaluation. The principal measurements to be recorded are heat shield temperature, char, flux, and ablation and command module vibration and acoustics.

2. Flight Telemetry Data. All telemetry data will be received by IMCC from just before liftoff through recovery. The measurements to be telemetered are specified in Appendix B.

12.2 EXTERNAL DATA

The following external data will be acquired. Refer to PRD 3200 and the Operations Directive (OD) for details.

12.2.1 Engineering Sequential Photographic Coverage

Visible events data coverage from launch through boost phase of the flight is required for surveillance of Apollo structural interface areas, S-IB staging and ignition, and LES tower separation.

12.2.2 Radar and Optical Metric Data

Radar and/or optical tracking is required to provide Apollo vehicle position, velocity, acceleration, attitude, and other pertinent data covered in the PRD 3200 and the OD.

12.2.3 Meteorological Data

Launch weather requirements will be dictated by the launch vehicle mission rule (see Section 6.0). Surface and upper air conditions at launch are required for post-flight data analysis.

12.3 DATA ACQUISITION SYSTEM

The data acquisition system consists of the onboard tape recorder, the telemetry system (which included the ground telemetry network), onboard and external cameras, and the radar and optical tracking system. A description of the onboard recorder and telemetry system is presented in Section 4.2. The ground system is defined in Section 9.5. A common time reference will be used for correlation of all data acquisition systems. Refer to the PRD 3200 and OD for details.

12.4 DATA DISPOSITION, FORMAT, AND PROCESSING

12.4.1 Checkout Data

The format requirements for checkout data will be provided by the applicable OTP's. These data will consist primarily of a tabulation of manually recorded readings and analog oscillograms made from the tape recordings. The detailed plan for disposition and processing will be provided at a later date and will be coordinated by System Integration of Apollo Engineering and ATO Data Engineering.

12.4.2 Flight Data

Format requirements for the presentation of the flight test data will be provided at a later date. This information will include special engineering requirements regarding the method of presentation, grouping of oscillogram traces, data computation, need date, etc. The detailed plan for data disposition and processing of all flight and range data will be coordinated by Systems Integration of Apollo Engineering and ATO Data Engineering.

12.5 DATA ANALYSIS PLANS

The data analysis plans will be supplied at a later date. The plans will include general analysis plans for each subsystem and use of data (e.g., computer programs, scanning, flight reconstruction, error analysis, etc.).

13.0 TEST REPORTS

13.1 MSC REPORTS

The analysis of the spacecraft mission and the writing of the post-launch report will be completed at MSC, Houston, Texas, by an analysis and report team (A and R team). This effort will be supported by MSC, MSFC, KSC, GSFC, and contractor personnel.

13.1.1 One-Hour Report

The MSC one-hour report will be an unclassified TWX prepared by the one-hour report group, approved by the MSC-ASPO manager (PA) and the MSC-FO manager (HA), and issued to NASA management approximately one-hour after lift-off. The report will contain a brief notification of the time and place of the launch and pertinent comments relating to the success or failure of the mission.

13.1.2 Flight Status Report (48-Hour)

The MSC flight status report will be a confidential TWX prepared by the flight status report group, approved by the PA, and issued to NASA management within approximately 48-hours after mission termination. This report will include subsystem performance analysis details and significant occurrences based on analysis of the quick-look data. If required, additional flight status reports will be written at 24-hour intervals to report additional significant facts.

13.1.3 Post-Launch Report

The post-launch report will be the primary effort for the A and R team and will be a confidential document issued by MSC-ASPO for distribution within the NASA organization and to ASPO contractors 24 calendar days after launch, so that it will be quickly available to personnel concerned with planning future missions. The post-launch report will present results from an evaluation of the data available to the team during the analysis and writing period at MSC-FO and MSC-Houston.

Post-launch report supplements will be prepared as required. These reports will present the results of additional study and analysis, will be prepared by MSC and the contractor personnel, and will be issued by MSC-ASPO for the same distribution as for the post-launch report.

13.2 NAA-ATO FIELD REPORTS

After the Spacecraft 011 flight, preliminary information will be made available for S&ID internal use at Downey in the form of one-hour, 24-hour and 5-day field reports.

