PRELIMINARY FOR INTERNAL REVIEW

PRELIMINARY RELIABILITY PROFILE

FOR MISSION 204A

GENERAL ELECTRIC

FN-412-D (12-50)

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FOR MISSION 204A

General Electric Company
Apollo Support Department
Houston, Texas

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ABBREVIATIONS

AMR	Atlantic Missile Range
CM	Command Module
CSM	Command and Service Module
CM-RCS	Command Module-Reaction Control Subsystem
ECS	Environmental Control Subsystem
FDAI	Flight Direction and Attitude Indicator
G&N	Guidance and Navigation
IMU	Inertial Measurement Unit
IU	Instrument Unit
L/V	Launch Vehicle
MSFN	Manned Spacecraft Flight Network
PLA	Planned Landing Area
RCS	Reaction Control Subsystem
S-IB	First Booster Stage
S-IVB	Second Booster Stage
S/C	Spacecraft
SCS	Stabilization Control Subsystem
SM	Service Module
SM-RCS	Service Module - Reaction Control Subsystem
SM-SPS	Service Module - Service Propulsion Subsystem
SPS	Service Propulsion Subsystem
S/V	Space Vehicle
TVC	Thrust Vector Control

1.0 INTRODUCTION AND PURPOSE

The primary objective in formulating and maintaining this Reliability Profile is to establish a common and logical framework from which realistic Mission Success and Crew Safety models and estimates can be made as required by the current Spacecraft Reliability Analysis Program Work Plan of Contract NASw-410. The Preliminary Reference Trajectory for Mission 204, reference 12.3, was used as the primary guideline in developing this report. This report is preliminary in nature and will be revised and expanded as dictated by modeling requirements. The next planned revision will be written when the Reference Trajectory for Mission 204 is available.

This profile document is intended to be used in conjunction with the referenced documents and items contained in those documents have not necessarily been included here.

2.0 MISSION PLAN

The Initial Mission Directive for Apollo Flight Mission 204 contains three alternative missions as follows:

Mission 204A - Manned, open-ended, but planned for 14 days duration

Mission 204B - Unmanned, high heat load, long-range lob shot

Mission 204C - Unmanned, high heat rate, short-range lob shot

Mission 204A is designated as the Primary Mission for Launch Vehicle SA-204 and Spacecraft CSM 012, and will be flown if Missions 201 and 202 are successful. Missions 204B and 204C are designated as Contingency Missions which may be flown, depending on the results obtained on Missions 201, 202, and 203.

Spacecraft CSM 012 and Launch Vehicle SA-204 will have the capability to perform each of the above alternative missions. The Spacecraft and Launch Vehicle will be configured for the manned mission at delivery to the Atlantic Missile Range. The planning will be such that either Mission 204A, 204B, or 204C can be selected five and one-half months prior to the scheduled launch of Mission 204, and not result in a delay in the launch of 204. However, if 201 is successful, a decision to delete alternate Mission 204C will be made one month after the launch of 201.

Mission 204A will be used for this preliminary reliability profile.

3.0 SPACECRAFT TEST OBJECTIVES

The test objectives indicated herein are those necessary for the successful completion of Mission 204A. The objectives are categorized along system and subsystem test requirements.

In general, the purpose of the mission is to verify Spacecraft and Crew Operations for a mission of up to 14 days duration and to determine CSM subsystems performance in earth orbital environments.

The launch vehicle will demonstrate S-IVB and Instrumentation Unit checkout in orbit; also, the mission shall demonstrate performance of crew/spacecraft /mission support facilities.

3.1 Primary Objectives

3.1.2 Verify the performance of the environmental control subsystem (ECS), apecifically:

(a) Verify ECS capability to provide a habitable environment in pressure suit and cabin circuits during suited and partially suited modes, and to provide satisfactory equipment temperature control during normal operation.

(b) Demonstrate performance of waste management section.

(c) Demonstrate performance of water separators.

3.1.2 Determine the performance of the stablization and control subsystem, specifically:

(a) Determine rate damping in all three axes to the limit cycle in minimum deadband during SCS attitude hold and observe the gravity gradient effect with the Y or Z axis perpendicular to the orbit plane for a minimum of 2 orbits with an initial spacecraft attitude offset of 5^{o}.

(b) Demonstrate the adequacy of navigational sighting capability with minimum impulse controller for all three axes.

(c) Demonstrate proportional rate commands utilizing hand controller and determine the minimum proportional rate obtainable.

(d) Evaluate thrust vector control (TVC) performance (minimum 5 seconds) with an initial gimbal position error of 1^{o} in the pitch or yaw axis introduced through the gimbal positon set dials.

(e) Demonstrate the use of the FDAI displays for both ordered single axis maneuvers and coordinated spacecraft attitude maneuvers by pilot manual control.

(f) Verify spacecraft SCS automatic control performance with one thruster quad disabled for a minimum of one orbit.

(g) Demonstrate operation of emergency direct attitude control in all three axes.

(h) Demonstrate operation of manual TVC.

(i) Demonstrate spacecraft attitude maneuvers using the Attitude Gyro Coupling Unit (AGCU) and comparing the final attitude readout to the G&N attitude.

3.1.3 Demonstrate operation of the guidance and navigation subsystem, specifically:

(a) Demonstrate in-flight IMU alignment with the G&N optics (sextant).

(b) Demonstrate orbital determination with G&N optics (scanning telescope) and computer by earth landmark tracking.

(c) Demonstrate updating of the on-board computer from the MSFN via up-data link and voice modes.

(d) Demonstrate capability of G&N to compute attitude error for display during boost monitor mode for astronaut monitoring.

(e) Demonstrate performance of G&N entry-lift vector control, thrust vector control and attitude control.

3.1.4 Verify the performance of the service propulsion subsystem, specifically:

(a) Verify 8 SPS starts with a minimum of one hour between burns to include minimum delta V ($\approx$ 1.5 sec.), restart after approximately 4.5 hours of coast, restart after SPS cold soak, restart after SPS thermal soak, and deorbit impulse.

(b) Verify operation of redundant SPS engine bi-propellant valve.

(c) Verify operation and accuracy of primary and secondary propellant guaging display.

(d) Verify closed-loop operation of Flight Stability Monitor.

3.1.5 Verify the thermal design of the CSM in accordance with established attitude criteria for a minimum of two days (except as otherwise noted) in earth orbital environment, specifically:

(a) Verify the thermal response of the heat shield.

(b) Verify SM/RCS valve and injector thermal response characteristics, with RCS thermal control active, for cold soak and heat soak.

(c) Verify CM/RCS thermal response characteristics and thermal control.

(d) Verify ECS and EPS radiator performance.

(e) Obtain data on cabin and secondary structure condensation areas and degree of condensation as a function of S/C attitude and duration.

3.1.6 Verify operation of alternate parallel SM/RCS pressurization regulators.

3.1.7 Verify the performance of the communications system, specifically:

(a) Verify S/C-MSFN S-band TLM,voice, TV, updata, and tracking compatibility and capability.

(b) Verify S-band and C-band antenna switch operation.

(c) Demonstrate capability of VHF/FM data dump.

(d) Verify S-Band and VHF antenna patterns by single plane maneuvers of roll (360°), tumble (360°), and yaw (360°) of S/C when within coverage of a ground station.

(e) Verify HF voice orbital communications capability.

(f) Verify compatibility of MSFN updata link for updating Central Timing Equipment (CTE) by VHF and S-band links.

3.1.8 Verify the performance of the electrical power subsystem (EPS), specifically:

(a) Verify operation of the fuel cells at minimum self-sustaining power.

(b) Verify fuel cell water separation during earth orbital environment and water purity.

(c) Verify automatic and manual operation of the cryogenic pressure control.

(d) Verify redundant switching functions in the EPS distribution function.

(e) Verify the non-essential buss-power supply disconnect functions.

