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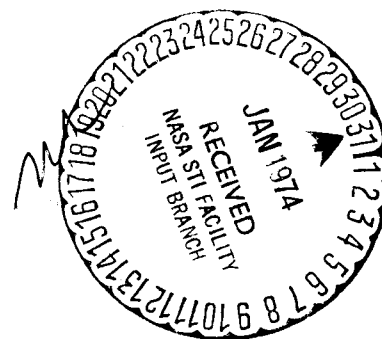
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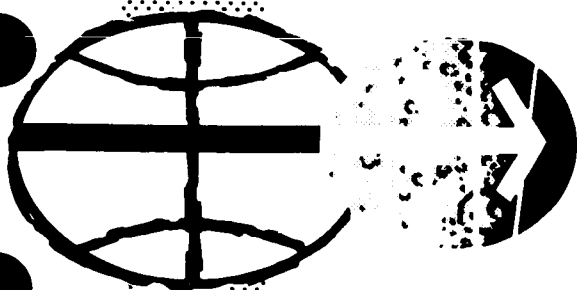
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OPERATIONAL SUPPORT PLAN FOR
THE REAL-TIME AUXILIARY
COMPUTING FACILITY
APOLLO 10 FLIGHT ANNEX



FLIGHT ANALYSIS BRANCH
MISSION PLANNING AND ANALYSIS DIVISION
MANNED SPACECRAFT CENTER
HOUSTON, TEXAS



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PROJECT APOLLO
OPERATIONAL SUPPORT PLAN FOR THE REAL-TIME
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APOLLO 10 FLIGHT ANNEX

By D. C. McDougall
Mission Operations Section
TRW Systems Group

May 12, 1969

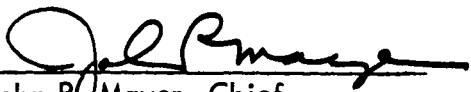
MISSION PLANNING AND ANALYSIS DIVISION
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NOMENCLATURE

ABDP	Apollo Block Data Program
ACR	Auxiliary Computing Room
ACRA	Atlantic continuous recovery area
AGOP	Apollo Generalized Optics Program
AGS	abort guidance system
AOS	acquisition of signal
AOT	alignment optical telescope
AP	abort processor
APS	ascent propulsion system
AREPMR	analytic return-to-earth abort processor
ARIA	Apollo Range Instrumentation Aircraft
ARMACR	Apollo Reference Mission Auxiliary Computing Room
ARRS	Apollo Real-Time Rendezvous Support Program
ARS	Apollo Reentry Simulation Program
ATP	arbitrary contingency lines
BAP	best adaptive path
CDH	constant differential height
CDR	commander
CLA	contingency landing area
CM	command module
CMC	command module computer
COAS	crew optical alignment sight
CRYOPWR	cryogenics electrical power
CSI	concentric sequence initiation

NOMENCLATURE

CSM	command and service module
DAP	digital autopilot
DKI	docking initiation
DMT	detailed maneuver table
DOI	descent orbit injection
DPS	descent propulsion system
ECI	earth centered inertial
ECS	environmental control system
EI	entry interface
ESM	earth, sun, moon
EMS	entry monitoring system
EMU	extravehicular mobility unit
EPO	earth parking orbit
EPS	electrical power system
ETR	Eastern Test Range
EVA	extravehicular activity
FAB	Flight Analysis Branch
FDAI	flight director attitude indicator
FDO	Flight Dynamics Officer
FTP	fixed throttle point
GEMMV	General Electric Missile and Satellite Multi-Vehicle Program
g. e. t.	ground elapsed time

NOMENCLATURE

GMT	Greenwich mean time
G&N	guidance and navigation
GNCS	guidance and navigation control system
GPMP	general purpose maneuver processor
HOPE	Houston Operations Prediction Estimator
ID	identification
IMU	inertial measurement unit
ITT	International Telephone and Telegraph
IU	instrument unit
IVT	intravehicular transfer
LAB	Landing Analysis Branch
LCG	liquid cooled garment
LEC	Lockheed Electronics Company
LET	launch escape tower
L/D	lift-to drag
LGC	lunar module guidance computer
LHe	liquid helium
LiOH	lithium hydroxide
LLM	lunar landing mission
LM	lunar module
LMP	Lunar module plot
LOI	lunar orbit insertion
LOS	loss of signal
LOST	lunar optical support table

NOMENCLATURE

LOX	liquid oxygen
LPD	landing point designator
LPO	lunar parking orbit
LV/LH	local vertical/local horizontal
MCC	midcourse correction
MCI	moon centered inertial
MPAD	Mission Planning and Analysis Division
MPSO	Mission Planning Support Office
MPT	mission plan table
MRS	Mass Properties, Reaction Control System, Service Propulsion System Program
MSFN	Manned Space Flight Network
MTVC	manual thrust vector control
NCC	Corrective combination
NSR	Slow rate catchup
OMAB	Orbital Mission Analysis Branch
OST	optical support table
PAO	Public Affairs Officer
PC	plane change
PDI	powered descent initiation
PGNCS	primary guidance and navigation control system
PLA	primary landing area
PLSS	portable life support system
PTC	passive thermal control
PVT	pressure-volume-temperature
RCS	reaction control system

NOMENCLATURE

REFSMMAT	reference to stable member matrix
REM	roentgen equivalent man
RTACF	Real-Time Auxiliary Computing Facility
RTCC	Real-Time Computing Complex
RTFIP	Real-Time Forward Iterator Program
S-IC	first stage of the Saturn V launch vehicle
S-II	second stage of the Saturn V launch vehicle
S-IVB	third stage of the Saturn V launch vehicle
SCS	stabilization control system
SEENA	Spacecraft Electrical Energy Network Analysis
SHe	Supercritical Helium Program
SLA	spacecraft lunar module adapter
SM	service module
SPAN	Solar Particle Alert Network
SPS	service propulsion system
SST	starsighting table
T&D	transposition and docking
TEI	transearth injection
TERRA	earth referenced return-to-earth abort program
TLC	translunar coast
TLI	translunar insertion
TPI	terminal phase initiation

NOMENCLATURE

TPF	terminal phase finalization
TRW	TRW Systems Group
ΔV	delta velocity
V- γ	Velocity - gamma

DEFINITIONS

Blackout Data	Blackout data include the ground elapsed time, latitude, and longitude of each entrance and exit of the blackout region for both S-band and VHF.
CDH	Second maneuver of the concentric flight plan logic. This maneuver establishes coelliptic orbits between the passive and active vehicles.
CSI	First maneuver of the concentric flight plan logic. This maneuver establishes the phase and height offset at the time of the second maneuver (CDH) in the concentric flight plan logic.
Entry Data	Entry data are generally intended to imply inertial velocity, flight-path angle, latitude, and longitude at the 400,000-foot altitude, blackout data, maximum g-load, landing point, and footprint data.
Entry Interface	Entry interface is considered to be the dividing line between the tenable atmosphere and space and is considered to occur at an altitude of 400,000 feet.
Footprint Data	Footprint data include the latitude and longitude for both a zero and a full lift entry.
Horizon Monitor	Horizon monitor refers to a spacecraft attitude which is maintained 31.7 degrees above or below the visual horizon.
Maneuver Data or Maneuver Specification	Maneuver data or specification consist of those quantities which are necessary to completely define a maneuver. These quantities may include engine ignition, imparted velocity, guidance mode, etc.
NCC	First maneuver of the two-impulse logic. This maneuver establishes a common node, phase offset, and height offset at the time of the second maneuver (NSR) in the two-impulse logic.

DEFINITIONS (Continued)

NSR	Second maneuver of the two-impulse logic. This maneuver establishes coelliptic and coplanar orbits between the passive and active vehicle.
Radar Data	Radar data include the ground elapsed time of acquisition and loss, minimum and maximum elevation angles, minimum range, tracking duration, and revolution number of each pass over a station.
REFSMMAT	REFSMMAT relates the Besselian coordinate system to the IMU stable member coordinate system.
State Vector	The term "State Vector" is meant to include the following quantities. 1. Vector identification 2. Lift-off time in GMT 3. Vector time in GMT 4. Position vector components 5. Velocity vector components 6. Revolution number 7. Spacecraft weight
TPF	Terminal phase finalization maneuver for planning purpose this maneuver is considered to result in zero velocity differences between the active and passive vehicles. In actuality terminal phase finalization is accomplished by the Apollo crew through a series of mid-course and braking maneuvers.
TPI	Terminal phase initiation maneuver. This maneuver establishes an intercept trajectory between the active and passive vehicle.

OPERATIONAL SUPPORT PLAN FOR
THE REAL-TIME AUXILIARY
COMPUTING FACILITY
APOLLO 10 FLIGHT ANNEX

By D. C. McDougall
Mission Operations Section
TRW Systems Group

1. INTRODUCTION

1.1 Purpose

The Apollo 10 Flight Annex summarizes those aspects of the operational support plan peculiar to the Apollo 10 mission. Included in this document are a description of the support requirements of the Real-Time Auxiliary Computing Facility (RTACF) for the Apollo 10 mission and a brief description of the programs that will be used for this support. It is intended to provide a central source of RTACF mission support information pertaining to this mission.

1.2 Method of Presentation

The Apollo 10 Flight Annex is divided into six sections, the first of which is the introduction. The second section states the main objectives of the mission and briefly outlines the major nominal mission events. The third section contains a description of the RTACF computing requirements for mission support, while the fourth section presents a description of the processors that will be used to satisfy these requirements. The fifth section presents the RTACF nominal mission timeline, and the sixth section lists the personnel assigned to the RTACF in support of the Apollo 10 mission.



2. GENERAL MISSION DESCRIPTION

2.1 Introduction

A description of the Apollo 10 mission profile is presented in this section. The nominal mission description as well as the summary of events tables have been derived from the Apollo 10 Spacecraft Operational Trajectory (Reference 1).

2.2 Nominal Mission

Apollo 10, the third manned Saturn V mission, is planned as an 8-day lunar orbital flight. It will include the first test of the lunar module (LM) in the lunar environment. The moon-centered operations will simulate as closely as possible the lunar landing mission (LLM) timeline, a G-mission type of rendezvous, and will pass over a prime landing site. Table I contains a listing of the nominal mission events, and Table II contains a listing of the Saturn launch vehicle events.

The nominal trajectory is based on a planned lift-off from pad 39B at 16:49 (hr:min) Greenwich mean time (GMT) on 18 May 1969. Launch azimuth is 72 degrees from true North.* The mission profile has been divided into six periods of activity.

2.2.1 First period of activity. - The first period of activity extends from lift-off to 2.5 hours ground elapsed time (g. e. t.) and consists of the following events:

a. Launch: The launch time for this mission is optimized to provide two translunar injection (TLI) opportunities. Both occur over the Pacific and yield acceptable lighting at the lunar landing target site. They occur on revolutions 2 and 3.

b. Earth parking orbit (EPO): Insertion into EPO occurs at 11:21.6 (min:sec) g. e. t. The EPO is a near-circular, 103-nautical mile orbit inclined 32.6 degrees. The total time in EPO is 2:20 (hr:min).

c. Postinsertion activities: Following insertion, the S-IVB maintains the cutoff attitude for 20 seconds before maneuvering the complete configuration to a local horizontal attitude. The S-IVB then imparts an orbital rate to the command service module (CSM)/S-IVB combination. The flight crew then conducts a series of systems checks and configures the CSM for TLI.

* The data used in preparation of this document were computed independently from the actual launch vehicle trajectory. Reference 2 presents the official launch vehicle trajectory data.

d. Translunar injection: The TLI burn is initiated near the western coast of Australia during the second revolution in EPO. It is initiated in darkness, and the vehicle enters sunlight approximately midway through the burn. The burn itself is biased for a 2-meter per second overburn to compensate for the service propulsion system (SPS) evasive maneuver that is performed after the LM ejection. The burn at 2:33 (hr:min) g. e. t. lasts approximately 300 seconds and consumes 149,000 pounds of fuel.

2.2.2 Second period of activity. - The second period of activity extends from TLI shutdown to 76 hours, g. e. t. It is the period referred to as translunar coast (TLC).

a. Free-return circumlunar trajectory: Free-return touchdown assumes perfect execution of TLI and no correction maneuvers. It is planned to occur in the Indian Ocean southeast of Madagascar. A more desirable landing position can be ensured by application of a corrective maneuver at an acceptable time during either the translunar or transearth coast phases of the circumlunar trajectory. The free-return trajectory is characterized by the following:

Pericynthion

Time (hr:min:sec, g. e. t.)	76:11:50.3
Altitude (n mi)	59.3
Selenographic latitude (deg)	1.97
Selenographic longitude (deg)	178.5

Return vacuum perigee altitude (n mi)	15.2
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Transit time from TLI to entry interface (hr:min:sec)	147:44:12
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Earth entry

Time (hr:min:sec, g. e. t.)	150:15:48.2
Altitude (n mi)	65.8
Geodetic latitude (deg)	-18.0
Longitude (deg)	45.2
Inclination (deg)	33.5

Touchdown

Geodetic latitude (deg)	67.1
Longitude (deg)	-27.9

b. SPS evasive maneuver: At approximately TLI plus 110 minutes (after the S-IVB has been separated), the SPS is fired to decrease the probability of S-IVB recontact and to avoid the ice particles expected to be expelled by the S-IVB during the liquid oxygen (LOX) dump. The current profile combines an early SPS confidence burn with the evasive maneuver. This SPS burn will have a ΔV of 20 ft/sec and will be approximately 3 seconds in duration. To achieve a burn of this magnitude without jeopardizing the reaction control system (RCS) capability to return to a free-return circumlunar mission, the TLI burnout conditions will be biased for a 2-meter per second overspeed at burn termination. The SPS evasive maneuver then will be performed in a direction which will compensate for the TLI bias. The attitude will be pitched down 75 degrees with respect to the local horizontal. This attitude will provide for spacecraft high gain S-band coverage with the steerable antenna, and a roll of approximately ± 60 degrees (based on the CSM/S-IVB separation attitude) provides for visual monitoring of the S-IVB during the evasive maneuver burn. At approximately 2 hours after TLI, the S-IVB is ground commanded to assume a local horizontal attitude for the LOX blowdown. The local horizontal attitude components are the following: pitch, 194 degrees; yaw, 0 degree; roll, 180 degrees. The magnitude of the ΔV that results from the LOX dump is expected to be approximately 120 ft/sec.

The LOX dump maneuver is designed to reduce the probability of spacecraft recontact with the S-IVB and also to prevent S-IVB impact with the earth or moon. Nominally, the LOX dump maneuver results in a sling-shot trajectory; the S-IVB will pass behind the trailing edge of the moon and will be accelerated by the lunar gravitational field. The result is a heliocentric orbit which avoids either earth or lunar impact.

c. Translunar coast: Passive thermal control (PTC) attitude will be maintained throughout most of the translunar coast phase. Four mid-course correction (MCC) maneuver points have been defined at the following times:

1. TLI plus 7 hours (MCC-1)
2. TLI plus 24 hours (MCC-2)
3. Lunar orbit insertion (LOI) minus 22 hours (MCC-3)
4. LOI minus 5 hours (MCC-4)

The third midcourse correction (MCC-3) will be the prime maneuver to establish the desired lunar approach trajectory. The first two maneuvers will not be performed unless the magnitude of the MCC-3 maneuver exceeds 25 ft/sec. The MCC-1 or MCC-2 maneuver or both will then be performed only if their values exceed the SPS minimum impulse (~ 3 ft/sec). The MCC-1 and MCC-2 residuals will not be trimmed.

To avoid use of the SPS for the MCC-4 maneuver, the MCC-3 maneuver will be performed if the predicted magnitude of MCC-4 is greater than 3 ft/sec using the SPS. Residuals will be trimmed to within 0.5 ft/sec. If MCC-3 is less than 3 ft/sec and if LOI-1 targeting cannot absorb the uncorrected approach dispersions without a shift greater than 45 degrees in the line of apsides of the 60- by 170-nautical mile orbit, MCC-3 will be performed with the service module (SM) RCS; however, if LOI-1 targeting can absorb the dispersions with less than a 45-degree apsidal shift, MCC-3 will not be performed.

The MCC-4 maneuver will not be performed if the dispersions can be absorbed by the LOI-1 targeting with apsidal rotation less than 45 degrees; otherwise, the maneuver will be performed with the SPS if the ΔV is greater than 3 ft/sec or the SM RCS if the ΔV is less than 3 ft/sec. The residual will be trimmed to within 1 ft/sec if the SPS is required for the MCC-4 maneuver.

The maneuvers are guidance and navigation control system (GNCS) controlled and use external ΔV guidance. Unless gimbal lock problems occur, the pad inertial measurement unit (IMU) alignment (REFSMMAT) will be used for the MCC-1, the Real-Time Computer Complex (RTCC) REFSMMAT for MCC-2 and MCC-3, and the descent REFSMMAT for MCC-4. The CSM remains in sunlight during the entire translunar coast phase. The duration of the phase is 73:37 (hr:min).

2.2.3 Third period of activity. - The third period of activity extends from the LOI-1 maneuver to 98:30 (hr:min) g. e. t. when the CSM and LM are undocked for the rendezvous performance. It includes the following events:

a. Lunar orbit insertion: Insertion into lunar orbit is accomplished using two SPS burns. The first burn (LOI-1) places the CSM into an approximately 60- by 170-nautical mile lunar parking orbit (LPO). The external ΔV guided burn is started at 75:45:43.17 (hr:min:sec) g. e. t. and provides approximately 2,978 ft/sec ΔV . This burn consumes 23,561 pounds of propellant. The REFSMMAT used for the LOI-1 burn as well as for all other burns in LPO is the landing site alignment at the nominal G-mission landing time relative to descent orbit insertion (DOI).

A coplanar circularization burn (LOI-2) is performed to place the CSM in approximately a 60-nautical mile circular LPO after two revolutions in the 60- by 170-nautical mile orbit. The target altitude of the orbit (60 nautical miles) is measured relative to the lunar target site and not relative to the mean lunar radius.

The landing REFSMMAT is used, and the CSM is oriented heads down. The burn is initiated near pericyynthion of the second revolution. More detailed information is given in Reference 1. The burn is external ΔV guided; ignition is at 80:10:45.52 g. e. t. and provides 138.5 ft/sec ΔV . It consumes 35 pounds of propellant.

b. CSM/LM coast from LOI-2 to undocking: At a g. e. t. of 81:45 (hr:min) or at about 1:34 (hr:min) after LOI-2, the crew begins preparation for intravehicular transfer (IVT) to the LM. In the LM, general house-keeping and equipment storage is performed. After about 2:40 (hr:min) in the LM, the commander (CDR) and lunar module pilot (LMP) perform IVT to the CSM and close the hatch. At a g. e. t. of approximately 84:40 (hr:min), landmark tracking is performed on a psuedo-site with a sun elevation angle of approximately 3 degrees. An inertial hold is initiated at a g. e. t. of 86 hours for an 8-hour crew rest period. The rest period is ended at 94 hours, g. e. t. After a 1-hour eat period, the CDR and LMP perform IVT to enter the LM and begin LM checkout. At 95:50 (hr:min) g. e. t. (revolution 11), landmark tracking is performed on the target site. The LM checkout is completed, and undocking occurs at 98:30 (hr:min) g. e. t. during revolution 12 or approximately 4:30 (hr:min) after wakeup.

c. LM undock and CSM separation: Undocking will occur at 98:05:15.62 g. e. t., 30 minutes prior to the RCS separation burn. After the CSM undocks from the LM, the CSM will perform stationkeeping at a distance of 30 feet from the LM for LM inspection. After completion of the inspection, the LM will perform stationkeeping while the CMP prepares for the RCS separation. At approximately a 180-degree central angle prior to DOI, the CSM performs a 2.5-ft/sec radially downward separation maneuver which places the LM and CSM in equiperiod orbits. Rendezvous will be accomplished from the equiperiod orbits if the DOI maneuver is not performed.

2.2.4 Fourth period of activity. - The fourth period of activity extends from the CSM/LM separation burn through 106:30 (hr:min) g. e. t. and includes the rendezvous sequence.

a. Sequence summary: The basic objective of the rendezvous sequence on the Apollo 10 mission is to simulate as nearly as possible the LLM rendezvous profile after LM insertion following ascent from the lunar surface. After separation of the LM and CSM, the rendezvous activities are initiated by the CSM separation maneuver (minifootball) at 98:35:23 g. e. t. Then the LM must perform a DOI maneuver and a phasing maneuver to establish the proper relative conditions (LM 49.4 nautical miles below and 270.0 nautical miles behind the CSM) at the simulated insertion point over the target site (34°E). After the insertion maneuver has been completed, the LM will compute and execute the coelliptic sequence that is planned for the LLM rendezvous. The sequence of concentric sequence initiation (CSI), constant delta height (CDH), and terminal phase initiation (TPI) will result in LM approach, braking, and docking at approximately 106:20:00 g. e. t., which completes the 7.5-hour exercise that began with separation. The relative motion plot for the nominal Apollo 10 mission rendezvous is presented in Figure 1. A detailed discussion of the rendezvous activities follows.

b. Descent orbit insertion: After the CSM separation maneuver, the LM will fine align the platform and will align the abort guidance system (AGS) to the primary guidance and navigation control system (PGNCS) in preparation for the DOI maneuver. The DOI maneuver is ground computed to be executed 195 degrees prior to the target site, which duplicates the same maneuver required in the LLM. The DOI maneuver is an external ΔV maneuver performed with the descent propulsion system (DPS) in a horizontal retrograde direction so that the resultant LM pericyynthion is 50,000 feet (referenced to the landing site radius) and is 15 degrees uprange from the landing site. The 71.1-ft/sec maneuver is performed at 99:33:57 g. e. t. with 10-percent thrust for 15 seconds and 40-percent thrust for 12.7 seconds.

c. Phasing: After the DOI maneuver, the LM will prepare for a landing radar test to be conducted as the vehicle passes over the target site at 34°E longitude and at an altitude of approximately 50,000 feet. Because the LM will lead the CSM (Figure 1) during this first pass over the site, a phasing maneuver is performed approximately 10 minutes after the site is passed to place the LM in a dwell orbit so that eventually the LM will fall behind the CSM and will trail the CSM by approximately 270 nautical miles at the time of the second pass, at which time the lunar landing mission relative profile can be simulated. The phasing maneuver is a ground-computed maneuver with an external ΔV of 195.4 ft/sec initiated with the DPS at 100:46:22 g. e. t. By use of the two-impulse processor, the maneuver is targeted to establish the nominal LLM phase and height offset relative to the CSM at the time of insertion. The posigrade burn at a 26.1-degree pitch above the local horizontal will place the LM in a 194.4- by 9.8-nautical mile orbit. The DPS burn will be started at 10-percent thrust for 26 seconds and will be increased to full throttle (92.5 percent) for 16 seconds.

d. Insertion: During the LM phasing orbit, the LM and CSM will conduct onboard tracking to determine the orbits. The onboard tracking data may be used by the ground to update the required insertion maneuver. The insertion maneuver will initiate the sequence that is designed to simulate the in-orbit ascent rendezvous of the LM after the lunar lift-off on the LLM. Prior to insertion, the LM will stage the DPS so that the burn may be executed with the ascent propulsion system (APS). Current plans call for the staging to occur approximately 10 minutes prior to insertion. While in a retrograde attitude, the LM will thrust posigrade 2 ft/sec with the -X RCS jets, will stage, and immediately will null this ΔV with 2 ft/sec retrograde with the +X jets. The result of the separation maneuver is to send the descent stage ahead of and above the ascent stage so that no recontact can occur after the ascent stage performs insertion. At 102:43:18 g. e. t., the APS thrusts at a 152.6-degree pitch for 15 seconds to impart a retrograde ΔV of 207 ft/sec and to place the LM into a 43.6- by 9.8-nautical mile orbit. Apocynthion occurs 51 minutes later. The 43.6- by 9.8-nautical mile orbit is identical to the orbit planned after

LM insertion in the LLM, and the insertion maneuver is scheduled 5 minutes prior to entry into darkness to duplicate the LLM lighting conditions. The insertion maneuver is also targeted by the ground two-impulse processor which establishes the nominal CSI offset (LM 14.7 nautical miles below and 148 nautical miles behind the CSM) at the nominal time.

e. CSI: After insertion, the LM will realign the platform and will begin radar tracking of the CSM to determine the orbits of the vehicles for onboard computation of the coelliptic sequence. The CSI maneuver will be scheduled for the apocynthion at 103:33:46 g. e. t. and will be calculated to cause TPI to occur at the midpoint of darkness approximately 94 minutes later. The nominal relative condition will be such that the CSI will place the LM in a 46.2- by 42.9-nautical mile orbit, 15.0 nautical miles below the CSM orbit at the time of CDH, one-half an orbital period after the CSI. The CSI will be performed with the four +X RCS jets so that the interconnect can be opened and APS propellant can be used. The 32.1-second burn is horizontal and adds a posigrade ΔV of 50.5 ft/sec.

f. Plane change technique: An out-of-plane component, which nominally is not required, will be applied in conjunction with the CSI if an out-of-plane velocity is detected prior to the CSI. The out-of-plane component will be targeted to null to zero the out-of-plane velocity, which will force the existence of a common node approximately 90 degrees later where the separate plane change (PC) maneuver is scheduled. At PC, the out-of-plane velocity is again nulled to zero, and a coplanar situation is established. If the out-of-plane situation is not determined soon enough to begin the PC at the CSI, the nodal shift would be initiated at the time of PC and completed in conjunction with the CDH. However, the CSI-PC sequence is more economical than the PC-CDH sequence because the in-plane component at the CSI is considerably larger than the in-plane component at CDH.

g. CDH: After the CSI, the LM will continue to track the CSM and will compute the required CDH maneuver to be done at 104:31:42 g. e. t. Normally, the CDH will be a small radial burn designed to place the LM in coelliptic orbit with the orbit of the CSM. If the CSM orbit was perfectly circular, the CDH would be zero; however, because of the simulated 61- by 58-nautical mile CSM orbit, a downward ΔV of 3.4 ft/sec is required. The burn will be performed with the four +X thrusters and the burn places the LM in a 46.2- by 42.9-nautical mile orbit, 15 nautical miles below the CSM orbit and coelliptic with it.

h. TPI: Radar tracking continues after CDH so that the LM may compute the required burn (TPI) when the elevation angle to the CSM reaches 26.6 degrees above the LM local horizontal. Nominally, the maneuver should occur approximately 36.5 minutes after CDH, when the LM is 23 minutes into darkness. TPI will be calculated to start the LM on an intercepting orbit; theoretically, rendezvous would occur after 130 degrees of CSM central angle travel. The 24.8-ft/sec burn is planned to be executed with the four +X jets to use the APS propellant through the interconnect. However, this arrangement may cause a temporary loss of radar lock, which is not considered to be a problem. The TPI ignition is at 105:09:00 g. e. t. , and the burn duration is approximately 15 seconds.

i. Midcourse corrections and braking: The LM will track the CSM after TPI and will perform nominally zero midcourse correction maneuvers 15 minutes later and 30 minutes later. The braking schedule assumed for this trajectory simulation calls for a reduction in range rate to 15 ft/sec at the 1-nautical mile gate, to 5 ft/sec at 1000 feet, and to 0.25 ft/sec at 300 feet. The braking schedule may be changed slightly after further simulations; line-of-sight corrections will be made as required. Final approach and stationkeeping should occur at approximately 105:55:00 g. e. t., approximately 25 minutes after the vehicles enter sunlight on the backside of the moon. Docking should begin at approximately 106:20:00 g. e. t. to complete 7.5 hours of rendezvous activities.

2.2.5 Fifth period of activity. - The fifth period of activity extends to approximately 137:70 (hr:min) g. e. t. and includes the following events:

a. APS burn to depletion: At 108:38:57.92 g. e. t., the unmanned LM is jettisoned in attitude hold, and the CSM performs a radially upward separation maneuver of approximately 2 ft/sec, which will place the CSM above and behind the LM at the time of the APS burn.

The ullage maneuver and the APS burn are initiated under PGNCs control, and after confirmation of burn initiation, a command is sent to transfer control to the AGS. The ascent stage will have been in attitude hold since jettison. Whether either or both of the RCS interconnects will be open will depend on the RCS usage up to that time in the mission and will probably be a real-time decision. The need for attitude hold to be maintained during the burn and as long, thereafter, as possible could be satisfied either with one interconnect open or with both closed. However, if the RCS margin is low enough at the time of jettison, both interconnects would be open to assure attitude control during the entire burn. In this case, ascent stage tracking after the burn would not be assured. The burn will be initiated at 108:39 (hr:min) g. e. t. After burning approximately 3 minutes, it will have provided 3837 ft/sec ΔV to place the APS in a hyperbolic trajectory with respect to the earth-moon system and assures a heliocentric orbit.

b. Coast to transearth injection (TEI): Shortly after the APS burn to depletion, an inertial attitude hold is initiated for an 8-hour crew rest period. After the rest period, four consecutive revolutions of landmark tracking will be performed, which will involve tracking either three or four landmarks per sunlight pass. After these four revolutions of tracking have been performed, one revolution of terminator-to-terminator stereo strip photography will be performed.

c. TEI: The TEI burn is initiated at 137:20:22.41 g. e. t. and is targeted for a 55-hour transearth flight time. The 2:48 (min:sec) SPS burn provides 3622.5 ft/sec ΔV and a 56-nautical mile perigee.

2.2.6 Sixth period of activity. - The sixth period of activity extends from TEI cutoff until splashdown at 165°W longitude and 20.25°S latitude. Three midcourse decision points have been defined for the transearth phase.

- a. MCC-5, TEI plus 17 hours
- b. MCC-6, Entry interface (EI) minus 15 hours
- c. MCC-7, EI minus 3 hours

The maneuvers will be targeted for corridor control only. The midcourse strategy, which includes the threshold values for each maneuver, is contained in References 3 and 4. The CSM remains in sunlight from TEI until approximately 21 minutes prior to EI. The last ground station coverage is by Honeysuckle, which terminates at 0 degrees elevation approximately 3 minutes prior to EI.

Nominal entry interface occurs at 191:50:32 g. e. t. Inertial velocity, flight-path angle, and azimuth at this point are 36,210 ft/sec, 6.49 degrees below the local horizontal, and 98.56 degrees, respectively. The nominal touchdown target location is 1350 nautical miles downrange from EI. The drogue parachutes are deployed at 23,300 feet, and the main parachutes are deployed at 10,500 feet. Nominal splashdown occurs 14:00 (min:sec) after EI.

Table I. Apollo 10 Sequence of Major Events

<u>Event</u>	<u>Time, **</u> (hr:min:sec, g. e. t.)	<u>Data Summary</u>	
Earth orbit insertion	11:24.0	Latitude (deg N)	32.8
		Longitude (deg W)	-54.2
		Inclination (deg)	32.7
Translunar injection	2:33:27.4	Burn time (sec)	321.6
		Plane change (deg)	1.34
SPS evasive maneuver	4:28:47.6	Altitude (n mi)	16,657
		ΔV (ft/sec)	19.7
		Burn time (sec)	2.8
		Propellant used (lb)	183.7
Midcourse correction ***	9:38:46.4	Altitude (n mi)	47,706
		ΔV (ft/sec)	57.0
		Burn time (sec)	8.1
		Propellant used (lb)	529.8
Free return, circumlunar pericyynthion	75:49:40.2	Altitude (n mi)	58.4
		Selenographic latitude (deg)	0.5
		Longitude (deg)	177.8
Free-return entry	149:34:46.4	Altitude (n mi)	65.8
		Longitude (deg)	65.0
		Latitude (deg)	-13.7
		Flight-path angle (deg)	-6.8
		Velocity (ft/sec)	36,140.5
		Equatorial inclination (deg)	35.6
		Vacuum perigee altitude (n mi)	15.7
Lunar orbit insertion	75:45:43.2	Mass at ignition (lb)	92,427.9
		Burn time (sec)	361.5
		SPS propellant used (lb)	23,560.7
		Inclination of LPO (deg)	1.2
		ΔV (ft/sec)	2978
LOI-2	80:10:45.5	Mass at ignition (lb)	68,821.2
		Burn duration (sec)	14.4
		SPS propellant used (lb)	935.3
		ΔV (ft/sec)	138.5
Undocking	98:05:15.6		
LM separation (minifootball)	98:35:15.6	Mass at ignition (lb)	36,484.4
		RCS burn time (sec)	6.9
		Propellant used (lb)	10.2
DOI	99:33:57	Ignition longitude (deg)	-139.7
		ΔV (ft/sec)	71.1
		Burn duration (sec)	29.9
CSM pass over target	100:38:30.9	Sun elevation at site (deg)	10.4
Phasing	100:46:21	Ignition longitude (deg)	-11.3
		ΔV (ft/sec)	195.6
		Burn duration (sec)	45.3

* Launch occurs at 12:48:35 EDT with a 72-degree launch azimuth.

** Time refers to g. e. t. of ignition for burns.

*** Nominal maneuver designed to change lunar parking orbit orientation to be compatible with G-mission lunar orbit.

Table I. Apollo 10 Sequence of Major Events (Continued)

<u>Event</u>	<u>Time, **</u> <u>(hr:min:sec, g.e.t.)</u>	<u>Data Summary</u>
Insertion	102:43:18	Ignition longitude (deg) 19.0 ΔV (ft/sec) 206.9 Burn duration (sec) 15.4
CSI	103:33:46	Ignition longitude (deg) -142.0 ΔV (ft/sec) 50.5 Burn duration (sec) 32.2
CDH	104:31:42	Ignition longitude (deg) 37.5 ΔV (ft/sec) 3.4 Burn duration (sec) 2.2
TPI	105:08:57	Ignition longitude (deg) -78.7 ΔV (ft/sec) 25.4 Burn duration (sec) 16.1
Fourth braking	105:54:24	Ignition longitude (deg) 141.2 ΔV (ft/sec) 4.6 Burn duration (sec) 5.9
CSM separation maneuver following LM jettison	108:09:24	Mass at ignition (lb) 36,674.0 RCS burn time (sec) 5.5 Propellant used (lb) 8.2 ΔV (ft/sec) 2.0
APS burn to depletion	108:38:57	Ignition longitude (deg) 0.0 Burn duration (sec) 214.5 Mass at ignition (lb) 7,600.0 Propellant used (lb) 2,451.0 ΔV (ft/sec) 3,836.0
Transearth injection	137:20:22.4	Mass at ignition (lb) 36,574.9 Burn time (sec) 168.9 Plane change (sec) -0.2 Propellant used (lb) 11,003.5
Entry interface	191:50:32.2	Velocity (ft/sec) 36,309.7 Flight-path angle (deg) -6.52 Latitude (deg) -22.71 Longitude (deg) 173.8 Time from TEI (hr:min) 54:27
Splashdown	192:04:27	Latitude (deg) -15.11 Longitude (deg) -165.0 Local time (a.m.) 5:53 Time of sunrise (a.m.) 6:17

* Launch occurs at 12:48:35 EDT with a 72-degree launch azimuth.

