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MSC INTERNAL NOTE

JANUARY 30, 1970

APOLLO MISSION TECHNIQUES
MISSION H-2 AND SUBSEQUENT
LUNAR ORBIT ACTIVITIES.

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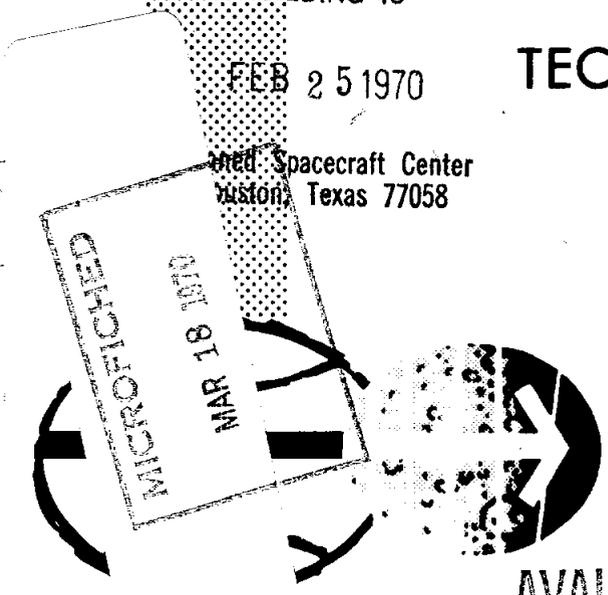
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TECHNIQUES DESCRIPTION

FINAL ISSUE.

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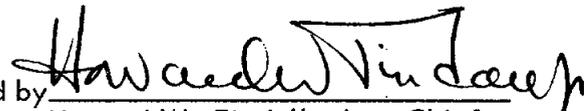
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NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
MANNED SPACECRAFT CENTER
HOUSTON, TEXAS

Approved by



Howard W. Tindall, Jr., Chief
Apollo Data Priority Coordination

FOREWORD

This Lunar Orbit Activities Mission Techniques Document is one of several documents describing the H-2 (Apollo 13) mission. The others are as follows:

- Launch Aborts
- Earth Parking Orbit and Translunar Injection
- Translunar Midcourse Corrections and Lunar Orbit Insertion
- Lunar Descent and Subsequent Rendezvous
- Descent Aborts
- Lunar Surface Phase
- Manual Ascent
- Lunar Ascent
- Transearth Injection, Midcourse Corrections, and Entry
- Tracking Data Selection Controllers Procedures
- Contingency Procedures

These documents contain the officially approved guidance and control sequence of events, data flow, and real-time decision logic for the H-2 mission. The purpose of these documents is to ensure compatibility of all related MSC and supporting contractor activities.

For each mission phase, a Data Priority Working Group has been established under the direction of the Chief, Apollo Data Priority Coordination, ASPO. These groups, which are comprised of representatives of MSC and support contractors, hold frequent meetings to coordinate their various associated activities, develop agreed upon mission techniques, and document them for ASPO. After formal review by ASPO, E&DD, FCOD, FOD, GAEC, MDC, MIT, NR, and TRW, a document such as this one is issued.

This is the final issue of the Lunar Orbit Activities Techniques Document. The mission dependent data limits and trajectory parameter values are applicable to the H-2 mission; the sequence of events, data flow, and decision logic are applicable to the H-2 and subsequent Apollo missions.

CONTENTS

	Page
1. INTRODUCTION	1-1
2. FUNCTIONAL DESCRIPTION OF THE LUNAR ORBIT ACTIVITIES	2-1
3. GENERAL TECHNIQUES DESCRIPTION	3-1
3.1 Go/No-Go Decisions	3-1
3.2 LM Rendezvous Navigation	3-1
3.3 CSM Rendezvous Navigation and Evaluation	3-6
3.4 LM and CSM State Vector Correction Limits During Rendezvous Navigation	3-7
3.5 CSM Activities During LM Maneuvers	3-8
4. DETAILED TECHNIQUES DISCUSSION	4-1
4.1 Docked Activities	4-1
4.1.1 Descent Orbit Insertion	4-1
4.1.2 Prerest Period LM Partial Activation and Pseudo Landing-Site Observations	4-3
4.1.3 State Vector and REFSMMAT Updates	4-3
4.1.4 Docked IMU Alignment and Drift Check	4-4
4.1.5 CMC, LGC and AEA Clock Synchronization	4-5
4.1.6 AGS Initialization and Alignment	4-6
4.2 Undocked Activities	4-6
4.2.1 Undocking/Separation of the CSM from the LM	4-7
4.2.2 PIPA Bias Calibration	4-7
4.2.3 AGS Initialization and Calibration	4-8
4.2.4 IMU Alignments and Drift Checks	4-8
4.2.5 RR and VHF Verification	4-9
4.2.6 CSM CIRC Maneuver	4-9
4.2.7 Landing Site Observations	4-9
4.2.8 Descent Targeting	4-12
4.2.8.1 State Vector and \underline{R}_{LS} Updates	4-12
4.2.8.2 MSFN Consistency Check	4-13
4.2.8.3 Propagation Bias Computation	4-16
4.2.8.4 Adjustments to \underline{R}_{LS} Using Sextant Data	4-16
4.2.8.5 Adjustments to \underline{R}_{LS} After PDI	4-18

CONTENTS (Continued)