(f) Verify operation of the battery charger.

(g) Verify operation of the low noise inverters.

3.1.9 Verify operation of the mechanical, structural, adapter, and earth landing subsystems, specifically:

(a) Verify operation of astro sextant door during orbital periods by cycling the door (unlatch-open-close-latch) as follows:

2 cycles at maximum high temperature.

2 cycles at maximum low temperature.

2 cycles of opening at maximum high temperature and closing at maximum low temperature.

(b) Verify post-landing operation of the following:

side crew access hatches

uprighting system if S/C is in stable I I (inverted) position

(c) Verify results of SLA panel deployment.

(d) Verify correct altitude display operation.

3.1.10 Demonstrate the following crew subsystem and activities:

(a) Demonstrate crew ability to perform "out of window" attitude orientation for retro and initial entry attitude.

(b) Demonstrate adequacy of displays and controls and crew compartment lighting including accessibility, readability, and handling ease.

(c) Verify the adequacy of the couch/restraint system with respect to launch and landing loads and positioning for crew control/dispaly access, and out-of-window visibility.

3.1.11 Obtain data on food provisioning for a long duration mission.

3.1.12 Demonstrate depressurized cabin operation including CSM subsystems operation and crew compatiblity and interference.

3.1.13 Determine the use of consumables for a long duration mission.

3.1.14 Verify bio-medical life instrumentation and its adequacy to monitor crew bio-medical status by comparision with preflight clinical information.

3.2 Secondary Objectives

3.2.1 Verify CSM/LEM VHF voice capability (earth located LEM VHF prototype).

3.2.2 Verify manual control of the ECS glycol evaporator.

3.2.3 Verify adequate operation of the ECS redundant valving.

3.2.4 Demonstrate manual SPS shutdown and startup on one of the first three SPS burns.

3.2.5 Determine attitude deadband excursions for wide deadband for a minimum of one orbit.

3.2.6 Demonstrate simulated translunar coast work/rest cycles to include subsystem checkout and monitoring, waste management, food preparation, and out-of-couch restraints for a minimum of one day during normal operation and verify task analysis time.

3.2.7 Obtain data (astronaut sensed) on acoustic environment.

3.2.8 Demonstrate operation of G&N optics (sextant) and computer for moon landmark/star navigation angle measurements.

3.2.9 Obtain data on sensed signs of equipment structure resonant vibration.

3.2.10 Demonstrate the following:

(a) Demonstrate simulated one-man lunar orbit and transearth coast operation for a minimum duration of one day for each function.

(b) Demonstrate simulated transearth injection activities.

3.2.11 Demonstrate backup modes of in-flight IMU alignment with the G&N optics (scanning telescope).

3.2.12 Demonstrate satisfactory operation of G&N subsystem under prevailing S/C boost, inflight, and entry manned environmental conditions.

3.2.13 Demonstrate don-doff capability (one time) of a Block I pressure garment assembly under zero g environment.

3.2.14 Demonstrate efficiency of the thermal blanket to maintain body at equilibrium temperature for a minimum of one hour.

3.2.15 Demonstrate radiation survey meter compatibility.

3.2.16 Determine the various physiological effects as applicable of launch, long duration orbital flight in a zero g CM environment, and entry by the following medical experiments:

(a) Cardiovascular reflex conditioning

(b) Bene demineralization

(c) Vestibular effects

(d) Exercise ergometer

(e) Inflight cardiac output

(f) Inflight vector cardiogiom

(g) Measurement of metabolic rate during flight.

(h) Inflight pulmonary functions.

3.2.17 Determine earth formulations by synoptic terraine photography.

4.0 MISSION DESCRIPTION

Apollo Mission SA-204 (AFRM 012) is the first manned Apollo flight and will be launched from Merritt Island Launch Complex 37B. The flight will consist of the following phases:

4.1 S-IB Boost

Vertical rise is maintained for 10 seconds after which the launch vehicle will perform a roll maneuver to achieve the programmed flight azimuth of 72 degrees from north after initially being aligned to a launch azimuth of 90 degrees from north. Simultaneously with the roll maneuver, the launch vehicle will initiate the S-IB pitch or tilt program. The tilt program is terminated at T_O+52 seconds, and a gravity tilt started. Maximum dynamic pressure is reached at approximately 80 seconds after lift-off. The four inboard engines are cut off at T_O+139.6 seconds, and the four outboard engines at T_O+146.1 seconds.

4.2 Staging and S-IV Boost

S-IB/S-IVB physical spearation occurs and the S-IVB, J-2 engine ignition occurs about 2 seconds later. The launch escape tower will be jettisoned 20 seconds after J-2 ignition. During the S-IVB burn, the L/V iterative guidance will be steering the launch vehicle to the desired 105 nautical mile orbital insertion conditions which occur at T_O+601.01 seconds.

4.3 S-IVB/CSM Transportation and Separation

During the second revolution, the CSM will separate from the pre-programmed inertially stabilized S-IVB attitude and translate in the +X direction for several hundred yards. The CSM will then orient itself to the optimum attitude for purposes of taking pictures of the resulting configuration. This maneuver will occur over the southern part of the United States during desirable lighting conditions in the second and third revolutions.

4.4 CSM Orbital

4.4.1 First Orbital Plane Change

During the second day on the 18th. revolution, about a 2 degree change in the spacecraft orbit plane inclination will be performed with the first SPS burn near 1.1 degrees north and 17.4 degrees west. Burn time is approximately 35 seconds.

4.4.2 First Orbital Transfer and Circulation

An orbital transfer from the 97-105 nautical miles initial orbit to an altitude of 130 nautical miles will be performed by a 7 second SPS burn corresponding to a Δ V of approximately 184 feet/second. The transfer coast phase will take approximately 27 minutes terminating with a second SPS burn of 3.1 seconds circularizing the orbit at 130 nautical miles.

4.4.3 Second Orbital Plane Change

During the third day of the mission on the 39th. revolution, a plane changes of about 1 degree is initiated for the purpose of verifying an SPS start after 12 hours SPS cold soak.

4.4.4 Third Orbital Plane Change

During the fourth day on the 57th. revolution, a plane change of approximately 0.8 degrees is initiated for the purpose of verifying an SPS start after approximately 12 hours SPS solar soak.

4.4.5 Second Orbital Transfer and Minimum Impulse Maneuver

During the 157th. revolution, the orbital altitude will be changed from an altitude of near circular 125 nautical miles to a slightly elliptic orbit of 105 nautical miles perigee and 125 nautical miles apogee. This will be accomplished with a 1 second SPS burn at perigee and then applying a minimum impulse of - - - pounds-second at apogee.

4.5 Deorbit

The Deorbit maneuver will be initiated approximately 13.8 days after lift-off. The SPS deorbit burn will last 18 seconds and consume pounds of propellant, resulting in a ΔV of about 559 feet/second.

4.6 Pre-Entry Sequence

Approximately 5 minutes prior to the 400,000 foot entry interface the SM is jettisoned. Prior to the jettison time, the CSM is oriented to a separation attitude of approximately 60 degrees above the local horizontal and maintained at this attitude until the separation sequences. The SM RCS - X translation jets are used to obtain a separation velocity of at least 10 feet per second. After SM jettison, the CM is oriented to approximately a 150 degrees pitch attitude to bring the heat shield forward until the entry interface is reached.

4.7 Entry

The CM will enter the atmosphere at 400,000 feet and the attitude will be controlled from that point for a guided lifting re-entry.

4.8 Impact Points

The selected landing site is near Hawaii.