** Time refers to g.e.t. of ignition for burns.

Table II. Apollo 10 Launch Vehicle Operational Trajectory
Sequence of Events

<u>Event</u>	<u>Time from lift-off (min:sec)</u>
Lift-off	0:00.0
S-IC inboard engine cutoff	2:15.0
S-IC outboard engine cutoff	2:39.9
S-IC/S-II separation	2:40.7
Second plane separation	3:10.7
Jettison launch escape system	3:16.2
S-II center engine cutoff	7:39.2
S-II thrust termination	9:13.8
S-II/S-IVB separation	9:14.8
S-IVB ignition	9:17.9
S-IVB guidance cutoff	11:43.2
Parking orbit insertion	11:53.2

3. APPROVED RTACF SUPPORT REQUIREMENTS FOR APOLLO 10

3.1 Introduction

The requirements that the Real-Time Auxiliary Computing Facility will support for the Apollo 10 mission are given in Reference 5 and are briefly summarized in this section. These requirements have been assigned to the RTACF as outlined in Section 6 of the Operational Support Plan (Reference 6) and have been discussed and mutually agreed upon by the Mission Planning and Analysis Division and the Flight Control Division.

3.2 General Requirements

3.2.1 Trajectory Related. -

3.2.1.1 Ephemeris tape generation: Generate two types of ephemeris tapes. The standard ephemeris tape which consists mainly of a state vector, REFSMMAT and gimbal angles for two vehicles, and the postflight ephemeris tape consisting of 113 trajectory related parameters for one vehicle. Also, the ephemeris tapes must reflect any maneuvers present in the mission plan table (MPT).

3.2.1.2 Ground track: Determine ground track data (latitude, longitude, altitude, revolution number, azimuth, and corresponding g. e. t.) for the earth or moon. In addition, produce a plot of the ground track.

3.2.1.3 Relative motion: Given the state vectors of the active and passive vehicle, current mission plan, time interval and REFSMMAT, determine the relative motion quantities (gimbal and/or flight director attitude indicator (FDAI) angles, range, range rate, azimuth, elevation, curvilinear position coordinates, and corresponding g. e. t.).

3.2.1.4 Spacecraft daylight darkness: Given a state vector, time interval, and any maneuvers to be performed during the interval, determine the spacecraft time and position of sunrise, sunset terminator rise, and terminator set. Also, compute the spacecraft attitude to point the X-axis at the sun.

3.2.1.5 Spacecraft moonsighting: Given a state vector, time interval, and any maneuvers to be performed during the interval, determine the time and spacecraft position of moonrise and moonset.

3.2.1.6 Landmark sighting: Given the landmark number, time interval to perform the landmark search, and a state vector, determine the following quantities: the spacecraft acquisition and loss time, slant range and azimuth at acquisition, minimum slant range, maximum elevation angle, and duration of the pass.

3.2.1.7 Spacecraft starsighting: Given a star identification number, revolution number, and a state vector, determine the times of starrise and starset relative to the spacecraft and the time and position in which the star-earth-spacecraft central angle is a minimum (closest approach).

3.2.1.8 Pointing data: Given a state vector, target identification, REFSMMAT, and time interval to perform the target search, determine the spacecraft-to-target look angles (gimbal angles and local vertical/local horizontal angles), and the target-to-spacecraft look angles (elevation angle and azimuth angle). Also, compute the spacecraft acquisition and loss times, maximum elevation angle, minimum slant range, altitude, and elapsed time of the pass.

3.2.1.9 Closest approach: Determine the spacecraft closest approach to a specified ground target, given target identification, revolution number, and a state vector.

3.2.1.10 Telescope data: Compute trajectory parameters required by various optical telescopes to acquire and track the CSM/LM. The output will be in the form of punched cards for transmission through an IBM 066/068 Card Transceiver.

3.2.1.11 Spacecraft-to-sun alignment: Determine the CSM attitude so that the liquid waste dump nozzle, the electrical power system radiator, and the environmental control system radiators receive optimum heating from the sun.

3.2.1.12 Checkout monitor: Given a state vector and a time, propagate the state vector to the specified time and output the data in the checkout monitor format.

3.2.1.13 Radio transmission time delay: Given a state vector, time of radio transmission, and radar station identification, determine the radio transmission time delay.

3.2.1.14 CSM horizon alignment check: Determine the IMU inner gimbal angle required to align a horizon alignment mark on the command module (CM) window to the horizon at a selected time, given the spacecraft state vector, REFSMMAT, and the vehicle yaw and roll angles.

3.2.1.15 LM horizon alignment check: Determine the LM IMU outer gimbal angle (yaw) required to align the LM Z-axis in the local vertical plane at a selected time, given the spacecraft inner and middle gimbal angles. Also, determine the position of the horizon on the landing point designator (LPD) set of lines on the LM window.

3.2.1.16 Docking alignment: Determine the LM REFSMMAT and the LM or CSM gimbal angles in the docked CSM/LM configuration.

3.2.1.17 Attitude for preferred REFSMMAT: Given a vehicle REFSMMAT and a preferred REFSMMAT, determine the gimbal angles required for the vehicle REFSMMAT which would define 0, 0, 0 gimbal angles for the preferred REFSMMAT.

3.2.1.18 FDO orbit and space digitals: Compute and display in the proper format the Flight Dynamics Officer (FDO) orbit digitals and space digitals.

3.2.1.19 PTC attitude: Determine the attitude so that the X-body axis of the spacecraft is perpendicular to the sun and earth lines of sight. If this attitude results in a middle gimbal angle constraint violation, generate all the possible alignment so that this constraint will not be violated. Also, determine the earth-moon plane spacecraft alignment.

3.2.1.20 Maneuver evaluation: Given a preburn and postburn state vector, burn profile, maneuver REFSMMAT, and an impulsive maneuver guess time, determine the actual maneuver that was performed by the spacecraft.

3.2.1.21 ESM look angles: Determine the look angles (aspect and cone angles) to the earth, sun, and moon (ESM) at a specified time.

3.2.1.22 True of date vector generation: Given any state vector and a specified time, determine the true of date state vector.

3.2.1.23 Radar tracking and summary: Determine the time history of the radar station parameters (azimuth, elevation, slant range, and the associated rates). Also, display the summary of this time history. The summary will include acquisition and loss times, slant range, and azimuth at acquisition, minimum slant range, maximum elevation angle, and the duration of the pass.

3.2.2 Optical sighting and antenna pointing. -

3.2.2.1 CMC sextant IMU alignment: Determine the Command Module Computer (CMC) REFSMMAT, given two stars, their location in the sextant field of view, and the corresponding spacecraft IMU gimbal angles.

3.2.2.2 CMC COAS IMU alignment: Determine the CMC REFSMMAT, given two stars, their location in the crewman optical alignment sight (COAS) field of view, and the corresponding spacecraft IMU gimbal angles.

3.2.2.3 LGC AOT IMU alignment: Determine the Lunar Module Guidance Computer (LGC) REFSMMAT given two stars, their location in the alignment optical telescope (AOT) field of view, and the corresponding spacecraft IMU gimbal angles.

3.2.2.4 LGC COAS IMU alignment: Determine the LGC REFSMMAT, given two stars, their location in the crew optical alignment sight field of view, and the corresponding spacecraft IMU gimbal angles.

3.2.2.5 CSM star search: Given the current CMC IMU alignment, the current spacecraft IMU gimbal angles, and a search interval, locate up to 10 stars that will be in the telescope field of view and the position of these stars with respect to the telescope reticle pattern as well as the stars acquisition of sight (AOS) and loss of sight (LOS).

3.2.2.6 LM star search: Given the current LGC IMU alignment, the vehicle REFSMMAT, and a search interval, find up to 10 stars that will be in the LM optics field of view during the search interval as well as the AOS and LOS of each star. The optics used, AOT or COAS, and their detent position or axis orientation must also be defined.

3.2.2.7 Star location: Determine the optics angles for the CSM, sextant, the CSM COAS, the LM AOT, or the LM COAS to center each star in the appropriate field of view, given the star identifications, the current REFSMMAT, and the spacecraft attitude.

3.2.2.8 CSM celestial target sighting (fixed sextant): Given a celestial target location, REFSMMAT, and fixed sextant configuration, determine the spacecraft IMU gimbal angles required to center the target in the sextant field of view. Also, compute the central angle and time of closest approach, the time of arrival at the line of sight to the target, and the earliest point at which the line of sight does not pass through the atmosphere of the earth.

3.2.2.9 CSM celestial target sighting (movable sextant): Determine the sextant shaft and trunnion angles for viewing a celestial target, given a celestial target location, REFSMMAT, and IMU gimbal angles.

3.2.2.10 LM celestial target sighting (fixed AOT): Given the spacecraft attitude, the AOT detent position, and the time of sighting, determine the right ascension and declination of the center of the field of view of the AOT at the time of sighting.

3.2.2.11 LM celestial target sighting (movable AOT): Given the right ascension and declination of the target point, the AOT detent position, and time of sighting, determine the IMU gimbal angles of the spacecraft at the time of sighting.

3.2.2.12 Ground target sighting (fixed sextant): Determine the spacecraft IMU gimbal angles, time of arrival at the desired line of sight to the target, and the time and central angle of closest approach, given the target location, REFSMMAT, and desired sextant configuration.

3.2.2.13 Ground target sighting (movable sextant): Determine the required sextant shaft and trunnion angles for viewing a ground target, time of arrival at the desired line of sight to the target, and the time and central angle of closest approach, given the target location, spacecraft attitude and desired sextant configuration.

3.2.2.14 Point AOT with CSM: Given the CMC REFSMMAT, the docking angle, and state vector, determine the CMC gimbal angles required to center a specified star in the AOT field of view at a specified time. The detent position of the AOT must also be specified.

3.2.2.15 Cislunar navigation: Given the CSM REFSMMAT, state vector, and landmark and/or star identifications, determine the inertial attitude which will align the CSM optical system in the null position to the earth or moon horizon or some specified landmark on the earth or moon. Also determine the sextant shaft and trunnion angle of the stars.

3.2.2.16 Steerable antenna data: Given the spacecraft attitude, REFSMMAT, and time of sighting of a specified earth target, determine the pitch and yaw angles of any one of the onboard antennas necessary to point the antenna at a specified earth target. Alternately, given the pitch and yaw angles of the antenna, determine the spacecraft attitude necessary to point the antenna at the selected target.

3.2.3 Systems. -

3.2.3.1 Mass properties and aerodynamics: Given the weights, center of gravity, mixture ratio, moments of inertia of the consumables tanks, and any miscellaneous items to be considered, determine any of the following: aerodynamics for CM entry, center of gravity location of either the LM, CM, or CSM, mass properties tables for different vehicle configurations, and digital autopilot (DAP) command loads.

3.2.3.2 SM RCS predicted propellant profile: Determine the complete SM reaction control system propellant budget, given the spacecraft mass properties, the control mode for each maneuver, the RCS jet selection, and a timeline of maneuvers.

3.2.3.3 SM RCS propellant status: Determine the current SM RCS propellant available using the primary or auxiliary system, given the quad selection, the corresponding helium pressures and temperatures, the tank expulsion efficiencies, and the RCS oxidizer to fuel mixture ratios.

3.2.3.4 LM RCS predicted propellant profile: Determine the LM RCS propellant consumables budget, given spacecraft weight, propellant weight, and a mission event timeline.

3.2.3.5 DPS SHe pressure profile: Determine the pressure profile of the DPS supercritical helium (SHe) fuel pressure system for each DPS maneuver. The profile will include the maximum helium pressure, the helium pressure at the end of a burn, the helium mass remaining at the end of a burn, and a tabulation of pressure versus time during the coast period prior to the maneuver, during the maneuver, and for a period of time following the maneuver.

3.2.3.6 LM electrical power system profile: Given the premission LM power distribution network parameters (on magnetic tape), the mission timeline, and any new component configuration, determine the capability of the LM electrical power system to support the mission.

3.2.3.7 LM telemetry diagnostics: Determine the common failure points in the telemetry downlink system, given the code number of the failed telemetry points.

3.2.3.8 CSM cryogenic electrical power: Given the electrical power load, the temperature and pressure of oxygen and hydrogen, compute the electrical power profile and cryogenic profiles for oxygen and hydrogen.

3.2.4 Command load. -

3.2.4.1 CMC uplink data: Given a set of data in engineering units to be uplinked to the CMC, determine the octal equivalent of these data in the format and scaling acceptable to the CMC. Conversion of the following sets of data will be required: REFSMMAT, orbital external ΔV data, and deorbit external ΔV data.

3.2.4.2 LGC uplink data: Given a set of data in engineering units to be uplinked to the LGC, determine the octal equivalent of these data in the format and scaling acceptable to the LGC. Conversion of the following sets of data will be required: REFSMMAT and orbital external ΔV data.

3.2.4.3 Engineering units/octal conversion: Given the octal scale factor and precision, convert either engineering units to octal or octal to engineering units.

3.2.4.4 Navigation vector updates (CMC, LGC, AGS, and IU): Given a state vector and navigation update time, output the state vector at the update time in engineering units and either the CMC, LGC, or the instrument unit (IU) format. For the AGS the output will be engineering units only.

3.2.5 Miscellaneous. -

3.2.5.1 PAO data: Display in proper format quantities associated with mission events for the Public Affairs Officer (PAO).

3.2.5.2 Earth illuminance and sun impingement on the CSM optics: Determine the amount of reflected earth-light in lumens per square foot on the scanning telescope of the CM. Also determine if sunlight will impinge on the CM telescope, given the state vector and shaft angle of the scanning telescope.

3.2.5.3 Vehicle gimbal angles conversion: Given the lift-off gimbal angles of the IU, the CM and the LM IMU, determine the three sets of gimbal angles, at a later time, prior to the transposition and docking, given any one of the three sets of angles.

3.2.5.4 LM gimbal angles FDAI angles conversion: Given a set of either flight director attitude indicator angles or LM gimbal angles, determine the corresponding set of LM gimbal angles or FDAI angles.

3.2.5.5 Radiation evaluation: Given a state vector and time interval of the required computation, determine the geomagnetic parameters, the radiation dose rates (REM per hour), and the cumulative radiation dose (REMS) in the CM and LM.

3.2.5.6 Solar activity: Given solar flare data transmitted from the radio and optical telescopes in the Solar Particle Alert Network (SPAN), reduce the data to obtain graphs of the radio frequency burst profile and particle density as a function of time in the vicinity of the earth-moon system.

3.2.5.7 State vector coordinate transformation: The state vector coordinate transformation requirements are as follows:

- a. Conversions of a launch pad inertial (IU) state vector to an earth centered inertial (ECI) Besselian state vector

- b. Conversion of an AGS state vector (stable member) to an ECI Besselian state vector

- c. Conversion of a Besselian moon centered inertial (MCI) state vector to a Besselian ECI state vector

3.2.5.8 Flight control work schedule: Display, in graphical form, mission and trajectory related events that occur in a specified interval of time during the mission. Any mission anomalies or alternate procedures that might develop must be reflected in the display.

3.2.5.9 RTCC restart conversion: Convert a numeric or alphanumeric onboard spacecraft state vector to an RTCC state vector.

3.2.5.10 Camera exposure data: There is no information available for the camera exposure data requirement at the time of this writing.

3.2.5.11 Model data: Compute the following quantities to be used for the Apollo mission simulation model: right ascension and declination of the sun, earth, moon, and velocity vector; the position vector in nautical miles and the velocity vector in nautical miles per hour; the time in g. e. t., GMT, and Central Standard Time; and the current REFSMMAT and gimbal angles. All the previous mentioned quantities are for both the LM and CSM. In addition, determine the LM FDAI angles.

3.2.5.12 Star availability: Determine the stars and the time they will be available for sighting by the spacecraft.

3.3 Mission Phase Related Requirements

3.3.1 Prelaunch and launch. -

3.3.1.1 Mode I launch abort: Determine the spacecraft landing points should an abort be necessary during the first 90 seconds of powered flight, based on wind profiles from 24 to 1 hour prior to launch.

3.3.1.2 Lift-off REFSMMAT: Determine the lift-off REFSMMAT, given the flight azimuth and time of guidance reference release.

3.3.1.3 Estimated nodal targets: Determine the estimated integrated nodal targets (height of the node above the lunar landing site, latitude and longitude of the node in the earth-moon plane system, and time of the node). Also, compute the nominal post-TLI state vector in classical orbital elements.

3.3.2 Earth orbit. -

3.3.2.1 General purpose maneuvers and earth rendezvous: This requirement is the same as the lunar general purpose maneuvers and lunar rendezvous planning requirement presented in Sections 3.3.5.1 and 3.3.5.3.

The requirement to maintain earth orbital maneuver planning capability on Apollo 10 is necessary only in case of some contingency prior to TLI that requires execution of an earth orbit alternate mission.

3.3.2.2 Orbital lifetime: Determine lifetime of the vehicle (CSM, S-IVB, or LM) in days, hours, and minutes, g. e. t., given a state vector and vehicle aerodynamics.

3.3.2.3 K-factor: Determine the atmospheric density K-factor, given the vehicle weight, drag coefficient, effective aerodynamics cross-sectional area, and two or more state vectors.

3.3.2.4 S-IVB venting simulation: Given a state vector and a time, simulate the S-IVB venting and determine any requested trajectory parameters.

3.3.2.5 Computed events: Determine the orbital events (apogee, perigee, ascending node, and revolution number) and related times of these events, given an initial state vector, time interval to be considered, and any maneuvers to be considered during the interval.

3.3.3 Earth deorbit. -

3.3.3.1 PLA: Determine the deorbit ignition time and the time to reverse bank angle from the initial bank angle in order to achieve a target latitude and longitude in one of the primary landing areas (PLA). The maneuver attitude, ΔV , and entry profile will be specified. Also, it may be necessary to include a separation maneuver prior to a deorbit which is at a specified time or a fixed delta time before deorbit ignition.

3.3.3.2 CLA: Determine the deorbit ignition time to achieve a target longitude in one of the contingency landing areas (CLA), given the maneuver attitude, ΔV , and entry profile. Also, it may be necessary to include a separation maneuver prior to a deorbit which is at a specified time or fixed delta time before deorbit ignition.

3.3.3.3 Block data: Generate block data which consists of contingency retrofire information for a period of anywhere from four to eight revolutions. This retrofire data may include SPS or SM RCS PLA's and CLA's, as well as SM RCS apogee deorbits for each revolution of the block. The state vector for each deorbit solution computed should reflect any orbital maneuvers which are currently planned during the interval for which the block of deorbit data is to be generated.

3.3.3.4 Hybrid deorbit: Determine the SM and CM RCS deorbit ignition times in order to achieve a target longitude, given the incremental velocities and spacecraft attitudes for each maneuver, REFSMMAT, and an entry profile consisting of a lift vector orientation to a specified g-level, followed by a zero-lift entry to drogue chute deployment. Also determine the landing point based on the actual hybrid deorbit performed. The SM RCS maneuver will be considered as having been performed nominally, and the actual incremental velocities will be used to define the CM RCS maneuver. The entry profile will be the same as that mentioned previously.

3.3.3.5 Hybrid deorbit without S-IVB/CSM separation: Given a state vector before S-IVB/CSM separation, the time of initiation of the LOX dump maneuver, and the CMC REFSMMAT, determine the CM RCS deorbit ΔV at apogee and the IMU gimbals angles at ignition to achieve a target longitude. The entry profile will nominally consist of a lift vector down to a 1-g deceleration followed by a rolling entry to drogue chute deployment.

3.3.4 Post-TLI through LOI-1 (translunar phase). -

3.3.4.1 TLI plus 10 minutes fixed attitude abort: Determine the abort ΔV , gimbal angles, and terminator reference angles for a fixed attitude, time critical, unspecified area, TLI plus 10 minutes abort, given an initial state vector, REFSMMAT, time of TLI cutoff (or ignition time), velocity and flight-path angle constraint at entry, initial pitch and yaw trim angles of the CSM, and the pitch angle below the line of sight to the far horizon. This type of abort is generated during the initial earth parking abort to verify or update the onboard crew charts which are determined preflight.

3.3.4.2 CSM attitude for docking maneuver: Given the CSM and LM REFSMMATS and the IU gimbal angles for the docking maneuver, determine the CSM attitude for the docking maneuver.

3.3.4.3 TLI ΔV comparison: Display, in graphical form, the differences in the velocity components sensed by the IMU and the IU during the TLI burn.

3.3.4.4 Midcourse maneuver planning: Determine translunar midcourse maneuvers to produce reoptimization of the planned mission, a circumlunar flyby mission plan as a degradation of the planned mission, a fuel critical return to earth mission plan, and a return to the nominal mission.

3.3.4.5 Return to earth: Determine the ignition time, gimbal angles, abort ΔV , entry interface and landing quantities, given an initial state vector, REFSMMAT, velocity and flight-path angle constraint at entry interface, mode of abort, and maximum and minimum landing time and landing area (if any) to be considered. Determine the entry monitoring system initialization quantities, the guided entry profile, and backup guidance quantities required to reach a target landing point given the guidance mode and desired targets.

3.3.4.6 LOI-1 maneuver planning: Given a state vector, the desired lunar parking orbit shape, the landing site location, approach azimuth, and appropriate constraints, determine LOI maneuvers in the following four categories: lunar orbit shape priority, lunar landing site priority, and basic and minimum ΔV .

3.3.4.7 LOI-1 crew chart update: Given the direction in which to apply the abort ΔV relative to the earth-moon plane for the LOI-1 plus 15 minute abort, determine the gimbal angles required to align the spacecraft to perform the maneuver. In addition, calculate the abort velocity increment necessary to satisfy the flight-path angle constraint at entry interface for selected premature LOI-1 shutdown times.

3.3.4.8 LOI-1 monitor: Following the targeting of the LOI-1 maneuver after a midcourse correction, simulate the maneuver and determine the effect of a 10-degree drift about the pitch IMU axis on the height of pericynthion. The 10-degree drift represents the maximum allowable drift before a manual takeover by the Apollo crew. If the dispersion causes an unacceptable pericynthion altitude, another MCC will be made.

3.3.4.9 Earth horizon radius: Determine the earth horizon radius (altitude above the Fisher ellipsoid) that the astronaut uses for the earth horizon reference.

3.3.5 Lunar operations. -

3.3.5.1 General purpose maneuvers: Given a state vector and a set of orbital conditions, compute the impulsive maneuver to obtain the desired orbital conditions.

3.3.5.2 Lunar descent planning: Determine the impulsive orbital adjustment maneuvers required to shape the lunar orbit prior to DOI. The DOI maneuver is performed to insert the vehicle in the nominal descent orbit with a 180-degree transfer to the target pericynthion location.

3.3.5.3 Lunar rendezvous planning: Determine the series of maneuvers required to accomplish a rendezvous plan and generate the lunar rendezvous plan table, rendezvous evaluation display, two-impulse rendezvous table (multiple solution), two-impulse rendezvous table (single solution), CSM insertion burn display, FDO mission plan table, and the FDO detailed plan table. Generate any other orbital maneuvers required to successfully complete the mission.

3.3.5.4 Transearth injection maneuver: The requirement to calculate the transearth injection maneuver is the same as the return-to-earth requirement presented in Section 3.3.4.5.

3.3.5.5 Landing site determination and MSFN evaluation in lunar orbit: There is no information available, at the time of this writing, for the landing site determination and Manned Spaceflight Network (MSFN) trajectory evaluation in lunar parking orbit requirement.

3.3.6 Post-TEI through entry interface (transearth phase). -

3.3.6.1 Return to earth: The return to earth requirement in the transearth phase is the same as the return to earth requirement in the translunar phase (Section 3.3.4.5).

3.3.6.2 Optimized midcourse: Determine the optimum ignition time, ΔV , gimbal angles, and entry interface conditions in the precision mode, given a vector during the transearth coast phase of the mission, desired inertial flight-path angle at entry interface, an estimate of the ignition time, and an approximate landing time.

3.3.7 Atmospheric entry. -

3.3.7.1 Guided and backup entry modes: Determine the entry monitoring system (EMS) guided entry and backup entry quantities required to reach a target landing point, given a state vector at entry interface. The state vector at entry interface will be generated by a RTACF processor that will simulate any maneuver which occurs before entry, given a preburn state vector, or will coast a vector to entry interface.

3.3.7.2 Landing point update: Determine the entry target landing point and update the CMC prior to entry interface.

3.3.7.3 Apollo Range Instrumentation Aircraft (ARIA) data: Determine the state vector at 425,000 feet and entry quantities for the ARIA.

4. RTACF PROCESSORS FOR APOLLO 10

4.1 Introduction

This section of the Flight Annex presents a description of the processors and their inputs and outputs that are used to satisfy the requirements listed in the previous section. To provide a more meaningful description of the processors, the requirements are restated along with any additional pertinent information concerning the requirements. The processors are divided into two categories, the general category that includes those processors that may be used at any time during a mission and the mission phase related category that includes those processors that could only be used during specific mission phases.

In this section, the word processor is used to refer to a particular configuration of a program or programs that are used to fulfill a requirement. The outputs required for a processor or program are displayed on RTACF summary sheets, except for information that is plotted or punched by peripheral equipment or obtained through normal program print.

4.2 General Processors

4.2.1 Trajectory related. - The trajectory related processors are those processors which satisfy requirements that primarily involve trajectory propagation and trajectory related output quantities. The two programs that are presently used in the RTACF to do the actual trajectory propagation are the Apollo RTACF Rendezvous Simulation (ARRS) Program and the Apollo Reference Mission Auxiliary Computing Room (ARMACR) Program.

4.2.1.1 Ephemeris tape generation: Generate two types of ephemeris tapes: the standard ephemeris tape which consists mainly of a state vector, REFSMMAT, and gimbal angles for two vehicles, and the post-flight ephemeris tape consisting of 113 trajectory related parameters for one vehicle. The ephemeris tapes must be able to reflect any maneuvers present in the mission plan table.

This requirement is fulfilled by any processor utilizing the ARMACR program. The ephemeris tapes, in general, are used with other processors that do not have the capability to propagate state vectors.

Inputs required:

- a. State vectors
- b. REFSMMAT
- c. Time increment and duration

- d. Maneuver specification
- e. Gimbal angles

Outputs required:

The outputs are in the form of binary word records stored on magnetic tape. The quantities for the standard ephemeris tape are shown in Table III and the quantities for the postflight ephemeris tape are shown in Table IV.

4.2.1.2 Ground track: Determine ground track data (latitude, longitude, altitude, revolution number, azimuth, and corresponding g. e. t.) for the earth or moon. In addition, produce a plot of the ground track.

The ground track requirement is satisfied by the ground track processor which utilizes the ARMACR program in conjunction with the Ground Track Plotter Program. ARMACR produces a continuous ground track, integrating through the current mission plan table and outputs the ground track summary sheet and a magnetic tape. The magnetic tape contains the data that the Ground Track Plotter Program reformats for use by the CalComp plotter.

Inputs required:

- a. State vector
- b. Time interval and duration
- c. Maneuver specification

Outputs required:

The outputs are displayed on the ground track summary sheet and the ground track plot shown in Figures 2 and 3, respectively.

4.2.1.3 Relative motion: Given the state vectors of the active and passive vehicle, current mission plan, time interval, and REFSMMATS, determine the relative motion quantities (gimbal and/or FDAI angles, range, range rate, azimuth, elevation, curvilinear position coordinates, and corresponding g. e. t.).

This requirement is satisfied in earth or lunar orbit by the ARRS program. In translunar or transearth coast and atmospheric entry, the requirement is satisfied by the ARMACR program. The ARRS program has the capability to calculate gimbal angles and FDAI angles for any selected active vehicle axis directed toward the passive vehicle, whereas, ARMACR will calculate gimbal angles only for the active vehicle positive X-axis directed toward the passive vehicle.

Inputs required:

- a. State vectors for both vehicles
- b. Time increment and interval
- c. Axis to be pointed at the passive vehicle (ARRS only)
- d. REFSMMATS
- e. Active vehicle
- f. Maneuver specification

Outputs required:

The outputs required are displayed on the relative motion summary sheet shown in Figure 4.

4.2.1.4 Spacecraft daylight darkness: Given a state vector, time interval, and any maneuvers to be performed during the interval, determine the spacecraft time and position of sunrise, sunset, terminator rise, and terminator set. Compute the spacecraft attitude to point the X-axis at the sun.

This requirement is satisfied with the Work Schedule Program in conjunction with the ARMACR program. The ARMACR program generates the standard ephemeris tape (Section 4.2.1.1) from which the Work Schedule Program computes the required quantities. The capability also exists in the ARRS program to output a daylight darkness table.

Inputs required:

- a. State vector
- b. Time increment and interval
- c. Maneuver specification

Outputs required:

The outputs required are displayed on the spacecraft daylight-darkness summary sheet shown in Figure 5. The output from the ARRS program is displayed on the spacecraft daylight-darkness table summary sheet shown in Figure 6.

4.2.1.5 Spacecraft moonsighting: Given a state vector, time interval, and any maneuvers to be performed during the interval, determine the time and spacecraft position of moonrise and moonset.

This requirement is satisfied by the Work Schedule Program and the ARMACR program. The ARMACR program generates the standard ephemeris tape (Section 4.2.1.1) from which the Work Schedule Program computes the required quantities.

Inputs required:

- a. State vector
- b. Time increment and interval
- c. Maneuver specification

Outputs required:

The output quantities are displayed on the moonsighting summary sheet shown in Figure 7.

4.2.1.6 Landmark sighting: Given the landmark number, time interval to perform the landmark search, and a state vector, determine the following quantities: the spacecraft acquisition and loss time, slant range and azimuth at acquisition, minimum slant range, maximum elevation angle, and duration of the pass.

This requirement is satisfied by the Work Schedule Program in conjunction with the ARMACR program. The ARMACR program generates the standard ephemeris tape (Section 4.2.1.1) from which the Work Schedule Program computes the required quantities.

Inputs required:

- a. State vector
- b. Time interval
- c. Maneuver specification
- d. Landmark number

Outputs required:

The output quantities are displayed on the landmark sighting summary sheet shown in Figure 8.

4.2.1.7 Spacecraft starsighting: Given a star identification number, revolution number, and a state vector, determine the times of star-rise and starset relative to the spacecraft and the time and position in which the star-earth-spacecraft central angle is a minimum (closest approach).

This requirement is satisfied by the Work Schedule Program in conjunction with the ARMACR program. The ARMACR program generates the standard ephemeris tape (Section 4.2.1.1) from which the Work Schedule Program computes the required quantities.

Inputs required:

- a. State vector
- b. Time interval
- c. Star identification number
- d. Maneuver specification

Outputs required:

The output quantities are displayed in the starsighting summary sheet shown in Figure 9.

4.2.1.8 Pointing data: Given a state vector, target identification, REFSMMAT, and time interval to perform the target search, determine the spacecraft to target look angles (gimbal angles and local vertical/local horizontal angles) and the target to spacecraft look angles (elevation angle and azimuth angle). Also, compute the spacecraft acquisition and loss times, maximum elevation angle, minimum slant range, altitude, and elapsed time of the pass.

This requirement is satisfied by the Work Schedule Program in conjunction with the ARMACR program. The ARMACR program generates the standard ephemeris tape (Section 4.2.1.1) from which the Work Schedule Program computes the required quantities.

Inputs required:

- a. State vector
- b. Target identification
- c. Maneuver specification

Outputs required:

The output quantities are displayed on the pointing data summary sheet shown in Figure 10.

4.2.1.9 Closest approach: Determine the spacecraft closest approach to a specified ground target, given target identification, revolution number, and a state vector.

This requirement is satisfied by the Work Schedule Program in conjunction with the ARMACR program. The ARMACR program generates the standard ephemeris tape (Section 4.2.1.1) from which the Work Schedule Program computes the required quantities.

Inputs required:

- a. State vector
- b. Time interval
- c. Target identification
- d. Maneuver specification

Outputs required:

The output quantities are displayed on the closest approach summary sheet shown in Figure 11.

4.2.1.10 Telescope data: Compute trajectory parameters required by various optical telescopes to acquire and track the CSM/LM. The output will be in the form of punched cards for transmission through an IBM 066/068 Card Transceiver.

This requirement will be fulfilled using the ARMACR program and the data can be output with any ARMACR processor.

Inputs required:

- a. State vector
- b. Time the data will be required

Outputs required:

- a. Vehicle identification
- b. GMT (hr:min:sec)
- c. ECI right ascension
- d. ECI declination
- e. ECI radius vector magnitude
- f. Selenographic latitude and longitude
- g. MCI radius vector magnitude

4.2.1.11 Spacecraft-to-sun alignment: Determine the CSM attitude so the liquid waste dump nozzle, the electrical power system radiator, and the environmental control system radiators receive optimum heating from the sun.

This requirement is satisfied by a monitor system processor. The processor computes the spacecraft gimbal angles necessary to expose a specified area of the spacecraft to direct sunlight for heating purposes.

Inputs required:

- a. Area of spacecraft to be exposed to the sun
- b. REFSMMAT
- c. Right ascension and declination of the sun

Outputs required:

IMU gimbal angles required to orient the specified areas to the sun

4.2.1.12 Checkout monitor: Given a state vector and a time, propagate the state vector to the specified time and output the data in the checkout monitor format.

This requirement is fulfilled by the ARMACR program.

Inputs required:

- a. State vector
- b. Time

Outputs required:

The output is displayed on the checkout monitor summary sheet shown in Figure 12.

4.2.1.13 Radio transmission time delay: Given a state vector, time of radio transmission, and radar station identification, determine the radio transmission delay.

This requirement is satisfied by the ARMACR program.

Inputs required:

- a. State vector
- b. Time
- c. Radar station identification

Outputs required:

The radio transmission time delay is output on the radar tracking summary sheet shown in Figure 13.

4.2.1.14 CSM horizon alignment check: Determine the inertial measurement unit inner gimbal angle required to align a horizon alignment mark on the CM window to the horizon at a selected time, given the spacecraft state vector, REFSMMAT, and the vehicle yaw and roll angles.

The CSM horizon alignment processor which utilizes the ARMACR program, will accept the middle and outer CM gimbal angles and compute the inner gimbal angles required to align the vehicle to a horizon alignment mark on the CM window.

Inputs required:

- a. State vector
- b. REFSMMAT
- c. Horizon monitor attitude
- d. Time of computation
- e. Spacecraft weight
- f. IMU roll and yaw gimbal angles

Outputs required:

IMU inner gimbal angle at the specified time

4.2.1.15 LM horizon alignment check: Determine the LM IMU outer gimbal angle (yaw) required to align the LM Z-axis in the local vertical plane at a selected time, given the spacecraft inner and middle gimbal angles. Also determine the position of the horizon on the landing point designator set of lines on the LM window.

This requirement is fulfilled by using an option of the Apollo Generalized Optics Program (AGOP) in conjunction with the ARMACR program. The ARMACR program generates the standard ephemeris tape (Section 4.2.1.1) from which the AGOP accepts the LM IMU inner and middle gimbal angles and computes the outer gimbal angles required to align the Z-body axis in the local vertical/local horizontal plane.