	Page
4.3 Postorbit Insertion Activities	4-19
4.3.1 Pre-CSI Activities	4-19
4.3.2 CSI Maneuver	4-20
4.3.3 Plane Change Maneuver.	4-21
4.3.4 CDH Maneuver.	4-21
4.3.5 TPI Maneuver	4-22
4.3.6 TPM Maneuvers.	4-23
4.3.7 TPF.	4-23
4.4 Postrendezvous Activities	4-24
4.4.1 LM Jettison and Deorbit	4-24
4.4.2 CSM Plane Change	4-24
4.4.3 Photography and Landmark Tracking.	4-25
5. LUNAR ORBIT ACTIVITIES FLOW CHARTS.	5-1
REFERENCES	R-1

TABLES

	Page
3-1 Mandatory Guidance, Navigation, and Control Systems. . . .	3-2
3-2 Onboard Tracking Schedule for Mission H-2.	3-3
4-1 CSM External ΔV (P30) PAD	4-10
4-2 LM PDI ₁ PAD.	4-14
4-3 CSM Target ΔV (P76) PAD	4-15

ILLUSTRATIONS

2-1 Typical H Mission Lunar Orbit Activities Timeline	2-2
2-2 H-2 Mission Rendezvous Relative Motion (CSM Centered Curvilinear Coordinates).	2-8

NOMENCLATURE

AEA	abort electronics assembly
AGS	abort guidance system
AOS	acquisition of signal
AOT	alignment optical telescope
APS	ascent propulsion system
ASA	abort sensor assembly
ASCP	attitude set control panel
ATT	attitude
AZ	azimuth angle
BMAG	body-mounted attitude gyros
BSS	boresight star
BT	burn time (time from engine on to engine off)
BT F/A	burn time forward/aft
BT L/R	burn time left/right
BT U/D	burn time up/down
CDH	concentric delta height
CDR	commander
CDU	coupling data unit
CES	control electronics system
CIRC	CSM circularization
CM	command module
CMC	command module computer
CMP	command module pilot
COAS	crewman optical alignment sight (boresight)

NOMENCLATURE (Continued)

CSI	concentric sequence initiation
CSM	command and service module
DAP	digital autopilot
DECA	descent engine control assembly
DEDA	data entry and display assembly
DOI	descent orbit insertion
DPS	descent propulsion system
DSKY	display and keyboard (CMC)
ECI	earth-centered inertial (reference frame)
EDA	electronic display assembly (SCS)
EL, E	elevation angle
EMS	entry monitoring system
FDAI	flight director attitude indicator
FOV	field of view
GDC	gyro display coupler
g.e.t.	ground elapsed time
GETI	ground elapsed time of ignition (MCC-H nomenclature)
GETMID	desired g.e.t. of TPI (midpoint of darkness)
G&N	guidance and navigation
GNCS	CSM primary guidance, navigation, and control system
H	altitude
HA	height of apogee
HP	height of perigee
IAVG	moment of inertia (average of body Y- and Z-axes)

NOMENCLATURE (Continued)

IMU	inertial measurement unit
IXX	moment of inertia about body X-axis
LDS	landing site
LEB	lower equipment bay
LGC	LM guidance computer
LM	lunar module
LMP	lunar module pilot
LOI	lunar orbit insertion
LOS	line of sight or loss of signal
MCC	midcourse correction
MCC-H	Mission Control Center - Houston
MSFN	Manned Space Flight Network
N	number of apsidal crossing
OI	orbit insertion
ORDEAL	orbital rate display - earth and lunar
OUA	optical unit assembly (SXT, SCT)
P	pitch gimbal angle at burn attitude
P05	PGNCS Power-up Program
P20	RDZ Navigation Program
P27	LGC (CMC) Update Program
P30	External ΔV Program
P32	CSI Targeting Program
P33	CDH Targeting Program
P34	TPI Targeting Program

NOMENCLATURE (Continued)

P35	TPM Targeting Program
P40	DPS (or SPS) Thrust Program
P41	RCS Thrust Program
P42	APS Thrust Program
P47	Thrust Monitor Program
P51	IMU Orientation Determination Program
P52	IMU Realign Program
PAD	data voiced to crew from ground
PC	plane change
PDI	powered descent initiation
PGNCS	LM primary guidance, navigation, and control system
PIPA	pulse integrating pendulous accelerometer
R	range
R	roll gimbal angle at burn attitude
\dot{R}	range rate
RCS	reaction control system
RDZ	rendezvous
REFSMMAT	transformation matrix from ECI reference frame to IMU inertial coordinates
RHC	rotational hand controller
RJ/EC	reaction jet and engine on/off control assembly
\underline{R}_{LS}	landing site position vector
RR	rendezvous radar
RSS	root sum square

NOMENCLATURE (Continued)

RTCC	Real-Time Computer Complex
SC	spacecraft
SCS	stabilization and control system
SCT	scanning telescope
SPS	service propulsion system
SXT	sextant
SXTS	sextant star
TD	desired g.e.t. of TPI (same as GETMID)
TEI	transearth injection
TFI	time from ignition
TGO	time to go to event
TGT	target
THC	translational hand controller
TIG	time of ignition (CMC nomenclature)
TPF	terminal phase final
TPI	terminal phase initiation
TPM	terminal phase midcourse
TRN	trunnion angle of SXT
Vgm	total velocity to be gained
Vgx	X-component of velocity to be gained in body axes
Vgy	Y-component of velocity to be gained in body axes
Vgz	Z-component of velocity to be gained in body axes
Vgx (LV)	X-component of velocity to be gained in local vertical axes