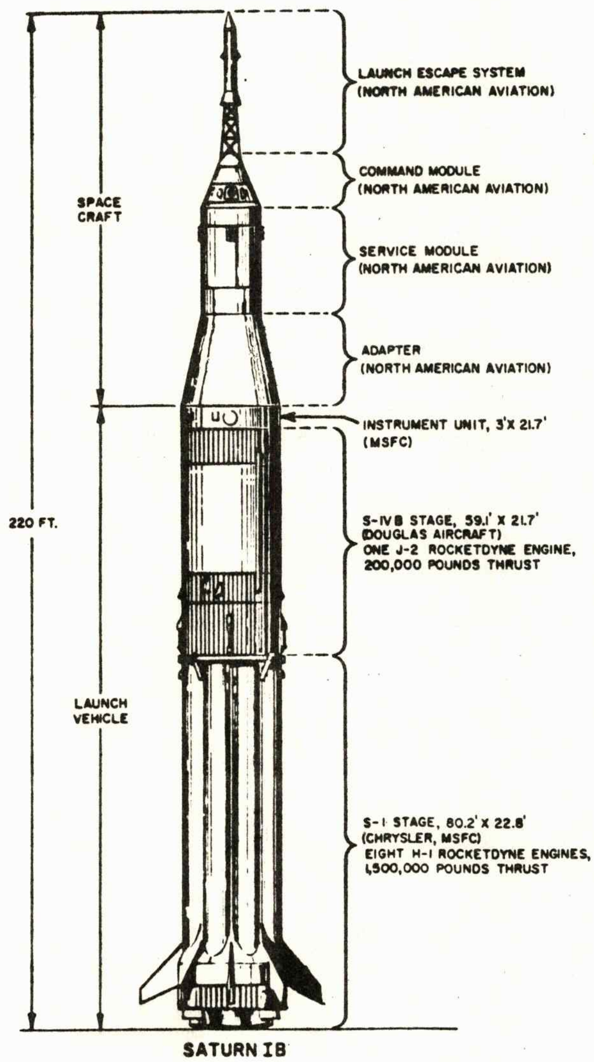
5.0 CONFIGURATION

5.1 Spacecraft Configuration

The spacecraft configuration for Mission 204A shall be per reference documents 12.4, 12.5, and 12.6. The payload requirement (spacecraft injected weight) shall be commensurate with current SA-204 launch vehicle capability and is to be achieved by adjusting SPS service module capability and is to be achieved by adjusting SPS service module propellants as required but without altering the service propulsion test objectives.

5.2 Launch Vehicle Configuration

The launch vehicle for Spacecraft 012 is a Saturn C-IB, designated SA-204, and consits of a first stage (S-IB), second stage (S-IVB), and an instrument unit.



APOLLO FLIGHT MISSION 204

6.0 PROFILE SUMMARY

After the Saturn IB launch to a near 105 nautical mile circular parking orbit (97-105), the CSM will separate from the preprogrammed inertially stabilized S-IVB attitude in the second revolution and translate in the +X direction for several hundred yards at which time the CSM will pitch around for purposes of taking pictures of the resulting configuration. In the selected preliminary reference trajectory, three orbital plane changes (change in inclination) are performed. The first inclination change is performed in the 18th. revolution near the descending node at latitude 1.1 degrees and West longitude of 17.4 degrees. The SPS burn is of 35 seconds duration and results in a change in inclination of approximately 2 degrees. The line of nodes is changed by approximately 0.0009 degrees. The second change in orbital inclination occurs in the 39th. revolution near latitude - 20.0 degrees and 18 degrees West longitude. The burn duration is 20 seconds and results in an inclination change of $\approx$ 1.0 degree and a shift of $\approx$ 1.3 degrees in the line of nodes. The third plane change is also of 20 seconds duration and results in an inclination change of 1.0 degree and changes the line of nodes by $\approx$ 1.7 degrees. This maneuver is performed at 26.0 degrees lattitude and 18 degrees West longitude. The first orbital altitude change is a combination of a 7 second off-optimum burn that changes the orbital altitude from $\approx$ 97-105 nautical miles to 97-130 nautical miles followed by a 3.1 second burn that circularizes the orbit at 130 nautical miles. The off-optimum burn of 7 seconds not only changes the orbital altitude, but results in a change of 0.0001 degrees in inclination and a change in the line of nodes of 0.06 degrees. The second orbital altitude change is initiated in the 157 th. revolution with a 1.0

second SPS burn followed by a minimum impulse of pounds second, approximately 0.23 seconds burn, which results in a slightly elliptic orbit of 105-125 nautical miles. The final or eight SPS burn of 18 seconds duration is the deorbit maneuver for landing in the vicinity of Hawaii. This burn is in the 210th. revolution or approximately 13.8 days after lift-off.

7.0 SEQUENCE OF EVENTS

Sequence of events for Apollo Mission SA-204 (AFRM 012); to be launched from Pad 37B Merritt Island launch area at a firing azimuth of 90° from North. See Section 4 for detailed description of the orbit maneuvers.

Event	Duration Sec.	Initiation Time Hr.Min.Sec.	Revolution Number
Align IMU to Launch Azimuth	-	-11:00:00	-
Final G&N Systems Check-Spacecraft closeout procedure is begun	-	-1:00:00	-
SI-B Engines Ignite	-	-0:00:01	-
T_0-Lift Off	-	0:00:00	-
Initiate Roll From 90°? 100° Firing Azimuth to 72° Launch Azimuth and Initiate Pitch Program	-	0:00:10	-
Initiate Gravity Turn	-	0:00:52	-
Maximum Dynamic Pressure	-	0:01:20	-
Terminate Gravity Turn	-	Not Avail.	-
Inboard Engines Cut-Off	-	0:02:20	-
Outboard Engines Cut-Off	-	0:02:26	-
SI-B/S-IV-B Separation SIV-B Ignition	-	0:02:28	-
LES Jettison	-	0:02:48	-
Programmed M/R Change		0:07:28	-
Orbital Insertion by MSFC Iterative Guidance Terminal Steering Near Circular 105 N.M. (97-105)	-	0:10:01	1
CSM/S-IVB Separation CSM Translation	19.3	3:10:10	2
CSM 180° Pitch (.2 Deg./Sec.)	0.9	3:11:27	3
*1st. Navigation Sighting	900.0	26:09:01	17
*1st. Inflight IMU Alignment	1800.0	26:24:01	17
*Orientation for SPS Burn	120.0	26:45:01	18

Event	Duration Sec.	Initiation Time Hr.Min.Sec.	Revolution Number
Ullage Mode	15.0	26:56:01	18
Orbital Inclination Change (1st. SPS Burn)	35	26:56:16	18
2nd. Navigation Sighting	900.00	30:57:35	20
2nd. IMU Alignment	1800.0	31:12:35	21
Orientation for SPS Burn	120.0	31:42:35	21
Ullage Mode	15.0	31:44:35	21
Transfer to 130 N.M (2nd. SPS Burn)	7.0	31:44:50	21
Ullage Mode	15.0	32:11:33	21
Circularize at 130 N.M. (3rd. SPS Burn)	3.1	32:11:48	21
1st. Cold Soak Orientation	900.0	47:18:21	31
3rd. Navigational Sighting	900.0	59:18:21	38
3rd. IMU Alignment	1800.0	59:33:21	38
Orientation for SPS	120.0	60:03:21	39
Ullage Mode	15.0	60:05:21	39
Orbital Inclination Change 1° (4th. SPS Burn)	20.0	60:05:36	39
Solar Soak Orientation	900.0	75:50:55	49
4th. Navigation Sighting	900.0	87:50:55	56
4th. IMU Alignment	1800.0	88:05:55	56
Orientation for SPS Burn	120.0	88:35:55	57
Ullage Mode	15.0	88:37:55	57
Orbital Inclination Change 1° (5th. SPS Burn)	20.0	88:38:10	57
5th. Navigation Sighting	900.0	89:40:44	57
6th. Navigation Sighting	900.0	126:20:44	81
3rd. Thermal Orientation	900.0	133:14:44	85
5th. IMU Alignment	1800.0	144:03:44	90