13.2.1 One-Hour Report

This report will be based on information available immediately after the test operation and, in general, will include a brief summary of the countdown, launch time, brief description of the mission, and pertinent observations and comments by NAA personnel at the launch site.

13.2.2 Twenty-Four Hour Flight Report

This report will include a brief description of the countdown and recovery operations, a preliminary statement on the accomplishment of spacecraft objectives, a description of flight events and trajectory, and results of the quick-look data review.

13.2.3 Five-Day Report

This report will include a more detailed coverage of the subjects contained in the 24-hour flight report. In addition, it will contain results of post-recovery examination, a preflight history, a problem summary, and a list of the test data received and expected from the flight.

13.3 NAA ENGINEERING REPORTS

13.3.1 Checkout Tests

Engineering will prepare a summary report on each OTP/OCP performed on the spacecraft and its associated equipment. These reports will contain a general test description and list the engineering problems encountered, corrective actions taken, and recommendations for future vehicles and operations. Reports covering integrated subsystem checkout will be transmitted to RASPO-Downey.

13.3.2 Flight Tests

Engineering will perform an analysis of spacecraft data from the flight and will prepare a documented, integrated evaluation of the results. This material will be delivered to NASA-MSC for review and subsequent publication by NASA-MSC as one or more post-launch report supplements. The first supplement will be published 60 days after the flight. Additional supplements, if required, will be published at 30-day intervals for a maximum of 120 days after the flight.

14.0 RANGE AND PAD SAFETY

14.1 RANGE SAFETY

Range safety requirements will be in accordance with the General Range Safety Plan, Volume II, AFMTC Pamphlet No. 80-2. This document supplements the general safety policies and procedures prescribed in AFMTC Regulation 80-9, Range Safety Policy for Missiles and AFMTC Regulation 80-7, Airborne Flight Termination Systems.

14.2 PAD SAFETY

All provisions of the General Range Safety Plan, Volume I, AFMTC Pamphlet No. 80-2, are applicable unless specifically excepted, modified, or supplemented.

15.0 GROUND SUPPORT EQUIPMENT REQUIREMENTS

The operations on Spacecraft 011 will mark the first use of a complete complement of ground support equipment. Units not previously used in support of the first spacecraft (Spacecraft 009) are those required for the G&N subsystem, fuel cells, expanded ECS capability, and the mission control programmer. Details on the GSE required for Spacecraft 011 for handling, servicing, checkout, and related functions are available in SID 62-417, GSE Planning and Requirements List. The referenced document identifies each unit of GSE, shows the test site and vehicle each unit of GSE will support, and the total number of units required. The four groups of GSE that are required for spacecraft support are as follows:

1. Checkout equipment: consisting of subsystems control and monitoring devices, interconnecting cables, power distribution systems, acceptance checkout equipment, etc.

2. Handling equipment: consisting of slings, dollies, workstands, etc.

3. Auxiliary equipment: consisting of protective covers, warning streamers, simulators, etc.

4. Servicing equipment: consisting of battery chargers, fluid transfer units, leak test units, etc.

15.1 DOWNEY CHECKOUT OPERATIONS

The Downey checkout operations for Spacecraft 011 will require 94 models of auxiliary and handling GSE, 13 models of servicing GSE, and 71 models of checkout GSE.

15.2 FIELD CHECKOUT OPERATIONS

The field checkout operations for Spacecraft 011 will require 140 models of auxiliary and handling GSE, 52 models of servicing GSE, and 138 models of checkout GSE.

16.0 GENERAL REQUIREMENTS

16.1 STRUCTURAL DESIGN CRITERIA

The structural design of Spacecraft 011 is in conformance with the structural design criteria specified in Apollo Requirements Manual, ARM-6 (SID 64-183), Structural Loads and Criteria, and MC 999-0051A, Apollo Environmental Design and Test Requirements. ARM-6 presents the structural loads that are consistent with the structural design criteria regulating the detail design of the Apollo spacecraft. Load data are shown for the command module, service module, adapter, and launch escape subsystem. In addition, structural criteria for use in the Apollo GSE design are contained within the document. MC 999-0051A gives the environmental criteria used, in conjunction with ARM-6, in the design of the spacecraft and GSE equipment.