NOMENCLATURE (Continued)

V_{gy} (LV)	Y-component of velocity to be gained in local vertical axes
V_{gz} (LV)	Z-component of velocity to be gained in local vertical axes
VHF	very high frequency
V_m	accumulated delta velocity
Y	yaw gimbal angle at burn attitude
\dot{Y}	out-of-plane velocity
ΔH	difference in height of LM and CSM
ΔTT	transfer duration from TPI to intercept
ΔV	delta velocity
ΔV F/A	ΔV forward/aft required for TPI (in LOS coordinates)
ΔV L/R	ΔV left/right required for TPI (in LOS coordinates)
ΔV U/D	ΔV up/down required for TPI (in LOS coordinates)

1. INTRODUCTION

This document presents the guidance and navigation sequence of events, data flow, and real-time decisions for the Mission H-1 (Apollo 12) lunar orbit activities. A functional description of activities is presented in Section 2. Section 3 presents a general discussion of techniques for both spacecraft. Section 4 presents a discussion of the sequence of events in chronological order.

The flow charts (Section 5) document the approved mission techniques for verification and usage of guidance and navigation (G & N) data and identify the backup procedures to be used if the G & N fails. The charts do not attempt to delineate all routine procedures except as they differ from standard practice or provide additional information. System comparison limits for each decision point were obtained from various analysis reports which are referenced in the notes.

The lunar orbit activities are divided into four distinct phases. The first phase starts with the lunar parking orbit maneuver; the second starts at undocking and terminates prior to the initiation of powered descent; the third phase starts at the completion of powered ascent; and the fourth phase starts after rendezvous and ends when preparation for the transearth injection (TEI) is initiated. Those activities between the third and fourth phases (powered descent, lunar surface activities, and lunar ascent) are covered in separate documents.

Mission H-2 differs significantly from the previous Apollo lunar landing missions (missions G and H-1) in that the descent orbit insertion (DOI) maneuver is performed by the command and service module (CSM) at the time of the previous second lunar orbit insertion (LOI-2) maneuver. The LOI and DOI maneuver sequence is now simply LOI and DOI instead of LOI-1 and LOI-2 and DOI performed by the LM after undocking. Undocking occurs one revolution earlier in order for the CSM to recircularize into a 60-nautical mile orbit to perform landmark tracking prior to powered descent initiation (PDI). The earlier undocking also allows two undocked P52 alignments (as opposed to one on previous missions) between undocking and PDI.

2. FUNCTIONAL DESCRIPTION OF THE LUNAR ORBIT ACTIVITIES

This section presents a brief description of the nominal sequence of events while in lunar orbit. The lunar orbit activities (Figure 2-1) consist of four phases as shown below:

- Phase 1 - DOI to undocking
- Phase 2 - Undocking to PDI
- Phase 3 - Rendezvous/docking
- Phase 4 - Docking to transearth injection (TEI)

Prior to phase 1, lunar orbit insertion (LOI) is performed in such a manner that the docked CSM/lunar module (LM) spacecraft is inserted into an intermediate lunar orbit of 60 by 170 nautical miles. Descent orbit insertion (DOI) is performed 4 hours after LOI, so that the CSM/LM will be in a nominal 58.9-by 8.3-nautical mile orbit about 22 hours later at PDI.*

In phase 1 the crew/MCC-H will verify that no CSM overburn has occurred at DOI, since an overburn of only 1 second will result in a lunar impact. After a safe orbit has been assured, the command module pilot (CMP) will observe/monitor landmarks to provide real-time training and familiarization with the terrain. The initial LM activation is performed by the commander (CDR) and lunar module pilot (LMP) beginning after loss-of-signal (LOS) during revolution 3. Following a sleep period for all crew members, the LMP and CDR reenter the LM and complete the LM activation and systems checkout. Landing site/landmark observations are again performed by the CMP prior to and after undocking and after CSM circularization (CIRC). These sightings are used to obtain a better relative position of the landing site with respect to the orbit.

* If after a period of MSFN tracking, the propagated PDI altitude falls outside the range, 30,000 feet to 70,000 feet, the crew will perform an MCC-H targeted RCS trim maneuver following the post-DOI rest period. If the PDI altitude is greater than 70,000 feet, the trim burn will be targeted to give a PDI altitude of 50,000 feet; if the PDI altitude is less than 30,000 feet, the trim burn will be targeted to raise the PDI altitude above 30,000 feet.