Event	Duration Sec.	Initiation Time Hr.Min.Sec.	Revolution Number
7th. Navigation Sighting	900.0	150:50:44	97
2nd. Cold Soak Orientation	900.0	170:49:44	109
8th. Navigation Sighting	900.0	198:38:44	126
Attitude Hold	17.1 Hr.	223:27:44	142
5th. Thermal Orientation	900.0	240:00:44	152
9th. Navigation Sighting	900.0	246:37:44	157
6th. IMU Alignment	1800.0	247:14:48	157
Orientation for SPS Burn	120.0	247:42:48	157
Ullage Mode	15.0	247:44:03	157
Underspeed to (105-130 Ellipse) (6th. SPS Burn)	.97	247:44:18	157
7th. IMU Alignment	1800.0	247:56:03	157
Orientation for SPS Burn	120.0	248:26:03	158
Ullage Mode	15.0	248:28:03	158
SPS Minimum Impulse (105-125 Ellipse) (7th. SPS Burn)	0.23	248:28:18	158
8th. IMU Alignment	1800.0	266:24:33	169
10th. Navigation Sighting	900.0	266:54:33	169
11th. Navigation Sighting	900.0	279:44:33	178
12th. Navigation Sighting	900.0	329:59:51	210
9th. IMU Alignment	1800.0	330:14:57	210
Ullage Mode	15.0	330:44:57	210
Deorbit (8th. SPS Burn)	18.0	330:45:12	210
CSM Oriented for Separation	-----	330:45:30	-
CM/SM Separation (Vel.= 10 Ft./Sec.)	-----	330:47:42	-
CM Orientation for Re-Entry	-----	330:48:00	-
Enter Atmosphere (400,000 Ft.)	-----	330:52:42	-
Chute Deployment	-----	331:04:07	-
Earth Touchdown	-----	331:10:16	-
Astronaut Recovery	-----	331:20:38	-

* All navigation sightings, IMU alignments, and thermal orientations require approximately 2 lbs. of RCS propellant using 2 quads to perform the maneuver. Approximately 3 seconds of burn time is used in a minimum impulse mode resulting in rotational rates of 2 degrees/second.

8.0 MISSION SUBPHASE BREAKOUT

The subphase breakout of a mission dictates mission model validity. The subphase is defined as an interval of time in the mission during which the system model and the reliability parameters are fixed. Therefore, a subphase break should occur in a mission whenever there is a system configuration change, a vehicle configuration description change, an abort option change, an operational mode change, or an environment change. Other subphases may be introduced for reasons of model construction or validity. Two long coast periods appear in this mission and past experience indicates that the crew safety model might be appreciably degraded if these long coast periods were left as a subphase. These long coasts were, then, arbitrarily subdivided into 24 hour increments and these subphases are defined as "time breaks". It may be found, in modeling, that these subphases are still too long and may need to be subdivided more frequently; or that the crew safety model is not greatly degraded by long subphases and this subdivision is not required. These adjustments will be made accordingly.

Subphase Designator	Events Defining Subphase	Event Time (h) (m) (s)	Subphase Time (h)(m)(s)	Abort Mode Capability	Subphase Break Assumption	Comments
	S-IB Ignition command	-0:00:06				
S-IB ingition cycle			0:00:06	LES abort, apogee less than 12K ft.	Abort mode change	
	Lift off (t=0)	0:00:00				
S-IB Vertical flight			0:00:10	LES abort, apogee less than 12K ft.	Operational change	
	Initiate roll and pitch programs	0:00:10				
S-IB Boost I			0:00:15	LES abort, apogee less than 12K ft.	Operational change	
	t + 25 seconds	0:00:25				
S-IB Boost II			0:00:21	LES abort, apogee less than 25K ft.	Abort mode change	Gravity turn is initiated at t + 52 seconds
	t + 46 seconds	0:00:46				
S-IB Boost III			0:00:49	LES abort, apogee less than 120K ft.	Abort mode change	Region of max dynamic pressures, max q occurs at t+70 secs
	t + 95 seconds	0:01:35				
S-IB Boost IV			0:00:45	LES abort, apogee greater than 120K ft.	Abort change	
	S-IB Inboard engines cutoff command	0:02:20				Out board engines cutoff 6 seconds after the inboard engine cutoff
S-IB/S-IVB Separation			0:00:08	LES abort, apogee greater than 120K ft.	Configuration change	
	S-IVB ignition	0:02:28				
S-IVB Boost I			0:00:20	LES abort, apogee greater than 120K ft.	Operational mode change	
	LES jettison	0:02:48				

EARTH ASCENT (Con't)

Subphase Designator	Events Defining Subphase	Event Time (h) (m) (s)	Subphase Time (h)(m)(s)	Abort Mode Capability	Subphase Break Assumption	Comments
	LES jettison	0:02:48				
S-IVB Boost II			0:04:40	SPS suborbital abort/abort - to - orbit	Abort mode change	
	Programmed M/R change	0:07:28				
S-IVB Boost III			0:02:33	SPS suborbital abort/abort - to - orbit	Operational mode change	
	S-IVB cutoff	0:10:01				Orbital insertion (97-105) near circular velocity= 25,582 fps

EARTH ORBIT

Subphase Designator	Events Defining Subphase	Event Time (h)(m)(s)	Subphase Time (h)(m)(s)	Abort Mode Capability	Subphase Break Assumption	Comments
	S-IVB Cutoff	0:10:01				
CSM/S-IVB Separation			3:00:09	SPS De-orbit to planned landing area; SM-RCS backup.	Abort Mode Change	
	CSM Translation	3:10:10				A 19.3 sec. RCS burn is used to accomplish A+X translation.
Earth Orbit Coast 1			23:46:06	See Above	Operational Change	CSM is oriented (180°) for observation and picture taking of S-IVB (1.30 lbs. RCS propellant used)
	SPS Ignition Command	26:56:16				*1st. navigational sighting, 1st. IMU alignment. Orientation for SPS burn. ** Ullage mode precede SPS burn.
SPS Burn #1 (orbital plane change)			0:00:35	See Above	Operational Change	
	SPS Cutoff	26:56:51				
Earth Orbit Coast 2			5:47:59	See Above	Operational Change	2nd. navigational sightings. 2nd. IMU alignment. Orientation for SPS burn.
	SPS Ignition Command	31:44:50				Ullage mode precedes SPS burn
SPS Burn #2 (Initiate transfer to 130 N.M.Orbit)			0:00:07	See Above	Operational Change	
	SPS Cutoff	31:44:57				

* RCS minimum impulse maneuvers are required for IMU alignments, navigation sightings, SPS burn orientations, and thermal orientations. The rotational rate is .2 deg/sec., requiring ≈ 2 lbs. of RCS propellant each.

** An RCS ullage burn (+X translation)is required preceding each SPS burn to settle propellants approximately 20 lbs. (15 sec.) of RCS propellant is used.