Inputs required:

- a. State vector
- b. REFSMMAT

- c. Inner and middle LM IMU gimbal angles
- d. Time of alignment

Outputs required:

- a. Yaw (outer) gimbal angle
- b. Position of horizon on landing point designator

The outputs are displayed on the optical support table (OST) summary sheet shown in Figure 14.

4.2.1.16 Docking alignment: Determine the LM REFSMMAT and the LM or CSM gimbal angles in the docked CSM/LM configuration.

The docking alignment processor which utilizes the ARMACR program has three options: option 1 computes the LM REFSMMAT, option 2 computes the LM attitude (gimbal angles and FDAI angles), and option 3 computes the CSM attitude (gimbal angles).

Inputs required:

Option 1:

- a. CSM REFSMMAT
- b. CSM gimbal angles
- c. Docking angle
- d. LM gimbal angles or FDAI angles

Option 2:

- a. CSM REFSMMAT
- b. LM REFSMMAT
- c. CSM gimbal angles
- d. Docking angle

Option 3:

- a. CSM REFSMMAT
- b. LM REFSMMAT
- c. LM gimbal angles or FDAI angles
- d. Docking angle

Outputs required:

The outputs required for this processor are displayed on the docking alignment processor summary sheet shown in Figure 15.

4.2.1.17 Attitude for preferred REFSMMAT: Given a vehicle REFSMMAT and a preferred REFSMMAT, determine the gimbal angles required for the vehicle REFSMMAT which would define 0, 0, 0 gimbal angles for the preferred REFSMMAT.

This requirement can be fulfilled using the ARMACR program or the AGOP program. In practice, either program accepts the current and desired REFSMMATS, computes the gimbal angles for the current REFSMMAT, which define 0, 0, 0 gimbal angles for the preferred REFSMMAT and outputs a set of gimbal angles and FDAI angles which used in conjunction with the current REFSMMAT defines the spacecraft attitude necessary to switch to the preferred REFSMMAT and read 0, 0, 0 gimbal angles.

Inputs required:

- a. Current REFSMMAT
- b. Preferred REFSMMAT

Outputs required:

LM gimbal angles and FDAI angles. The outputs are displayed on the REFSMMAT to REFSMMAT conversion summary sheet shown in Figure 16.

4.2.1.18 FDO orbit and space digitals: Compute and display in the proper format the Flight Dynamics Officer orbit and space digitals.

The FDO orbit digitals and space digitals processors, which are routes through the ARRS program, are used to satisfy this requirement. These displays allow evaluation of state vectors in any phase of the mission.

FDO orbit digital processor takes a state vector and computes orbital parameters associated with the given state vector at a specified time. Orbital parameters are displayed for the subsequent apoapsis and periapsis.

Inputs required:

- a. State vector
- b. Threshold time

Outputs required:

The FDO orbit digital summary sheet is presented in Figure 17.

The space digitals processor computes and provides parameters necessary to evaluate and monitor trajectories involving earth-moon relationships. The space digitals displays orbital parameters for a specified time (g. e. t.) or a specified maneuver. These orbital parameters are computed at the input vector or maneuver time, at the next apoapsis after vector or maneuver time, at the closest approach point to the moon, at the node defined by the planes of the approach hyperbola and the desired lunar parking orbit, at entry interface, and at vacuum perigee. The space digitals processor, also, computes the time (g. e. t.) that the spacecraft will pass through the earth and lunar spheres of influence.

Inputs required:

State vector

Outputs required:

The space digital summary sheet is presented in Figure 18.

4.2.1.19 Passive thermal control attitude: Determine CSM attitude such that the X-body axis is perpendicular to the sun and earth line of sight. Also determine the earth-moon plane alignment and the middle gimbal angle constrained scan.

This requirement is fulfilled by the PTC processor which utilizes the ARMACR program. The object of the PTC mode is to hold the X-body axis as close to perpendicular to the sun line of sight as possible while rolling at approximately one revolution per hour to satisfy thermal constraints. While in the PTC mode, communication with the earth must be maintained and the middle gimbal angle must not exceed some specified value.

First, the processor aligns the X-body axis perpendicular to the sun line of sight and the Z-body axis as close as possible to the radius vector from the spacecraft to the earth. Then for this alignment, the gimbal angles for the input REFSMMAT and the REFSMMAT for zero gimbal angles are calculated.

The earth-moon plane alignment is computed next by aligning the X-body axis perpendicular to the earth-moon plane and the Z-body axis as close as possible to the radius vector from the spacecraft to the earth. The REFSMMAT for zero gimbal angles is then computed.

The middle gimbal angle constrained scan is generated finally by setting the inner gimbal angle to zero, the middle gimbal angle to the constrained value, and the outer gimbal angle such that the angle between the Z-body axis and the radius vector to the earth is a minimum. The inner gimbal angle is incremented by two degrees and the alignments are recomputed until the full 360-degree cone of alignments are generated. The middle gimbal angle is then negated and the complimentary cone of alignments is generated.

Inputs required:

- a. State vector
- b. REFSMMAT
- c. Middle gimbal angle
- d. Desired time of computation

Outputs required:

- a. Gimbal angles
- b. Preferred REFSMMATS

The outputs for this processor are displayed on the PTC summary sheet shown in Figure 19.

4.2.1.20 Maneuver evaluation: Given a preburn and postburn state vector, burn profile, maneuver REFSMMAT, and an impulsive maneuver guess time, determine the actual maneuver that was performed by the spacecraft.

This requirement is satisfied by a combination of the maneuver evaluation processor which utilizes the ARMACR program and the MPT processor which is one route through the ARRS program. The maneuver evaluation processor determines the time of minimum range between the preburn and postburn state vectors and transfers the two state vectors to the MPT processor via an interface tape. A finite burn is then fitted to the two state vectors by the MPT processor such that the state vector after the finite burn corresponds to the postburn state vectors at the time calculated by the maneuver evaluation processor.

Inputs required:

- a. Preburn and postburn state vectors
- b. REFSMMAT
- c. Impulsive maneuver guess time
- d. Thrust profile

Outputs required:

The output quantities are displayed on the detailed maneuver table (DMT) summary sheet shown in Figure 20.

4.2.1.21 ESM look angles: Determine the look angles (aspect and cone angles) to the ESM at a specified time.

This requirement is satisfied by the ARMACR program.

Inputs required:

- a. State vector
- b. Body attitude
- c. Time of computation

Outputs required:

The outputs are displayed on the ESM look angle summary sheet shown in Figure 21.

4.2.1.22 True of date vector generation: Given any state vector and a specified time, determine the true of date state vector.

This requirement is satisfied by the ARMACR program.

Inputs required:

- a. State vector
- b. Time of computation

Outputs required:

The outputs are displayed on the true of date vector conversion summary sheet shown in Figure 22.

4.2.1.23 Radar tracking and summary: Determine the time history of the radar station parameters (azimuth, elevation, slant range, and the associated rates) and also display the summary of this time history. The summary will include acquisition and loss time, slant range and azimuth at acquisition, minimum slant range, maximum elevation angle, and the duration of the pass.

Radar tracking and summary are determined using any of the ARMACR processors. Also, the ARRS program has the ability to determine station acquisition and loss time, maximum elevation angle, and duration of the pass.

Inputs required:

- a. State vector
- b. Station identification number
- c. Time increment and duration

Outputs required:

The outputs are displayed on the radar tracking summary sheet and the radar summary sheet shown in Figures 23 and 24.

4.2.2 Optical sighting and antenna pointing. - The optical equipment aboard the CM consists of the following instruments: a scanning telescope, a sextant, and a boresight. The scanning telescope and sextant are interconnected with equipment common to the inertial and computer subsystems to form the primary onboard optical navigational subsystem. The scanning telescope is a single line-of-sight, unit power, wide field instrument. It is used for landmark tracking, in conjunction with IMU attitude reference, as an acquisition instrument for the sextant, for backup alignment of the inertial attitude sensors, and as a general viewing instrument. The sextant is a highly accurate, dual line-of-sight, 28-power telescope with a 1.8-degree field of view. The sextant is used primarily to determine star to landmark or horizon angle navigation measurements, to compute star measurements for IMU alignment, and as a high power general viewing instrument.

The boresight is not part of the optical navigation subsystem and has no coupling with either the inertial or computer subsystems. It is used as a general viewing instrument and as a backup for IMU alignment if the optical subsystem should fail. The boresight is a compact, low magnification, 5-degree field-of-view telescope which has a null position line of sight parallel to the spacecraft roll axis.

Two additional instruments make up the optical equipment onboard the LM: the AOT and the COAS. The AOT is a unity power telescope with a 60-degree field of view and has three sighting (detent) positions; the 0-degree position is in the X-Z body plane and 45 degrees above the Z axis. The other positions are 60 degrees to the right (+60) or left (-60) of the 0-degree position. When the CSM and LM are undocked, three other positions are available. They are rear facing (grouped about the -Z axis) and spaced as previously described. The function of the AOT is to supplement the rendezvous radar by measuring azimuth and elevation angles to stars for alignment of the LM IMU stable member. The COAS is a 10-degree field-of-view instrument with only one degree of freedom. However, it may be placed on the side of the commander in either the overhead window (X-axis mount) or the forward window (Z-axis mount).

4.2.2.1 CMC sextant IMU alignment: Determine the CMC REFSMMAT, given two stars, their location in the sextant field-of-view, and the corresponding spacecraft IMU gimbal angles.

This requirement is fulfilled by the AGOP in conjunction with the ARMACR program. The ARMACR program produces the standard ephemeris tape (Section 4.2.1.1) from which AGOP computes the required quantities.

Inputs required:

- a. State vector
- b. Spacecraft attitude in IMU gimbal angles
- c. Two-star identification
- d. The sextant shaft and trunnion angles of each star
- e. Time of star sighting

Outputs required:

REFSMMAT associated with the new IMU alignment

The outputs are displayed on the optical support table 2 (OST-2) summary sheet shown in Figure 25.

4.2.2.2 CMC COAS IMU alignment: Determine the CMC REFSMMAT, given two stars, their location in the COAS field of view, and the corresponding spacecraft IMU gimbal angles.

This requirement is fulfilled by the AGOP in conjunction with the ARMACR program. The ARMACR program produces the standard ephemeris tape (Section 4.2.1.1) from which AGOP computes the required quantities.

Inputs required:

- a. State vector
- b. Spacecraft attitude (CSM IMU gimbal angles or FDAI angles)
- c. Spacecraft axis along which the COAS is mounted
- d. COAS reticle elevation angle for each star
- e. COAS reticle X-position angle for each star
- f. Time of the two-star sightings
- g. Two-star identifications

Outputs required:

REFSMMAT associated with the new alignment

The outputs are displayed on the OST-2 summary sheet shown in Figure 25.

4. 2. 2. 3 LGC AOT IMU alignment: Determine the LGC REFSMMAT, given two stars, their location in the alignment optical telescope field of view, and the corresponding spacecraft IMU gimbal angles.

This requirement is fulfilled by the AGOP in conjunction with the ARMACR program. The ARMACR program produces the standard ephemeris tape (Section 4. 2. 1. 1) from which AGOP computes the required quantities.

Inputs required:

- a. State vector
- b. Spacecraft attitude (LM IMU gimbal angles or FDAI angles)
- c. AOT detent position (0, +60, and -60 degrees)
- d. Time of the two-star sightings
- e. AOT A-1 (cursor) and A-2 (spiral) angles of each star
- f. Axis ($\pm X$, $\pm Y$) on the AOT lens from which to measure the cursor angle
- g. Two-star identifications

Outputs required:

REFSMMAT associated with the new alignment

The outputs for this processor are displayed on the OST-2 summary sheet shown in Figure 25.

4. 2. 2. 4 LGC COAS IMU alignment: Determine the LGC REFSMMAT, given two stars, their location in the crew optical alignment sight field of view, and the corresponding spacecraft IMU gimbal angles.

This requirement is fulfilled by the AGOP in conjunction with the ARMACR program. The ARMACR program produces the standard ephemeris tape (Section 4. 2. 1. 1) from which AGOP computes the required quantities.

Inputs required:

- a. State vector
- b. Spacecraft attitude (LM IMU gimbal angles or FDAI angles)
- c. Spacecraft axis along which the COAS is mounted
- d. COAS reticle elevation angle for each star

- e. COAS reticle X-position angle for each star
- f. Time of the two-star sightings
- g. Two-star identifications

Outputs required:

REFSMMAT associated with the new alignments

The outputs are displayed on the OST-2 summary sheet shown in Figure 25.

4.2.2.5 CSM star search: Given the current CMC IMU alignment, the current spacecraft IMU gimbal angles, and a search interval, locate up to 10 stars that will be in the telescope field of view and the position of these stars with respect to the telescope recticle pattern as well as the star acquisition of sight and loss of sight.

This requirement is fulfilled by the AGOP in conjunction with the ARMACR program. The ARMACR program produces the standard ephemeris tape (Section 4.2.1.1) from which AGOP computes the required quantities.

Inputs required:

- a. State vector
- b. REFSMMAT
- c. IMU gimbal angles
- d. Time interval to be searched

Outputs required:

Acquisition and loss times of the stars found

The outputs are displayed on the OST-1 summary sheet shown in Figure 14.

4.2.2.6 LM star search: Given the current LGC IMU alignment, the vehicle REFSMMAT, and a search interval, find up to 10 stars that will be in the LM optics field of view during the search interval as well as the AOS and LOS of each star. The optics used, AOT or COAS, and their detent position or axis orientation must also be defined.

This requirement is fulfilled by the AGOP in conjunction with the ARMACR program. The ARMACR program produces the standard ephemeris tape (Section 4.2.1.1) from which AGOP computes the required quantities.

Inputs required:

- a. State vector
- b. REFSMMAT
- c. IMU gimbal angles or FDAI angles
- d. Instrument to be used for sighting
- e. Time interval to be searched
- f. Detent position if AOT is used
- g. Axis along which COAS is mounted (if used)

Outputs required:

Acquisition and loss times of the stars found

The outputs are displayed on the OST-1 summary sheet shown in Figure 14.

4.2.2.7 Star location: Determine the optics angles for the CSM sextant, the CSM COAS, the LM AOT, or the LM COAS to center each star in the appropriate field of view, given the star identifications, the current REFSMMAT, and the spacecraft attitude.

This requirement is fulfilled by the AGOP in conjunction with the ARMACR program. The ARMACR program produces the standard ephemeris tape (Section 4.2.1.1) from which AGOP computes the required quantities.

Inputs required:

- a. State vector
- b. Desired optical system
- c. Star identifications
- d. REFSMMAT
- e. Spacecraft attitude

Outputs required:

Optics angles

The outputs are displayed on the OST-1 summary sheet shown in Figure 14.

4.2.2.8 CSM celestial target sighting (fixed sextant): Given a celestial target location, REFSMMAT, and a fixed sextant configuration, determine the spacecraft IMU gimbal angles required to center the target in the sextant field of view. Also, compute the central angle and time of closest approach, the time of arrival at the line of sight to the target, and the earliest point at which the line of sight does not pass through the atmosphere of the earth.

This requirement is fulfilled by the AGOP in conjunction with the ARMACR program. The ARMACR program produces the standard ephemeris tape (Section 4.2.1.1) from which AGOP computes the required quantities.

Inputs required:

- a. State vector
- b. REFSMMAT
- c. Target identification or location
- d. Sextant shaft and trunnion angles

Outputs required:

- a. IMU gimbal angles
- b. Time of arrival at the line of sight
- c. Minimum target-earth-spacecraft central angle (closest approach)
- d. Time of closest approach
- e. Right ascension and declination of the line of sight
- f. Earliest subsatellite longitude at which the line of sight does not pass through the atmosphere of the earth

The outputs are displayed on the starsighting table (SST) summary sheet shown in Figure 26.

4.2.2.9 CSM celestial target sighting (movable sextant): Determine the sextant shaft and trunnion angles for viewing a celestial target, given a celestial target location, spacecraft attitude and state vector.

This requirement is fulfilled by the AGOP in conjunction with the ARMACR program. The ARMACR program produces the standard ephemeris tape (Section 4.2.1.1) from which AGOP computes the required quantities.

Inputs required:

- a. State vector
- b. Target identification or location
- c. Spacecraft attitude

Outputs required:

- a. Sextant shaft and trunnion angles
- b. Time of arrival at the line of sight
- c. Minimum target-earth-spacecraft central angle (closest approach)
- d. Time of closest approach
- e. Right ascension and declination of the line of sight
- f. Earliest subsatellite longitude at which the line of sight does not pass through the atmosphere of the earth

The outputs are displayed on the SST summary sheet shown in Figure 26.

4. 2. 2. 10 LM celestial target sighting (fixed AOT): Given the spacecraft attitude, the AOT detent position, and the time of sighting, determine the right ascension and declination of the center of the field of view of the AOT at the time of sighting.

This requirement is fulfilled by the AGOP in conjunction with the ARMACR program. The ARMACR program produces the standard ephemeris tape (Section 4. 2. 1. 1) from which AGOP computes the required quantities.

Inputs required:

- a. State vector
- b. REFSMMAT
- c. Time of sighting
- d. IMU gimbal angles
- e. AOT detent position

Outputs required:

- a. Right ascension and declination of the center of the AOT field of view
- b. Central angle

The outputs for this processor are displayed on the SST summary sheet shown in Figure 26.

4.2.2.11 LM celestial target sighting (movable AOT): Given the right ascension and declination of the target point, the AOT detent position, and time of sighting, determine the IMU gimbal angles of the spacecraft at the time of sighting.

This requirement is fulfilled by the AGOP in conjunction with the ARMACR program. The ARMACR program produces the standard ephemeris tape (Section 4.2.1.1) from which AGOP computes the required quantities.

Inputs required:

- a. State vector
- b. REFSMMAT
- c. Time of sighting
- d. Right ascension and declination of the celestial target point
- e. AOT detent position

Outputs required:

- a. IMU gimbal angles
- b. Central angle

The outputs are displayed on the SST summary sheet shown in Figure 26.

4.2.2.12 Ground target sighting (fixed sextant): Determine the spacecraft IMU gimbal angles, time of arrival at the desired line of sight to the target, and the time and central angle of closest approach, given the target location, REFSMMAT, and desired sextant configuration.

This requirement is fulfilled by the AGOP in conjunction with the ARMACR program. The ARMACR program produces the standard ephemeris tape (Section 4.2.1.1) from which AGOP computes the required quantities.

Inputs required:

- a. State vector
- b. REFSMMAT
- c. Target identification or location
- d. Elevation angle of line of sight to the target
- e. Sextant shaft and trunnion angles

Outputs required:

- a. IMU gimbal angles
- b. Time of arrival at the desired line of sight to the ground target
- c. Central angle and time of closest approach

The outputs are displayed on the SST summary sheet shown in Figure 26.

4.2.2.13 Ground target sighting (movable sextant): Determine the required sextant shaft and trunnion angles for viewing a ground target, time of arrival at the desired line of sight to the target, and the time and central angle of closest approach, given the target location, spacecraft attitude, and desired sextant configuration.

This requirement is fulfilled by the AGOP in conjunction with the ARMACR program. The ARMACR program produces the standard ephemeris tape (Section 4.2.1.1) from which AGOP computes the required quantities.

Inputs required:

- a. State vector
- b. REFSMMAT
- c. Target identification or location
- d. Elevation angle of line of sight to target
- e. Spacecraft attitude

Outputs required:

- a. Shaft and trunnion angles
- b. Time of arrival at the desired line of sight

c. Central angle and time of closest approach

The outputs are displayed on the SST summary sheet shown in Figure 26.

4.2.2.14 Point AOT with CSM: Given the CMC REFSMMAT, the docking angle, and a state vector, determine the CMC gimbal angles required to center a specified star in the AOT field of view at a specified time. The detent position of the AOT must also be specified.

This requirement is fulfilled by the AGOP in conjunction with the ARMACR program. The ARMACR program produces the standard ephemeris tape (Section 4.2.1.1) from which AGOP computes the required quantities.

Inputs required:

- a. State vector
- b. CSM REFSMMAT
- c. Search interval
- d. AOT detent position
- e. Star identification
- f. Docking angle

Outputs required:

CSM IMU gimbal angles at the time of star sighting

The outputs are displayed on the SST summary sheet shown in Figure 26.

4.2.2.15 Cislunar navigation: Given the CSM REFSMMAT, state vector, and landmark and/or star identifications, determine the inertial attitude which will align the CSM optical system in the null position, to the earth or moon horizon or some specified landmark on the earth or moon. Also determine the sextant shaft and trunnion angle of the stars.

This requirement is fulfilled by an AGOP option in conjunction with the ARMACR program. The ARMACR program produces the standard ephemeris tape (Section 4.2.1.1) from which AGOP computes the required quantities.

The normal mode of this AGOP option computes a roll, pitch, and yaw associated with the REFSMMAT such that only a vehicle pitch or optics trunnion angle change is necessary. The Z-axis (or 0 optics axis) is pointed at the horizon or landmark, the X-axis is then fixed in the near

horizon-star-spacecraft plane; this fixes the Y-axis position. If the Y-axis position is such that gimbal lock has been attained, then it is necessary to run this AGOP option in the pseudo mode.

In the pseudo mode a yaw angle is input (which should be the maximum allowable while avoiding gimbal lock). The Z-axis or zero optics is pointed at the horizon or landmark, the Y-axis is then placed at the input yaw angle, thus the X-axis position is determined (not in the horizon-star-spacecraft plane). If the maximum allowable yaw angle is close to that obtained in the normal mode, then the attitudes achieved in each mode will be nearly identical.

Inputs required:

- a. State vector
- b. CSM REFSMMAT
- c. Landmark and/or star(s)
- d. Desired yaw for pseudo method

Outputs required:

- a. IMU gimbal angles
- b. Sextant shaft and trunnion angles

The outputs are displayed on the cislunar navigation summary shown in Figure 27.

4. 2. 2.16 Steerable antenna data: Given the spacecraft attitude, state vector, REFSMMAT, and time of sighting of a specified earth target, determine the pitch and yaw angles of any one of the onboard antennas necessary to point the antenna at a specified earth target. Alternately, given the pitch and yaw angles of the antenna, determine the spacecraft attitude necessary to point the antenna at the selected target.

This requirement is fulfilled by the AGOP in conjunction with the ARMACR program. The ARMACR program produces the standard ephemeris tape (Section 4. 2. 1. 1) from which AGOP computes the required quantities.

The CSM antenna is the deep-space or high-gain (also S-band) antenna; the LM antenna is a steerable S-band antenna. Both antennas are gimballed in pitch and yaw. This processor will accept a state vector and spacecraft attitude (either CSM or LM) and compute the pitch and yaw angles necessary to point the specified antenna at a specified ground based target site. This computation also outputs the related azimuth and elevation angles required for the ground station to acquire the spacecraft. Also, the capability exists for fixing the position (pitch and yaw) of the

specified antenna and computing the necessary spacecraft IMU gimbal angles for pointing the antenna at the specified ground station.

Inputs required for movable antenna option:

- a. State vector
- b. Vehicle identification
- c. Instrument identification
- d. Site identification or location (longitude, geodetic latitude, and altitude)
- e. Spacecraft gimbal angles
- f. REFSMMAT
- g. Minimum elevation angle at which the site can acquire the spacecraft

Inputs required for fixed antenna option:

- a. Vehicle identification
- b. Instrument identification
- c. Site identification or location (longitude, geodetic latitude, and altitude)
- d. Instrument attitude (pitch and yaw)
- e. Minimum elevation angle at which the site can acquire the spacecraft

Outputs required for the movable and fixed antenna option:

The outputs required for this processor are displayed on the steerable antenna pointing summary sheet shown in Figure 28.

4.2.3 Systems. - The systems programs will be used to update the consumables profiles and the CSM and LM mass properties to reflect this consumption as well as the result of vehicle reconfigurations. Eight programs are employed in the RTACF to generate these data. One program computes the mass properties of the CSM/LM for a specified configuration and can compute the CM trim aerodynamics for entry. Three programs are used to determine the RCS propellant status on either the CSM or LM, two programs monitor the electrical power system (EPS) of the CSM or LM, one program is for diagnosis of the LM telemetry system, and the other program monitors the supercritical helium pressure in the descent propulsion system.

4.2.3.1 Mass properties and aerodynamics: Given the weights, center of gravity, mixture ratio, moments of inertia of the consumables tanks, and any miscellaneous items to be considered, determine any of the following: aerodynamics for CM entry, center of gravity location of either the LM, CM, or CSM, mass properties tables for different vehicle configurations, and DAP command load.

This requirement is fulfilled by the Mass Properties Program which has four options that can be exercised. These options are the CM entry aerodynamics, CM, CSM, LM, or CSM/LM center-of-gravity locations, mass properties, and DAP command loads.

The aerodynamics option computes the trim aerodynamic coefficients of the CM as a function of Mach number.

Inputs required:

- a. Weight of present CM configuration
- b. X, Y, and Z components of the center of gravity

Outputs required:

Aerodynamics for CM entry

The outputs required are displayed on the aerodynamics update summary sheet shown in Figure 29.

DAP command load: This option computes those mass properties required for uplink to the DAP. The quantities are converted to the proper units and octal format acceptable to the CMC or LGC DAP programs.

Inputs required:

- a. Dry weight of CM, SM, and LM ascent and descent stages
- b. Weight of consumables
- c. CM, SM, ascent stage, and descent stage center-of-gravity locations and moments of inertia
- d. SPS and DPS thrust levels
- e. Weight, center-of-gravity location, and moments of inertia of any items to be considered

Outputs required:

- a. Mass properties
- b. Weight

- c. Pitch trim angles
- d. Yaw trim angles

The outputs required are displayed on the DAP command load summary sheet shown in Figure 30.

4.2.3.2. SM RCS predicted propellant profile: Determine the complete SM RCS propellant budget, given the spacecraft mass properties, the control mode for each maneuver, the RCS jet selection, and a timeline of maneuvers.

This requirement is fulfilled by the Mass Properties, RCS/SPS (MRS) Program which will generate a complete RCS propellant budget using premission data supplied for individual maneuver propellant consumption and internally computed mass properties characteristics. During the mission, as propellant is expended and vehicle configuration modified, the RCS portion of the program accepts inputs from the mass properties portion for use in its computations. In addition to mass properties, the RCS portion uses a form of flight time-line which is input by the user and fixed data which are stored in the program.

Inputs required:

- a. Spacecraft weight
- b. Total RCS propellant
- c. Spacecraft mass properties
- d. Timeline of mission events

Outputs required:

- a. Spacecraft mass properties
- b. Propellant remaining, after each event, per quad

The outputs are displayed on the MRS summary sheet shown in Figure 31.

4.2.3.3 SM RCS propellant status: Determine the current SM RCS propellant available using the primary or auxiliary system, given the quad selection, the corresponding helium pressures and temperatures, the tank expulsion efficiencies, and the RCS oxidizer to fuel mixture ratios.

The Pressure Volume Temperature (PVT) Program satisfies this requirement by determining the amount of SM RCS oxidizer and fuel remaining in each tank and how much of this can be considered usable

propellant. From telemetered values of helium temperature and pressure, the program employs the gas equation to determine the volume of helium used to pressurize the fuel-oxidizer system. Once the volume of helium is determined in each tank, the amount of fuel or oxidizer is computed from the known total volume of each tank. The amount of usable propellant is then determined from the oxidizer-to-fuel mixture ratio being used and the expulsion efficiency of each tank.

Inputs required:

- a. Volume of each oxidizer, fuel, and helium tank
- b. Volume of all connecting lines
- c. Initial pressure and temperature of the helium tank
- d. Initial weight of fuel and oxidizer in each tank
- e. Helium source pressure and temperature at the time of the PVT calculation
- f. Fuel and oxidizer manifold pressure
- g. Oxidizer-to-fuel mixture ratio
- h. Weight of oxidizer and fuel remaining in the passive system (primary or auxiliary)
- i. Expulsion efficiency of each tank

Outputs required:

- a. Weight of fuel expended (primary and auxiliary)
- b. Weight of oxidizer expended (primary and auxiliary)
- c. Weight of fuel remaining (primary and auxiliary)
- d. Weight of oxidizer remaining (primary and auxiliary)
- e. Weight of usable propellant

The outputs are displayed on the PVT summary sheet shown in Figure 32.

4.2.3.4 LM RCS predicted propellant profile: Determine the LM RCS propellant consumables budget, given spacecraft weight, propellant weight, and a mission event timeline.

The LM RCS consumables program fulfills this requirement by generating LM RCS propellant consumption budgets in response to input mission event timelines. The RCS propellant consumption data and mass properties data used to compute the budgets are stored in the program preflight, but can be updated in real-time.

Inputs required:

- a. Total LM weight
- b. Total CSM weight
- c. Spacecraft configuration
- d. LM RCS, DPS, APS propellant
- e. Mission event timeline

Outputs required:

- a. Event
- b. Time of the event
- c. Spacecraft weight
- d. Propellant remaining in systems a and b

The outputs required are displayed on the LM RCS propellant budget summary sheet shown in Figure 33.

4. 2. 3. 5 DPS supercritical helium pressure profile: Determine the pressure profile of the DPS supercritical helium fuel pressure system for each DPS maneuver. The profile will include the maximum helium pressure, the helium pressure at the end of a burn, the helium mass remaining at the end of a burn, and a tabulation of pressure versus time during the coast period prior to the maneuver, during the maneuver, and for a period of time following the maneuver.

The SHe program satisfies this requirement by monitoring the LM DPS SHe propellant tank pressurization system. This system provides pressure for the DPS engine fuel feed. As fuel is used by the engine, it is replaced by helium that is stored as a high-density gas in the supercritical state. The helium pressure is maintained by increasing the temperature in its storage bottle as the fuel is used.

Heat is transferred from the fuel to the helium bottle by means of heat exchangers. This program accepts the mission profile, consisting of various burns and coast phases, and computes the corresponding helium pressure resulting from heat transfer between the systems.

Inputs required:

- a. Total propellant weight
- b. Initial helium weight
- c. Initial helium pressure
- d. LM dry weight
- e. CSM total weight
- f. Pressure rise rate
- g. Burn thrust versus time profile

Outputs required:

- a. Maximum helium tank pressure
- b. Helium tank pressure at the end of the burn
- c. Helium mass remaining at the end of the burn

The outputs are displayed on the SHe summary sheet shown in Figure 34.

4.2.3.6 LM electrical power system profile: Given the premission LM power distribution network parameters (on magnetic tape), the mission timeline, and any new component configuration, determine the capability of the LM EPS to support the mission.

The Spacecraft Electrical Energy Network Analysis (SEENA) Program is used to satisfy this requirement. The program will be used to determine the capability of the LM EPS to support the various phases of the mission. It determines: (1) the total energy drain on the LM power supply for any configuration of the onboard electrical equipment and (2) the remaining energy available based on an event timeline. This timeline of chronological events defines changes in the EPS switching status which in turn defines the total load on the system. The premission data (power distribution network, battery characteristics, and power requirements of components) are stored internally, and as these components are added to or removed from the circuit by on-line card input to the program, new total energy drain and the resultant energy available are computed by the program and are output as printed data of consumables (energy consumed, battery charge status) and steady state conditions (components ON, voltages, currents, etc.) of the EPS circuit. In addition, tapes may be written (optional) for data plotting and for thermal data to be input to an environmental control system (ECS) computer program.

Inputs required:

- a. Battery voltage, current, and capacity
- b. Equipment configuration
- c. Equipment loads
- d. Mission timeline

Outputs required:

- a. Total current used
- b. Total ampere-hours used

The outputs are displayed on the LM SEENA summary sheet shown in Figure 35.

4.2.3.7 LM telemetry diagnostics: Determine the common failure points in the telemetry downlink system, given the code number of the failed telemetry points.

The LM Telemetry Diagnostics Program will be used to fulfill this requirement. It will accept a list of onboard sensors that have detected failures on the LM and output a summary sheet report that designates the common failure mode of the list of input parameters.

Inputs required:

List of sensors

Outputs required:

Common failure mode

The outputs are displayed on the LM telemetry summary sheet shown in Figure 36.

4.2.3.8 CSM cryogenic electrical power: Given the electrical power load, the temperature and pressure of oxygen and hydrogen, compute the electrical power profile and cryogenic profiles for oxygen and hydrogen.

The CSM Cryogenic Electrical Power (CRYOPWR) Program will satisfy this requirement by analytically simulating the temperature, the pressure, and the quantity remaining in the oxygen and hydrogen tanks as a function of an electrical load profile and an oxygen profile for metabolic needs. The CRYOPWR program provides the user the capability of determining the heater cycling frequency and the resultant change in the electrical load profile due to the current drawn by the heaters. The program also provides a prediction of the remaining quantities of cryogenics.

Inputs required:

- a. Power load
- b. Oxygen temperature
- c. Hydrogen temperature
- d. Oxygen pressure
- e. Hydrogen pressure

Outputs required:

- a. Time history of the current used
- b. Time history of the temperature of oxygen
- c. Time history of the temperature of hydrogen
- d. Time history of the pressure of oxygen
- e. Time history of the pressure of hydrogen

4.2.4 Command load. - The RTACF possesses the capability of generating command loads in octal with the proper scaling and format to be directly uplinked to the CMC or LGC. The capability also exists for receiving certain down-linked quantities from the CMC or LGC and converting them to the appropriate engineering units. A program was developed to perform these special conversions, as well as any number of general conversions from engineering units to octal or from octal to engineering units. In addition, a processor was developed to generate CMC, LGC, AGS, or S-IVB state vector updates at a given time from a state vector.

4.2.4.1 CMC uplink data: Given a set of data in engineering units to be uplinked to the CMC, determine the octal equivalent of these data in the format and scaling acceptable to the CMC. Conversion of the following sets of data will be required: REFSMMAT, orbital external ΔV data, and deorbit external ΔV data.

This requirement is satisfied with a monitor system processor.

Inputs required for a REFSMMAT update:

Elements of REFSMMAT

Outputs required:

The outputs required are displayed on the REFSMMAT update summary sheet shown in Figure 37.

Inputs required for an orbital external ΔV update:

- a. Maneuver time
- b. External ΔV components
- c. Maneuver weight

Outputs required:

The outputs required are displayed on the orbital external ΔV summary sheet shown in Figure 38.

Inputs required for deorbit external ΔV update:

- a. Maneuver time
- b. External ΔV components
- c. Maneuver weight
- d. Latitude and longitude of the target

Outputs required:

The outputs required are displayed on the deorbit external ΔV summary sheet shown in Figure 39.

4.2.4.2 LGC uplink data: Given a set of data in engineering units to be uplinked to the LGC, determine the octal equivalent of these data in the format and scaling acceptable to the LGC. Conversion of the following sets of data will be required: navigation vector update, REFSMMAT, and orbital external ΔV data.

This requirement is satisfied with a monitor system processor.

Inputs required for a REFSMMAT update:

Elements of REFSMMAT

Outputs required:

The outputs required are displayed on the REFSMMAT update summary sheet shown in Figure 37.