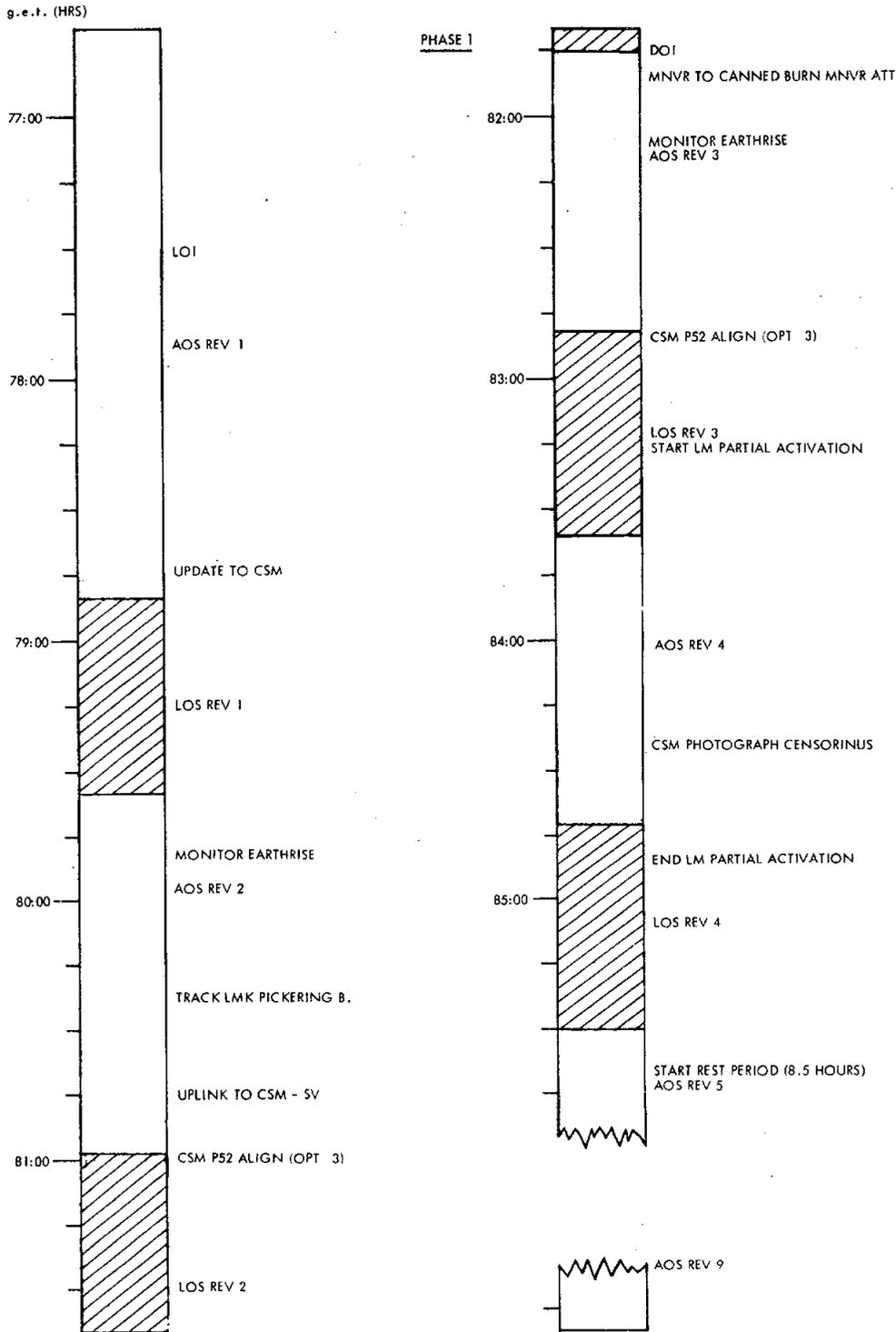


Figure 2-1. Typical H Mission Lunar Orbit Activities Timeline

g.e.t. (HRS)