EARTH ORBIT

Subphase Designator	Events Defining Subphase	Event Time (h)(m)(s)	Subphase Time (h)(m)(s)	Abort Mode Capability	Subphase Break Assumption	Comments
	SPS Cutoff	31:44:57				
Orbit Transfer Coast			0:26:51	See Above	Operational Change	
	SPS Ignition Command	32:11:48				Ullage mode precedes SPS burn
SPS Burn #3 (Circularize at 130 N.M.)			0:00:03.1	See Above	Operational Change	
	SPS Cutoff	32:11:51				
Earth Orbit Coast 3			27:53:45	See Above	Operational Change	1st. cold soak orientation. 3rd. navigational sighting. 3rd. IMU alignment orientation for SPS.
	SPS Ignition Command	60:05:36				Ullage mode precedes SPS burn.
SPS Burn #4 (orbital plane change)			0:00:20	See Above	Operational Change	
	SPS Cutoff	60:05:56				
Earth Orbit Coast 4			28:32:14	See Above	Operational Change	1st. solar soak orientation. 4th. navigation sighting. 4th. IMU alignment. Orientation for SPS burn.
	SPS Ignition Command	88:38:10				Ullage mode precedes SPS burn
SPS Burn #5 (orbital plane change)			0:00:20	See Above	Operational Change	
	SPS Cutoff	88:38:30				

EARTH ORBIT

Subphase Designator	Events Defining Subphase	Event Time (h)(m)(s)	Subphase Time (h)(m)(s)	Abort Mode Capability	Subphase Break Assumption	Comments
	SPS Cutoff	88:38:30				
Earth Orbit Coast 5			24:00:00	See Above	Operational Change	5th. navigational sighting.
	5th. SPS Burn + 24 Hr.	112:38:30				
Earth Orbit Coast 6			24:00:00	See Above	Time Break	6th.navigational sighting. 3rd. thermal orientation
	5th. SPS Burn + 48 Hr.	136:38:30				
Earth Orbit Coast 7			24:00:00	See Above	Time Break	5th. IMU alignment. 7th. navigational sighting.
	5th. SPS Burn + 72 Hr.	160:38:30				
Earth Orbit Coast 8			24:00:00	See Above	Time Break	2nd. cold soak orientation
	5th. SPS Burn + 96 Hr.	184:38:30				
Earth Orbit Coast 9			24:00:00	See Above	Time Break	8th. navigational sighting
	5th. SPS Burn + 120 Hr.	208:38:30				
Earth Orbit Coast 10			24:00:00	See Above	Time Break	Attitude Hold (17.1 Hr.)
	5th. SPS Burn + 144 Hr.	232:38:30				

Subphase Designator	Events Defining Subphase	Event Time (h)(m)(s)	Subphase Time (h)(m)(s)	Abort Mode Capability	Subphase Break Assumption	Comments
	5th. SPS Burn + 144 Hr.	232:38:30				
Earth Orbit Coast 11			15:05:48	See Above	Operational Change	5th. thermal orientation. 9th. navigation sighting. 6th. IMU alignment Orientation for SPS burn.
	SPS Ignition Command	247:44:18				Ullage mode preceding SPS burn.
SPS Burn #6 (105-130 N.M. Ellipse)			0:00:01	See Above	Operational Change	
	SPS Cutoff	247:44:19				
Earth Orbit Coast 12			0:43:59	See Above	Operational Change	7th. IMU alignment orientation for SPS burn.
	SPS Ignition Command	248:28:18				Ullage mode precedes SPS burn
SPS Burn #7 (105-125 N.M. Ellipse)			(min.impulse) 0:00:00.23	See Above	Operational Change	
	SPS Cutoff	248:28:18				
Earth Orbit Coast 13			24:00:00	See Above	Time Break	8th. IMU alignment. 10th. navigation sighting.
	7th. SPS Burn + 24 Hr.	272:28:18				
Earth Orbit Coast 14			24:00:00	See Above	Operational Change	12th. navigational sighting. 9th. IMU alignment.
	7th. SPS Burn + 48 Hr.	296 :28:18				

EARTH ORBIT

Subphase Designator	Events Defining Subphase	Event Time (h)(m)(s)	Subphase Time (h)(m)(s)	Abort Mode Capability	Subphase Break Assumption	Comments
	7th. SPS Burn + 48 Hr.	296:28:18				
Earth Orbit Coast 15			24:00:00	See Above	Time Break	
	7th. SPS Burn + 72 Hr.	320:28:18				
Earth Orbit Coast 16			10:16:54	See Above	Operational Change	12th. navigational sighting. 9th. IMU alignment.
	SPS Ignition Command	330:45:12				Ullage mode precedes SPS burn

ENTRY

Subphase Designator	Events Defining Subphase	Event Time (h)(m)(s)	Subphase Time (h) (m) (s)	Abort Mode Capability	Subphase Break Assumption	Comments
	SPS Ignition Command	330:45:12				
SPS Burn #8 (Deorbit)			0:00:18	No Abort Capability	Operational Change	
	SPS Cutoff	330:45:30				
CM/SM Separation			0:07:12	See Above	Configuration Change	CSM oriented for separation after separation CM oriented for re-entry.
	Enter Atmosphere	330:52:42				Altitude 400,000 ft.
Atmospheric Entry			0:11:25	See Above	Operational Change	Aerodynamic heating environment changes
	Chute Deployment	331:04:07				
Chute Ride			0:06:09	See Above	Operational Change	
	Touchdown	331:10:16				

9.0 ABORT MODE GROUND RULES

9.1 General

The abort mode ground rules established for this profile are based on the assumptions that, 1) alternate mission capability will not be considered, and 2) multiple aborts will not be considered.

These ground rules are to be considered preliminary and will be used to establish the framework around which the finalized ground rules will be developed.

The detailed abort phase ground rules in the following sub-sections should be used by the modeling personnel in conjunction with the information in the Abort Mode Description and Sequence List and the Abort Logic Diagrams to construct their required abort models.

9.2 Launch Escape System (LES) Abort Phase

This phase begins when the flight crew enables the LES during the countdown and ends when the LES is jettisoned during Saturn S-IVB stage burn period.

9.2.1 Launch Vehicle Engine shutdown shall precede LES ignition for all aborts except those initiated during the first 40 seconds of flight. During the first 40 seconds, engine shutdown will be inhibited by an AMR Range Safety constraint.

9.2.2 Aborts initiated below an altitude of 120,000 feet will have an automatic sequencing of spacecraft events.

9.2.3 Aborts initiated above an altitude of 120,000 feet will require the astronauts to jettison the LES manually after escape motor thrust tail-off and to orient the spacecraft to a heat shield forward attitude.

9.3 Suborbit and/or Abort-To-Orbit Phase

This phase begins when the LES is jettisoned during the Saturn S-IVB burn period and ends when earth orbit insertion maneuver is completed.

9.3.1 Launch vehicle engine shutdown shall precede initiation of spacecraft abort action.

9.3.2 The SM Propulsion and S/C Guidance can be used for this phase.

9.3.3 The CSM abort separation procedure will utilize both a SM-RCS and SPS burn of fixed initiation time, Δ V, and altitude.

9.3.4 The SM shall be separated from the CM at some fixed increment of time from SM engine shutdown after the completion of suborbital abort and de-orbit maneuvers.

9.3.5 Two primary landing areas will be designated for launch abort landing area control.

(a) Atlantic Missile Range

(b) Indian Ocean

9.3.6 Suborbital abort will be the "primary" abort choice from LES jettison until such time that a suborbital abort would result in a landing outside the primary landing areas, and the abort-to-orbit will be considered as "secondary". Aborts at such time that a primary landing area cannot be reached will constitute abort-to-orbit as "primary" and suborbital aborts as "secondary".

(a) The CM will fly a maximum lift trajectory during entry to avoid excessive decelerations.

(b) The abort to orbit mode requires that a fixed amount of SM propellant be reserved for deorbit capability.

(c) The deorbit maneuver will be accomplished by using either the SPS or SM-RCS.

9.3.7 All time critical aborts will be suborbital.

9.4 Service Module Deorbit Abort Phase

This phase begins with the confirmation of earth orbit insertion and ends with the initiation of the nominal deorbit maneuver.

1. The Service Module SPS (primary mode) or RCS (secondary mode) shall be used for the deorbit maneuver from an earth orbit.
2. The deorbit maneuver will utilize a single propulsion burn at a fixed retro-grade attitude and for a specified ΔV.
3. The CSM separation procedure will utilize a service module RCS translation and roll maneuver.
4. When time permits, the deorbit maneuver will be delayed until that time required for entry to one of the best available preflight selected recover areas.