Inputs required for an orbital external ΔV update:

- a. Maneuver time
- b. External ΔV components
- c. Maneuver weight

Outputs required:

The outputs required are displayed on the orbital external ΔV summary sheet shown in Figure 38.

4.2.4.3 Engineering units/octal conversion: Given the octal scale factor and precision, convert either engineering units to octal or octal to engineering units.

This requirement is satisfied with a monitor system processor. The processor also has the capability to convert from one set of engineering units to another set by specifying the multiplier.

Inputs required:

- a. Number to be converted (octal or engineering units)
- b. Scale factor
- c. Octal precision
- d. Multiplier (if necessary)

Outputs required:

The outputs required are displayed on the general octal conversion summary sheet shown in Figure 40.

4.2.4.4 Navigation vector updates (CMC, LGC, AGS and IU): Given a state vector and navigation update time, output the state vector at the update time in engineering units in either the CMC, LGC, or S-IVB format. For the Abort Guidance System the output will be engineering units only.

This requirement is satisfied by any of the ARMACR processors.

Inputs required:

- a. State vector
- b. Time of navigation vector update

Outputs required:

The outputs will be displayed on either the CMC/LGC or S-IVB navigation vector update summary sheet shown in Figures 41 and 42, respectively. The AGS navigation update summary sheet is shown in Figure 43.

4.2.5 Miscellaneous. - The processors in this section could be used for any phase of the mission and could not be grouped into any particular category.

4.2.5.1 PAO data: Display in proper format quantities associated with mission events for the PAO.

This requirement is satisfied with any ARMACR processor. The inputs are dependent on what data quantities were requested, and the output is displayed on the PAO data summary sheet shown in Figure 44.

4.2.5.2 Earth illuminance and sun impingement on the CSM optics: Determine the amount of reflected earth-light in lumens per square foot on the scanning telescope of the CM. Also determine if sunlight will impinge on the CM telescope.

This requirement will be satisfied by the illuminance program in conjunction with the ARMACR program. The ARMACR program generates the standard ephemeris tape (Section 4.2.1.1) and the illuminance program then determines the amount of reflected earth-light which will impinge on the scanning telescope of the CM for specified attitudes of the CM and shaft angle of the CM optics. A message will also be output if sunlight will impinge on the optics at the specified attitude.

Inputs required:

- a. State vector
- b. Time interval and duration
- c. Maneuver specifications
- d. Shaft angle of the scanning telescope

Outputs required:

Illuminance on the scanning telescope and a message if sunlight impinges on the CSM optics.

4.2.5.3 Vehicle gimbal angles conversion: Given the lift-off gimbal angles of the IU, the CM, and the LM IMU, determine the three sets of gimbal angles, at a later time, prior to the transposition and docking, given any one of the three sets.

This requirement is satisfied by a monitor system processor which will accept the lift-off gimbal angles of the IU, the CM IMU, and the LM IMU and construct matrices relating the three. Any one of the three sets may be input at a later time, prior to transposition and docking. The remaining two sets will be output based on the original matrices constructed.

Inputs required:

- a. IU gimbal angles at lift-off
- b. CM IMU gimbal angles at lift-off
- c. LM IMU gimbal angles at lift-off
- d. Any one of the three sets at a later time

Outputs required:

Three sets of gimbal (IU, CM IMU, LM IMU) angles

4.2.5.4 LM gimbal angles/FDAI angles conversion: Given a set of either FDAI angles or LM gimbal angles, determine the corresponding set of LM gimbal angles or FDAI angles.

This monitor system processor accepts either FDAI angles or LM gimbal angles and outputs the corresponding angles based on a fixed relationship between the two sets of angles. This fixed relationship is defined by a set of equations, not a rotation matrix.

Inputs required:

A set of either FDAI angles or LM gimbal angles

Outputs required:

The corresponding LM gimbal angles or FDAI angles

4.2.5.5 Radiation evaluation: Given a state vector and time interval of the required computation, determine the geomagnetic parameters, the radiation dose rates (REM per hour), and the cumulative radiation dose (REMS) in the CM and LM.

This requirement is satisfied by the radiation program in conjunction with the ARMACR program. The ARMACR program produces the standard ephemeris tape (Section 4.2.1.1) and the radiation program computes the required quantities.

Inputs required:

- a. State vector
- b. Time interval and duration
- c. Maneuver specifications

Outputs required:

The outputs will be displayed on the radiation evaluation summary sheet shown in Figure 45.

4.2.5.6 Solar activity: Given solar flare data transmitted from the radio and optical telescopes in the Solar Particle Alert Network, reduce the data to obtain graphs of the radio frequency burst profile and particle density as a function of time in the vicinity of the earth-moon system.

This requirement is fulfilled by the SPAN program. It will process solar flare data received from the optical and radio telescopes in the SPAN. The input data will be received in the form of a punched paper tape which is generated at the space environment console. The SPAN processor will first transfer these data from the paper tape to a magnetic tape. It will then process the magnetic tape and display the resulting data in a form from which the radiation hazard in the vicinity of the earth can be determined.

4.2.5.7 State vector coordinate transformation: The state vector coordinate transformation requirements are as follows:

- a. Conversions of a launch pad inertial state vector to an ECI Besselian state vector
- b. Conversion of an AGS state vector (stable member) to an ECI Besselian state vector
- c. Conversion of a Besselian MCI state vector to a Besselian ECI state vector

The state vector coordinate transformation requirements can all be fulfilled with any ARMACR processor with the exception of the AGS state vector to ECI Besselian state vector. The AGS state vector conversion must be executed with a program on the monitor system.

Inputs required:

- a. State vector
- b. Vector coordinate system
- c. REFSMMAT (AGS conversion only)

Outputs required:

State vector in ECI Besselian coordinate system

4.2.5.8 Flight control work schedule: Display, in graphical form, mission and trajectory related events that occur in a specified interval of time during the mission. Also, any mission anomalies or alternate procedures that might develop must be reflected in the display.

This requirement is satisfied by the Work Schedule Program in conjunction with any of the ARMACR processors. The ARMACR program produces the standard ephemeris tape (See Section 4.2.1.1) from which the work schedule processor computes the required quantities and produces a plot tape. This plot tape is then converted to the work schedule format.

Inputs required:

- a. State vectors
- b. Maneuver specification
- c. Time duration
- d. Time scale
- e. Current statements
- f. Tracking station identification
- g. Computed events required

Outputs required:

- a. Time of radar station acquisition and loss
- b. Sunrise, sunset, moonrise, and moonset
- c. Current statements
- d. Computed events
 1. Time, geodetic latitude and longitude, and altitude of apogee and perigee
 2. Altitude in nautical miles above a spherical earth at specified times
 3. Ascending node times and right ascension
 4. Revolution number

The work schedule display is shown in Figure 46.

4.2.5.9 RTCC restart conversion: Convert a numeric or alpha-numeric onboard spacecraft state vector to an RTCC state vector.

This requirement is satisfied with a monitor system processor. The output is displayed on the command load navigation update summary sheet shown in Figure 47.

4.2.5.10 Camera exposure data: There is no information available for the camera exposure data requirement at the time of this writing.

4.2.5.11 Model data: Compute the following quantities to be used for the Apollo mission simulation model: right ascension and declination of the sun, earth, moon, and velocity vector; the position vector in nautical miles and the velocity vector in nautical miles per hour; the time in g.e.t., GMT, and Central Standard Time; and the current REFSMMAT and gimbal angles. All the previous mentioned quantities are for both the LM and CSM. In addition, determine the LM FDAI angles.

This requirement will be satisfied by the model data processor which utilizes the ARMACR program. There is no additional information available at the time of this writing for the model data processor.

4.2.5.12 Star availability: Determine the stars and the time they will be available for sighting by the spacecraft.

This requirement will be satisfied by the Lunar Trajectory Propagation Orion Program.

Inputs required:

- a. State vector
- b. Time interval

Outputs required:

The outputs will be a listing of available stars.

4.3 Mission Phase Related Processors

4.3.1 Prelaunch and launch. - Prelaunch is the interval prior to lift-off that the RTACF is required to support. The launch phase is the interval from lift-off to earth parking orbit insertion. The Mode I abort processor is run prelaunch, while the lift-off REFSMMAT and estimated nodal targets processors are run as soon as lift-off time and the flight azimuth have been established.

4.3.1.1 Mode I launch abort: Determine the spacecraft landing points should an abort be necessary during the first 90 seconds of powered flight based on wind profiles from 1 to 24 hours prior to launch.

This requirement is satisfied by the Mode I launch abort processor which utilizes the General Electric Missile and Satellite Multi-Vehicle Program (GEMMV). The Mode I abort region is defined from lift-off until the launch escape tower (LET) jettison which occurs 15 seconds after S-IVB ignition. The Mode I landing area is completely contained in the Atlantic continuous recovery area (ACRA), which extends downrange 3200 nautical miles along the flight azimuth.

The actual wind profiles encountered from 1 to 24 hours prior to launch are employed in this processor to predict the spacecraft landing points should a Mode I abort be performed during the first 90 seconds of the flight. Selected abort times are considered in this time interval to determine which aborts will result in a land impact about the launch area.

Inputs required:

- a. Wind profile from the Kennedy Space Center
- b. CM entry weight

Outputs required:

The outputs required for this processor will be displayed on the Mode I launch-abort summary sheet shown in Figure 48.

4.3.1.2 Lift-off REFSMMAT: Determine the lift-off REFSMMAT, given the flight azimuth and time of guidance reference release.

This requirement is satisfied by an ARMACR processor. The lift-off REFSMMAT is calculated such that the Z-stable member axis is aligned with the negative geocentric radius vector of the launch site, the X-stable member axis is aligned along the launch azimuth, and the gimbal angles are zero.

Inputs required:

- a. Flight azimuth
- b. Time of spacecraft guidance reference release

Outputs required:

The outputs required will be a lift-off REFSMMAT.

4.3.1.3 Estimated nodal targets: Determine the estimated integrated nodal targets (height of the node above the lunar landing site, latitude and longitude of the node in the earth-moon plane system, and time of the node). Also, compute the nominal post-TLI state vector in classical orbital elements.

This requirement is satisfied with the midcourse maneuver processor which utilizes the Real-Time Forward Iterator Program (RTFIP).

Inputs required:

- a. Launch azimuth
- b. Injection opportunity

Outputs required:

The outputs are displayed on the midcourse tradeoff summary sheet shown in Figure 49.

4.3.2 Earth orbit, - The processors in this section are used for the earth-orbit phase, which is the interval from earth orbit insertion to trans-lunar injection. The processors could also be used if a contingency developed that resulted in an earth-orbit alternate mission.

4.3.2.1 Earth general purpose maneuvers and earth rendezvous planning: This processor is the same as the lunar general purpose maneuver and lunar rendezvous planning processors presented in Sections 4.3.5.1 and 4.3.5.3, respectively.

The requirement to maintain earth capability on Apollo 10 is necessary only in case of some contingency, prior to TLI, that requires execution of an earth orbit alternate mission.

4.3.2.2 Orbital lifetime: Determine lifetime of the vehicle (CSM, S-IVB, or LM) in days, hours, and minutes, g.e.t., given a state vector and vehicle aerodynamics.

This requirement is satisfied by the Orbital Lifetime Program.

Inputs required:

- a. State vector
- b. Atmosphere model to be used

Outputs required:

Lifetime in days, hours, and minutes from lift-off and from the time of the vector.

4.3.2.3 K-factor: Determine the atmospheric density K-factor, given the vehicle weight, drag coefficient, effective aerodynamics cross-sectional area, and two or more state vectors.

This requirement can be satisfied by two processors, a monitor system processor or the K-factor processor which utilizes the ARMACR program. The K-factor, which is an atmospheric density multiplier, is determined by propagating one input state vector to the time of one or more succeeding state vectors. The value of the K-factor is adjusted until the propagated vector and succeeding state vectors agree to some specified accuracy.

Inputs required:

- a. Two or more state vectors for the same vehicle
- b. Spacecraft weight
- c. Spacecraft drag coefficient and reference area

Outputs required:

- a. Value of K-factor
- b. Probable error in K-factor

4.3.2.4 S-IVB venting simulation: Given a state vector and a time, simulate the S-IVB venting and calculate any requested trajectory parameters.

This requirement is satisfied by the S-IVB venting processor which utilizes the ARMACR program.

Inputs required:

- a. State vector
- b. S-IVB venting model
- c. Time

Outputs required:

State vector propagated through the S-IVB venting interval and any requested trajectory parameters.

4.3.2.5 Computed events: Determine the orbital events (apogee, perigee, ascending node, and revolution number) and related times of these events, given an initial state vector, time interval to be considered, and any maneuvers to be considered during the interval.

This requirement is satisfied by the Work Schedule Program in conjunction with any of the ARMACR processors. The ARMACR Program produces the standard ephemeris tape (Section 4.2.1.1) from which the Work Schedule Program computes the required quantities.

Inputs required:

- a. State vector
- b. Time duration
- c. Maneuver specifications

Outputs required:

- a. Revolution number and g.e.t. of Cape Crossing
- b. GMT and g.e.t. of apogee and perigee
- c. Geodetic latitude and longitude of apogee and perigee
- d. Apogee and perigee altitude referenced to spherical earth
- e. GMT and g.e.t. of the ascending node
- f. Right ascension of the ascending node
- g. Longitude of the ascending node

The outputs are displayed on the computed events summary sheet shown in Figure 50.

4.3.3 Earth deorbit. - The deorbit processors in this section would only be used for an earth orbit alternate mission.

4.3.3.1 Primary landing area: Determine the deorbit ignition time and the time to reverse bank angle from the initial bank angle in order to achieve a target latitude and longitude in one of the primary landing areas. The maneuver attitude, ΔV and entry profile will be specified. Also, it may be necessary to include a separation maneuver prior to a deorbit which is at a specified time or a fixed delta time before deorbit ignition.

This requirement is satisfied by an option of the Apollo Block Data Program (ABDP). This program option employs an iterative technique to determine the SPS or RCS deorbit ignition time and time to reverse bank to land in a PLA. The deorbit maneuver is determined by the following quantities: an initial vehicle attitude can be input with respect to the local vertical/local horizontal (LV/LH) system or aft-looking line of sight to the horizon. The attitude may also be specified by indicating the initial thrust vector attitude with respect to the LV/LH system. The spacecraft orientation, during the maneuver, is maintained inertial by either the stabilization and control system (SCS) or external ΔV guidance mode. The maneuver thrust may be terminated after a specified value of one of the following conditions has been satisfied; an incremental velocity change, external ΔV onboard targets, a burn duration, or a velocity and flight-path angle constraint at entry interface. The entry profile consists of the following sequence: a constant lift vector orientation to a specified g-level, a constant bank angle to the time to reverse bank, and the negative of the previous bank angle to drogue chute deployment.

Inputs required:

- a. State vector
- b. CM weight at entry
- c. Deorbit SPS or RCS maneuver data
- d. Entry profile
- e. Latitude and longitude of target
- f. Separation maneuver specification if necessary

Outputs required:

- a. Event times
- b. Burn quantities
- c. Entry quantities
- d. Landing points

The outputs are displayed on the ABDP deorbit summary sheet shown in Figure 51.

4.3.3.2 Contingency landing area: Determine the deorbit ignition time to achieve a target longitude in one of the CLA's, given the maneuver attitude, ΔV , and entry profile. It may be necessary to include a separation maneuver prior to a deorbit which is at a specified time before deorbit ignition.

This requirement is satisfied by either the CLA processor utilizing the ARMACR program or an option of the ABDP. Either CLA processor employs an iterative technique to determine SPS or RCS deorbit ignition time to hit a target longitude using a fixed entry profile. The entry profile consists of a specified lift vector orientation from entry interface to a specified g-level, followed by a constant bank angle to drogue chute deployment. The deorbit maneuver is determined by the following quantities: an initial attitude, a guidance mode, and a terminating value for the maneuver. The initial vehicle attitude can be input with respect to the LV/LH system or aft-looking line of sight to the horizon. The attitude may also be specified by indicating the initial thrust vector attitude with respect to the LV/LH system. The spacecraft orientation, during the maneuver, is maintained inertial by either the SCS or external ΔV guidance mode. The maneuver thrust may be terminated after a specified value of one of the following conditions has been satisfied: an incremental velocity change, external ΔV onboard targets, a burn duration, or a velocity and flight-path angle (V - γ) constraint at entry interface.

Inputs required:

- a. State vector
- b. CM weight at entry
- c. Deorbit SPS or RCS maneuver data
- d. Entry profile
- e. Longitude of target
- f. Separation maneuver specifications, if necessary

Outputs required:

- a. Event times
- b. Burn quantities
- c. Entry quantities
- d. Landing points

The outputs for the ABDP are displayed on the ABDP deorbit summary sheet shown in Figure 51. The outputs for the ARMACR program are displayed on the standard deorbit summary sheet shown in Figure 52.

4.3.3.3 Block data: Generate block data which consists of contingency retrofire information for a period of anywhere from four to eight revolutions. This retrofire data may include SPS or SM RCS PLA's and

CLA's as well as SM RCS apogee deorbit for each revolution of the block. The state vector for each deorbit solution computed should reflect any orbital maneuvers which are currently planned during the interval for which the block of deorbit data is to be generated.

This requirement will be satisfied by the ABDP which uses an iterative technique to determine deorbit solutions to achieve a target landing point. In the execution of the ABDP, an orbital maneuver ephemeris is first established which reflects up to eight orbital maneuvers. Each orbital maneuver may be defined by specifying the main engine ignition time, propulsion system, guidance mode, and spacecraft attitude and incremental velocity or external ΔV components. For each deorbit solution to be computed, the following quantities must be specified: spacecraft attitude, propulsion system, guidance mode, incremental deorbit velocity or V- γ constraint at entry interface, entry profile, and target landing point. The specific inputs required to execute the ABDP are described in the Apollo Block Data Program User's Manual - Revision 1 (Reference 7). The output data from the ABDP are displayed on the three summary sheets shown in Figures 51, 53, and 54.

4.3.3.4 Hybrid deorbit: Determine the SM and CM RCS deorbit ignition times in order to achieve a target longitude, given the incremental velocities and spacecraft attitudes for each maneuver, REFSMMAT, and an entry profile consisting of a lift vector orientation to a specified g-level, followed by a zero-lift entry to drogue chute deployment. Also determine the landing point based on the actual hybrid deorbit performed.

This requirement will be satisfied by the hybrid deorbit processor which utilizes the GEMMV program. The hybrid deorbit processor employs an iterative scheme to determine the RCS ignition times in order to achieve a target longitude. The SM RCS burn duration is based on a fixed incremental velocity to be realized by the SM RCS burn. The spacecraft maintains an inertial attitude which is specified in the LV/LH plane at the centroid of the hybrid deorbit. A 60-second coast between the SM and CM RCS burns allows time to perform the CM/SM separation and the reorientation of the CM for the CM RCS maneuver. The CM attitude is maintained so that the thrust vector alignment is the same as the SM RCS thrust vector alignment. During entry, a specified lift vector orientation is maintained to a fixed g-level, followed by a specified lift entry profile to drogue chute deployment.

The processor can also be used to evaluate the actual hybrid deorbit performed. The SM RCS maneuver is assumed to have been performed nominally while the CM RCS maneuver is defined by the actual incremental velocities achieved. The processor remains essentially unchanged except that the SM RCS ignition time and actual velocity increments are input and the landing point is determined.

Inputs required:

- a. State vector
- b. CM weight at entry
- c. Deorbit maneuver data
- d. Entry profile
- e. Longitude of target

Outputs required:

- a. Event times
- b. Burn quantities
- c. Entry quantities
- d. Landing points

The outputs are displayed on the standard deorbit summary sheet shown in Figure 52.

4.3.3.5 Hybrid deorbit without S-IVB/CSM separation: Given a state vector before S-IVB/CSM separation, the time of initiation of the LOX dump maneuver, and the CMC REFSMMAT, determine the CM RCS deorbit, ΔV at apogee, and the IMU gimbal angles at ignition to achieve a target longitude. The entry profile will nominally consist of a lift vector down to a 1-g deceleration followed by a rolling entry to drogue chute deployment.

This requirement is satisfied by the hybrid deorbit processor which utilizes the ARMACR program. This hybrid deorbit processor uses an iterative technique to determine the CM ΔV required to achieve the target longitude, given the ignition time and ΔV of the LOX dump maneuver and the time of the CM burn initiation (apogee). The LOX dump maneuver will be performed in the constant orbital rate mode which maintains a constant pitch attitude to the local horizontal. After the LOX dump maneuver, the CM separates from the SM/S-IVB and assumes a thrust vector attitude which is specified in the LV/LH plane at the centroid of the hybrid deorbit. The CM RCS maneuver is performed by maintaining the thrust vector attitude inertial.

Inputs required:

- a. State vector
- b. S-IVB/CSM weight
- c. CM weight at entry
- d. REFSMMAT or IMU gimbal angles
- e. Time of LOX dump initiation
- f. Entry profile
- g. Longitude of target
- h. Duration of LOX dump maneuver
- i. Apogee at which CM burn is to be performed

Outputs required:

- a. Event times
- b. Burn quantities
- c. Entry quantities
- d. Landing points
- e. Time of CM RCS ignition
- f. ΔV of CM RCS maneuver

The outputs of this processor will be displayed on the standard summary sheet shown in Figure 52.

4.3.4 Post-TLI through LOI-1 (translunar phase). - The processors in this section will be used after the TLI maneuver has been performed through the first lunar orbit insertion maneuver.

4.3.4.1 TLI plus 10 minutes fixed attitude abort: Determine the abort ΔV , gimbal angles, and terminator reference angles for a fixed attitude, time critical, unspecified area, (TLI plus 10 minutes abort, given an initial state vector, REFSMMAT, time of TLI cutoff (or ignition time), velocity and flight-path angle constraint at entry, initial pitch and yaw trim angles of the CSM, and the pitch angle below the line of sight to the far horizon. This type of abort is generated during the initial earth parking orbit to verify or update the onboard crew charts which are determined preflight.

This requirement is satisfied by the earth referenced return-to-earth abort program, TERRA.

Inputs required:

- a. State vector
- b. TLI cutoff time or actual ignition time
- c. Minimum and maximum landing time
- d. Maximum ΔV available
- e. Propulsion system
- f. Ullage specification (zero, two, or four jet)
- g. Vehicle configuration
- h. REFSMMAT
- i. Entry interface V- γ target line
- j. Type of precision entry desired
- k. Gimbal angles or specified pitch angle below line of sight to the horizon if attitude is specified
- l. Desired aerodynamics and entry weight
- m. Initial pitch and yaw trim angles for the CSM

Outputs required:

- a. Ignition time
- b. Landing time
- c. True anomaly at ignition
- d. Roll, pitch, yaw at ignition, LV/LH attitudes
- e. Roll, pitch, yaw at ignition, platform gimbal angles
- f. V_c , ΔV required along the X-body axis
- g. Burn time, including ullage and tailoff
- h. V_T , total ΔV of the maneuver, including ullage and tailoff

- i. ΔT , ullage duration
- j. Time of entry interface
- k. Entry interface latitude and longitude
- l. Entry interface inertial velocity and flight-path angle

The outputs for this processor will be displayed on the abort scan table and return-to-earth digitals summary sheet shown in Figures 55 and 56.

4.3.4.2 CSM attitude for docking maneuver: Given the CSM and LM REFSMMAT and the gimbal angles for the docking maneuver, determine the CSM attitude for the docking maneuver.

This requirement is satisfied by using two separate processors. The first step requires the execution of the gimbal angle conversion processor (Section 4.2.5.3) to obtain the LM gimbal angles from the IU gimbal angles. The second step requires the execution of the docking alignment processor (Section 4.2.1.16) to obtain the CSM attitude from the CSM and LM REFSMMAT's and the LGC gimbal angles.

Inputs required:

- a. IU, LGC, and CMC lift-off gimbal angles
- b. IU gimbal angles for the docking maneuver
- c. LGC and CMC REFSMMAT

Outputs required:

The outputs for this requirement are displayed on the docking alignment summary sheet shown in Figure 57.

4.3.4.3 TLI ΔV comparison: Display, in graphical form, the differences in the velocity components sensed by the IMU and the IU during the TLI burn.

This requirement is satisfied by the TLI ΔV comparison program. This program interprets the down-linked telemetry tapes from the RTCC and produces a plot tape suitable for the CalComp plotter. The graphs produced consist of the differences in the velocity components and the difference in total velocity which was sensed by the IU and IMU plotted against time during the TLI burn. These plots are used to evaluate the performance of the IMU and supply some of the information for the LOI-1 go/no-go decision.

Inputs required:

- a. Time of lift-off
- b. Launch azimuth
- c. Time to begin the plot
- d. Telemetry tapes containing the IU and IMU data during LOI-1

Outputs required:

The outputs are a graph of ΔV_X , ΔV_Y , ΔV_Z , and ΔV_T versus time.

4.3.4.4 Midcourse maneuver planning: Determine translunar midcourse maneuvers to produce reoptimization of the planned mission, a circumlunar flyby mission plan as a degradation of the planned mission, a fuel critical return-to-earth mission plan, and a return to the nominal mission.

This requirement is satisfied by the midcourse correction processor which utilizes RTFIP and the MPT processor. After the generation of an impulsive MCC maneuver by the midcourse correction processor, the maneuver information is transferred to the MPT processor via an interface tape. The MPT processor fits a finite burn such that the resultant orbit after the finite burn corresponds to the resultant orbit after the impulsive maneuver. The midcourse correction processor options are presented below:

- a. Free return fixed lunar parking orbit orientation best adaptive path (BAP)

This option is used to reoptimize a free-return mission to the pre-flight azimuth over the lunar landing site.

Inputs required for the midcourse correction processor:

1. State vector
2. Spacecraft weight
3. LM weight
4. SPS fuel weight
5. Launch azimuth
6. Injection opportunity
7. Ground elapsed time of midcourse maneuver
8. Thrust profile

b. Free return free lunar parking orbit orientation BAP

This option is used if the fixed lunar parking orbit orientation mode requires an excessive midcourse maneuver.

Additional inputs required:

1. Minimum selenographic approach azimuth
2. Maximum selenographic approach azimuth

c. Constrained circumlunar flyby

This option is used to verify a discrete flyby, computed by the RTCC, or to compute a return to nominal circumlunar flyby.

Additional inputs required:

1. Height of pericyynthion
2. Inclination of free return

d. Optimum circumlunar flyby

This option is used to compute fuel critical circumlunar flyby maneuvers.

Additional inputs required:

1. Maximum inclination of return
2. Minimum pericynthion altitude
3. Maximum pericynthion altitude

e. Circumlunar flyby return to desired landing longitude

This option is used to compute the fuel critical circumlunar midcourse to return to a desired earth landing longitude. Inclination of free return is incremented until the desired landing longitude is achieved.

Additional inputs required:

1. Inclination of free return
2. Minimum height of pericynthion
3. Maximum height of pericynthion

f. Return to nominal

This option is used to return to the nominal position and time of LOI (X, Y, Z, and T) as defined by the best translunar midcourse. This option will be used for small maneuvers only.

Inputs required:

There are no additional inputs required for this option.

Outputs required:

The outputs required for the midcourse correction processor are displayed on the midcourse tradeoff display and midcourse summary sheet shown in Figures 49 and 58, respectively. The outputs for the MPT processor are displayed on the detailed maneuver table summary sheet shown in Figure 20.

4. 3. 4. 5 Return to earth: Determine the ignition time, gimbal angles, abort ΔV , entry interface and landing quantities, given an initial state vector, REFSMMAT, velocity and flight-path angle constraint at entry interface, mode of abort, and maximum and minimum landing time and landing area (if any) to be considered. Also, determine the entry monitoring system initialization quantities, the guided entry profile, and backup entry quantities required to reach a target landing point given the guidance mode and desired targets.

This requirement is fulfilled for aborts in earth reference by TERRA and for aborts in moon reference by a combination of the Analytic Return to Earth Program Moon Referenced (AREPMR), the abort processor (AP) which utilizes RTFIP, and ARMACR. The processors are designed to calculate optimum time or fuel critical abort trajectories or tradeoff displays which satisfy general operational and trajectory constraints. These computed abort trajectories can either be targeted to arbitrary contingency lines (ATP) or have no landing constraints at all (unspecified area). The processors use conic solutions to initialize precision logic and include effects of an oblate earth and a finite burn simulation. In addition, the processors output pertinent maneuver quantities, entry interface quantities, landing conditions, and a state vector at entry interface which, along with other quantities, on an interface tape can be used to initialize the Apollo Reentry Simulation (ARS) Program (Section 4. 3. 7. 1).

In addition, near earth maneuver attitudes can be specified through gimbal angles, with respect to the line of sight to the near or far horizon or unconstrained in attitude.

Inputs required:

- a. State vector
- b. Ignition time (actual, impulsive, minimum or maximum)
- c. Minimum and maximum landing time
- d. Maximum ΔV available
- e. Target longitude (ATP)
- f. Propulsion system
- g. Ullage specification (zero, two, or four jet)
- h. Vehicle configuration
- i. REFSMMAT
- j. Entry interface V- γ target line
- k. Type of precision entry desired
- l. Gimbal angles or specified pitch angle below line of sight to the horizon if attitude is specified
- m. Remote earth only - specification of direct, circumlunar, or processor choice of return mode which will result in the shortest flight time
- n. Remote earth only - specification of the minimum allowable pericynthion
- o. Desired aerodynamics and entry weight

Outputs required:

- a. Ignition time
- b. Landing time
- c. True anomaly at ignition
- d. Roll, pitch, yaw at ignition, LV/LH attitudes

- e. Roll, pitch, yaw at ignition, platform gimbal angles
- f. V_c , ΔV required along the X-body axis
- g. Burn time, including ullage and tailoff
- h. V_T , total ΔV of the maneuver, including ullage and tailoff
- i. ΔT , ullage duration
- j. Resultant pericythion altitude, if maneuver is performed pre-pericynthion

The outputs for this processor will be displayed on the abort scan table and return-to-earth digitals summary sheet shown in Figures 55 and 56. Also, the atmospheric entry quantities are displayed on the ARS summary sheet shown in Figure 59.

4.3.4.6 LOI-1 maneuver planning: Given a state vector, the desired lunar parking orbit shape, the landing site location, approach azimuth, and appropriate constraints, determine LOI-1 maneuvers in the following four categories: lunar orbit shape priority, lunar landing site priority, and basic and minimum ΔV .

This requirement is fulfilled by the LOI and ARRS programs. The LOI program generates up to ten impulsive maneuvers to provide a trade-off of possible solutions. The ten possible solutions are listed below:

Lunar orbit shape priority:

- a. SVO - Minimum ΔV with landing site azimuth within acceptable limits
- b. SIN - Nearest landing site azimuth to desired azimuth within maximum ΔV constraint and within acceptable azimuth limits
- c. SIT - Closest approach to the lunar landing site (minimum wedge angle between LPO plane and nearest acceptable wedge angle) within the input ΔV .

Lunar landing site priority:

- d. AVO - Minimum ΔV with unconstrained parking orbit pericynthion
- e. AIN - Azimuth as close as possible to desired azimuth within ΔV constraint; height of pericynthion unconstrained

f. AIP - Nearest height of pericyynthion to desired, within ΔV constraint

Basic:

g. BVM - Minimum allowable azimuth

h. BVN - Desired azimuth

i. BVX - Maximum allowable azimuth

Minimum ΔV :

j. CV - Desired orbit shape, but may not pass over landing site

After the generation of the impulsive solutions, one is selected to be transferred to ARRS via an interface tape, and the ARRS MPT processor fits a finite burn such that the resultant orbit after the finite burn corresponds to the resultant orbit after the impulsive maneuver.

Inputs required:

a. CSM state vector

b. SPS thrust profile

c. Lunar parking orbit shape before LOI-1

d. Lunar parking orbit shape after LOI-1

e. Maximum, minimum, and desired azimuths of approach over lunar landing site

f. Maximum ΔV allowable

g. Time from LOI-1 to first pass over landing site

h. Lunar landing site coordinates

i. CSM dry weight

j. LM weight

k. SPS specific impulse

l. Weight changes while coasting in lunar orbit

m. Earth landing longitude

- n. Inertial entry range
- o. Time from entry to landing

Outputs required:

The outputs of the LOI program are displayed on the LOI-1 planning and LOI-1 external ΔV summary sheets shown in Figures 60 and 61, respectively. The outputs of the MPT processor are displayed on the mission planning table and detailed maneuver table summary sheets shown in Figures 20 and 62.

4.3.4.7 LOI-1 crew chart update: Given the direction in which to apply the abort ΔV relative to the earth-moon plane for the LOI-1 plus 15-minute abort, determine the gimbal angles required to align the spacecraft to perform the maneuver. In addition, calculate the abort velocity increment necessary to satisfy the flight-path angle constraint at entry interface for selected premature LOI-1 shutdown times.

The LOI-1 plus 15-minute abort mode will be employed in the event of an impending SPS failure during the LOI-1 maneuver. The abort mode assumes that the SPS is still operable but only for a relatively short period. This mode of failure necessitates an immediate return-to-earth abort which is scheduled to occur at 15 minutes after LOI-1 ignition. The abort maneuver is specified to the Apollo crew by the attitude in gimbal angles at which to do the maneuver, the time of the maneuver (15 minutes after LOI ignition), and the abort ΔV from an onboard chart of velocity as a function of LOI-1 burn time.

The RTACF is required to compute the attitude and ΔV of the abort to determine whether the planned LOI-1 burn will necessitate an update of the onboard crew charts. If an update is necessary, the RTACF will supply the abort gimbal angles and the ΔV of the maneuver for several LOI-1 burn times to enable the crew to construct an updated chart.

This requirement is satisfied by the ARMACR LOI-1 crew chart update processor. It determines the CSM attitude for the abort by aligning the thrust vector in the direction of the abort ΔV vector at LOI-1 ignition. The CSM attitude is then computed using average estimated trim angles at the abort maneuver time. The processor then simulates the abort maneuver, which maintains the thrust vector inertial, after a premature LOI-1 cutoff and determines the abort ΔV needed to attain a specified flight-path angle at entry interface.

Inputs required:

- a. State vector
- b. REFSMMAT
- c. Abort thrust vector direction

- d. LOI-1 ignition time
- e. LOI-1 external ΔV 's
- f. Premature LOI-1 shutdown times
- g. Estimated trim angles at the abort time
- h. Flight-path angle constraint at entry interface

Outputs required:

- a. Gimbal angles at abort time
- b. ΔV of abort

4.3.4.8 LOI-1 monitor: Following the targeting of the LOI-1 maneuver after a midcourse correction, simulate the maneuver and determine the effect of a 10-degree drift about the pitch IMU axis on the height of pericynthion. The 10-degree drift represents the maximum allowable drift before a manual takeover by the Apollo crew. If the dispersion causes an unacceptable pericynthion altitude, another MCC will be made.