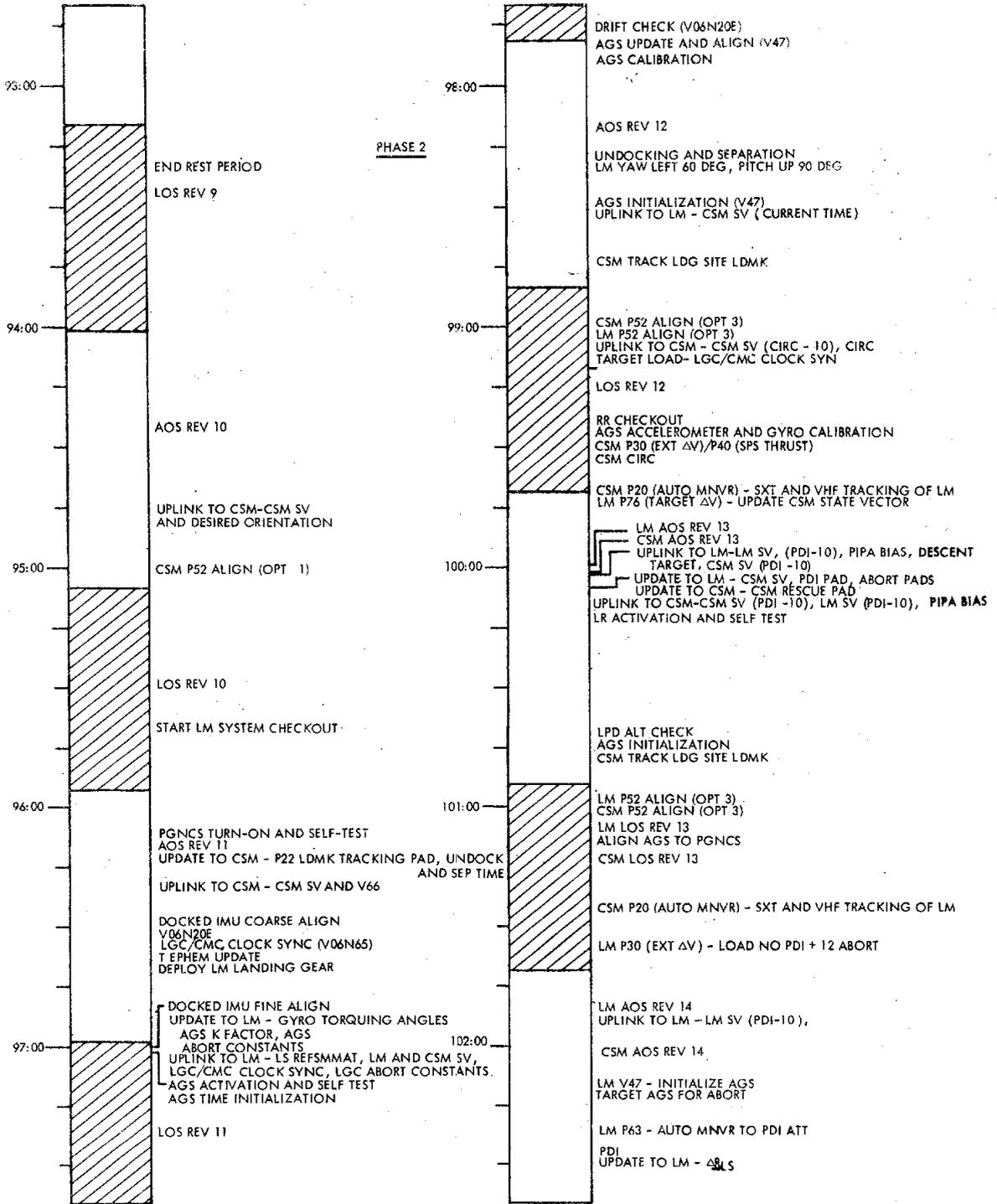


Figure 2-1. Typical H Mission Lunar Orbit Activities Timeline (Continued)

g.e.t. (HRS)

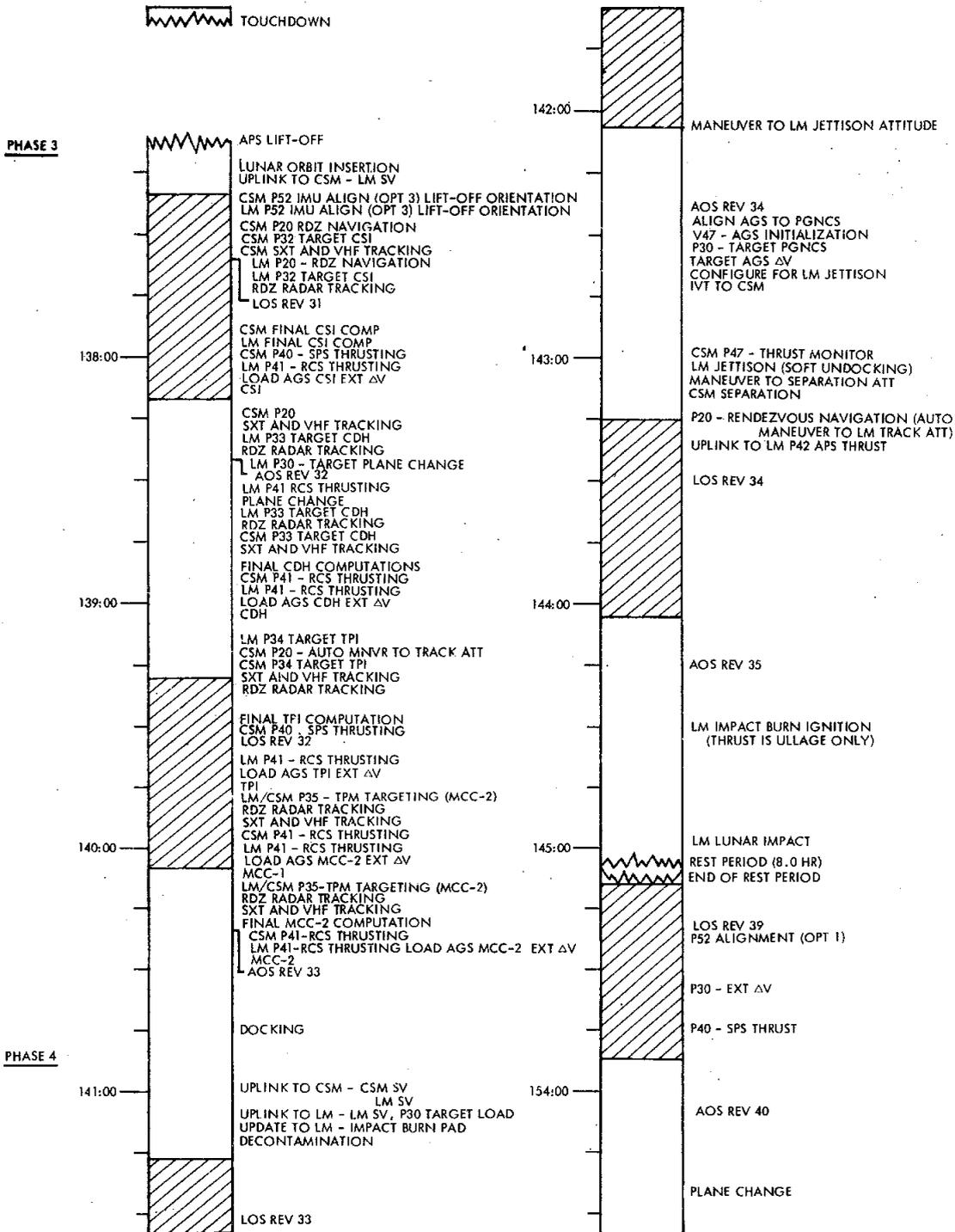


Figure 2-1. Typical H Mission Lunar Orbit Activities Timeline (Continued)