10.0 ABORT MODE DESCRIPTION AND SEQUENCE LIST

10.1 General

Abort mode selection and development for this initial effort was based on the following operational constraints:

10.1.1 Available propulsion capability (primary)

10.1.2 Allowable time constraint

10.1.3 Planned landing area availability

A short narrative explaining the uniqueness of any individual mode along with the sequence of primary events required to insure a successful earth return is shown in this section. As more analytical studies are completed the corresponding mode listing and associated description and sequence list will be revised.

The event list/times shown are intended to be representative and consequently many of the times are not shown. The actual times for these abort lists will depend on the abort initiation time and the resulting trajectory requirements. As the modeling activity progresses; individual abort lists with actual times will be generated according to the mission model abort requirements.

10.2 Launch Phase Abort Modes

10.2.1 Les Aborts

This phase begins at liftoff minus 6 seconds and ends when the LES is jettisoned at 20 seconds after S-IVB ignition.

The operations events which follow an abort up to LES jettison are controlled by sequential timers and barometric switches. At the time of abort the launch escape and pitch control motors are fired simultaneously. The pitch motor burns for approximately $\frac{1}{2}$ second while the escape motor thrusts for about 3.5 seconds. At 11 seconds after abort, the canards are deployed. If at 14 seconds after abort a 25,000 foot barometric switch (closed prelaunch) is still closed, the tower jettison motor separates the LES (including CM apex cover) from CM. If the switch is open, LES jettison is delayed until reaching 25,000 feet during descent when the barometric switch is closed again. Drogue parachute deployment occurs 2 seconds after LES jettison. Main parachute deployment is controlled by a timer and barometric switch similar to LES jettison. If at 12 seconds after drogue deployment a 12,000 foot barometric switch is still closed, the drogue chutes are disconnected and the main chutes are deployed. If the switch is open, the drogue chutes are retained until reaching 12,000 feet on descent when the switch is closed again. Above approximately 120,000 feet the canard subsystem is ineffective and the pilot must manually orient the CM in a heat shield forward attitude. To accomplish this function, the LES must be manually jettisoned by the pilot after launch escape motor thrust tail-off (about 8 seconds). The CM apex cover is left on. Automatic RCS rate damping begins at tower jettison. Upon reaching 25,000 feet on descent, the apex cover is jettisoned upon closing of the previously discussed barometric switch. Two seconds later the drogue parachutes are deployed. At 12,000 feet another barometric switch closed which disconnects the drogue chutes and deployes the main chutes.

ABORT MODE	ELAPSED TIME HR:MIN:SEC	SEQUENCE OF PRIMARY EVENTS
Apogee 12K Ft.	: 0	Abort initiation
	: 0	Launch escape & pitch motors ignite
	: 0.5	Pitch motor cutoff
	: 3.5	Launch escape motor cutoff
	:11	Canard deployment
	:14	LES jettison (including apex cover)
	:16	Drogue chute deployment
	:28	Main chute deployment (continue with nominal)
Apogee 12K Ft. 25K Ft.	: 0	Abort Initiation
	: 0	Launch escape & pitch motors ignite
	: 0.5	Pitch motor cutoff
	: 3.5	Launch escape motor cutoff
	:11	Canard deployment
	:14	LES jettison (including apex cover)
	:16	Drogue chute deployment
	time at 12K Ft.	Main chute deployment (continue with nominal
Apogee 25K Ft.	: 0	Abort decision
	: 0	Launch escape & pitch motor ignition
	: 0.5	Pitch motor cutoff
	: 3.5	Launch escape motor cutoff
	:11	Canard deployment
	time at 25K Ft.	LES jettison (including apex cover)
	time at 25K Ft. +2 sec.	Drogue chute deployment
	time at 12K Ft.	Main chute deployment (continue with nominal)
Apogee 120K Ft.	: 0	Abort decision
	: 0	Launch escape & pitch motor ignite
	: 0.5	Pitch motor cutoff
	: 3.5	Launch escape motor cutoff
	: 8	LES manual jettison
	: 8.1	Begin RCS rate damping
	1:40	CM orientation & rate damping complete
	time at 25K Ft.	Jettison apex cover
	time at 25K Ft. +2 sec.	Drogue chute deployment
	time at 12K Ft.	Main chute deployment (continue with nominal mission)

10.2.2 Suborbital Abort Modes

A suborbital abort may be performed any time between LES jettison and earth orbit insertion. The Service Module RCS is the only method available for separating the CSM from the launch vehicle. An initial separation of about 45 inches must be provided by the RCS before the SPS engine nozzle is clear of the adapter and can be ignited. It may be necessary to fire the SPS as soon as possible to damp out any existing pitch and yaw rates. After the separation and damping maneuver a turn around to the required thrust attitude is achieved by pulsing the Service Module RCS. Depending on the range control required the SPS can be burned for variable length of time in either a posigrade or retrograde attitude or it may not be ignited at all. During the first t seconds of flight the thrust to weight ratio is so low that the use of SM SPS for range maneuvering is ineffective. As a result of no SPS shaping and the CM flying a maximum lift trajectory, abort initiation in the first t seconds will dictate a landing within the confines of the Atlantic Missile Range (AMR). If the Service Module SPS is not used for range control after t seconds of flight, it is possible for the CM to land somewhere in the continent of Africa. To avoid this undesirable landing area, SPS thrusting will be used to land the CM in the Indian Ocean recovery area. For this abort the CM will fly a constant bank angle entry utilizing approximately half lift capability.

After a range control maneuver the CSM must then be oriented to the SM/CM separation attitude. Since the CM has no translatory thrust capability, the SM-RCS must provide the separation velocity. Once the CM is free, it is then capable of entering ballistically or with a variable lift.

ABORT MODE	ELAPSED TIME HR:MIN:SEC	SEQUENCE OF PRIMARY EVENTS
Suborbital Abort to AMR	: 0	Abort initiation
	: 0	Booster cutoff
	: 0	Activate SM RCS (translation maneuver)
	: 0	Activate separation devices (S/C-L/V)
	: 3	Begin separation
	: 4	Initiate RCS roll control
	: 9	SPS ignition
	:12	SM RCS cutoff (translation)
	:19	SPS cutoff
	To be Supplied	S/C orientation using SM RCS
		Activate separation devices
		Start translation maneuver (SM-RCS-X)
		Ignite SM RCS roll jets
		CM entry orientation (continue with nominal)
Suborbital Abort to Indian Ocean	: 0	Abort initiation
	: 0	Booster cutoff
	: 0	Activate SM RCS (translation maneuver)
	: 0	Activate separation devices (S/C-L/V)
	: 3	Begin separation
	: 4	Initiate roll control
	: 9	SPS ignition
	:12	SM RCS cutoff
	:19	SPS cutoff
	:27	Attain attitude for range thrusting
	To be Supplied	SPS ignition
		SPS cutoff
		Activate SM RCS (S/C orientation)
		Activate separation devices
		Start translation maneuver (RCS-X)
		Ignite SM RCS roll jets
		CM entry orientation (continue with nominal)

10.2.3 Abort to Orbit Modes

If it is not desired to abort to an immediate earth entry, the SPS could be used for aborts to orbit. The time range for these aborts extends from LES jettison to completion of the earth orbit insertion maneuver.

An abort of this type should be considered only when it can be accomplished with enough SPS propellant remaining to successfully deorbit.

ABORT MODES	ELAPSED TIME HR:MIN:SEC	SEQUENCE OF PRIMARY EVENTS
SPS Abort to Orbit	To be supplied	Abort initiation Booster cutoff Activate separation devices Ignite SM RCS (translation maneuver) Begin separation SM-SPS ignition SM-RCS cutoff SM-SPS cutoff (orbit attained) (see sec. following for abort from orbit).

10.3 Earth Orbit Abort Modes

The only abort possible during earth orbit is to deorbit by a retro-maneuver. The propulsion for the retro-maneuver may be provided by either the Service Module - SPS (primary mode) or Service Module - RCS (secondary mode).