During the LOI-1 maneuver, the Apollo crew will manually takeover control of the burn if the IMU drift exceeds 10 degrees. The LOI-1 maneuver must, therefore, be planned such that a manual takeover at the 10-degree drift attitude will result in an acceptable altitude (greater than 20 nautical miles) at pericynthion. The LOI-1 monitor processor was developed to determine the effects of both a positive and negative IMU drift upon the pericynthion altitude using the planned LOI-1 maneuver. An acceptable pericynthion altitude may result in the cancellation of any further midcourse corrections whereas an unacceptable pericynthion will dictate a further MCC.

The LOI-1 monitor processor which utilizes the ARMACR program simulates the nominal LOI-1 maneuver followed by two IMU perturbed maneuvers. The perturbation is simulated by a drift rate applied to the IMU pitch axis which results in a 10-degree pitch misalignment at the end of the burn.

Inputs required:

- a. External ΔV 's for LOI-1 maneuver
- b. Time of LOI-1 maneuver
- c. State vector
- d. REFSMMAT
- e. Delta time of burn

Outputs required:

- a. Apogee and perigee altitudes of a +10-degree drift maneuver
- b. Apogee and perigee altitudes of a -10-degree drift maneuver

4.3.4.9 Earth horizon radius: Determine the earth horizon radius (altitude above the Fisher ellipsoid) that the astronaut uses for the earth horizon reference.

This requirement will be satisfied by the Houston Operations Prediction Estimator (HOPE) Program.

Inputs required:

- a. State vector
- b. Tape containing astronauts navigation sightings

Outputs required:

The output for this processor is the height of the reference horizon above the Fisher ellipsoid.

4.3.5 Lunar operations.- The processors in this section will be used for maneuver planning for the second lunar orbit insertion maneuver through transearth injection.

4.3.5.1 General purpose maneuver: Given a state vector and a set of orbital conditions, compute impulsive maneuvers to achieve desired orbital conditions.

The general purpose maneuver processor in the ARRS program consists of 53 routines. These routines will be used to support the Apollo 10 mission. A listing of the 53 routines is presented in Table V. The specific inputs required for these routines are described in Reference 8.

Inputs required:

- a. State vector
- b. GPMP routine desired
- c. Active vehicle
- d. Desired orbital elements
- e. Burn quantities

Outputs required:

- a. Resulting apogee and perigee quantities
- b. Resulting maneuver orbital elements
- c. Resulting burn quantities

The outputs are presented on the general purpose maneuver planning display shown in Figure 63.

4.3.5.2 Lunar descent planning: Determine impulsive orbital adjustment maneuvers required to shape the lunar orbit prior to DOI and the DOI maneuver which is performed to insert the vehicle in the nominal descent orbit with a 180-degree transfer to the target pericyynthion location.

The lunar descent planning processor in the ARRS program has the capability to compute circularization, plane change, apside shaft, and dual Hohmann transfer maneuvers so that the DOI maneuver results in the nominal descent orbit with a 180-degree transfer to the target pericyynthion location. A listing of the routines is presented in Table VI.

Inputs required:

- a. Preburn state vector
- b. Maneuver sequence desired
- c. Landing site location
- d. Target pericyynthion location
- e. Altitude at an apsis
- f. Desired descent azimuth
- g. Number of dwell orbits desired between DOI and powered descent ignition.
- h. Threshold time for orbital adjustment maneuvers
- i. Threshold time for DOI maneuver

Outputs required:

- a. Orbital adjustment maneuver quantities
- b. Landing site location
- c. DOI maneuver quantities

The outputs are presented on the lunar descent planning display shown in Figure 64.

4.3.5.3 Lunar rendezvous planning: Determine the series of maneuvers required to accomplish a rendezvous plan and generate the lunar rendezvous plan table, rendezvous evaluation display, two-impulse rendezvous table (multiple solution), two-impulse rendezvous table (single solution), CSM insertion burn display, mission plan table, and the detailed maneuver table. Also, generate any other orbital maneuvers required to successfully complete the mission.

The Apollo Real-Time Rendezvous Support Program was designed to provide a dual purpose tool for both mission planning and rendezvous mission support in the RTACF. The ARRS program is composed of a number of processors required to support the lunar mission. The processors that will be of concern to the Apollo 10 lunar rendezvous and a listing of the processors used in the nominal mission plan and the three abort modes are given below.

The nominal lunar rendezvous maneuver sequence is:

- a. LM phasing maneuver (two-impulse and terminal phase processor)
- b. LM insertion maneuver (two-impulse and terminal phase processor)
- c. LM CSI maneuver (concentric rendezvous processor)
- d. LM CDH (concentric rendezvous processor)
- e. LM TPI (two-impulse and terminal phase processor)
- f. LM TPF (two-impulse and terminal phase processor)

The powered descent initiation (PDI) abort maneuver sequence which occurs after the nominal DOI maneuver is:

- a. LM phase adjustment maneuver (docking initiation (DKI) processor)
- b. LM height adjustment maneuver (DKI processor)
- c. LM coelliptic maneuver (concentric rendezvous processor)
- d. LM TPI maneuver (two-impulse and terminal phase processor)
- e. LM TPF maneuver (two-impulse and terminal phase processor)

The CSM insertion rescue maneuver sequence which occurs after the nominal phasing maneuver is:

- a. CSM insertion maneuver (CSM insertion processor)

- b. CSM CSI maneuver (concentric rendezvous processor)
- c. CSM CDH maneuver (concentric rendezvous processor)
- d. LM TPI maneuver (two-impulse and terminal phase processor)
- e. LM TPF maneuver (two-impulse and terminal phase processor)

The Rescue 2 maneuver sequence which occurs after the nominal DOI maneuver is:

- a. CSM height adjustment maneuver (DKI processor)
- b. CSM phase adjustment maneuver (DKI processor)
- c. CSM coelliptic maneuver (DKI processor)
- d. CSM TPI maneuver (two-impulse and terminal phase processor)
- e. CSM TPF maneuver (two-impulse and terminal phase processor).

The general inputs and output for the ARRS processors for the Apollo 10 rendezvous are given below.

The docking initiation processor computes a maneuver plan, using a combination of phase and height maneuvers with the end result being that a rendezvous situation is created.

Inputs required:

- a. Active vehicle and passive vehicle state vectors
- b. Travel angle
- c. Elevation angle at TPI
- d. Phase and height offsets
- e. Phase and height adjustment maneuver point
- f. PC maneuver point
- g. Maneuver sequence control flag
- h. TPI on time or location
- i. Desired phase angle at NSR
- j. Initial counter-line and counter-line time
- k. Initial and final rendezvous counter-line

Outputs required:

The outputs are displayed on the lunar rendezvous plan table and the rendezvous evaluation display summary sheets shown in Figure 65 and 66, respectively.

The concentric rendezvous (CSI-CDH) processor computes a rendezvous plan by using the concentric flight plan logic.

Inputs required:

- a. Active and passive vehicle state vectors
- b. Travel angle
- c. Elevation angle at TPI
- d. Height offset
- e. CSI time
- f. CDH on time or location
- g. TPI on time or location

Outputs required:

The outputs are displayed on the lunar rendezvous plan table and the rendezvous evaluation display summary sheets shown in Figure 65 and 66, respectively.

The two-impulse and terminal phase (NCC-NSR and/or TPI-TPF) processor computes a set of maneuvers by specifying when the maneuvers should be performed and by specifying the conditions, such as phase and height offsets, at the final maneuver point.

Inputs required:

- a. Active and passive vehicle state vectors
- b. Travel angle
- c. Elevation angle at TPI
- d. Phase and height offsets
- e. Two-impulse maneuver times control flag (BOFI/FIFI/SEFI)
- f. Time of NCC maneuver
- g. Time of nominal NSR maneuver
- h. NSR start time

- i. Terminal phase initiation time or location
- j. Terminal phase final time or location

Outputs required:

The outputs are displayed on the two-impulse rendezvous table (multiple solution) and the two impulse rendezvous table (single solution) summary sheets shown in Figures 67 and 68, respectively.

The CSM insertion processor computes an orbital adjustment maneuver with the result being that a rendezvous situation is created.

Inputs required:

- a. Active and passive vehicle state vectors
- b. Travel angle
- c. Elevation angle at TPI
- d. CSM insertion control flag
- e. Height offset
- f. LM insertion time
- g. Delta time between LM and CSM insertion
- h. Time of CSI maneuver
- i. TPI time or location

Outputs required:

The outputs are displayed on the CSM insertion burn summary sheet shown in Figure 69.

The MPT processor is designed to accept impulsive vectors before and after maneuvers and compute the required burn quantities to achieve the impulsive orbit after the maneuver.

Inputs required:

- 1. Before and after impulsive vectors
- 2. Active vehicle
- 3. Thrust profile and engine quantities

Outputs required:

The outputs are displayed on the FDO mission plan table and the FDO detailed maneuver table shown in Figures 20 and 62, respectively.

4.3.5.4 TEI maneuver: The processors to calculate the transearth injection maneuver is the same as the return-to-earth processors presented in Section 4.3.4.5, except only the moon referenced processor will be used.

4.3.5.5 Landing site determination and MSFN evaluation in lunar orbit: There is no information available, at the time of this writing, for the landing site determination and MSFN evaluation in lunar parking orbit processor.

4.3.6 Post-TEI through entry interface (transearth phase). - The processors in this section will be used to ensure proper entry interface conditions after the TEI maneuver.

4.3.6.1 Return to earth: The return-to-earth processor in the transearth phase is the same as the return-to-earth processor in the trans-lunar phase (Section 4.3.4.5).

4.3.6.2 Optimized midcourse: Determine the optimum ignition time, ΔV , gimbal angles, and entry interface conditions in the precision mode, given a vector during the transearth coast phase of the mission, desired inertial flight-path angle at entry interface, an estimate of the ignition time, and an approximate landing time.

The optimized midcourse processor is used to fulfill this requirement. If, in addition, the return-to-earth digitals are required in an earth-referenced midcourse maneuver, a TERRA optimized midcourse processor may be used. Both processors are designed to optimize the midcourse ΔV in the precision mode necessary to meet specified entry interface conditions landing conditions, or both. After the conic optimums are used to initialize the precision logic, the processors reoptimize in the precision mode.

Inputs required:

- a. State vector on transearth coast phase of the mission
- b. Ignition time
- c. Approximate landing time
- d. Inertial flight-path angle at entry
- e. Propulsion system
- f. Inertial downrange and crossrange

Outputs required:

- a. Ignition time
- b. Midcourse ΔV
- c. True anomaly of the maneuver
- d. Entry interface velocity and flight-path angle
- e. Entry interface latitude, longitude, and g. e. t.
- f. Landing latitude, longitude, and g. e. t.
- g. Perigee altitude
- h. Burn time
- i. External ΔV targets

The outputs are displayed on the abort scan table summary sheet shown in Figure 55.

If, in addition, return-to-earth digitals are desired:

Inputs required:

Inputs (a-f) are identical to those above with the following additions:

- g. Longitude of landing, if specified area
- h. REFSMMAT

Outputs required:

Outputs (a-i) are identical to those above with the following additions:

- j. Roll, pitch, yaw at ignition, LV/LH attitudes
- k. Roll, pitch, yaw at ignition, platform gimbal angles

The outputs are displayed on the return-to-earth digitals summary sheet shown in Figure 56.

4.3.7 Atmospheric entry. - The processors in this section will be used whenever atmospheric entry information is required.

4.3.7.1 Guided and backup entry modes: Determine entry monitoring system data, the guided entry, and backup entry quantities required to reach a target landing point, given a state vector at entry interface. The state vector at entry interface will be generated by an RTACF processor that will simulate required maneuvers, given a preburn state vector, or will propagate a postburn vector to entry interface.

The entry GNCS used for Apollo 10 ensures a safe return to the surface of the earth and provides landing point control. In order to ensure a safe return, midcourse correction, abort maneuvers, or both are performed to place the vehicle within the bounds of the operational entry corridor. Given a state vector within the entry corridor, the guidance and navigation (G&N) in the onboard computer generates the steering commands to achieve the target landing point. These steering commands or IMU gimbal angles are sampled by the onboard control system or digital autopilot and are used to update the vehicle attitude angles and rates. As a backup capability to the GNCS, the CM commander monitors the entry DSKY quantities using the EMS display, FDAI, and g-meter. In case of a GNCS failure, the EMS quantities are then used. In case of GNCS and EMS failure, the backup is a constant g-level for high speed entries or backup bank angle and time to reverse bank angle for low speed entries. A description of the guided (GNCS) and the backup entry techniques are presented below.

a. Guidance, navigation, and control system steering mode

In this steering mode, the ARS program uses the CMC entry logic to compute the entry steering commands and the DAP to simulate the entry trajectory required to achieve the target landing point.

Inputs required:

1. Steering mode option flag
2. Guidance constant LAD (maximum lift-to drag (L/D) command)
3. Guidance constant LOD (nominal L/D in final phase)
4. EMS initialization
5. Entry weight
6. Footprint option flag
7. Primary g-level
8. Target landing point

b. Constant-g steering mode

The constant-g entry consists of modulating the lift vector in a manner that results in a constant load factor profile after initial entry into the atmosphere. Initial entry into the atmosphere can be made with a lift

vector up or lift vector down attitude. The initial lift vector orientation is maintained from entry interface to the primary g-level. The lift vector is then modulated in a manner such that a constant g-level is maintained until drogue chute deployment. The constant g-level can be input or computed by the ARS program to achieve the target landing point.

Inputs required:

1. Steering mode option flag
2. Guidance constant LAD (maximum L/D command)
3. Guidance constant LOD (nominal L/D in final phase)
4. Entry weight
5. Footprint option flag
6. Primary g-level
7. Initial bank angle to primary g-level
8. Constant-g level (constant-g mode only)
9. Target landing point (constant-g iterator mode only)
10. Roll direction switch flag

c. Bank-reverse-bank steering mode

In this steering mode, an initial bank angle is maintained from entry interface to the primary g-level. The initial bank angle is then changed to the backup angle and maintained until the time to reverse bank at which time the bank angle is reversed. The reversed bank angle is flown until drogue chute deployment. The ARS program computes the necessary backup bank angle and time to reverse bank to achieve the target landing point.

Inputs required:

1. Steering mode option flag
2. Guidance constants LAD (maximum L/D command)
3. Guidance constants LOD (nominal L/D in final phase)
4. Entry weight
5. Footprint option flag
6. Primary g-level
7. Initial bank angle to primary g-level

8. Target landing point
9. Roll direction switch flag

d. Rolling steering mode

In this steering mode, the initial bank angle is maintained from entry interface to the primary g-level at which time a constant roll rate is initiated until drogue chute deployment.

Inputs required:

1. Steering mode option flag
2. Entry weight
3. Primary g-level
4. Initial bank angle to primary g-level
5. Constant roll rate

Outputs required:

1. Event times
2. EMS quantities
3. Burn quantities
4. Gimbal angles
5. Entry quantities
6. Landing points
7. Guidance support
8. Backup quantities
9. P-61 quantities

The outputs required are displayed on the ARS summary sheet shown in Figure 59.

4.3.7.2 Landing point update: Determine the entry target landing point and update the CMC prior to entry interface.

The entry target landing point processor which utilizes the ARMACR program computes the entry target landing point by integrating on the entry lift multiplier to achieve a given relative range.

Inputs required:

- a. State vector
- b. Entry aerodynamics
- c. Entry lift multiplier
- d. Relative range

Outputs required:

The outputs for this processor are the landing points corresponding to a specified relative range.

4.3.7.3 ARIA data: Determine the state vector at 425,000 feet and entry quantities for ARIA.

This requirement is satisfied with the ARMACR program in conjunction with the ARS program (Section 4.3.7.1).

Inputs required:

State vector

Outputs required:

The required ARIA data are presented in Table 7.

Table III. Standard Ephemeris Tape Record

<u>Vehicle 1</u>	
<u>Word</u>	
1	GET (sec)
2	Rev number
3	ECI Besselian X-position component (ft)
4	ECI Besselian Y-position component (ft)
5	ECI Besselian Z-position component (ft)
6	ECI Besselian $\dot{X}$ -velocity component (ft/sec)
7	ECI Besselian $\dot{Y}$ -velocity component (ft/sec)
8	ECI Besselian $\dot{Z}$ -velocity component (ft/sec)
9	Altitude above spherical earth (n mi)
10-16	Not presently used
17	Central body reference - flag 0 for earth centered, 1 for moon centered
18-26	$T_{SM\ 2I}$ - transpose of REFSMMAT
27-32	Not to be used
33	Radius magnitude (ft)
34	Right ascension of Greenwich (deg)
35	Longitude of vehicle one (deg)
36	Geodetic latitude of vehicle (deg)
37	Altitude of vehicle one above oblate earth (ft)
38	Outer gimbal angle (deg)
39	Inner gimbal angle (deg)
40	Middle gimbal angle (deg)

Table III. Standard Ephemeris Tape Record (Continued)

<u>Vehicle 2</u>	
<u>Word</u>	
41	GET (sec)
42	Rev number
43	ECI Besselian X-position component (ft)
44	ECI Besselian Y-position component (ft)
45	ECI Besselian Z-position component (ft)
46	ECI Besselian $\dot{X}$ -velocity component (ft/sec)
47	ECI Besselian $\dot{Y}$ -velocity component (ft/sec)
48	ECI Besselian $\dot{Z}$ -velocity component (ft/sec)
49	Altitude above spherical earth (n mi)
50-56	Not presently used
57	Central body reference flag - 0 for earth centered, 1 for moon centered
58-66	$T_{SM\ 2I}$ transpose of REFSMMAT
67-72	Not to be used
73	Radius magnitude (ft)
74	Right ascension of Greenwich (deg)
75	Longitude of vehicle two (deg)
76	Geodetic latitude of vehicle two (deg)
77	Altitude of vehicle two above oblate earth (ft)
78	Outer gimbal angle
79	Inner gimbal angle
80	Middle gimbal angle

Table IV. Postflight Ephemeris Tape Record

Earth Orbit			
<u>Word</u>	<u>Definition</u>	<u>Units</u>	<u>ARMACR Variable</u>
1	L/O time	sec	LTIME
2	Current GMT	sec	GMTT
3	GET current	hr	T
4	GETS	sec	ELT
5-7	Not used		
8	Geocentric latitude	deg	GLAT
9	Geodetic latitude	deg	LAT
10	Longitude	deg	LON
11	Geodetic altitude	ft	ALT
12	Inertial velocity	ft/sec	V
13	Inertial γ	deg	PTH
14	Inertial A_3	deg	AZ
15	GHA	deg	GHA
16-17	Not used		*
19	Relative velocity	ft/sec	VE
20	Relative γ	deg	PTE
21	Relative azimuth	deg	AZE
22-24	Not used	deg	*
25	Sun look angle (IMU)	deg	IALS
26	Sun look angle, φ	deg	IBTS
27-28	Not used		
29	X_i	ft	X
30	Y_i	ft	Y
31	Z_i	ft	Z
32	DX_i	ft/sec	DX
33	DY_i	ft/sec	DY
34	DZ_i	ft/sec	DZ
35-42	Not used		*

Table IV. Postflight Ephemeris Tape Record (Continued)

<u>Word</u>	<u>Definition</u>	<u>Units</u>	<u>ARMACR Variable</u>
43	X_n (S-IVB)	ft	XIM
44	Y_n	ft	YIM
45	Z_n	ft	ZIM
46	$\dot{X}_n$	ft/sec	VXIM
47	$\dot{Y}_n$	ft/sec	VYIM
48	$\dot{Z}_n$	ft/sec	VZIM
49-63	Not used		
64	$\ddot{X}_1$	ft/sec ²	DDXI
65	$\ddot{Y}_1$	ft/sec ²	DDYI
66	$\ddot{Z}_1$	ft/sec ²	DDZI
67-69	Not used		
70	Rev		REVN1
71-73	$\ddot{X}_n - \ddot{Z}_n$	ft/sec ²	CC
74	Apogee radius	ft	APO
75	Perigee radius	ft	RP
76-79	Not used		
80	Eccentricity		ECC
81	Inclination	deg	INC
82	Right ascension of ascending node	deg	RAN
83	Argument of perigee	deg	APF
84	True anomaly	deg	TA
85	Not used		
86	Declination	deg	DEC
87	Right ascension	deg	RA
88-89	Not used		
90	Eccentric anomaly	deg	EA

Table IV. Postflight Ephemeris Tape Record (Continued)

<u>Word</u>	<u>Definition</u>	<u>Units</u>	<u>ARMACR Variable</u>
91	Mean anomaly	deg	MA
92-104	Not used		
105	Total acceleration	ft/sec ²	ACC
106-130	Not used		
110-111	Earth α , β (IMU)	deg	IALE, IBTE
112-113	Moon α , β (IMU)	deg	IALM, IBTM

For Lunar Orbit

Change the following:

<u>Word</u>	<u>Definition</u>	<u>Units</u>	<u>ARMACR Variable</u>
22	Selenocentric latitude	deg	LATS
23	Selenocentric longitude	deg	LONS
24	Selenocentric radius	ft	RL
25	Selenocentric inertial velocity	ft/sec	VL
26	Selenocentric flight-path angle	deg	PTHL
27	Selenocentric azimuth	deg	AZL
74	Apogee radius	ft	APOS
75	Perigee radius	ft	RPS
80	Eccentricity		ECCS
81	Inclination	deg	INCS
82	Right ascension of ascending node	deg	RANS
83	Argument of perigee	deg	APFS
84	True anomaly	deg	TAS
86	Declination	deg	DECL

Table IV. Postflight Ephemeris Tape Record (Continued)

<u>Word</u>	<u>Definition</u>	<u>Units</u>	<u>ARMACR Variable</u>
87	Right ascension	deg	RAL
90	Eccentric anomaly	deg	EAS
91	Mean anomaly	deg	MAS
92	X_s	ft	XL
93	Y_s	ft	YL
94	Z_s	ft	ZL
95	$\dot{X}_s$	ft/sec	DXL
96	$\dot{Y}_s$	ft/sec	DYL
97	$\dot{Z}_s$	ft/sec	DZL
99	$\ddot{X}_s$	ft/sec ²	DDXI
100	$\ddot{Y}_s$	ft/sec ²	DDYI
101	$\ddot{Z}_s$	ft/sec ²	DDZI

Transearth - Translunar

Change the following:

<u>Word</u>	<u>Definition</u>	<u>Units</u>	<u>ARMACR Variable</u>
57	Sun θ	deg	IALS
58	Sun φ	deg	IBTS

Table V. Maneuvers Performed by the General Purpose
Maneuver Processor

<u>Maneuver Number</u>	<u>Maneuver Display Code</u>	<u>Definition and Required Input</u>
1	PCE	Plane change at equatorial crossing Input: DPC, threshold time
2	PCL	Plane change at a specified longitude Input: DPC, ALONG, threshold time
3	PCT	Plane change at a specified time Input: DPC, threshold time
4	CRL	Circularization at a specified longitude Input: ALONG, threshold time
5	CRH	Circularization at a specified height Input: HICIR, threshold time
6	HOL	Height maneuver at a specified longitude Input: ALONG, NM, KMLO, DHLONG, threshold time
7	HOT	Height maneuver at a specified time Input: DHLONG, NM, KMLO, thresh- old time
8	HAO	Height maneuver at apogee Input: DHLONG, NM, KMLO, thresh- old time
9	HPO	Height maneuver at perigee Input: DHLONG, NM, KMLO, thresh- old time
10	NST	Node shift maneuver at a specified time Input: SHNODE, threshold time
11	NSO	Optimum node shift maneuver Input: SHNODE, threshold time
12	HBT	Maneuver to change both apogee and perigee at a specified time Input: HAD, HPD, threshold time

Table V. Maneuvers Performed by the General Purpose
Maneuver Processor (Continued)

<u>Maneuver Number</u>	<u>Maneuver Display Code</u>	<u>Definition and Required Input</u>
13	HBH	Maneuver to change both apogee and perigee at a specified height Input: HICIR, HAD, HPD, thresh- old time
14	HBO	Optimum maneuver to change both apogee and perigee Input: HAD, HPD, threshold time
15	FCT	Input maneuver at a specified time Input: GPDELV, YAW, PITCH, NM, threshold time
16	FCL	Input maneuver at a specified longitude Input: ALONG, NM, GPDELV, YAW, PITCH, threshold time
17	FCH	Input maneuver at a specified altitude Input: HICIR, NM, GPDELV, YAW, PITCH, threshold time
18	FCA	Input maneuver at an apogee Input: NM, GPDELV, YAW, PITCH, threshold time
19	FCP	Input maneuver at a perigee Input: NM, GPDELV, YAW, PITCH, threshold time
20	FCE	Input maneuver at an equatorial crossing Input: NM, GPDELV, YAW, PITCH, threshold time
21	NHT	Combination maneuver to change both apogee and perigee and shift the node at a specified time Input: SHNODE, HAD, HPD, thresh- old time

Table V. Maneuvers Performed by the General Purpose
Maneuver Processor (Continued)

<u>Maneuver Number</u>	<u>Maneuver Display Code</u>	<u>Definition and Required Input</u>
22	NHL	Combination maneuver to change both apogee and perigee and shift the node at a specified longitude Input: ALONG, SHNODE, HAD, HPD, threshold time
23	SAL	Maneuver to shift line-of-apsides some angle at a specified longitude Input: ALF, ALONG, threshold time
24	SAA	Maneuver to shift line-of-apsides to a specified longitude Input: ALONG, threshold time
25	PHL	Combination height maneuver and a plane change at a specified longitude Input: NM, KMLO, DPC, ALONG, DHLONG, threshold time
26	PHT	Combination height maneuver and a plane change at a specified time Input: NM, KMLO, DPC, DHLONG, threshold time
27	PHA	Combination height maneuver and a plane change at an apogee Input: NM, KMLO, DPC, DHLONG, threshold time
28	PHP	Combination height maneuver and a plane change at a perigee Input: NM, KMLO, DPC, DHLONG, threshold time
29	CPL	Combination circularization maneuver and a plane change at a specified longitude Input: NM, DPC, ALONG, threshold time
30	CPH	Combination circularization maneuver and a plane change at a specified altitude Input: NM, DPC, HICIR, threshold time

Table V. Maneuvers Performed by the General Purpose
Maneuver Processor (Continued)

<u>Maneuver Number</u>	<u>Maneuver Display Code</u>	<u>Definition and Required Input</u>
31	SAT	Maneuver to shift line-of-apsides some angle at a specified time Input: ALF, threshold time
32	SAO	Maneuver to shift line-of-apsides some angle and keep the same apogee and perigee altitudes Input: ALF, KMLO, threshold time
33	HBL	Maneuver to change both apogee and perigee at a specified longitude Input: ALONG, HAD, HPD, threshold time
34	CNL	Circularization and node shift at a specified longitude Input: NM, ALONG, SHNODE, threshold time
35	CNH	Circularization and node shift at a specified height Input: NM, HICIR, SHNODE, threshold time
36	HNL	Height maneuver and node shift at a specified longitude Input: NM, KMLO, ALONG, DHLONG, threshold time
37	HNT	Height maneuver and node shift at a specified time Input: NM, KMLO, DHLONG, SHNODE, threshold time
38	HNA	Height maneuver and node shift at apogee Input: NM, KMLO, DHLONG, SHNODE, threshold time

Table V. Maneuvers Performed by the General Purpose
Maneuver Processor (Continued)

<u>Maneuver Number</u>	<u>Maneuver Display Code</u>	<u>Definition and Required Input</u>
39	HNP	Height Maneuver and node shift at perigee Input: NM, KMLO, DHLONG, SHNODE, threshold time
40	CRT	Circularization maneuver at a specified time Input: threshold time
41	CRA	Circularization maneuver at apogee Input: threshold time
42	CRP	Circularization maneuver at perigee Input: threshold time
43	CPT	Circularization and plane-change maneuver at a specified time Input: DPC, threshold time
44	CPA	Circularization and plane-change maneuver at apogee Input: DPC, threshold time
45	CPP	Circularization and plane-change maneuver at perigee Input: DPC, threshold time
46	CNT	Circularization and node shift maneuver at a specified time Input: SHNODE, Threshold time
47	CNA	Circularization and node shift maneuver at apogee Input: SHNODE, threshold time
48	CNP	Circularization and node shift maneuver at perigee Input: SHNODE, threshold time
49	PCH	Plane change at a height Input: DPC, HICIR, threshold time

Table V. Maneuvers Performed by the General Purpose
Maneuver Processor (Continued)

<u>Maneuver Number</u>	<u>Maneuver Display Code</u>	<u>Definition and Required Input</u>
50	NSH	Node shift at a height Input: HICIR, SHNODE, threshold time
51	NSL	Node shift at a longitude Input: along, SHNO SHNODE, threshold time
52	HOH	Height maneuver at a height Input: HICIR, DHLONG, NM, KMLO
53	HAS	Maneuver to change both apogee and perigee and place perigee a certain number of degrees from a specified longitude a cer- tain number of degrees from a specified longitude a certain number of revs later. Input: NM, KMLO, ALONG, ALF, HAD HPD, threshold time.

Table VI. Maneuvers Performed by the Lunar Descent Planning Processor

<u>Maneuver Mode No.</u>	<u>Maneuver Sequence No.</u>	<u>Maneuver Display Code</u>	<u>Definition</u>
1	-1	PC	Compute plane-change maneuver only.
1	0	PCC	Compute plane-change and circularization maneuver.
1	1	ASP-CIA	Compute plane-change maneuver combined with first maneuver of a CSM two maneuver sequence to circularize the CSM orbit at an input altitude.
1	2	PCCH	Compute plane-change and circularization maneuver at a specific altitude.
1	3	PCCT	Compute plane-change and circularization maneuver at an input time.
2	0	ASH	Compute CSM maneuver to establish an apsis and an input altitude at the DOI maneuver point
2	1	CIR	Compute CSM maneuver to circularize orbit at an input altitude.
3	-1	ASHT-CIA	Compute CSM two-maneuver sequence with the first maneuver performed at an input time and the second maneuver performed at an input altitude to circularize the orbit.
3	1	ASHA-CIA	Compute CSM two-maneuver sequence with the first maneuver performed at an apsis and the second maneuver performed at an input altitude to circularize the orbit.

Table VI. Maneuvers Performed by the Lunar Descent
Planning Processor (Continued)

<u>Maneuver Mode No.</u>	<u>Maneuver Sequence No.</u>	<u>Maneuver Display Code</u>	<u>Definition</u>
4	1	DOI	Computes the DOI maneuver only.
5	-1	PC, HO1, HO2	Compute CSM three-maneuver sequence so that the first maneuver is a plane change and the following pair is a double Hohmann to a circular orbit at an input altitude.
5	0	HO1, PC, HO2	Compute CSM three-maneuver sequence so that the first maneuver initiates a double Hohmann, the second is a plane-change, and the third completes the double Hohmann to a circular orbit at an input altitude.
5	1	HO1, HO2, PC	Compute CSM three-maneuver sequence so that the first two maneuvers constitute a double Hohmann to a circular orbit at an input altitude and the third is a plane change.

Table VII. ARIA Reentry Format

1. LAT ____:____:____.____	Target for entry
LONG ____:____:____.____	
2. TIME ____:____:____.____	Time and state vector at 425 K.
X _____._____._____.	Time is GET. State vector is
Y _____._____._____.	NMBY (ft and ft/sec).
Z _____._____._____.	
XDOT _____._____._____.	
YDOT _____._____._____.	
ZDOT _____._____._____.	
3. LAD. ____	
4. LOD. ____	
5. CGBIAS ____.	
6. PHI ____:____:____.____	Latitude and longitude at 425 K
LAMBDA ____:____:____.____	
7. CIOX _____._____.	
8. CMWT _____.	
10. GC ____.	
12. KSWCH ____	If a backup mode is not used for the entry, KSWCH must be blank and items 13-19 crossed off.
13. K1 _____._____.	
14. K2 _____._____.	
15. DOX ____.	
16. LADB ____	
17. RLDIR ____	
18. LATT ____:____:____.____	
19. LONGT ____:____:____.____	

General Notes:

1. All blanks must be filled.
2. Quantities that are N/A should be marked out.
3. Spaces marked ____ are for sign on quantity when at beginning of a series. Should be either a - sign or blank.
4. Spaces marked ____ are for direction (N, S, E, W) when at the end of a series.

Table VII. ARIA Reentry Format (Continued)

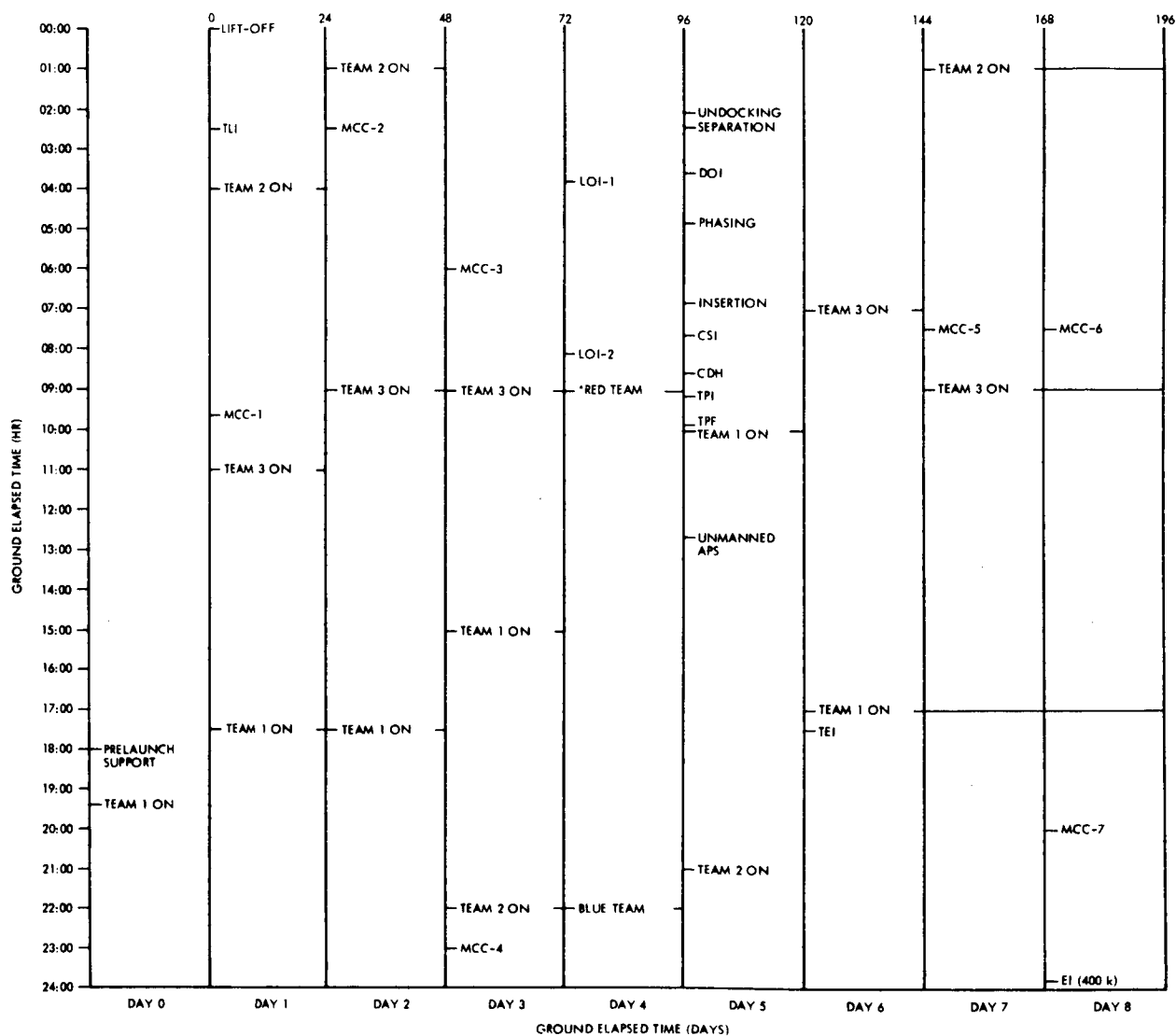
20. AERODYNAMICS (b) TABLE

MACH (b)(b) ALPHA (b)(b) CL (b)(b)(b)(b)(b) CD

0 0 . 2 0 (b)		(b)		(b)	
0 0 . 4 0					
0 0 . 7 0					
0 0 . 9 0					
0 1 . 1 0					
0 1 . 2 0					
0 1 . 3 5					
0 1 . 6 5					
0 2 . 0 0					
0 2 . 4 0					
0 3 . 0 0					
0 4 . 0 0					
1 0 . 0 0					
2 9 . 5 0					

5. APOLLO 10 RTACF NOMINAL MISSION TIMELINE

This section of the Flight Annex presents the RTACF nominal mission timeline. The timeline is not meant to display the schedule of RTACF activities during the mission but rather to present, in graphical form, a synopsis of the events that are of particular importance in planning the RTACF work schedule, which is to be published as a separate MSC internal note. The events and their corresponding times presented in the timeline include the major nominal mission events and the RTACF team schedule. The event times used in generating the timeline were based on the Apollo 10 Operational Trajectory (Revision 1).