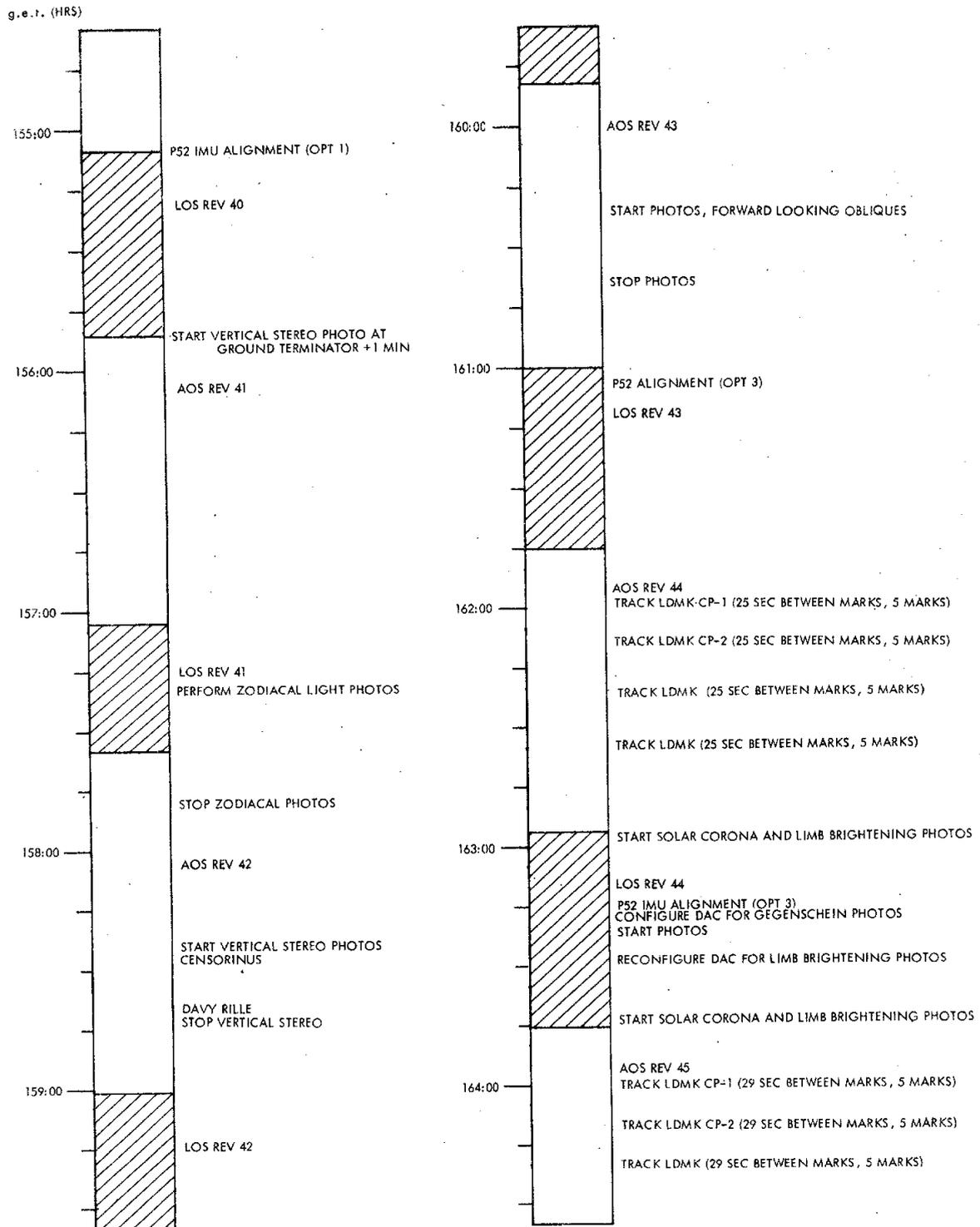


Figure 2-1. Typical H Mission Lunar Orbit Activities Timeline (Continued)

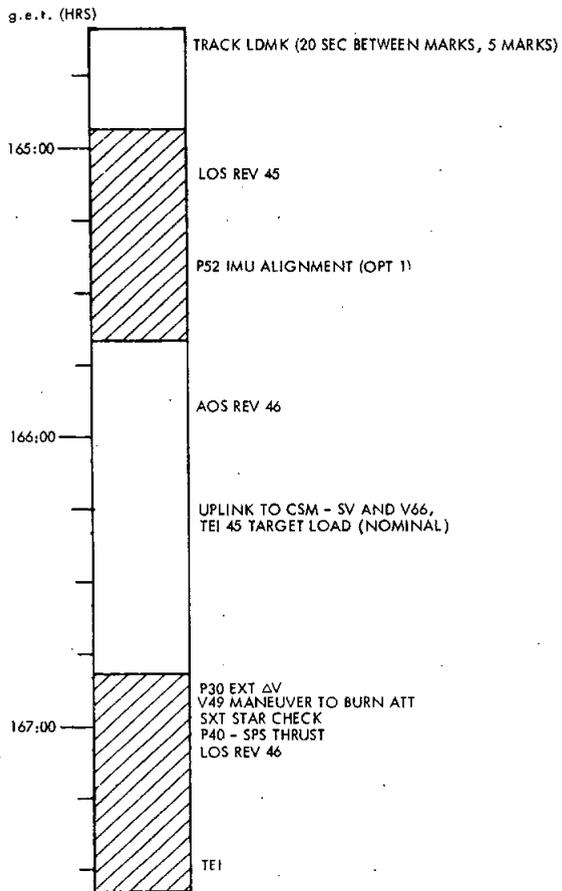


Figure 2-1. Typical H Mission Lunar Orbit Activities Timeline (Continued)