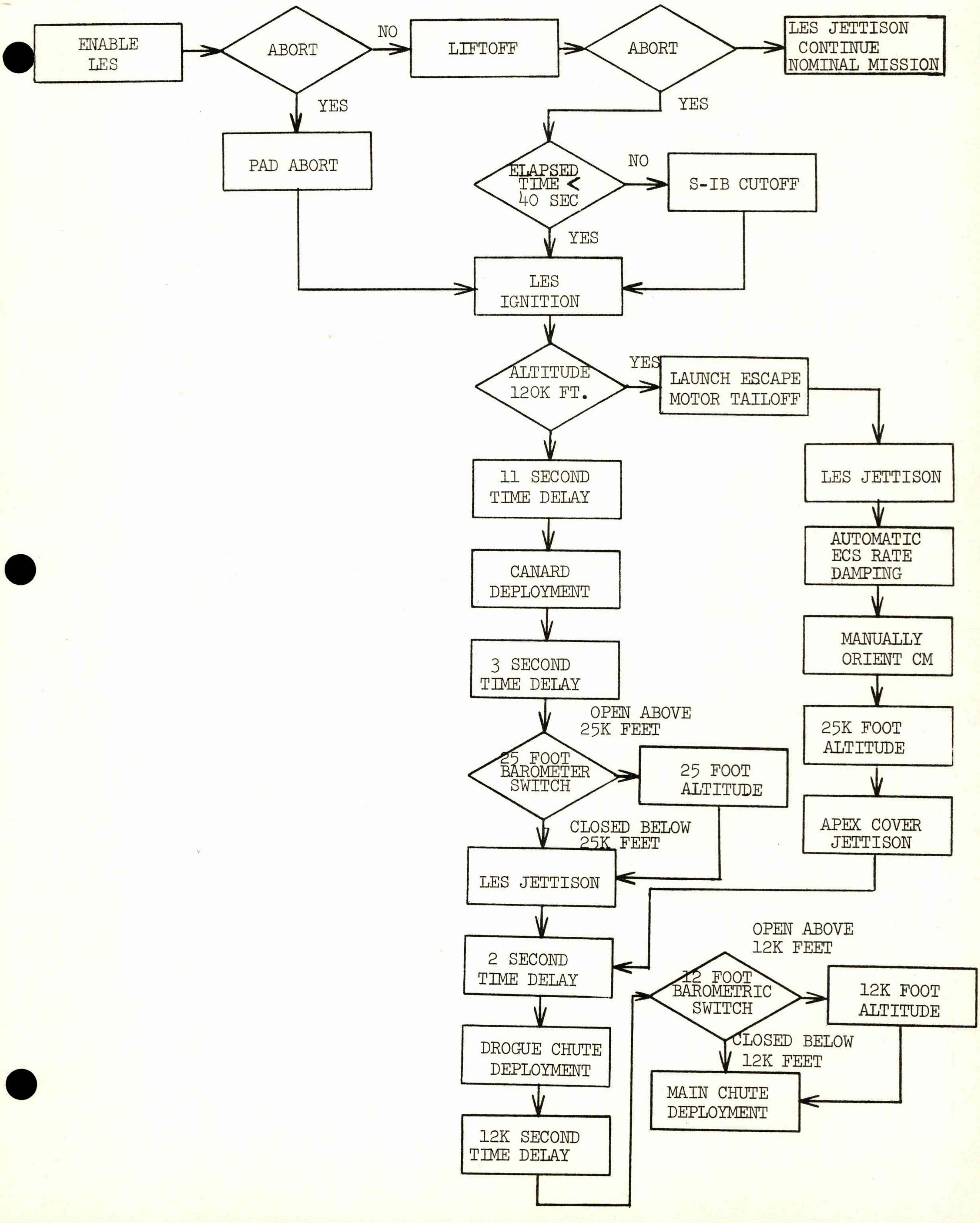
ABORT MODE	ELAPSED TIME HR:MIN:SEC	SEQUENCE OF PRIMARY EVENTS
Service Module - SPS Deorbit	To be supplied	Abort initiation Begin compulation of Retro-maneuver ΔV, firing angle and time of initiation for return to selected PLA Begin SM-RCS maneuver for retro-maneuver orientation End SM-RCS maneuver for retro-maneuver orientation SPS ignition SPS cutoff Begin SM-RCS orientation maneuver for CM-SM separation End SM-RCS orientation maneuver for CM-SM separation

(Con't)

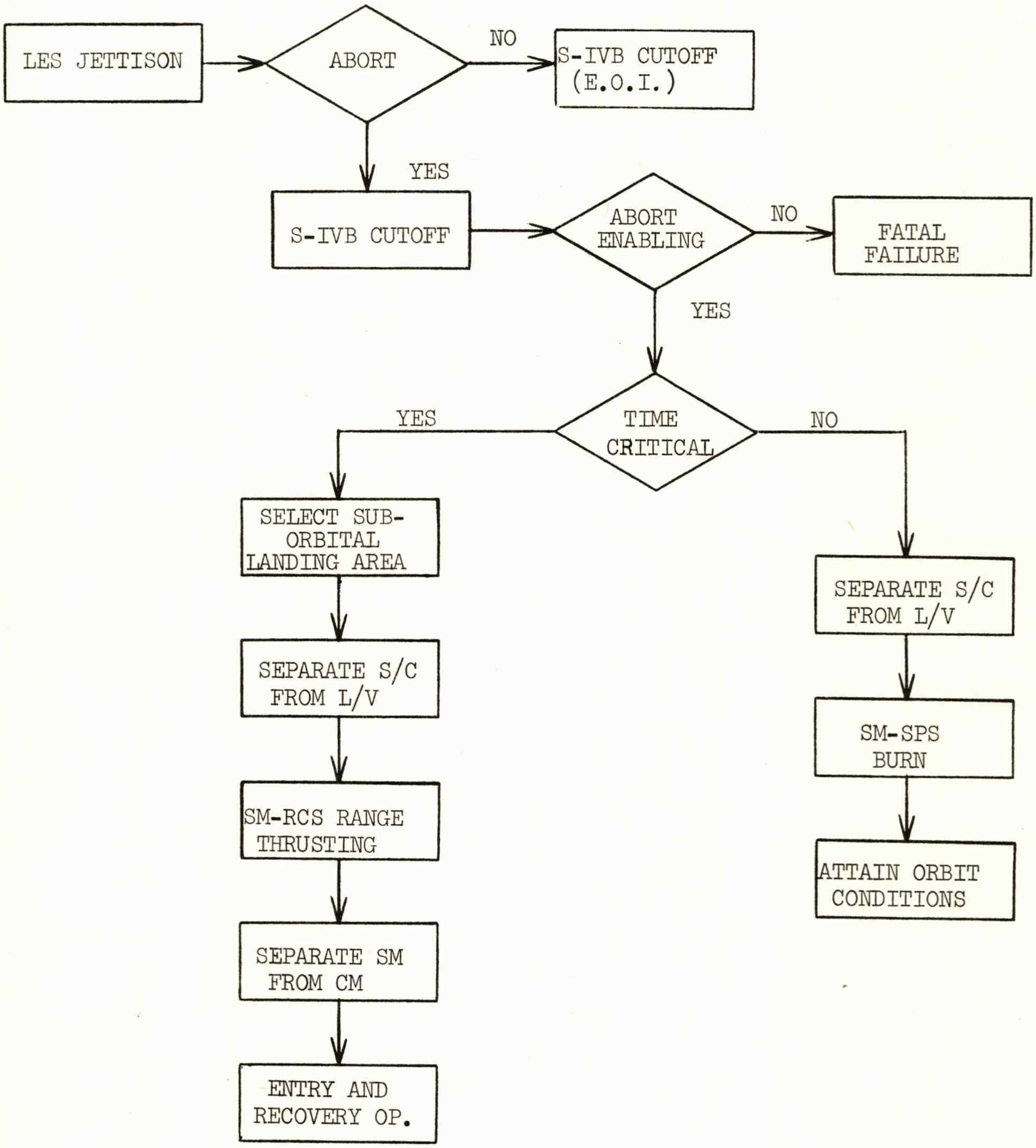
ABORT MODE	ELAPSED TIME HR:MIN:SEC	SEQUENCE OF PRIMARY EVENTS
	To be supplied	CM-SM separation devices ignited Begin SM-RCS separation maneuver Begin SM-RCS rotational (about X-axis) Begin CM-RCS re-entry orientation End CM-RCS re-entry orientation Re-entry at 400,000 feet (continue with nominal)
Service Module-RCS Deorbit	To be supplied	Abort initiation Begin computation of retro-manuever ΔV, firing angle and time of initiation for return to selected PLA Begin SM-RCS maneuver for retro-maneuver orientation End SM-RCS maneuver for retro-maneuver orientation Begin SM-RCS firing End SM-RCS firing Begin SM-RCS orientation maneuver for CM-SM separation End SM-RCS orientation maneuver for CM-SM separation CM pyro circuit armed CM-SM separation devices ignited Begin SM-RCS separation maneuver Begin SM-RCS rotational (about X-axis) CM pyro circuit secured Begin CM-RCS re-entry orientation End CM-RCS re-entry orientation Re-entry at 400,000 feet (continue with nominal)