* DUE TO EXCESSIVE WORK LOAD DURING THE RENDEZVOUS, THE THREE ACR TEAMS WILL BE COMBINED INTO TWO TEAMS, DENOTED BY RED AND BLUE.

Apollo 10 RTACF Nominal Mission Timeline

6. APOLLO 10 RTACF OPERATIONAL SUPPORT TEAM

The following individuals will man positions in the Flight Dynamics Staff Support Room and the Auxillary Computing Room during the Apollo 10 mission.

Flight Dynamics Staff Support Room

	<u>1st Team</u>	<u>2nd Team</u>	<u>3rd Team</u>
Trajectory Support Chief	S. D. Holzaephel (FAB)	L. D. Davis (FAB)	T. L. Turner (FAB)
Assistant Trajectory Support Chief	W. R. Pruett (FAB)	R. D. Davis (FAB)	H. Garcia (FAB)
Maneuver Specialist		K. Young (OMAB)	
Rendezvous Specialist		J. D. Alexander (OMAB)	
Entry Specialist	Jon Harpold (LAB)	John Burton (LAB)	Joe Rogers (LAB)
Engineering Aide	R. M. Hanson (ITT)	M. E. Crawford (ITT)	E. D. Langford (ITT)

Auxiliary Computing Room

ACR Chief	P. A. DiValerio (TRW)	C. D. Chenoweth (TRW)	L. Baker (TRW)
ACR Engineers	J. H. Kawasaki (TRW)	D. C. McDougall (TRW)	J. R. Lewis (TRW)
	F. J. Drake (TRW)	T. Richardson (TRW)	B. C. Schneider (FAB)
	D. W. Sager (TRW)	R. Moore (TRW)	V. R. Dragotta (TRW)
		R. T. Witton (TRW)	
Program Consultants	D. Dodson (LEC)	J. Bem (LEC)	S. Stacy (LEC)
	K. Christie (TRW)		M. Foster (TRW)
Run Coordinators	J. E. Frieble (LEC)	B. Ferguson (LEC)	R. Gregory (LEC)
	W. Baker (LEC)	L. Burney (LEC)	J. Parker (LEC)
Engineering Aide	L. A. Holden (ITT)	W. D. Gray (ITT)	C. M. Tully (LEC)

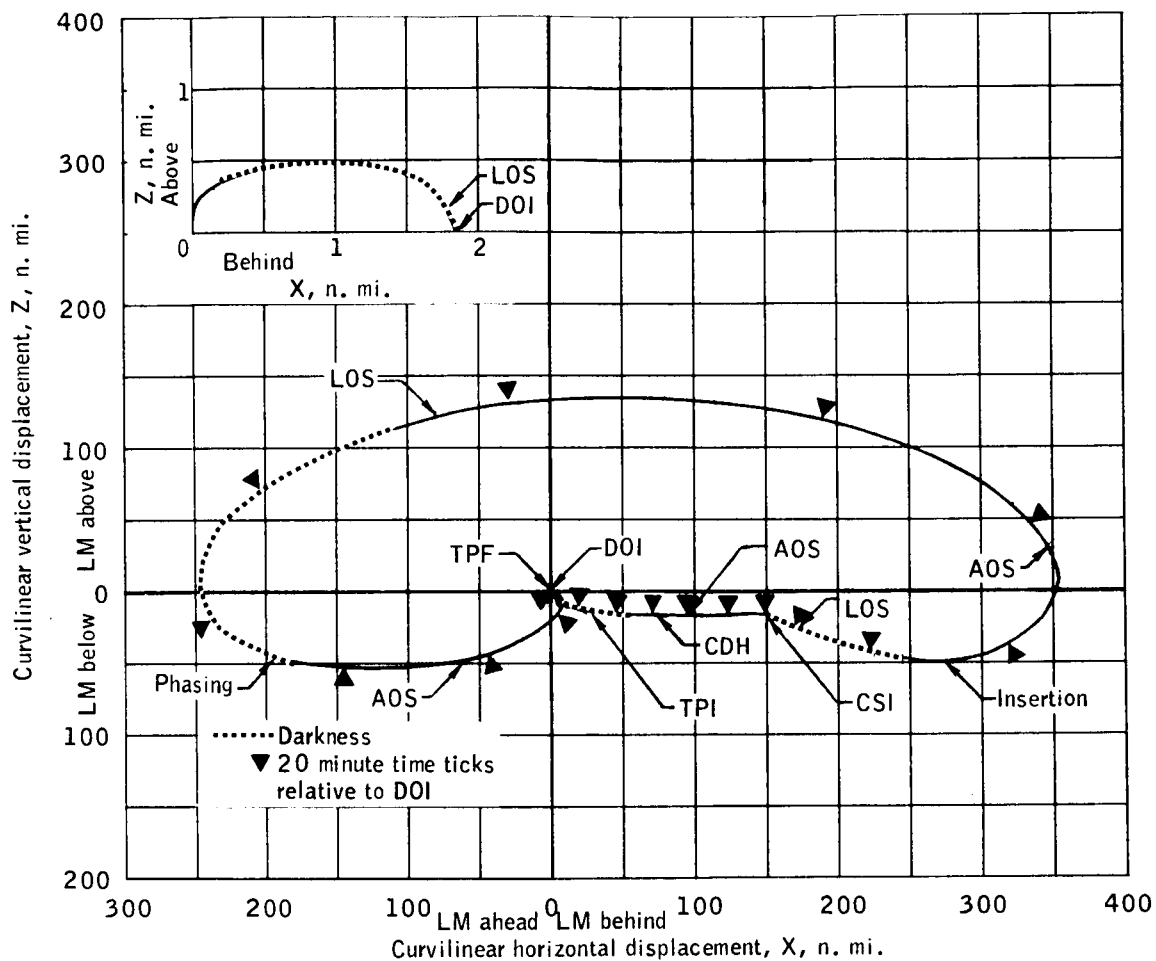


Figure 1. Rendezvous Sequence Relative Motion Plot

REV NO.	HRS	GET		DAY	GMT		GROUND TRACK			LONG DEG	MIN	ALT N MI	AZIMUTH DEGREES
		MIN	SEC		HRS	MIN	SEC	DEG	LAT MIN				
XX	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XXXXX	XXXXX
XX	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XXXXX	XXXXX
XX	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XXXXX	XXXXX
XX	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XXXXX	XXXXX
XX	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XXXXX	XXXXX
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Figure 2. Ground Track Summary Sheet

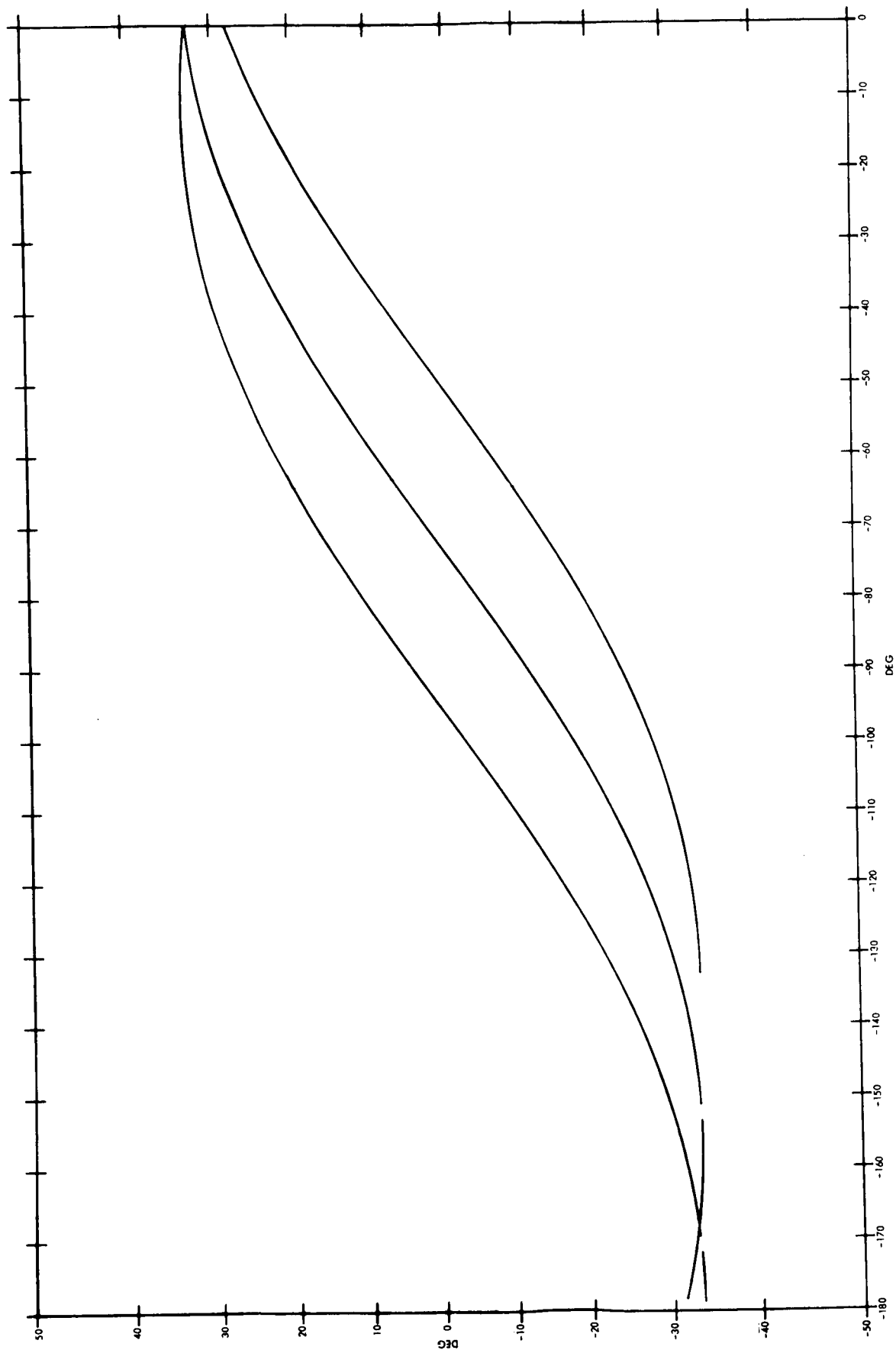


Figure 3. Ground Track Plot

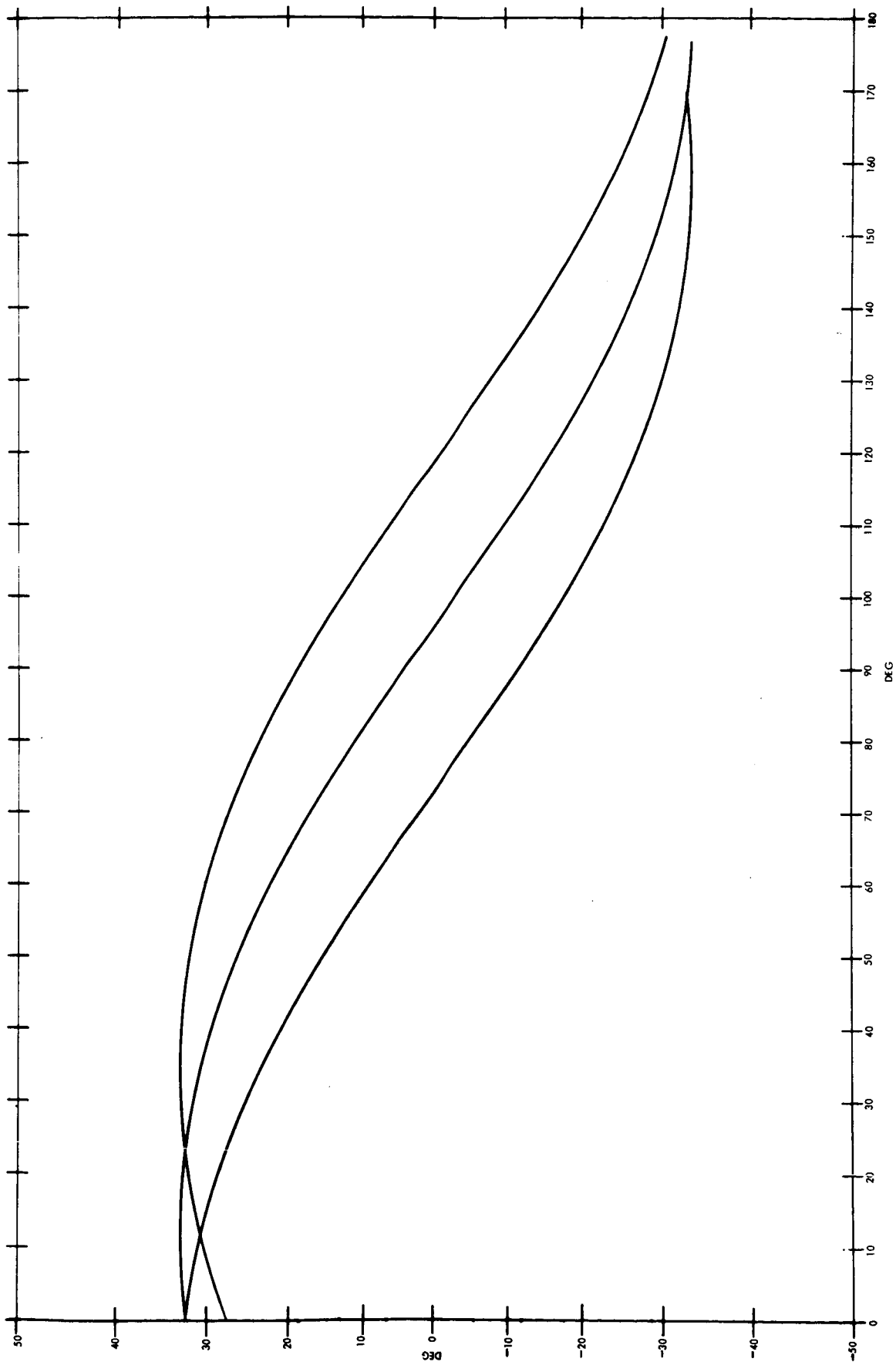


Figure 3. Ground Track Plot (Continued)

RELATIVE MOTION DIGITALS

CSM STA ID		VEH			LM STA ID		
REFSMMAT		X AXIS AT TARGET					
GET	R	RDOT	AZH	ELH	X/P	Z/Y	Y/R
XXX XX XX	XXXXXX	XXXXXX	XXXXXX	XXXXXX	XXXXXX	XXXXXX	XXXXXX
XXX XX XX	XXXXXX	XXXXXX	XXXXXX	XXXXXX	XXXXXX	XXXXXX	XXXXXX
.	.	.	.	.	.	.	.
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.	.	.	.	.	.	.	.

Figure 4. Relative Motion Summary Sheet

SPACECRAFT DAYLIGHT-DARKNESS

VEHICLE 1											
	GET	GMT			GEODETIC	LONGITUDE	ALTITUDE	SUN AZIM	PITCH	LV/LH	YAW
	HR	MIN	SEC	DAY	HR	MIN	DEG MIN	ANGLE	DEG		DEG
REVOLUTION X	XXX	XX	XX	XX	XX	XX	XXX XX	XXXX			
TERMINATOR SET	XXX	XX	XX	XX	XX	XX	XXX XX	XXXX			
SUNSET	XXX	XX	XX	XX	XX	XX	XXX XX	XXX	XXX		XXX
SUNRISE	XXX	XX	XX	XX	XX	XX	XXX XX	XXX	XXX		XXX
TERMINATOR RISE	XXX	XX	XX	XX	XX	XX	XXX XX				
REVOLUTION X	XXX	XX	XX	XX	XX	XX	XXX XX	XXX			
TERMINATOR SET	XXX	XX	XX	XX	XX	XX	XXX XX	XXX	XXX		XXX
SUNSET	XXX	XX	XX	XX	XX	XX	XXX XX	XXX	XXX		XXX
SUNRISE	XXX	XX	XX	XX	XX	XX	XXX XX				
TERMINATOR RISE	XXX	XX	XX	XX	XX	XX	XXX XX	XXX	XXX		XXX
REVOLUTION X	XXX	XX	XX	XX	XX	XX	XXX XX	XXX	XXX		XXX
TERMINATOR SET	XXX	XX	XX	XX	XX	XX	XXX XX				
	.	.	.	.	.	.	.	.	.	.	.
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	.	.	.	.	.	.	.	.	.	.	.

Figure 5. Daylight-Darkness Summary Sheet

DAYLIGHT/DARKNESS TABLE			
SUNRISE		SUNSET	
XXX XX	XXXXX	XXX XX	XXXXX
XXX XX	XXXXX	XXX XX	XXXXX
.	.	.	.
.	.	.	.
.	.	.	.

Figure 6. Daylight-Darkness Table Summary Sheet

SPACECRAFT MOON SIGHTINGS

VEHICLE 1

	GET			GMT			GEODETTIC LATITUDE		LONGITUDE		ALTITUDE N. MI.
	HR	MIN	SEC	DAY	HR	MIN	SEC	DEG	MIN	DEG	
MOONSET	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XXXX
REVOLUTION X	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XXXX
MOONRISE	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XXXX
MOONSET	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XXXX
REVOLUTION X	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XXXX
MOONRISE	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XXXX
MOONSET	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XXXX
REVOLUTION X	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XXXX
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Figure 7. Moonsighting Summary Sheet

LANDMARK SIGHTINGS

VEHICLE 1

LANDMARK NUMBER	REV NO.	ACQUISITION						LOSS OF LANDMARK						DELTA			ACQ AZ DEG	ACQ RANGE N. MI.	GRND MIN RANGE N. MI.	MAX ELEV DEG	GET OF MAX ELEV			ALT OF MAX ELEV N. MI.
		GET H	GET M	GET S	D	H	M	GMT	GET H	GET M	GET S	D	H	M	GMT	T					H	M	S	
XXX	XXX	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XXXXX	XXXXX	XX	XXX	XX	XX	XXXXX
XXX	XXX	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XXXXX	XXXXX	XX	XXX	XX	XX	XXXXX
XXX	XXX	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XXXXX	XXXXX	XX	XXX	XX	XX	XXXXX
XXX	XXX	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XXXXX	XXXXX	XX	XXX	XX	XX	XXXXX
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Figure 8. Landmark Sighting Summary Sheet

STAR SIGHTINGS

VEHICLE 1

STAR XX RISE STAR XX SET STAR XX RISE STAR XX SET	GET			RIGHT ASCENSION			DECLINATION			MAGNITUDE		CENTRAL ANGLE		ALTITUDE		AZIMUTH	
	HR	MIN	SEC	HR	MIN	SEC	DEG	MIN	SEC			DEG	MIN	N. MI.		DEGREES	
STAR XX RISE	XXX	XX	XX	XX	XX	XX	XXX	XX	XX	XXXXX		XXX	XX	XXXXX		XXX	
STAR XX SET	XXX	XX	XX	XX	XX	XX	XXX	XX	XX	XXXXX		XXX	XX	XXXXX		XXX	
STAR XX RISE	XXX	XX	XX	XX	XX	XX	XXX	XX	XX	XXXXX		XXX	XX	XXXXX		XXX	
STAR XX SET	XXX	XX	XX	XX	XX	XX	XXX	XX	XX	XXXXX		XXX	XX	XXXXX		XXX	
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Figure 9. Starsighting Summary Sheet

POINTING DATA

VEHICLE 1

TARGET XXXXX LATITUDE XXXXX DEG LONGITUDE XXXXX DEG ALTITUDE XXXXX N.MI.

REFSMMAT

```

*****
* X XXXXX XXXXX XXXXX X *
* Y XXXXX XXXXX XXXXX Y *
* Z XXXXX XXXXX XXXXX Z *
*****

```

REVOLUTION XXX

GET		GMT	ELAPSED TIME		MAX ELEV ANGLE	MINIMUM RANGE	ALT MIN RANGE		
HR:MIN:SEC	DAY:HR:MIN:SEC	GMT	MIN:SEC	DEG	DEG	N.MI.	N.MI.		
ACQ	XX XX XX XX XX XX	XX XX XX XX	XX:XX	XX:XX	XXXXX	XXXXX	XXXXX		
LOSS	XX XX XX XX XX XX	XX XX XX XX							
GET		GMT	R-DOT	R-ACC	AZIMUTH	ELEVATION	ROLL(0)	PITCH(1)	YAW
HR:MIN:SEC	DAY:HR:MIN:SEC	GMT	FT/SEC	FT/SEC2	DEG	DEG	DEG	DEG	DEG
XXX XX XX	XX XX XX XX XX XX	XX XX XX XX	XXXXX	XXXXX	XXXXX	XXXXX	XXX	XXX	XXX
XXX XX XX	XX XX XX XX XX XX	XX XX XX XX	XXXXX	XXXXX	XXXXX	XXXXX	XXX	XXX	XXX
XXX XX XX	XX XX XX XX XX XX	XX XX XX XX	XXXXX	XXXXX	XXXXX	XXXXX	XXX	XXX	XXX

LVLH

Figure 10. Pointing Data Summary Sheet

CLOSEST APPROACH

VEHICLE 1

TARGET ACR LAND

REV	DAY	HR	GMT	MIN	SEC	HR	GET	MIN	SEC	LATITUDE DEGREES XXXXX	LONGITUDE DEGREES XXXXX	ALTITUDE N. MI.	PITCH (I) DEG	GIMBAL ANGLES YAW (M) DEG	ROLL (O) DEG	PITCH DEG	LV/LH	YAW DEG
1	XX	XX	XX	XX	XX	XXX	XX	XX	XX	XXXXX	XXXXX	XXXXX	XXX	XXX	XXX	XXX	XXX	XXX
2	XX	XX	XX	XX	XX	XXX	XX	XX	XX	XXXXX	XXXXX	XXXXX	XXX	XXX	XXX	XXX	XXX	XXX
3	XX	XX	XX	XX	XX	XXX	XX	XX	XX	XXXXX	XXXXX	XXXXX	XXX	XXX	XXX	XXX	XXX	XXX
4	XX	XX	XX	XX	XX	XXX	XX	XX	XX	XXXXX	XXXXX	XXXXX	XXX	XXX	XXX	XXX	XXX	XXX
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Figure 11. Closest Approach Summary Sheet

CHECKOUT MONITOR

VEH CSM	R-DAY XX XX XX	K-FAC XXX
GET	VID	RF A
GMT XXX XX XX	XT XXXXX	YT XXXXX
		ZT XXXXX
		OPTION
X XXXXXXXXX	V XXXXX	A XXXXX
Y XXXXXXXXX	PTH XXXXX	E XXXXX
		WT XXXXX
		WC XXXXX
Z XXXXXXXXX	AZ XXXXX	I XXXXX
		SPS XXXXX
		RCSC XXXXX
DX XXXXXXXXX	LATC XXXXX	AP XXXXX
		WL XXXXX
		APS XXXXX
DY XXXXXXXXX	LONC XXXXX	RAN XXXXX
		DPS XXXXX
		RCSL XXXXX
DZ XXXXXXXXX	HS XXXXX	TA XXXXX
		J2 XXXXX
LO/C XXX XX XX	HA XXXXX	MA XXXXX
GRR/C XXX XX XX	HP XXXXX	
ZS/C XXX XX XX	HO XXXXX	X XXXXX
GRR/S XXX XX XX	HO XXXXX	Y XXXXX
LO/L XXX XX XX	LATD XXXXX	Z XXXXX
ZS/L XXX XX XX	LOND XXXXX	DX XXXXX
ZS/A XXX XX XX	R XXXXX	DY XXXXX
EPHB XXX XX XX	DECL XXXXX	DZ XXXXX
EPHE XXX XX XX	RAE XXX XX XX	SRAE XXX XX XX
THT XXX XX XX	DECE XXX XX XX	SDECE XXX XX XX
	RAM XXX XX XX	SRAM XXX XX XX
	DECM XXX XX XX	SDECM XXX XX XX

Figure 12. Checkout Monitor Summary Sheet

RADAR TRACKING

STATION ID	RA	DECL	RLA1	RLA2	RLAA	XXXXXXX
AZI	AZIR	ELV	ELVR	RANGE	RRATE	XXXXXXX
DLAYT						XXXXXXX

STATION ID	RA	DECL	RLA1	RLA2	RLAA	XXXXXXX
AZI	AZIR	ELV	ELVR	RANGE	RRATE	XXXXXXX
DLAYT						XXXXXXX

Figure 13. Radar Tracking Summary Sheet

MODE, X	OPTICAL SIGHTING TABLE				VEH XX			
***** BODY ATTITUDES *****								
LM OGA	XXXXX		CSM OGA	XXXXX				
LM IGA	XXXXX		CSM IGA	XXXXX				
LM MGA	XXXXX		CSM MGA	XXXXX				
***** BURN HORIZN CHECK *****								
MATRIX USED	XXXXX	GETHORX	XXX XX XX	OGA	XXXX	LPD	XXXX	
***** ALIGNMENT AND MANEUVER CHECK *****								
STAR	X	X	AOS			LOS		OCC
X/XXX	XXXXX	XXXXX	XXX	XX	XX	XXX	XX	XX
X/XXX	XXXXX	XXXXX	XXX	XX	XX	XXX	XX	XX
X/XXX	XXXXX	XXXXX	XXX	XX	XX	XXX	XX	XX
X/XXX	XXXXX	XXXXX	XXX	XX	XX	XXX	XX	XX
X/XXX	XXXXX	XXXXX	XXX	XX	XX	XXX	XX	XX
X/XXX	XXXXX	XXXXX	XXX	XX	XX	XXX	XX	XX
X/XXX	XXXXX	XXXXX	XXX	XX	XX	XXX	XX	XX
X/XXX	XXXXX	XXXXX	XXX	XX	XX	XXX	XX	XX
X/XXX	XXXXX	XXXXX	XXX	XX	XX	XXX	XX	XX
X/XXX	XXXXX	XXXXX	XXX	XX	XX	XXX	XX	XX
***** OPTICS ALIGNMENT *****								
STAR	RA	DEC	OGA	IGA	MGA	XXX	X	X
X/XXX	XXX XX XX	XXX XX XX	XXXX	XXXX	XXXX		XXXXX	XXXXX X
X/XXX	XXX XX XX	XXX XX XX	XXXX	XXXX	XXXX		XXXXX	XXXXX X
***** ALIGN ANGLES *****								
GIMB	OGA	XXXX	IGA	XXXX	MGA	XXXX		
FDAI	Y	XXXX	P	XXXX	R	XXXX		
MATRIX XXXXX								
XIXE	XXXXXXXX		XIYE	XXXXXXXX		XIZE	XXXXXXXX	
YIXE	XXXXXXXX		YIYE	XXXXXXXX		YIZE	XXXXXXXX	
ZIXE	XXXXXXXX		ZIYE	XXXXXXXX		ZIZE	XXXXXXXX	
MATRIX XXXXX								
XIXE	XXXXXXXX		XIYE	XXXXXXXX		XIZE	XXXXXXXX	
YIXE	XXXXXXXX		YIYE	XXXXXXXX		YIZE	XXXXXXXX	
ZIXE	XXXXXXXX		ZIYE	XXXXXXXX		ZIZE	XXXXXXXX	

Figure 14. Optical Support Table 1 Summary Sheet

RTACF

DOCKED LM REFSMMAT COMPUTATION

LM REFSMMAT IS COMPUTED

```

*****
* CSM *
*****

REFSMMAT  ID      IMU GIMBAL ANGLES
              IGA(P)  MGA(Y)  OGAR)
              DEG      DEG      DEG
XXXXXXXXX  XXX      XXXXX  XXXXX  XXXXX

              IMU GIMBAL ANGLES
              IGA(P)  MGA(R)  OGA(Y)
              DEG      DEG      DEG
XXXXXXXXX  XXXXX  XXXXX  XXXXX

REFSMMAT
XXXXXXXXX  XXXXX  XXXXX  XXXXX
XXXXXXXXX  XXXXX  XXXXX  XXXXX
XXXXXXXXX  XXXXX  XXXXX  XXXXX

```

Figure 15. Docking Alignment Processor Summary Sheet

```

REFSMMAT TO REFSMMAT CONVERSION - XXX

REFSMMAT  ID.      IMU GIMBAL ANGLES
              IGA(P)  MGA(Y)  OGA(R)
              DEG      DEG      DEG
XXXXXXXXX  XXX      XXXXX  XXXXX  XXXXX
XXXXXXXXX  XXX      XXXXX  XXXXX  XXXXX

```

Figure 16. REFSMMAT to REFSMMAT Conversion Summary Sheet

FIDO ORBIT DIGITALS

GET	XXX XX XXXX						
VEHICLE	XXX						
REV	XXX						
		LPP	XXX XX		REVL	XXX	
		PPP	XXX XX		GETL	XXX XX XXXX	
STA ID	XXX	GETCC	XXX XX XX		L	XXX XX	
LIFETIME	XXX	TAPP	XXXXXX		TO	XXXXX	
		LNPP	XXX XX		K	XXXXX	
					ORBWT	XXXXX	
					REQUESTED		
H	XXXXX				REV	XXX	
V	XXXXX				GETBY	XXX XX XXXX	
GAM	XXXXX						
A	XXXXX						
E	XXXXX						
I	XXXXX						
HA	XXXXX				HA	XXXXX	
PA	XXX XX				PA	XXX XX	
LA	XXX XX				LA	XXX XX	
GETA	XXX XX XXXX				GETA	XXX XX XXXX	
HP	XXXXX				HP	XXXXX	
PP	XXX XX				PP	XXX XX	
LP	XXX XX				LP	XXX XX	
GETP	XXX XX XXXX				GETP	XXX XX XXXX	

Figure 17. FDO Orbit Digital Summary Sheet

		SPACE DIGITALS			
STA ID	XXXX	WEIGHT	XXXXX	GETR	XXX XX XXXX
GMTV	XXX XX XXXX	GETV	XXX XX XXXX	GET AXIS	XXX XX XXXX
GET	XXX XX XXXX	V	XXXXX	PHI	XXX XX H XXXXX
REF	XXXX	GAM	LAM	XXX XX PSI	XXXXX
GET VECTOR 2					
GET VECTOR	XXX XX XXXX	GETS	XXX XX XXXX	GETSE	XXX XX XXXX
REF MOON	WT XXXXX	GETCA	XXX XX XXXX	GETEI	XXX XX XXXX
AREA	K	VCA	XXXXX	VEI	XXXXX
GETA	XXX XX XXXX	HCA	XXXXX	GEI	XXXXX
HA	XXXXX	PCA	XXX XX	PEI	XXX XX
HP	XXXXXX	LCA	XXX XX	LEI	XXX XX
H	XXXXXX	PSICA	XXXXXXXXXX	PSIEI	XXXXX
V	XXXXXX	GETMN	XXX XX XXXX	GETVP	XXX XX XXXX
GAM	XXXXXX	HMN	XXXXX	VVP	XXXXX
PSI	XXXXXX	PMN	XXX XX	HVP	XXX
PHI	XXX XX	LMN	XXX XX	PVP	
LAM	XXX XX	DMN	XXXXX	LVP	
HS	XXXXXX			PSI VP	
HO	XXXXX				
PHIO	XXX XX				
IEMP	XXXXX	A	XXXXX		
W	XXXXX	L	XXXXX		
OMG	XXXXX	E	XXXXX		IE
PRA	XXXXX	I	XXXXX		LN

Figure 18. Space Digital Summary Sheet

EMP REFMMMAT

XXXXXXXXXX	XXXXXXXXXX	XXXXXXXXXX
XXXXXXXXXX	XXXXXXXXXX	XXXXXXXXXX
XXXXXXXXXX	XXXXXXXXXX	XXXXXXXXXX

VECTOR FROM VEHICLE TO EARTH = XXXXXXXXXX XXXXXXXXXX XXXXXXXXXX

LOOK ANGLES

	EARTH	MOON	SUN
ALPHA	XXXXXX	XXXXXX	XXXXXX
BETA	XXXXXX	XXXXXX	XXXXXX

EARTH MOON PLANE ALIGNMENT

ROLL	PITCH	YAW	IGA	MGA	OGA	ALPHAS	ALPHAE	BETAS	BETAE
XXXXXX	XXXXXX	XXXXXX	XXXXXX	XXXXXX	XXXXXX	XXXXXX	XXXXXX	XXXXXX	XXXXXX

MIDDLE GIMBAL ANGLE CONSTRAINT ALIGNMENT

(EARTH REFERENCE)

PITCH	YAW	IGA	MGA	OGA	ALPHAS	ALPHAE	BETAS	BETAE
XXXXXX	XXXXXX	XXXXXX	XXXXXX	XXXXXX	XXXXXX	XXXXXX	XXXXXX	XXXXXX
XXXXXX	XXXXXX	XXXXXX	XXXXXX	XXXXXX	XXXXXX	XXXXXX	XXXXXX	XXXXXX
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Figure 19. PTC Summary Sheet

FDO DETAILED MANEUVER TABLE

C STA ID		L STA ID		STA ID		WT	
GMTV	XX XX XX XX	GMTV	XX XX XX XX	GMTV	XX XX XX XX	WT	XXXXX
GETV	XXX XX XX	GETV	XXX XX XX	GETV	XXX XX XX	WC	XXXXX
CODE		REF		GETR	XXX XX XX	WL	XXXXX
						WF	XXXXX
GETI		DT B		DT TO		REFSMAT	
PETI	XXX XX XX	DT U	XXX	OR	XXX	YB	XXX
DVM	XXX XX XX	VGX	XXX	IP	XXX	PB	XXX
DVREM	XXXX	VGY	XXX	MY	XXX	RB	XXX
DVC	XXXX	VGZ	XXX				
VF		H BI		HA		VP	
VS	XXXX	P BI	XXX	HP	XXX	THETA P	XXX
VD	XXXX	L BI	XXX	L AN	XXX	DELTA P	XXX
DH	XXXX	F BI	XXX	E	XXX	P LLS	XXX
PHASE DOT	XX			I	XXXX	L LLS	XXX
PHASE DOT	XX			WP	XXXX	R LLS	XXX
WEDGE ANG	XXX	UNTIL					
YD	XXX						
PGNS		AGS		TARGETS			
EXT DV CSM		EXT DV					
GETI	XXX XX XX	GETI	XXX XX XX	GETI	XXX XX XX	MVR	XXX XX XX
VX	XXX	VX	XXX	T F	XXX XX XX	GETI	XXX
VY	XXX	VY	XXX	X	XXX XX XX	APSYS	XXX
VZ	XXX	VZ	XXX	Y	XXX	ELEV	XXX XX XX
				Z	XXX	TPI	XXX XX XX
				C	XXX	DT	XXX XX XX
					XXX	OPTION	
WT AFTER XXXXX							

Figure 20. Detailed Maneuver Table Summary Sheet

EARTH/MOON/SUN LOOK ANGLES - CSM

REFSMMAT	ID	GIMBAL ANGLES						EARTH			MOON		
		GET	IGA(P)	MGA(Y)	OGA(R)	THETA	PHI	THETA	PHI	THETA	PHI	THETA	PHI
		HR MIN SEC	DEG	DEG	DEG	DEG	DEG	DEG	DEG	DEG	DEG	DEG	DEG
XXXXX	XXX	XXX XX XX	XXX	XXX	XXX	XXX	XXX	XXX	XXX	XXX	XXX	XXX	XXX

Figure 21. ESM Look Angle Summary Sheet

TRUE OF DATE VECTOR CONVERSION VECTOR ID.