Undocking (phase 2 initiation) will take place 2-1/4 revolutions prior to PDI, approximately 1 hour, 25 minutes prior to the service propulsion system (SPS) circularization burn (CIRC). The undocking/separation commences with a soft undocking being executed followed by a CSM 1-foot per second separation rate. The separation burn is a 1-foot per second radially inward maneuver that results in the CSM being in front of and above the LM three-quarters of a revolution later at the time of the CSM CIRC. After the CSM undocks and separates from the LM, the LM will yaw left 60 degrees, pitch up 90 degrees for visual sighting of the CSM through the LM forward window.

Following powered descent, lunar surface activities, and lunar ascent by the LM, the third phase of the lunar orbit activities is initiated.

The LM active rendezvous (phase 3) is initiated after orbit insertion (nominally 9 by 45 nautical miles) with the CSM leading the LM approximately 16 degrees central angle. Figure 2-2 shows the relative position of the LM with respect to the CSM from orbit insertion through the terminal phase of the rendezvous.

Approximately 50 minutes after LM orbit insertion, the concentric sequence initiation (CSI) maneuver is performed using the LM RCS.* This nominally 50-foot per second maneuver results in a near-circular orbit of approximately 45 nautical miles and establishes the proper phasing and differential altitude conditions for the CDH maneuver point. A plane change maneuver which aligns the LM and CSM orbital planes is executed (if required) 30 minutes before CDH. The concentric delta height (CDH) maneuver (nominally zero) is executed one-half of the LM orbital period (approximately 58 minutes) after CSI and is designed to place the LM and CSM into coelliptic orbits with a constant delta height of 15 nautical miles between the two orbits.

* Beginning with mission H-3 (Apollo 14), the CSI/CDH/TPI/TPF rendezvous sequence may be replaced by a direct rendezvous technique in which TPI is initiated about 38 minutes after insertion. This would require 2 hours less between insertion and docking.

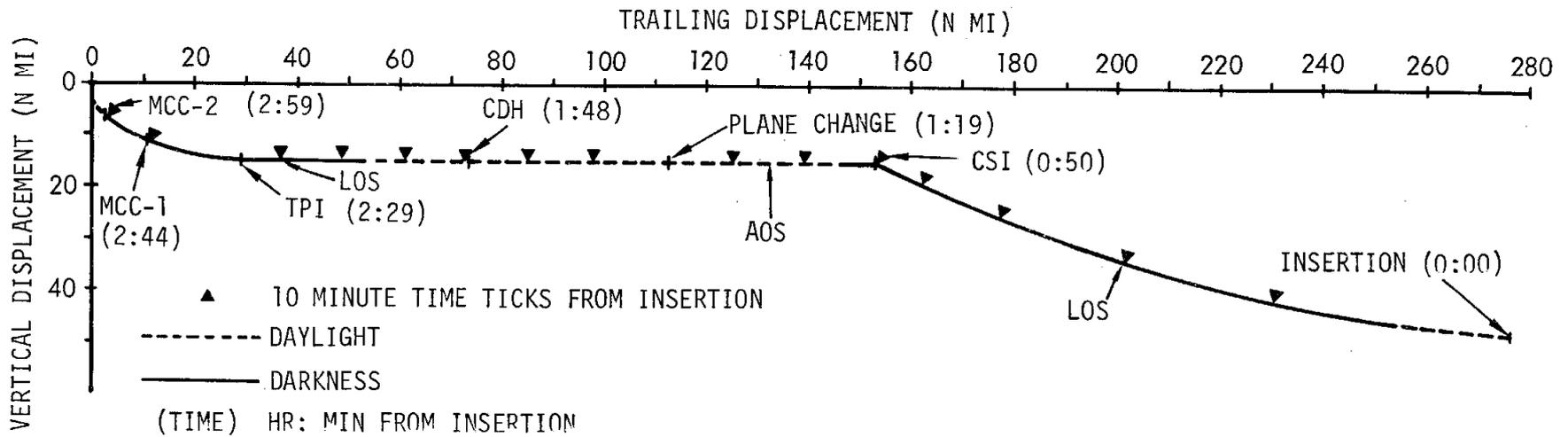


Figure 2-2. H-2 Mission Rendezvous Relative Motion (CSM Centered Curvilinear Coordinates) (for reference only)

When the elevation angle from the LM to the CSM reaches 26.6 degrees (approximately 39 minutes after CDH), the LM terminal phase initiation (TPI) maneuver is executed. This maneuver (nominally 26 feet per second) is designed to place the LM on a trajectory that will intercept the CSM orbit after 130 degrees of CSM travel. Two midcourse correction maneuvers are scheduled between TPI and braking: the first, 15 minutes after TPI, and the second, 30 minutes after TPI. Terminal phase final (TPF) braking is initiated about 33 minutes after TPI which circularizes the LM orbit at 60 nautical miles altitude. Docking is accomplished after completion of the braking maneuvers in which relative range and range rate are reduced to zero.