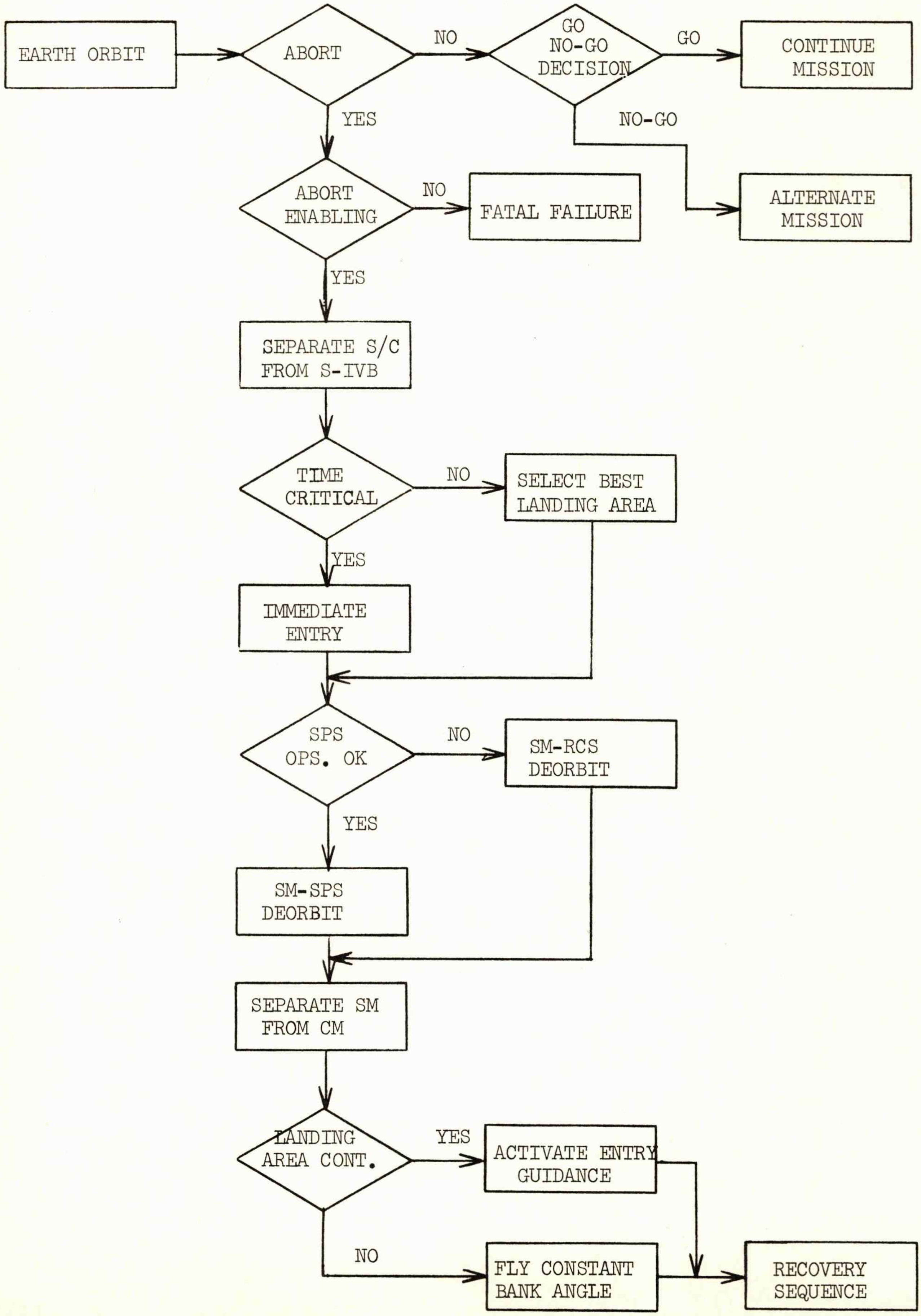
11.0 ABORT LOGIC DIAGRAMS

The primary purpose in developing the phase oriented logic charts shown on the following pages is to show, given an abort enabling failure, the functional relationships that exist between the systems and the corresponding abort options. Because of time limitations and the preliminary status of this report, all interactions of the major subsystems could not be considered, the propulsion system was, therefore, selected as being the most useful and representative at this time. Future revisions of this report will show on a mission level the logic involved to complete either a nominal or an aborted mission.


ENABLE LES
ABORT
NO
YES
LIFTOFF
ABORT
YES
LES JETTISON CONTINUE NOMINAL MISSION
PAD ABORT
ELAPSED TIME < 40 SEC
NO
S-IB CUTOFF
YES
LES IGNITION
ALTITUDE 120K FT.
YES
LAUNCH ESCAPE MOTOR TAILOFF
11 SECOND TIME DELAY
LES JETTISON
CANARD DEPLOYMENT
AUTOMATIC ECS RATE DAMPING
3 SECOND TIME DELAY
MANUALLY ORIENT CM
OPEN ABOVE 25K FEET
25 FOOT BAROMETER SWITCH
25 FOOT ALTITUDE
25K FOOT ALTITUDE
CLOSED BELOW 25K FEET
APEX COVER JETTISON
LES JETTISON
2 SECOND TIME DELAY
OPEN ABOVE 12K FEET
12 FOOT BAROMETRIC SWITCH
12K FOOT ALTITUDE
CLOSED BELOW 12K FEET
DROGUE CHUTE DEPLOYMENT
MAIN CHUTE DEPLOYMENT
12K SECOND TIME DELAY

SUBORBITAL AND ABORT TO ORBIT PHASE




EARTH ORBIT
ABORT
NO
GO
NO-GO
DECISION
GO
CONTINUE
MISSION
YES
NO-GO
ABORT
ENABLING
NO
FATAL FAILURE
ALTERNATE
MISSION
YES
SEPARATE S/C
FROM S-IVB
TIME
CRITICAL
NO
SELECT BEST
LANDING AREA
YES
IMMEDIATE
ENTRY
SPS
OPS. OK
NO
SM-RCS
DEORBIT
YES
SM-SPS
DEORBIT
SEPARATE SM
FROM CM
LANDING
AREA CONT.
YES
ACTIVATE ENTRY
GUIDANCE
NO
FLY CONSTANT
BANK ANGLE
RECOVERY
SEQUENCE

12.0 REFERENCES

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12.4 SID 64-1237 CSM Master End Item Specification (Block I) North American Aviation, Inc. December 1, 1964 - Confidential.

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