GREENWICH MEAN TIME				X	Y	Z	XDOT		YDOT		ZDOT		WEIGHT
YEAR	MO	DAY	HR MIN SEC	KM	KM	KM	KM/SEC	KM/SEC	KM/SEC	KM/SEC	KM/SEC	KM/SEC	LB.
XXXX	XXXX	XX XX	XX XXXX	XXXXXXXXXX	XXXXXXXXXX	XXXXXXXXXX	XXXXXXXXXX	XXXXXXXXXX	XXXXXXXXXX	XXXXXXXXXX	XXXXXXXXXX	XXXXXXXXXX	XXXXXXXXXX

Figure 22. True of Date Vector Conversion Summary Sheet

RADAR TRACKING

AZI	STATION ID	RA	XXXXXXXXXX	DECL	XXXXXXXXXX	RLAI	XXXXXXXXXX	RLA2	XXXXXXXXXX	RLAA	XXXXXXXXXX
	XXXXXXXXXX	AZIR	XXXXXXXXXX	ELV	XXXXXXXXXX	ELVR	XXXXXXXXXX	RANGE	XXXXXXXXXX	RRATE	XXXXXXXXXX
AZI	STATION ID	RA	XXXXXXXXXX	DECL	XXXXXXXXXX	RLAI	XXXXXXXXXX	RLA2	XXXXXXXXXX	RLAA	XXXXXXXXXX
	XXXXXXXXXX	AZIR	XXXXXXXXXX	ELV	XXXXXXXXXX	ELVR	XXXXXXXXXX	RANGE	XXXXXXXXXX	RRATE	XXXXXXXXXX
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Figure 23. Radar Tracking Summary Sheet

VEHICLE 1

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Figure 24. Radar Summary Sheet

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MODE X      OPTICAL SIGHTING TABLE      VEH XX

*****BODY ATTITUDES *****
LM OGA XXXX      CSM OGA XXXX
LM IGA XXXX      CSM IGA XXXX
LM MGA XXXX      CSM MGA XXXX

*****OPTICS ALIGNMENT *****
STAR      RA      DEC      OGA      IGA      MGA      XXX      X      X
X/XXX      XXX XX XX      XXX XX XX      XXXX      XXXX      CS XXXX      XXXX      1.
X/XXX      XXX XX XX      XXX XX XX      XXXX      XXXX      CS XXXX      XXXX      1.

*****ALIGN ANGLES *****
GIMB OGA XXXX      IGA XXXX      MGA XXXX
FDAI  R XXXX      P XXXX      Y XXXX

MATRIX XXXXX
XIYE XXXXXXXXXX      XIYE XXXXXXXXXX      XIZE XXXXXXXXXX
YIYE XXXXXXXXXX      YIYE XXXXXXXXXX      YIZE XXXXXXXXXX
ZIZE XXXXXXXXXX      ZIZE XXXXXXXXXX      ZIZE XXXXXXXXXX