16.2 ENVIRONMENTAL REQUIREMENTS

The Spacecraft 011 design is in conformance with the environmental requirements specified in MC 999-0051A, Apollo Environmental Design and Test Requirements. The environmental criteria upon which the spacecraft and ground support equipment design have been based are specified in the referenced document. The document categorizes all the equipment on the basis of common environmental requirements, which are essentially determined by performance location. The equipment experience with the various environments is defined in terms of time; i.e., these environments are related to successive operational phases—from ground transport and storage to final crew recovery.

16.3 HANDLING AND TRANSPORT REQUIREMENTS

The requirements for packaging and transportation of the spacecraft are contained in DIM-15 (SID 64-212), Packaging and Transportation. The reference document outlines the plans for preservation and packaging, transportation, and storage of the spacecraft. Preservation of each spacecraft subsystem is outlined, in addition to overall spacecraft packaging and preservation plans. Packaging and preservation equipment requirements are listed, and the transport methods for the spacecraft are outlined.

16.4 ELECTROMAGNETIC COMPATIBILITY REQUIREMENTS

The electromagnetic compatibility requirements are presented in ARM-12 (SID 64-189), Electromagnetic Compatibility. This document

contains electromagnetic interference, electrical bonding, and wiring requirements. Electromagnetic interference establishes the design and test requirements for the spacecraft to ensure that the subsystems will be electrically compatible with its operational environment and associated GSE; the bonding section establishes the design requirements for electrical bonding for the Apollo spacecraft; and the wiring section establishes the procedures used for the control of electromagnetic interference in the equipment interconnecting wiring. Installation procedures and requirements outlined are applicable to all electronic, electrical, and electromechanical equipment installed in the Apollo spacecraft and GSE that are electrically interconnected.

16.5 THERMAL ANALYSIS DATA

The thermal analysis data for the Apollo spacecraft are contained in ARM 2-5 (SID 64-170), Thermal Analysis Data Manual. The referenced document contains the thermal analysis data on the spacecraft structure and components for various boost, space flight, and entry environmental conditions. Predicted temperature and heat flux throughout the spacecraft and ablation rates of the command module heat shield are also presented for various missions and conditions.

16.6 AERODYNAMIC DATA

The aerodynamic data for the spacecraft are contained in ARM 2-1 (SID 64-174), Aerodynamic Data Manual. The referenced document presents the aerodynamic design information for the Apollo spacecraft, and aerodynamic force, moment, and damping data for all Apollo components and configurations during the atmospheric portion of any of the Apollo missions. The data are presented for appropriate angle-of-attack, roll angle, and Mach No. ranges as determined by mission requirements and for each mission phase, such as boost, abort, or entry and recovery.

REFERENCES

SID 62-417	GSE Planning and Requirements
SID 63-313	Apollo Command and Service Module System Specification (Block I)
SID 63-511	Apollo Measurement Requirements, Spacecraft 011
SID 63-703	Vehicle Contract End Item Specification, Spacecraft 011
SID 64-170	(ARM 2-5) Thermal Analysis Data Manual
SID 64-174	(ARM 2-1) Aerodynamics Data Manual
SID 64-183	(ARM-6) Structural Loads and Criteria
SID 64-189	(ARM-12) Electromagnetic Compatibility (To be published)
SID 64-212	(DIM-15) Packaging and Transportation
SID 64-330-1	Vehicle Ground Operations Requirements Plan, Spacecraft 011
SID 64-1237	Vehicle Model Specification—Basic, Block I
MC 999-0051A	Apollo Environmental Design and Test Requirements
PRD-3200	Program Requirements Document
AFMTC Pamphlet 80-2	General Range Safety Plan, Volumes I and II
AFMTC Regulation 80-7	Airborne Flight Termination Systems
AFMTC Regulation 80-9	Range Safety Policy for Missiles

APPENDIX A

DEFINITIONS OF TERMS

This appendix presents the definitions of terms used throughout this document.

TEST OBJECTIVES

First-Order Test Objectives

First-order test objectives comprise the main purpose for conducting the mission. The mission profile will be tailored to optimize the accomplishment of first-order test objectives.

Second-Order Test Objectives

Second-order test objectives consist of those specific tests which constrain (e.g., must be completed satisfactorily prior to) succeeding flights and of flight proof tests of spacecraft equipment. The minimum spacecraft subsystem and instrumentation configuration required for the mission is established by the first- and second-order test objectives.

Third-Order Test Objectives

Third-Order test objectives are those which support or enhance succeeding Apollo flights, those which supply supplementary data for overall spacecraft evaluation, and those scientific experiments assigned to the mission.

OBJECTIVE TERMINOLOGY

The definitions of terms used in conjunction with the flight test objectives in this document are as follows:

1. Demonstrate. Denotes the occurrence of an action or an event during a test. The accomplishment of an objective of this type requires a qualitative answer. The answer will be derived through the relation of this action or event to some other known information or occurrence. This category of objective implies that a minimum

of airborne instrumentation will be required and/or that the information will be obtained external to the spacecraft.

2. Verify. Denotes quantitative demonstration of safe functioning, achievement of minimum performance, and operational suitability of equipment. It also implies a quantitative investigation of overall operation, which indicates quantitatively that the system does in fact operate as designed, but does not imply that the instrumentation should necessarily be adequate to allow performance deficiencies to be isolated.

3. Determine. Denotes the measurement of performance of any system or subsystem. This category implies a quantitative investigation of overall operation, which includes, generally, instrumentation for measuring basic inputs and outputs of the system or subsystem. The information obtained should indicate to what extent the system is operating as designed. The instrumentation should allow performance deficiencies to be isolated to either the system or to the system inputs.

4. Evaluate. Denotes the measurement of performance of any system or subsystem as well as the performance and/or interaction of its components that are under investigation. The accomplishment of objectives of this type requires quantitative data on the performance of both the system or subsystem and its components. The performance levels will then be analyzed for their contribution toward performance of the system. This category will provide the most detailed information of any of these categories.

5. Obtain Data. Denotes the gathering of engineering information which is to be measured to augment the general knowledge required in the development of the overall spacecraft. This category may also be used for supplemental investigation, such as environmental studies and ground equipment studies. The degree of instrumentation is not implied by this definition.

6. Test. Denotes a specially prepared operation, or series of operations, the intent of which is to determine the resultant characteristics of a component or system when operated under specified, controlled conditions. It is characterized as a unique, developmental procedure; once performed successfully, it usually need not be repeated.

7. Proof. Denotes a test, the performance of which produces evidence that a precise design criteria, specification, or requirement is

met in a manner such that the design of the tested article(s) is considered to be confirmed.

SUBSYSTEM PRIORITIES

The spacecraft subsystems are assigned either primary or secondary priority. The priority assigned relates to the flight test objectives. During the initial test program, when incomplete subsystems are tested, this definition will apply to units and elements.

Primary Subsystems

Primary subsystems are those subsystems functionally required for the spacecraft to successfully accomplish first- and second-order test objectives. A positive indication of satisfactory operation of primary subsystems must be available prior to launch.

Secondary Subsystems

Secondary subsystems are those subsystems not functionally required for the spacecraft to complete its planned mission (first- and second-order objectives). Malfunction of secondary subsystems will not require a mandatory countdown hold or scrub.

CONSTRAINT

A constraint is the minimum mandatory testing that must be accomplished satisfactorily prior to the event being considered. A flight constraint not completed will hold the mission under consideration. Constraints are hard requirements and will not be arbitrarily removed.

EVALUATION CRITERIA

Evaluation criteria are the detailed, specific requirements and data required to establish the method and degree of compliance with the test objectives specified as (1) demonstrate, (2) determine, and (3) evaluate.

ELECTROMAGNETIC COMPATIBILITY (EMC)

EMC is the compatibility of systems or subsystems to function normally in each others presence and in the planned electromagnetic environment.

APPENDIX B

MISSION RULES

The mission rules will be provided at a later date.

APPENDIX C

OPERATIONAL TIME LINES

The operational time lines will be provided at a later date.