MATRIX XXXXX
XIYE XXXXXXXXXX      XIYE XXXXXXXXXX      XIZE XXXXXXXXXX
YIYE XXXXXXXXXX      YIYE XXXXXXXXXX      YIZE XXXXXXXXXX
ZIZE XXXXXXXXXX      ZIZE XXXXXXXXXX      ZIZE XXXXXXXXXX

```

Figure 25. Optical Support Table 2 Summary Sheet

STAR SIGHTING TABLE

VEHICLE CSM MODE X

TARGET	XXX XX XX	GA		SOLAR DATA
TGT RT ASC	XXX XX XX			AZMH XXXX
TGT DEC	XXX XX XX	OGA XXXX	IGA XXXX	ELEV XXXX

GND PT DATA

LAT	XXXX	OPTICS	XXX	OC	REV	XXXX
LON	XXXX	X	XXXXX	X	LON	XXXX
ALT	XXXX				LON GET	XXX XX XX
ELV	XXXX				CA	XXXXX
					GETCA	XXX XX XX

MATRIX XXXXX

XIXE	XXXXXXXXXX	XIYE	XXXXXXXXXX	XIZE	XXXXXXXXXX
YIXE	XXXXXXXXXX	YIYE	XXXXXXXXXX	YIZE	XXXXXXXXXX
ZIXE	XXXXXXXXXX	ZIYE	XXXXXXXXXX	ZIZE	XXXXXXXXXX

Figure 26. Starsighting Table Summary Sheet

MODE X	OST CISLUNAR NAVIGATION													
	GET		AXIS STAR		ID		HORZ		L ID		OPTICS ANGLES		INERTIAL ATTITUDE	
	HR:MIN:SEC	H-Z	DEC/OCT	N-F	DEC	TRN	SFT	TRN	P	Y	R	THETA	THETA	PHI
XXX XX XX	X	XX/XXX	XXX	XXX	XX	XXXXX	XXXXX	XXXXX	XXXX	XXXX	XXXX	XXXXX	XXXXX	XXXXX
XXX XX XX	X	XX/XXX	XXX	XXX	XX	XXXXX	XXXXX	XXXXX	XXXX	XXXX	XXXX	XXXXX	XXXXX	XXXXX
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Figure 27. Cislunar Navigation Summary Sheet

STEERABLE ANTENNA POINTING PROGRAM

MODE X TGT ID X ACTIVE VEH XXX POINTING VEH XXX

MATRIX XXXXX

XIXE XXXXXXXX XIYE XXXXXXXX XIZE XXXXXXXX

YIXE XXXXXXXX YIYE XXXXXXXX YIZE XXXXXXXX

ZIXE XXXXXXXX ZIYE XXXXXXXX ZIZE XXXXXXXX

MATRIX XXXXX

XIXE XXXXXXXX XIYE XXXXXXXX XIZE XXXXXXXX

YIXE XXXXXXXX YIYE XXXXXXXX YIZE XXXXXXXX

ZIXE XXXXXXXX ZIYE XXXXXXXX ZIZE XXXXXXXX

***** TARGET DATA *****

***** CSM ***** LM *****

GET	PCH	YAW	IGA	MGA	OGA	PCH	YAW	IGA	MGA	OGA	SLANT RANGE	AZM	ELV
XXX XX XX	XXXX	XXXX	XXXX	XXXX	XXXX	XXXX	XXXX	XXXX	XXXX	XXXX	XXXXXXX	XXXX	XXXX
XXX XX XX	XXXX	XXXX	XXXX	XXXX	XXXX	XXXX	XXXX	XXXX	XXXX	XXXX	XXXXXXX	XXXX	XXXX
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Figure 28. Steerable Antenna Pointing Summary Sheet

TRIM AERODYNAMIC COEFFICIENTS

XCG = XXXXX YCG = XXXXX ZCG = XXXXX

WEIGHT = XXXXX

BANK ANGLE BIAS = XXXXX DEG.

MACH NO.	ALPHA	CL	CD	CL/CD
XXXXX	XXXXX	XXXXX	XXXXX	XXXXX
XXXXX	XXXXX	XXXXX	XXXXX	XXXXX
XXXXX	XXXXX	XXXXX	XXXXX	XXXXX
XXXXX	XXXXX	XXXXX	XXXXX	XXXXX

Figure 29. Aerodynamics Update Summary Sheet

CSM (OR LGC) DAP UPLINK

PRA	ADD	OCTAL	ENG. U.	UPL. U.
IXX	XXXX	XXXXX	XXXXXXXX	XXXXXXXX
IAVG	XXXX	XXXXX	XXXXXXXX	XXXXXXXX
WEIGHT	XXXX	XXXXX	XXXXXXXX	XXXXXXXX
PITCH TRIM	XXXX	XXXXX	XXXXXXXX	XXXXXXXX
YAW TRIM	XXXX	XXXXX	XXXXXXXX	XXXXXXXX
TLX	XXXX	XXXXX	XXXXXXXX	XXXXXXXX

*** XCG = XXX YCG = XXX ZCG = XXX ***

Figure 30. Digital Autopilot Command Load Summary Sheet

NEXT INPUT BLOCK/EVENT

 TITLE
 MANV, CSI ATT LM

H M S
 XXX XX XX

BLOCK/TYPE
 XXX

 ***** TOTAL SPACECRAFT PROPERTIES *****

WEIGHT (LBS)	X	CG (IN)	Y	CG (IN)	Z	CG (IN)	IXX	IYY	IZZ	PXY	PZX	PYZ	AVE MIX RATIO	RDLE AVE MIXED RATIO
XXXXXXXXXX		XXXXX		XXXXX		XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX
EVENT TITLE		HRS	M	S	MODE	MANEUVER	RATE OR	PROPELLANT USED	TO DATE	PROP	LEFT	MARGIN		
X AXIS MANEUVER	A	XXX	XX	XX	XX	X AXIS	DEADBRAND	QUAD	EVENT					
						X,X	XXX DPS	A	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX
								B	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX
								C	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX
								D	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX
							TOTAL	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX

BLOCK XXX SUBTOTALS
 MANV, CSI ATT LM

XXX XX XX

QUADS A B C D TOTAL
 XXXXX XXXXX XXXXX XXXXX XXXXX

Figure 31. MRS Summary Sheet

SM RCS PROPELLANT COMPUTATION

	QUAD A		QUAD B		QUAD C		QUAD D	
TANK	XXX		XXX		XXX		XXX	
PH	XXXXX		XXXXX		XXXXX		XXXXX	
T	XXXXX		XXXXX		XXXXX		XXXXX	
DELT	XXXXX		XXXXX		XXXXX		XXXXX	
PO	XXXXX		XXXXX		XXXXX		XXXXX	
PF	XXXXX		XXXXX		XXXXX		XXXXX	
MR1	XXX		XXX		XXX		XXX	
MR2	XXX		XXX		XXX		XXX	

	QUAD A		QUAD B		QUAD C		QUAD D	
WFE	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX
WOE	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX
WFR	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX
WOR	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX
WPU	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX

Figure 32. PVT Summary Sheet

LM - RCS PROPELLANT BUDGET

TIME HRS M	EVENT TITLE	S/C WT (LBS)	LM RCS USED (LBS)	LM RCS LEFT (LBS)	LM RCS LEFT PCNT
XXX XX	OUTPUT PROPELLANT LOADINGS	XXXXXX	XX	XXX	XXX
XXX XX	REDLINE OFF	XXXXXX	XX	XXX	XXX
XXX XX	RCS HOT FIRE	XXXXXX	XX	XXX	XXX
XXX XX	PGNS 3AX ATTITUDE HOLD 1DB	XXXXXX	XX	XXX	XXX
XXX XX	PITCH	XXXXXX	XX	XXX	XXX
XXX XX	YAW	XXXXXX	XX	XXX	XXX
XXX XX	PGNS 3AX ATTITUDE HOLD 1DB	XXXXXX	XX	XXX	XXX
XXX XX	PITCH	XXXXXX	XX	XXX	XXX
XXX XX	YAW	XXXXXX	XX	XXX	XXX
XXX XX	2 JET ULLAGE	XXXXXX	XX	XXX	XXX
XXX XX	YAW MOMENT CONTROL DOCKED BURN	XXXXXX	XX	XXX	XXX
XXX XX	DOCKED DPS BURN	XXXXXX	XX	XXX	XXX
.	.	.	.	.	.
.	.	.	.	.	.

Figure 33. LM RCS Propellant Budget Summary Sheet

SUPERCritical HELIUM PROFILE

D	GMT		H	TIME GET		GET SEC	THRUST (LBS)	BOTTLE PRESSURE (PSI)	BOTTLE MASS (LBS)	BOTTLE TEMP (DEG F)	REG OUT TEMP (DEG F)	PROP WT (LBS)
	H	M	SEC	M	SEC							
XX	XX	XX	XX	XX	XX	XXXXXX	XXXXX	XXXX	XXXX	XXXX	XXX	XXXXXX
XX	XX	XX	XX	XX	XX	XXXXXX	XXXXX	XXXX	XXXX	XXXX	XXX	XXXXXX
XX	XX	XX	XX	XX	XX	XXXXXX	XXXXX	XXXX	XXXX	XXXX	XXX	XXXXXX
XX	XX	XX	XX	XX	XX	XXXXXX	XXXXX	XXXX	XXXX	XXXX	XXX	XXXXXX
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THE MAXIMUM PRESSURE OF XXXX.X PSI, OCCURS AT GET XXXXXX.X SECONDS.

Figure 34. SHe Summary Sheet

LM TELEMETRY DIAGNOSTICS

THERE ARE 2 FAILURES ASSOCIATED WITH SEQUENCER 7 PAM GATE NO 1, THEY ARE GC0202 GC0204
SEQUENCER 7 PAM GATE NO 1 IS ASSOCIATED WITH THE FOLLOWING PARAMETERS GC0202 GC0204
THERE ARE 1 FAILURES ASSOCIATED WITH SEQUENCER 7 PAM GATE NO 2, THEY ARE GF3585
SEQUENCER 7 PAM GATE NO 2 IS ASSOCIATED WITH THE FOLLOWING PARAMETERS GF3585
THERE ARE 1 FAILURES ASSOCIATED WITH SEQUENCER 9 PAM GATE NO 1, THEY ARE GC0203
SEQUENCER 9 PAM GATE NO 1 IS ASSOCIATED WITH THE FOLLOWING PARAMETERS GC0203 GF3571
THERE ARE 1 FAILURES ASSOCIATED WITH SEQUENCER 14 PAM GATE NO 3 , THEY ARE GC0155
SEQUENCER 14 PAM GATE NO 3 IS ASSOCIATED WITH THE FOLLOWING PARAMETERS GC0155 GR1085
THERE ARE 3 FAILURES ASSOCIATED WITH ROW 3. THEY ARE GF3585 GC0202 GC0204
ROW 3 IS ASSOCIATED WITH THE FOLLOWING PARAMETERS GC0202 GC0204 GF3585 GQ3015 GQ3018

Figure 36. LM Telemetry Summary Sheet

CSM LM REFSMMAT LOAD		
ID =		
FCT	DSKY	DECIMAL
XIXE	XXXXX	XXXXXXXXX
XIXE	XXXXX	
XIYE	XXXXX	XXXXXXXXX
XIYE	XXXXX	
XIZE	XXXXX	XXXXXXXXX
XIZE	XXXXX	
YIXE	XXXXX	XXXXXXXXX
YIXE	XXXXX	
YIYE	XXXXX	XXXXXXXXX
YIYE	XXXXX	
YIZE	XXXXX	XXXXXXXXX
YIZE	XXXXX	
ZIXE	XXXXX	XXXXXXXXX
ZIXE	XXXXX	
ZIYE	XXXXX	XXXXXXXXX
ZIYE	XXXXX	
ZIZE	XXXXX	XXXXXXXXX
ZIZE	XXXXX	

Figure 37. REFSMMAT Update Summary Sheet

CSM EXTERNAL DELTA V LOAD		
FCT	DSKY V71	DECIMAL
VGX	XXXXX	XXXXXXXXX
VGX	XXXXX	
VGY	XXXXX	XXXXXXXXX
VGY	XXXXX	
VGZ	XXXXX	XXXXXXXXX
VGZ	XXXXX	
TIGN	XXXXX	XXX XX XXXXX
TIGN	XXXXX	

Figure 38. Orbital External ΔV Summary Sheet

CMC RETROFIRE EXTERNAL DELTA-V UPDATE

TTI	STA ID	TYPE		LOAD NO		
CT	GET GEN	SITES PRI		B/U		
THRUSTER		OID	FCT	DSKY V71	DECIMAL	
AREA		1	INDEX			
		2	ADD			
GMTI		3	LAT	XXXXX	XXXXX	DEG
GETI		4	LAT	XXXXX		
		5	LONG	XXXXX	XXXXX	DEG
BT		6	LONG	XXXXX		
V(C)		7	TIGN	XXXXX	XXX XX XX	
		10	TIGN	XXXXX		
R(O)		11	VGX	XXXXX	XXXXX	FPS
P(I)		12	VGX	XXXXX		
Y(M)		13	VGX	XXXXX	XXXXX	FPS
		14	VGX	XXXXX		
H(P)		15	VGZ	XXXXX	XXXXX	FPS
RT400K		16	VGZ	XXXXX		
RETRB		17	WT	XXXXX	XXXXX	LBS

Figure 39. Deorbit External ΔV Summary Sheet

ENGINEERING UNITS/OCTAL CONVERSION

ENGINEERING NUMBER	OCTAL NUMBER	POWER OF TWO	PRECISION	SCALE FACTOR
XXXXXX	XXXXX XXXXX	XX	XX	XXXXX

Figure 40. General Octal Conversion Summary Sheet

RTACF^{CSM}_{LM} NAV UPDATE TO CMC

VECTOR ID

VECTOR GET XXX XX XX

VECTOR GMT XXX XX XX

X	XXXXX	XXXXXXXXX
X	XXXXX	
Y	XXXXX	XXXXXXXXX
Y	XXXXX	
Z	XXXXX	XXXXXXXXX
Z	XXXXX	
X-DOT	XXXXX	XXXXXXXXX
X-DOT	XXXXX	
Y-DOT	XXXXX	XXXXXXXXX
Y-DOT	XXXXX	
Z-DOT	XXXXX	XXXXXXXXX
Z-DOT	XXXXX	
T	XXXXX	XXX XX XX
T	XXXXX	

Figure 41. CMC/LGC Navigation Vector Update Summary Sheet

SATURN COMMAND LOAD

NAVIGATION UPDATE

LOAD NO XXXX			GETSV XX XX XX				
FCT	ENGLISH	METRIC	OCTAL LOAD DATA				
Z DOT	XXXXX	XXXXX	11	12	13	14	15
			XXX	XXX	XXX	XXX	XXX
X DOT	XXXXX	XXXXX	21	22	23	24	25
			XXX	XXX	XXX	XXX	XXX
Y DOT	XXXXX	XXXXX	31	32	33	34	35
			XXX	XXX	XXX	XXX	XXX
Z POS	XXXXX	XXXXX	41	42	43	44	45
			XXX	XXX	XXX	XXX	XXX
X POS	XXXXX	XXXXX	51	52	53	54	55
			XXX	XXX	XXX	XXX	XXX
Y POS	XXXXX	XXXXX	61	62	63	64	65
			XXX	XXX	XXX	XXX	XXX
TIME	XX XX XX	XXXXX	71	72	73	74	75
			XXX	XXX	XXX	XXX	XXX

Figure 42. S-IVB Navigation Vector Update Summary Sheet

RTACF LM NAV UPDATE TO AGS

VECTOR ID	
VECTOR GET	XXX XX XX
VECTOR GMT	XX XX XX XX
X	XXXXXXXX
X	
Y	XXXXXXXX
Y	
Z	XXXXXXXX
Z	
X-DOT	XXXXXXXX
X-DOT	
Y-DOT	XXXXXXXX
Y-DOT	
Z-DOT	XXXXXXXX
Z-DOT	
T	XXX XX XX
T	
GETK	XXX XX XX
GETK	

Figure 43. AGS Navigation Vector Update Summary Sheet

PAO DATA BASED ON VECTOR XXXX

XXXXX REFERENCED

TIME GET HR MIN SEC		
XXX XX XX		
TIME (GET-SECONDS)	XXXXXXXXXX	AZIMUTH (DEG)
LATITUDE (GEOGRAPHIC)	XXXXXXXXXX	APOPSIS ALTITUDE (N MI)
LONGITUDE	XXXXXXXXXX	PERAPSIS ALTITUDE (N MI)
ALTITUDE (FT)	XXXXXXXXXX	ORBITAL PERIOD (MIN)
ALTITUDE (N MI)	XXXXXXXXXX	INCLINATION (DEG)
VELOCITY (FT/SEC)	XXXXXXXXXX	WEIGHT (POUNDS)
FLIGHT PATH ANGLE (DEG)	XXXXXXXXXX	ECCENTRICITY
X (FT)	XXXXXXXXXX	XD (FT/SEC)
Y (FT)	XXXXXXXXXX	YD (FT/SEC)
Z (FT)	XXXXXXXXXX	ZD (FT/SEC)

Figure 44. PAO Data Summary Sheet

LONG DEG	LAT DEG	ALT NM	B GAUSS	L ER	DAY	HR	MIN	SEC	DOSE RATE (REM/HR)			CUMULATIVE DOSE (REM)		
									LM SKIN	CM SKIN	CM DEPTH	LM SKIN	CM SKIN	CM DEPTH
XXXX	XXXX	XXXX	XXXX	XXXX	XX	XX	XX	XX	XXXX	XXXX	XXXX	XXXX	XXXX	XXXX
XXXX	XXXX	XXXX	XXXX	XXXX	XX	XX	XX	XX	XXXX	XXXX	XXXX	XXXX	XXXX	XXXX
XXXX	XXXX	XXXX	XXXX	XXXX	XX	XX	XX	XX	XXXX	XXXX	XXXX	XXXX	XXXX	XXXX
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Figure 45. Radiation Evaluation Summary Sheet

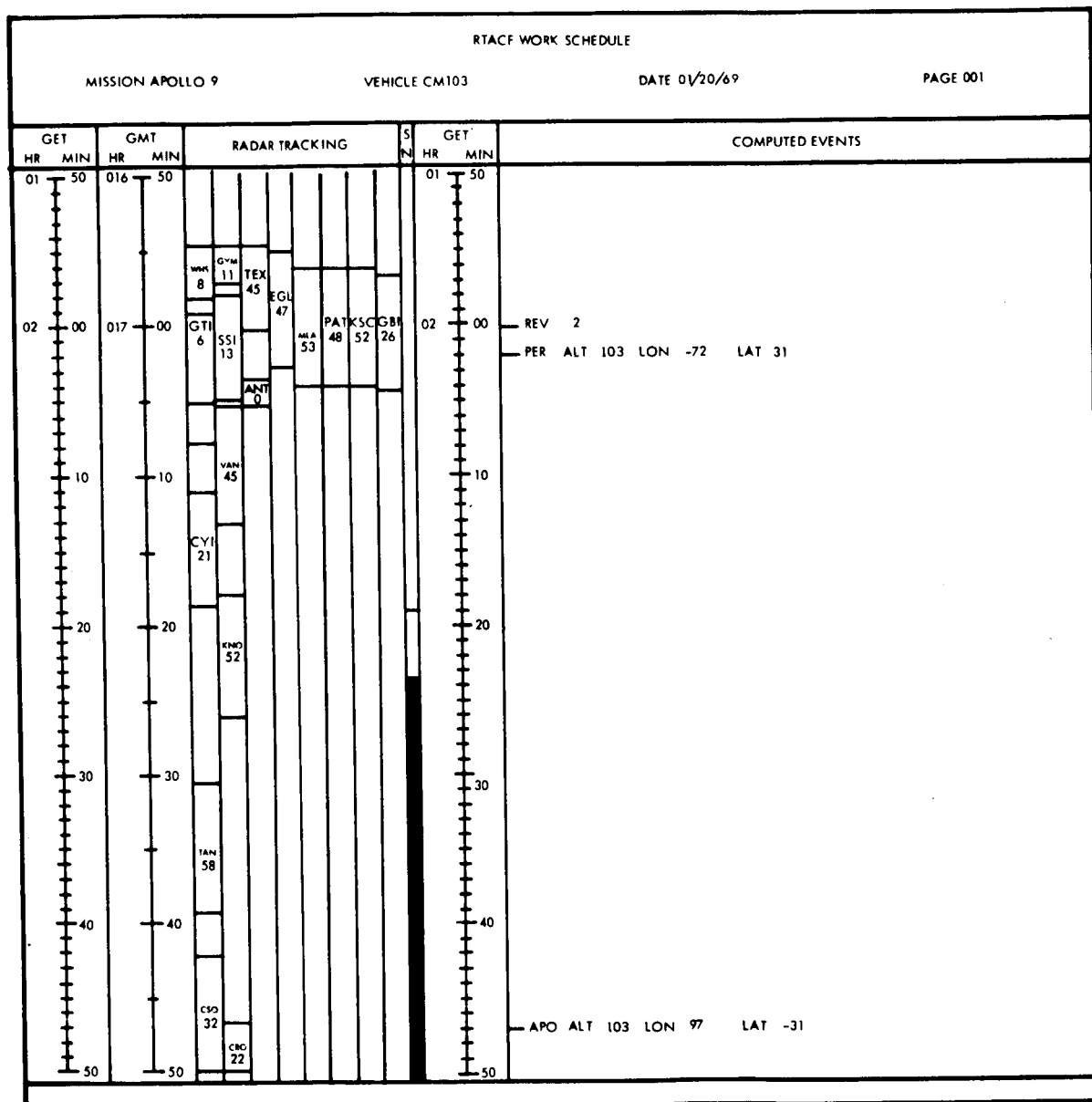


Figure 46. Work Schedule Display

NAV UPDATE TO CMC OR LGC				
LOAD NO	GET GEN		SITES	
STA ID			GMT ID	
OID	FCT	DSKY V71	VECTOR	
1	INDEX			
2	ADD			
3	VID			
4	X	XXXXX	XXXXXXXX	
5	X	XXXXX		
6	Y	XXXXX	XXXXXXXX	
7	Y	XXXXX		
10	Z	XXXXX	XXXXXXXX	
11	Z	XXXXX		
12	X-DOT	XXXXX	XXXXXXXX	
13	X-DOT	XXXXX		
14	Y-DOT	XXXXX	XXXXXXXX	
15	Y-DOT	XXXXX		
16	Z-DOT	XXXXX	XXXXXXXX	
17	Z-DOT	XXXXX		
20	T	XXXXX	XXX XX XX	
21	T	XXXXX		

Figure 47. Command Load Navigation Update Summary Sheet

[illegible]

MINUS SIGN INDICATES WEST OF NORTH (W/N)

Figure 48. Mode I Launch Abort Summary Sheet

MIDCOURSE TRADE OFF

FREE RETURN BAP FREE LPO

STAID		GMTV	XXX XX XX	GETR	XXX XX XX
		GETV	XXX XX XX		
VECID	XXXXX	MODE	XXX	RETURN FREE	
AZ MIN	XXXXX	GETLOI	XXX XX XX	GETTEI	XXX XX XX
AZ MAX	XXXXX	DV LOI	XXXXX	DV TEI	XXXXX
WEIGHT	XXXXX	AZ ACT	XXXXX	DV REM	XXXXX
GETMCC	XXXXX	I FR	XXXXX	GET LC	XXX XX XX
DV MCC	XXXXX	I PR	XXXXX	PHI IP	XXX XX
YAW	XXXXX	V EI	XXXXX	LAM IP	XXX XX
H PYCN	XXXXX	G EI	XXXXX	DV PC	XXXXX

RTCC TARGET UPDATES

FREE RETURN BAP FREE LPO

NODAL CONDITIONS FOR PLANNED LOI

PHI ND	XXXXX	LAM ND	XXXXX	GMT ND	XXX XX XX
--------	-------	--------	-------	--------	-----------

PERICYNTHION CONDITIONS FOR LUNAR FLYBY

PHI PC	XXXXX	LAM PC	XXXXX	GMT PC	XXX XX XX
HPYCN	XXXXX	IFR	XXXXX	GET PC	XXX XX XX

EXTERNAL DELTA VELOCITY

DV MCC	XXXXX	GMTIGN	XXX XX XX	GMTMCC	XXX XX XX
DVX	XXXXX	DVY	XXXXX	DVZ	XXXXX

Figure 49. Midcourse Tradeoff Summary Sheet

COMPUTED EVENTS

VEHICLE 1

	HR	GET		SEC	DAY	GMT		SEC	ALTITUDE N. MI.	GEODETIC LATITUDE		LONGITUDE		RIGHT ASCENSION		INCLINATION DEGREES
		HR	MIN			HR	MIN			DEG	MIN	DEG	MIN	HR	MIN	
ASC NODE	XXX	XX	XX	XX	XX	XX	XX	XX	XXXX	XX	XX	XXX	XX	XX	XX	XXX
PERIGEE	XXX	XX	XX	XX	XX	XX	XX	XX	XXXX	XX	XX	XXX	XX	XX	XX	XXX
REVOLUTION	XXX	XX	XX	XX	XX	XX	XX	XX	XXXX	XX	XX	XXX	XX			
APOGEE	XXX	XX	XX	XX	XX	XX	XX	XX	XXXX	XX	XX	XXX	XX			
ALTITUDE	XXX	XX	XX	XX	XX	XX	XX	XX	XXXX	XX	XX	XXX	XX			
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Figure 50. Computed Events Summary Sheet

[illegible]

DELTA V(T)	XXXXXX	FT/SEC	VELOCITY EI	XXXXXXXXXX	FT/SEC	TYPE	RCS
DELTA V(C)	XXXXXX	FT/SEC	GAMMA EI	XXXXX	DEG	RCS TIGN (GET)	XXX XXX XXX
ULLAGE DURATION	XXXXXX	SEC	WEIGHT EI	XXXXXXXXXX	LBS	DELTA V(T)	XXXXXXXXX FT/SEC
BURN DURATION	XXXXXX	SEC	LAT TARGET	XX XX	XX	ULLAGE DURATION	XXXXXXXXX SEC
INITIAL CSM WEIGHT	XXXXXX	LBS	LONG TARGET	XX XX	XX	BURN DURATION	XXXXXXXXX SEC
INITIAL THETA	XXXXXX	DEG	REV TARGET	X		PITCH (TV)	XXXXXXXXX DEG
ROLL (TV)	XXXXXX	DEG	L/D	XXX		YAW (TV)	XXXXXXXXX DEG
PITCH (TV)	XXXXXX	DEG				RESULTING APOGEE	XXXXXXXXX MILES
YAW (TV)	XXXXXX	DEG				RESULTING PERIGEE	XXXXXXXXX MILES
PTRIM	XXXXXX	DEG				REV NUMBER	X
YTRIM	XXXXXX	DEG					
DELTA V (TAILOFF)	XXXXXX	FT/SEC					
MANEUVER NO.	XX						

Figure 51. ABDP Deorbit Summary Sheet

VECTOR IDENTIFICATION

EVENT TIMES	RET			GMT				GET			LATITUDE		LONGITUDE	
	HR	MIN	SEC	DAY	HR	MIN	SEC	HR	MIN	SEC	DEG	MIN	DEG	MIN
RETROFIRE	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XXX	XX
BURN TERMINATION	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XXX	XX
ENTRY INTERFACE	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XXX	XX
BEGIN BLACKOUT, S BAND	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XXX	XX
BEGIN BLACKOUT, VHF	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XXX	XX
.20 G'S	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XXX	XX
REVERSE BANK	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XXX	XX
END BLACKOUT, VHF	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XXX	XX
END BLACKOUT, S BAND	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XXX	XX
DROGUE DEPLOY	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XXX	XX
MAIN DEPLOY	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XXX	XX
LANDING	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XXX	XX

BURN QUANTITIES			GIMBAL ANGLES		OUTER (ROLL)	INNER (PITCH)	MIDDLE (YAW)
DELTA VELOCITY(X)	XXXXX	FT/SEC	BURN		XXX	XXX	XXX
DELTA VELOCITY(C)	XXXXX	FT/SEC	ENTRY INTERFACE		XXX	XXX	XXX
DELTA VELOCITY(T)	XXXXX	FT/SEC	REVERSE BANK, BEGIN		XXX	XXX	XXX
DELTA TIME	XXXXX	SEC	REVERSE BANK, END		XXX	XXX	XXX
WEIGHT	XXXXX	POUNDS					
TRUE ANOMALY	XXXXX	DEG					
THRUST PITCH	XXXXX	DEG					
LATITUDE	XX	DEG XX MIN					
LONGITUDE	XXX	DEG XX MIN					
ALTITUDE	XXXXX	N MI					
EXTERNAL DELTA V							
DX	XXXXX	FT/SEC					
DY	XXXXX	FT/SEC					
DZ	XXXXX	FT/SEC					

LANDING POINTS			
LATITUDE	TARGET	XX DEG	XX MIN
LONGITUDE	TARGET	XXX DEG	XX MIN
LATITUDE	IMPACT	XX DEG	XX MIN
LONGITUDE	IMPACT	XXX DEG	XX MIN

REENTRY QUANTITIES	
VELOCITY(I) EI	XXXXX FT/SEC
GAMMA(I) EI	XXXXX DEGREES
WEIGHT	XXXXX POUNDS
BANK ANGLE	XXXXX DEG
LIFT	XXXXX
MAX G'S	XXXXX

Figure 52. Standard Deorbit Summary Sheet

APOLLO BLOCK DATA SUMMARY OF DEORBITS

AREA	TYPE XXXXXXXX		VECTOR IDENTIFICATION XXXXXXXX									
	GETB		RETEI		RETXG		RETRB		DELTA V(C)		DTBURN	
	HR	MIN SEC	MIN	SEC	MIN	SEC	MIN	SEC	FT/SEC	SEC	SEC	SEC
XXXXXX	XX	XX XX	XX	XX	XX	XX	XX	XX	XXXXX	XXXX		
XXXXXX	XX	XX XX	XX	XX	XX	XX	XX	XX	XXXXX	XXXX		
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Figure 53. ABDP Summary of Deorbits Summary Sheet

APOLLO BLOCK DATA MANEUVER SUMMARY SHEET

VECTOR IDENTIFICATION XXXXXXXXXX

MANEUVER NUMBER	MANEUVER TYPE	REV NUMBER	HR	MIN	SEC	TIGN(GET)	DELTA V(T) FT/SEC	PITCH DEG	YAW DEG	RESULTING APOGEE MILES	RESULTING PERIGEE MILES
1	RCS	X	XXX	XX	XXXXX	XXXXXX	XXXXXX	XXXXXX	XXXXXX	XXXXXX	XXXXXX
2	RCS-SPS	X	XXX	XX	XXXXX	XXXXXX	XXXXXX	XXXXXX	XXXXXX	XXXXXX	XXXXXX
3	SPS	X	XXX	XX	XXXXX	XXXXXX	XXXXXX	XXXXXX	XXXXXX	XXXXXX	XXXXXX
.		.	.	.	.	.	.	.	.	.	.
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8		X	XXX	XX	XXXXX	XXXXXX	XXXXXX	XXXXXX	XXXXXX	XXXXXX	XXXXXX

Figure 54. ABDP Maneuver Summary Sheet

RTACF ABORT SCAN TABLE		CASE X
STA ID		CSM WT XXXXX
GETV XXX XX XX		AREA XX
MANEUVER QUANTITIES		
GETI XXX XX XX		DELTA VT XXX XXX
ENTRY INTERFACE		
GETEI XXX XX XX		INCL XXX
VEI XXXXX		GAMMAEI XXX
LATEI XXX XX		LONGEI XXX XX
LANDING		
GETL XXX XX		
LATL XXX XX		LONL XXX XX
INERTIAL DOWNRANGE	XXXXX	N MI
INERTIAL CROSSRANGE	XXXXX	N MI

Figure 55. Abort Scan Table Summary Sheet

RTACF RETURN TO EARTH DIGITALS

STA ID		CSM WT	XXXXX
GETV	XXX XX XX	AREA	XX

MANEUVER QUANTITIES

GETI	XXX XX XX	DELTA VT	XXX
T-A ABORT	XXX	DELTA VC	XXX
REFSMMAT ID		DELTA TB	XXX
ULLAGE QUADS	X	DELTA TU	XXX
RLH	XXX	RO	XXX
PLH	XXX	PI	XXX
YLH	XXX	YM	XXX

ENTRY INTERFACE

GETEI	XXX XX XX	INCL	XXXXX
VEI	XXXXX	GAMMAEI	XXXXX
LATEI	XXX XX	LONGEI	XXX XX

LANDING

GETL	XXX XX XX		
LATL	XXX	LONL	XXX
INERTIAL DOWNRANGE		XXX XX N MI	
INERTIAL CROSSRANGE		XXX XX N MI	

EXTERNAL DELTA V TARGETS

P30	VX	XXXX	VY	XXXXX	VZ	XXXXX
P40	VF	XXXX	VS	XXXXX	VD	XXXXX

LOOK ANGLES

THETA E	XXX XX	PSIE	XXX XX
THETA S	XXX XX	PSIS	XXX XX

Figure 56. Return-to-Earth Digital Summary Sheet

RTACF
DOCKED CSM GIMBAL ANGLES COMPUTATION
CSM GIMBAL ANGLES ARE COMPUTED

* CSM *

* LM *

IMU GIMBAL ANGLES			IMU GIMBAL ANGLES			REFSMAT			ID
IGA(P)	MGA(Y)	OGA(R)	IGA(P)	MGA(R)	OGA(Y)	REFSMAT	ID		
DEG	DEG	DEG	DEG	DEG	DEG				
XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXXX	XXX		

Figure 57. Docking Alignment Summary Sheet

MIDCOURSE PROCESSOR
LUNAR ORBIT INSERTION DISPLAY

TA LOI	XXXXXXXX	DAZLOI	XXXXXXXX	SINDAL	XXXXXXXX
TA ND	XXXXXXXX	LATLOI	XXXXXXXX	LONLOI	XXXXXXXX
HT LOI	XXXXXXXX	AZ PCD	XXXXXXXX	VELPCF	XXXXXXXX
HT PC	XXXXXXXX	AZ PCR	XXXXXXXX	VELPCE	XXXXXXXX
HT ND	XXXXXXXX	A PC	XXXXXXXX	E PC	XXXXXXXX

STATE VECTOR AND UNIT VECTOR AT PERICYNTHION TIME XXXXXXXX

X PC	XXXXXXXX	Y PC	XXXXXXXX	Z PC	XXXXXXXX
XD PC	XXXXXXXX	YD PC	XXXXXXXX	ZD PC	XXXXXXXX
ZUNIT	XXXXXXXX	YUNIT	XXXXXXXX	ZUNIT	XXXXXXXX

POSITION UNIT VECTORS AT LOI AND NODE

LOI	XXXXXXXX	XXXXXXXX	XXXXXXXX
NODE	XXXXXXXX	XXXXXXXX	XXXXXXXX

UNIT NORMALS

LOI	XXXXXXXX	XXXXXXXX	XXXXXXXX
LLS'	XXXXXXXX	XXXXXXXX	XXXXXXXX
LLS''	XXXXXXXX	XXXXXXXX	XXXXXXXX

Figure 58. Midcourse Display Summary Sheet

SUMMARY SHEET - PRE-BURN ENTRY

EVENT TIMES	GMT				GET			RET			LATITUDE		LONGITUDE	
	DAY	HR	MIN	SEC	HR	MIN	SEC	HR	MIN	SEC	DEG	MIN	DEG	MIN
RETROFIRE	X	XXX	XX	XXXX	XXX	XX	XXXX	XXX	XX	XXXX	XXX	XXXX	XXX	XXXX
BURN TERMINATION	X	XXX	XX	XXXX	XXX	XX	XXXX	XXX	XX	XXXX	XXX	XXXX	XXX	XXXX
SEPARATION	X	XXX	XX	XXXX	XXX	XX	XXXX	XXX	XX	XXXX	XXX	XXXX	XXX	XXXX
ENTRY INTERFACE	X	XXX	XX	XXXX	XXX	XX	XXXX	XXX	XX	XXXX	XXX	XXXX	XXX	XXXX
BEGIN BLACKOUT (VHF)	X	XXX	XX	XXXX	XXX	XX	XXXX	XXX	XX	XXXX	XXX	XXXX	XXX	XXXX
BEGIN BLACKOUT (S)	X	XXX	XX	XXXX	XXX	XX	XXXX	XXX	XX	XXXX	XXX	XXXX	XXX	XXXX
.05G'S (EMS)	X	XXX	XX	XXXX	XXX	XX	XXXX	XXX	XX	XXXX	XXX	XXXX	XXX	XXXX
0.2G'S	X	XXX	XX	XXXX	XXX	XX	XXXX	XXX	XX	XXXX	XXX	XXXX	XXX	XXXX
REVERSE BANK	X	XXX	XX	XXXX	XXX	XX	XXXX	XXX	XX	XXXX	XXX	XXXX	XXX	XXXX
END BLACKOUT (S)	X	XXX	XX	XXXX	XXX	XX	XXXX	XXX	XX	XXXX	XXX	XXXX	XXX	XXXX
END BLACKOUT (VHF)	X	XXX	XX	XXXX	XXX	XX	XXXX	XXX	XX	XXXX	XXX	XXXX	XXX	XXXX
DROGUE DEPLOY	X	XXX	XX	XXXX	XXX	XX	XXXX	XXX	XX	XXXX	XXX	XXXX	XXX	XXXX
MAIN DEPLOY	X	XXX	XX	XXXX	XXX	XX	XXXX	XXX	XX	XXXX	XXX	XXXX	XXX	XXXX
LANDING	X	XXX	XX	XXXX	XXX	XX	XXXX	XXX	XX	XXXX	XXX	XXXX	XXX	XXXX
BURN QUANTITIES														
DELTA VELOCITY	=	XXXXXX FT/SEC												
DELTA TIME	=	XXXXXX MIN				XX SEC								
WEIGHT	=	XXXXXX POUNDS												
PERIGEE RESULTING	=	XXXXXX N.MILES												
TRUE ANOMALY	=	XXXXXX DEGREES												
ALTITUDE	=	XXXXXX N.MILES												
LATITUDE	=	XXXXXX DEG				XX MIN								
LONGITUDE	=	XXXXXX DEG				XX MIN								
EXTERNAL DELTA VX	=	XXXXXX												
EXTERNAL DELTA VY	=	XXXXXX												
EXTERNAL DELTA VZ	=	XXXXXX												
GIMBAL ANGLES														
	OUTER (ROLL)				INNER (PITCH)			MIDDLE (YAW)						
BURN	XXX				XXX			XXX						
SEPARATION	XXX				XXX			XXX						
400K	XXX				XXX			XXX						
.05G	XXX				XXX			XXX						
BACKUP BANK AT .05	XXX				XXX			XXX						
BACKUP BANK AT RB	XXX				XXX			XXX						

Figure 59. ARS Summary Sheet

SUMMARY SHEET - PRE-BURN ENTRY

REENTRY QUANTITIES

INITIAL BANK XXX DEG (RIGHT)
 BACKUP BANK XXX DEG (RIGHT)

HRS MIN SEC
 RETRB XX XX XX
 RET400K XX XX XX
 RET.05G XX XX XX
 RET.2G XX XX XX
 RET.05G XX XX XX
 RETBBO (S) XX XX XX
 RETEBO (S) XX XX XX
 RETDD XX XX XX
 RETMD XX XX XX

LANDING POINTS

LATITUDE LONGITUDE
 DEG MIN DEG MIN
 MIN LIFT XXX XX
 MAX LIFT XXX XX
 TARGET XXX XX
 G AND N XXX XX
 BACKUP XXX XX
 MISS DISTANCE XXXXX NM (NORTH)
 XXXXX NM (WEST)

BACKUP QUANTITIES

BANK-R-BANK

RT400K XXXXX N.MILES
 V400K XXXXX FT/SEC
 GAMMA400K XXXXX DEG
 (RP-RT).2G XXXXX N.MILES
 MAXG XXXXX G
 L/D XXXXX
 WEIGHT XXXXX LBS

INITIAL BANK XXX DEG (RIGHT)
 BACKUP BANK XXX DEG (RIGHT)
 RETRB XX HRS. XX MIN. XXX SEC.

CONSTANT G.

G LEVEL XXXXX G
 ROLL DIRECTION

EMS QUANTITIES

RET XX HRS. XX MIN. XXXX SEC
 V XXXXXXXX FT/SEC
 RNG XXXXXXXX N.MILES
 ALTD XXXXXXXX FT
 G XXXXXXXX G

GUIDANCE SUPPORT

RT400K XXXXX N.MILES
 VI XXXXX FT/SEC
 VL XXXXX FT/SEC
 A0 XXXXX G
 DL XXXXX G
 LEWD XXXXX
 ASKEP XXXXX N.MILES
 FINAL PHASE XXXXX N.MILES
 UPPHASE XXXXX N.MILES
 GAMMA CORRECTION XXXXX N.MILES
 DOWN CONTROL XXXXX N.MILES
 D0 XXXXX G
 MAX ALTD XXXXX FT
 MAX G1 XXXXX G
 MAX G2 XXXXX G
 MIN G XXXXX G
 VCIRC XX HRS. XX MIN. XXXX SEC

P - 61 QUANTITIES

RETEMS XX HRS. XX MIN. XXXX SEC
 VEMS XXXXX FT/SEC
 RNGEMS XXXXX N.MILES
 ALTDEMS XXXXX FT.
 MAXG XXXXX G

HEATING

MAX HEAT RATE XXXXX BTU/FT**2-SEC
 TOTAL HEAT LOAD XXXXX BTU/FT**2

Figure 59. ARS Summary Sheet (Continued)

***** LUNAR ORBIT INSERTION PLANNING *****

CSM STA ID

GET VECTOR XXX XX XX
 GMT VECTOR XXX XX XX
 LAT-LLS XXX XX
 LON-LLS XXX XX

AZ-LLS XXXXX
 HPLPO XXXXX
 HCIRC XXXXX
 HALPO XXXXX
 DVMAX XXXXX

CODE	GET-LOI	DV-LOI	ALT-NODE	THETA	ALT-PL	AZ-LLS	ETA-NODE	OUT-OF-PLN	DVRM
XXX X	XXX XX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX
XXX X	XXX XX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX
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Figure 60. LOI-1 Planning Summary Sheet

***** EXTERNAL DELTA V TARGETS *****

CODE	GET-IGN	VX	VY	VZ	GMT-IMP
XXX X	XXX XX XXX	XXXXX	XXXXX	XXXXX	XXX XX XX
XXX X	XXX XX XXX	XXXXX	XXXXX	XXXXX	XXX XX XX
.	.	.	.	.	.
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.	.	.	.	.	.

PSEUDO-LLS LONGITUDE = XXXXX

PSEUDO-LLS LATITUDE = XXXXX

Figure 61. LOI-1 External ΔV Target Summary Sheet

FDO MISSION PLAN TABLE

CSM STA ID			GETDKI XXX XX XX			LEM STA ID		
GETAV			XXX XX XX			GETAV	XXX XX XX	
ID			NPC	XXXXX	NC1	XXXXX	NCH2	XXXXX
M			NH	XXXXX	NCH1	XXXXX	NSR	XXXXX
GETBI			DT	DELTA V	DVREM	HA	HP	CODE
XXX XX XX				XXXXX	XXXXX	XXXXX	XXXXX	XXXXXX
			XXX XX XX					
XXX XX XX				XXXXX	XXXXX	XXXXX	XXXXX	XXXXX
			XXX XX XX					
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Figure 62. Mission Planning Table Summary Sheet

LUNAR FDO GENERAL PURPOSE MANEUVER TABLE

MAN VEH STA ID		WEIGHT	XXXXX
		CODE	XXXXX
		GETAV	XXX XX XX
		GETTH	XXX XX XX
GET A	XXX XX XX		
HA	XXXXX		
LONG A	XXX XX		
LAT A	XXX XX	GETI	XXX XX XX
GET P	XXX XX XX	DEL V MAN	XXXXX
HP	XXXXX	DEL TB MAN	XXXXX
LONG P	XXX XX	PIT MAN	XXXXX
LAT A	XXX XX	YAW MAN	XXXXX
A	XXXXX		
E	XXXXX	H MAN	XXXXX
I	XXXXX	LONG MAN	XXX XX
NODE AN	XXX XX	LAT MAN	XXX XX
DEL G	XXX XX		
TRUE A	XXXXX		
VX	XXXXX		
VY	XXXXX		
VZ	XXXXX		

Figure 63. General Purpose Maneuver Planning Table

LM DESCENT PLANNING												
STA ID	GMTV			XXX	XX	XX	MODE	X	LAT LLS	XXX	XX	
LM WT	XXXXXX			GETV	XXX	XX	XX		LONG LLS	XXX	XX	
MVR/REV	GETTH/GETIG			LIG/DV		AC/HPC		DEL/THPC		DVX/DVY		DVZ
DOI												
XXX	XXX	XX	XX	XXX	XX	XXXXXX		XXXXXX		XXXXX	XXXXX	XXXXXX
	XXX	XX	XX	XXX	XX	XXXXXX		XXXXXX		XXXXX	XXXXX	XXXXXX
XXX	XXX	XX	XX	XXX	XX	XXXXXX		XXXXXX		XXXXX	XXXXX	XXXXXX
	XXX	XX	XX	XXX	XX	XXXXXX		XXXXXX		XXXXX	XXXXX	XXXXXX
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Figure 64. Lunar Descent Planning Display Summary Sheet

LUNAR RENDEZVOUS PLAN TABLE

CSM STA		LM POSITION				LM STA		LM LIFETIME			
GMTV		XXX XX XX		XXX XX		GMTV		XXX XX XX		DV MAX	XXXXX
GETV		XXX XX XX		XXX XX		GETV		XXX XX XX		MIN H	XXXXX
MANVR VEH		XX		THRESH T	XXX XX XX	DTC SI		XXX XX XX		WT	XXXXX

ID	M	DH	DV CSM	DV	LM	DV DI	DV DT	DV CSI	DV CDH	DV PC	DV TP	GETPI
X	X	XXX	XXXXX	XXXXX	XXXXX	XX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXX XX XX
X	X	XXX	XXXXX	XXXXX	XXXXX	XX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXX XX XX
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LM TARGETING

GETLO	XXX XX XX	VF XXXXX	DEL Y ANG	XXXXX	AZP	XXXXX
WEDGE ANGLE	XXXXX	YS XXXXX	YD	XXXXX	AZB	XXXXX
		DEL R	XXXXX			

Figure 65. Lunar Rendezvous Plan Table Summary Sheet

RENDEZVOUS EVALUATION DISPLAY

ID		X	M		X	DT		DV	VEH	PURP	CODE	PHASE	HEIGHT	HA	HP
GET															
XXX XX XX								XXX	XX	XXX	XXXXXX	XXXXXX	XXXXXX	XXXXXX	XXXXXX
						XXX XX									
XXX XX XX								XXX	XX	XXX	XXXXXX	XXXXXX	XXXXXX	XXXXXX	XXXXXX

TARGET AT CDH

HA	XXXXXX	HP	XXXXXX	PHASE	XXXXXX	HEIGHT	XXXXXX
PITCH	YAW	VX	VY	VZ			
XXXXXX	XXXXXX	XXXXXX	XXXXXX	XXXXXX			
XXXXXX	XXXXXX	XXXXXX	XXXXXX	XXXXXX			

Figure 66. Rendezvous Evaluation Display Summary Sheet

TWO-IMPULSE RENDEZVOUS TABLE-MULTIPLE SOLUTION

LM STA ID		CSM STA ID
LM GETTHS	XXX XX XX	CSM GETTHS XXX XX XX
MAN VEH	XX	PHASE XXXXX
WT	XXXXXX	DEL H XXXXX
GET1	XXX XX XX	OPTION
GMT1	XXX XX XX	XXX XX UNTIL DAYLIGHT

DEL V1	DEL V2	GET2	PITCH	YAW	PITCH	L	C
XXXXXX	XXXXXX	XXX XX XX	XXXXXX	XXXXXX	XXXXXX	X	X
XXXXXX	XXXXXX	XXX XX XX	XXXXXX	XXXXXX	XXXXXX	X	X
.	.	.	.	.	.	.	.
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Figure 67. Two-Impulse Rendezvous Table (Multiple Solution Summary Sheet)

TWO-IMPULSE RENDEZVOUS TABLE-SINGLE SOLUTION

LM STA ID				CSM STA ID			
LM GETTHS		XXX XX XX		CSM GETTHS		XXX XX XX	
MAN VEH		XX		PHASE		XXXXXX	
MODE		XXX		DEL H		XXXXXX	
ID		X		DEL PITCH		XXXXXX	
GET	XXX XX XX			EHOR	XXXXX		GMT XXX XX XX
DV	XXXXX	VX	XXXXX			XD	XXXXX DT XXX XX
YAW	XXXXX	VY	XXXXX	YAW	XXXXX	YD	XXXXX DT XXX XX
PITCH	XXXXX	VZ	XXXXX	PITCH	XXXXX	ZD	XXXXX DT XXX XX
XXX XX UNTIL DAYLIGHT				HA	XXXXX		HP XXXXX
GET	XXX XX XX	TGT AZ	TGT EL	RANGE		R DOT	X Z Y
XXX XX XX		XXXXX	XXXXX	XXXXX		XXXXX	XXXXX XXXXX XXXXX XXXXX
XXX XX XX		XXXXX	XXXXX	XXXXX		XXXXX	XXXXX XXXXX XXXXX XXXXX
XXX XX XX		XXXXX	XXXXX	XXXXX		XXXXX	XXXXX XXXXX XXXXX XXXXX
XXX XX XX		XXXXX	XXXXX	XXXXX		XXXXX	XXXXX XXXXX XXXXX XXXXX
GET	XXX XX XX			EHOR	XXXXX		GMT XXX XX XX
DV	XXXXX	VX	XXXXX			XD	XXXXX DT XXX XX
YAW	XXXXX	VY	XXXXX	YAW	XXXXX	YD	XXXXX DT XXX XX
PITCH	XXXXX	VZ	XXXXX	PITCH	XXXXX	ZD	XXXXX DT XXX XX
XXX XX UNTIL DARKNESS				HA	XXXXX		HP XXXXX
GET	XXX XX XX	TGT AZ	TGT EL	RANGE		R DOT	X Z Y
XXX XX XX		XXXXX	XXXXX	XXXXX		XXXXX	XXXXX XXXXX XXXXX XXXXX
XXX XX XX		XXXXX	XXXXX	XXXXX		XXXXX	XXXXX XXXXX XXXXX XXXXX
XXX XX XX		XXXXX	XXXXX	XXXXX		XXXXX	XXXXX XXXXX XXXXX XXXXX
XXX XX XX		XXXXX	XXXXX	XXXXX		XXXXX	XXXXX XXXXX XXXXX XXXXX
GET	XXX XX XX	TGT AZ	TGT EL	RANGE		R DOT	X Z Y
XXX XX XX		XXXXX	XXXXX	XXXXX		XXXXX	XXXXX XXXXX XXXXX XXXXX
XXX XX XX		XXXXX	XXXXX	XXXXX		XXXXX	XXXXX XXXXX XXXXX XXXXX
XXX XX XX		XXXXX	XXXXX	XXXXX		XXXXX	XXXXX XXXXX XXXXX XXXXX
XXX XX XX		XXXXX	XXXXX	XXXXX		XXXXX	XXXXX XXXXX XXXXX XXXXX
GET	XXX XX XX	TGT AZ	TGT EL	RANGE		R DOT	X Z Y
XXX XX XX		XXXXX	XXXXX	XXXXX		XXXXX	XXXXX XXXXX XXXXX XXXXX
XXX XX XX		XXXXX	XXXXX	XXXXX		XXXXX	XXXXX XXXXX XXXXX XXXXX
XXX XX XX		XXXXX	XXXXX	XXXXX		XXXXX	XXXXX XXXXX XXXXX XXXXX
XXX XX XX		XXXXX	XXXXX	XXXXX		XXXXX	XXXXX XXXXX XXXXX XXXXX
GET	XXX XX XX	TGT AZ	TGT EL	RANGE		R DOT	X Z Y
XXX XX XX		XXXXX	XXXXX	XXXXX		XXXXX	XXXXX XXXXX XXXXX XXXXX
XXX XX XX		XXXXX	XXXXX	XXXXX		XXXXX	XXXXX XXXXX XXXXX XXXXX
XXX XX XX		XXXXX	XXXXX	XXXXX		XXXXX	XXXXX XXXXX XXXXX XXXXX
XXX XX XX		XXXXX	XXXXX	XXXXX		XXXXX	XXXXX XXXXX XXXXX XXXXX
GET	XXX XX XX	TGT AZ	TGT EL	RANGE		R DOT	X Z Y
XXX XX XX		XXXXX	XXXXX	XXXXX		XXXXX	XXXXX XXXXX XXXXX XXXXX
XXX XX XX		XXXXX	XXXXX	XXXXX		XXXXX	XXXXX XXXXX XXXXX XXXXX
XXX XX XX		XXXXX	XXXXX	XXXXX		XXXXX	XXXXX XXXXX XXXXX XXXXX
XXX XX XX		XXXXX	XXXXX	XXXXX		XXXXX	XXXXX XXXXX XXXXX XXXXX
GET	XXX XX XX	TGT AZ	TGT EL	RANGE		R DOT	X Z Y
XXX XX XX		XXXXX	XXXXX	XXXXX		XXXXX	XXXXX XXXXX XXXXX XXXXX
XXX XX XX		XXXXX	XXXXX	XXXXX		XXXXX	XXXXX XXXXX XXXXX XXXXX
XXX XX XX		XXXXX	XXXXX	XXXXX		XXXXX	XXXXX XXXXX XXXXX XXXXX
XXX XX XX		XXXXX	XXXXX	XXXXX		XXXXX	XXXXX XXXXX XXXXX XXXXX
GET	XXX XX XX	TGT AZ	TGT EL	RANGE		R DOT	X Z Y
XXX XX XX		XXXXX	XXXXX	XXXXX		XXXXX	XXXXX XXXXX XXXXX XXXXX
XXX XX XX		XXXXX	XXXXX	XXXXX		XXXXX	XXXXX XXXXX XXXXX XXXXX
XXX XX XX		XXXXX	XXXXX	XXXXX		XXXXX	XXXXX XXXXX XXXXX XXXXX
XXX XX XX		XXXXX	XXXXX	XXXXX		XXXXX	XXXXX XXXXX XXXXX XXXXX
GET	XXX XX XX	TGT AZ	TGT EL	RANGE		R DOT	X Z Y
XXX XX XX		XXXXX	XXXXX	XXXXX		XXXXX	XXXXX XXXXX XXXXX XXXXX
XXX XX XX		XXXXX	XXXXX	XXXXX		XXXXX	XXXXX XXXXX XXXXX XXXXX
XXX XX XX		XXXXX	XXXXX	XXXXX		XXXXX	XXXXX XXXXX XXXXX XXXXX
XXX XX XX		XXXXX	XXXXX	XXXXX		XXXXX	XXXXX XXXXX XXXXX XXXXX
GET	XXX XX XX	TGT AZ	TGT EL	RANGE		R DOT	X Z Y
XXX XX XX		XXXXX	XXXXX	XXXXX		XXXXX	XXXXX XXXXX XXXXX XXXXX
XXX XX XX		XXXXX	XXXXX	XXXXX		XXXXX	XXXXX XXXXX XXXXX XXXXX
XXX XX XX		XXXXX	XXXXX	XXXXX		XXXXX	XXXXX XXXXX XXXXX XXXXX
XXX XX XX		XXXXX	XXXXX	XXXXX		XXXXX	XXXXX XXXXX XXXXX XXXXX
GET	XXX XX XX	TGT AZ	TGT EL	RANGE		R DOT	X Z Y
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XXX XX XX		XXXXX	XXXXX	XXXXX		XXXXX	XXXXX XXXXX XXXXX XXXXX
XXX XX XX		XXXXX	XXXXX	XXXXX		XXXXX	XXXXX XXXXX XXXXX XXXXX
GET	XXX XX XX	TGT AZ	TGT EL	RANGE		R DOT	X Z Y
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XXX XX XX		XXXXX	XXXXX	XXXXX		XXXXX	XXXXX XXXXX XXXXX XXXXX
XXX XX XX		XXXXX	XXXXX	XXXXX		XXXXX	XXXXX XXXXX XXXXX XXXXX
GET	XXX XX XX	TGT AZ	TGT EL	RANGE		R DOT	X Z Y
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GET	XXX XX XX	TGT AZ	TGT EL	RANGE		R DOT	X Z Y
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XXX XX XX		XXXXX	XXXXX	XXXXX		XXXXX	XXXXX XXXXX XXXXX XXXXX
GET	XXX XX XX	TGT AZ	TGT EL	RANGE		R DOT	X Z Y
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XXX XX XX		XXXXX	XXXXX	XXXXX		XXXXX	XXXXX XXXXX XXXXX XXXXX
GET	XXX XX XX	TGT AZ	TGT EL	RANGE		R DOT	X Z Y
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GET	XXX XX XX	TGT AZ	TGT EL	RANGE		R DOT	X Z Y
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GET	XXX XX XX	TGT AZ	TGT EL	RANGE		R DOT	X Z Y
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GET	XXX XX XX	TGT AZ	TGT EL	RANGE		R DOT	X Z Y
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GET	XXX XX XX	TGT AZ	TGT EL	RANGE		R DOT	X Z Y
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GET	XXX XX XX	TGT AZ	TGT EL	RANGE		R DOT	X Z Y
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GET	XXX XX XX	TGT AZ	TGT EL	RANGE		R DOT	X Z Y
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GET	XXX XX XX	TGT AZ	TGT EL	RANGE		R DOT	X Z Y
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XXX XX XX		XXXXX	XXXXX	XXXXX		XXXXX	XXXXX XXXXX XXXXX XXXXX
XXX XX XX		XXXXX	XXXXX	XXXXX		XXXXX	XXXXX XXXXX XXXXX XXXXX
GET	XXX XX XX	TGT AZ	TGT EL	RANGE		R DOT	X Z Y
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GET	XXX XX XX	TGT AZ	TGT EL	RANGE		R DOT	X Z Y
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XXX XX XX		XXXXX	XXXXX	XXXXX		XXXXX	XXXXX XXXXX XXXXX XXXXX
XXX XX XX		XXXXX	XXXXX	XXXXX		XXXXX	XXXXX XXXXX XXXXX XXXXX
GET	XXX XX XX	TGT AZ	TGT EL	RANGE		R DOT	X Z Y
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XXX XX XX		XXXXX	XXXXX	XXXXX		XXXXX	XXXXX XXXXX XXXXX XXXXX
XXX XX XX		XXXXX	XXXXX	XXXXX		XXXXX	XXXXX XXXXX XXXXX XXXXX
XXX XX XX		XXXXX	XXXXX	XXXXX		XXXXX	XXXXX XXXXX XXXXX XXXXX
GET	XXX XX XX	TGT AZ	TGT EL	RANGE		R DOT	X Z Y
XXX XX XX		XXXXX	XXXXX	XXXXX		XXXXX	XXXXX XXXXX XXXXX XXXXX
XXX XX XX		XXXXX	XXXXX	XXXXX		XXXXX	XXXXX XXXXX XXXXX XXXXX
XXX XX XX		XXXXX	XXXXX	XXXXX		XXXXX	XXXXX XXXXX XXXXX XXXXX
XXX XX XX		XXXXX	XXXXX	XXXXX		XXXXX	XXXXX XXXXX XXXXX XXXXX
GET	XXX XX XX	TGT AZ	TGT EL	RANGE		R DOT	X Z Y
XXX XX XX		XXXXX	XXXXX	XXXXX		XXXXX	XXXXX XXXXX XXXXX XXXXX
XXX XX XX		XXXXX	XXXXX	XXXXX		XXXXX	XXXXX XXXXX XXXXX XXXXX
XXX XX XX		XXXXX	XXXXX	XXXXX		XXXXX	XXXXX XXXXX XXXXX XXXXX
XXX XX XX		XXXXX	XXXXX	XXXXX		XXXXX	XXXXX XXXXX XXXXX XXXXX
GET	XXX XX XX	TGT AZ	TGT EL	RANGE		R DOT	X Z Y
XXX XX XX		XXXXX	XXXXX	XXXXX		XXXXX	XXXXX XXXXX XXXXX XXXXX
XXX XX XX		XXXXX	XXXXX	XXXXX		XXXXX	XXXXX XXXXX XXXXX XXXXX
XXX XX XX		XXXXX	XXXXX	XXXXX		XXXXX	XXXXX XXXXX XXXXX XXXXX
XXX XX XX		XXXXX	XXXXX	XXXXX		XXXXX	XXXXX XXXXX XXXXX XXXXX
GET	XXX XX XX	TGT AZ	TGT EL	RANGE		R DOT	X Z Y
XXX XX XX		XXXXX	XXXXX	XXXXX		XXXXX	XXXXX XXXXX XXXXX XXXXX
XXX XX XX		XXXXX	XXXXX	XXXXX		XXXXX	XXXXX XXXXX XXXXX XXXXX
XXX XX XX		XXXXX	XXXXX	XXXXX		XXXXX	XXXXX XXXXX XXXXX XXXXX
XXX XX XX		XXXXX	XXXXX	XXXXX		XXXXX	XXXXX XXXXX XXXXX XXXXX
GET	XXX XX XX	TGT AZ	TGT EL	RANGE		R DOT	X Z Y
XXX XX XX		XXXXX	XXXXX	XXXXX		XXXXX	XXXXX XXXXX XXXXX XXXXX
XXX XX XX		XXXXX	XXXXX	XXXXX		XXXXX	XXXXX XXXXX XXXXX XXXXX
XXX XX XX		XXXXX	XXXXX	XXXXX		XXXXX	XXXXX XXXXX XXXXX XXXXX
XXX XX XX		XXXXX	XXXXX	XXXXX		XXXXX	XXXXX XXXXX XXXXX XXXXX
GET	XXX XX XX	TGT AZ	TGT EL	RANGE		R DOT	X Z Y
XXX XX XX		XXXXX	XXXXX	XXXXX		XXXXX	XXXXX XXXXX XXXXX XXXXX
XXX XX XX		XXXXX	XXXXX	XXXXX		XXXXX	XXXXX XXXXX XXXXX XXXXX
XXX XX XX		XXXXX	XXXXX	XXXXX		XXXXX	XXXXX XXXXX XXXXX XXXXX
XXX XX XX		XXXXX	XXXXX	XXXXX		XXXXX	XXXXX XXXXX XXXXX XXXXX
GET	XXX XX XX	TGT AZ	TGT EL	RANGE		R DOT	X Z Y
XXX XX XX		XXXXX	XXXXX	XXXXX		XXXXX	XXXXX XXXXX XXXXX XXXXX
XXX XX XX		XXXXX	XXXXX	XXXXX		XXXXX	XXXXX XXXXX XXXXX XXXXX
XXX XX XX		XXXXX	XXXXX	XXXXX		XXXXX	XXXXX XXXXX XXXXX XXXXX
XXX XX XX		XXXXX	XXXXX	XXXXX		XXXXX	XXXXX XXXXX XXXXX XXXXX
GET	XXX XX XX	TGT AZ	TGT EL	RANGE		R DOT	X Z Y
XXX XX XX		XXXXX	XXXXX	XXXXX		XXXXX	XXXXX XXXXX XXXXX XXXXX
XXX XX XX		XXXXX	XXXXX	XXXXX		XXXXX	XXXXX XXX

CSM INSERTION BURN DISPLAY

XXX PERFORMING CSI-CDH

TCSM INS BURN =	XXXXXXXXXX	XX XX XX XXXXX
TCSM INS BURN =	XXXXXXXXXX	XX XX XX XXXXX
X CSM INS =	XXXXXX	Z CSM INS = XXXXX
TCSI MANEUVER =	XXXXXXXXXX	XX XX XX XXXXX
TCSI MANEUVER =	XXXXXXXXXX	XX XX XX XXXXX
X CSI =	XXXXXX	
TCDH MANEUVER =	XXXXXXXXXX	XX XX XX XXXXX
TCSI MANEUVER =	XXXXXXXXXX	XX XX XX XXXXX
X CDH =	XXXXXX	
		Z CDH = XXXXX

Figure 69. CSM Insertion Burn Display Summary Sheet

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