Following the rendezvous and docking, the LM crew configures the LM for jettison and then transfers to the CSM. The CSM executes a 1-foot per second retrograde maneuver after LM jettison to provide adequate separation distance for the subsequent LM deorbit and lunar surface impact near Fra Mauro. After the sleep period, a CSM plane change will be executed to place the CSM into an orbit to photograph candidate exploration sites. Vertical stereo and zodiacal light photographs and photographs of forward looking obliques at selected lunar sites/areas will be taken.

3. GENERAL TECHNIQUES DESCRIPTION

3.1 GO/NO-GO DECISIONS

Go/no-go decisions are made at various points in the mission based on the systems required for completion of that mission phase. The guidance, navigation, and control systems required for initiation of undocking, separation, CIRC, and PDI are listed in Table 3-1 for both the LM and CSM. The basic requirements before initiating CIRC and PDI are that the LM should be able to perform the navigation, targeting, and burns required for powered descent, ascent, and rendezvous, and the CSM should be able to perform the navigation, targeting, and burns required for a LM rescue. Table 3-1 assumes single failures only; i. e., when a system is not required, it is assumed that all other systems are available (exceptions are called out in the notes).

3.2 LM RENDEZVOUS NAVIGATION

Rendezvous navigation is conducted in the LM using the Rendezvous Navigation Program (P20) and rendezvous radar (RR) data to update the LM state vector in the LM guidance computer (LGC). RR tracking is conducted per the tracking schedule shown in Table 3-2. The W-matrix is manually reinitialized at specific points as shown in the tracking schedule to reduce navigation errors. The W-matrix values are 2,000 feet in each position component, 2 feet per second in each velocity component, and 5 milliradians for each angle for all reinitializations except following orbit insertion. The W-matrix values manually entered for the reinitialization following orbit insertion are 10,000 feet, 10 feet per second, and 15 milliradians.

Throughout the nominal rendezvous sequence from orbit insertion plus 18 minutes to TPF, the LM will be held at the rendezvous tracking attitude to maintain RR lock. This means that all rendezvous maneuvers (including the plane change) will be accomplished with the Z-axis bore-sighted (or nearly so) on the CSM. The LM will execute CDH by thrusting out individual components of delta velocity (ΔV) while maintaining RR lock.

Table 3-1. Mandatory Guidance, Navigation, and Control Systems

<u>LM Systems</u>	<u>Undocking and Separation¹</u>	<u>CIRC</u>	<u>PDI</u>
PGNCS LGC	R	R	R
IMU	R	R	R
DSKY	R	R	R
AGS (ASA, AEA, DEDA)	R	R	R
CES	R	R	R
DPS/DECA	R	R	R
RR	R	R	NR
Landing radar	R	R	R
FDAI	R ²	R ²	R ²
AOT	R	NR ³	NR
Hand controllers ⁴	R	R	R
Flashing light on LM	NR	R	R
<u>CSM Systems</u>			
GNCS CMC	R	R	R
IMU	R	R	R
DSKY	R ²	R ²	R ²
Optics SXT	R ⁷	R	R
SCT	R	R	R
COAS	R	R	R
SCS BMAG	R ⁵	R ⁵	R ⁵
GDC	R	R	R
FDAI	R ²	R ²	R ²
SPS	R ⁶	R ⁶	R ⁶
Hand controllers	R	R	R
EMS ΔV counter	NR	NR	NR
VHF ranging	R ⁷	NR	NR
RR transponder	R	R	R

1. The separation maneuver and mini-football activities will be performed for all conditions allowing undocking.
2. Only one unit is required.
3. AOT is required until the pre-CIRC fine alignment is completed.
4. Translation and at least one RHC
5. One set of BMAG required.
6. If SPS has failed, TEI will be performed at the next opportunity.
7. Either optics or VHF is required.

Table 3-2. Onboard Tracking Schedule for Mission H-2 (Reference 1)

<u>LM Event</u>	<u>Time from Insertion (min)</u>	<u>CSM Event</u>	<u>Time from Insertion (min)</u>
Insertion	0	Insertion	0
V93 before first mark W = 10,000 ft, 10 ft/sec, 15 mrad		V93 before first mark W = 10,000 ft, 10 ft/sec	
Initiate tracking	18		
		Initiate tracking (SXT + VHF)	19
Cease tracking	40	Cease tracking	38
CSI	50	CSI	50
V93 before first mark W = 2,000 ft, 2 ft/sec, 5 mrad		V93 before first mark W = 2,000 ft, 2 ft/sec	
Initiate tracking	55		
		Initiate tracking (SXT + VHF)	56
Cease tracking	71		
		Cease tracking	78
Plane change	79	Plane change	79
V93 before first mark W = 2,000 ft, 2 ft/sec, 5 mrad		V93 before first mark W = 2,000 ft, 2 ft/sec	
Initiate tracking	81	Initiate tracking (SXT + VHF)	81

Table 3-2. Onboard Tracking Schedule for Mission H-2 (Reference 1) (Continued)

<u>LM Event</u>	<u>Time from Insertion (min)</u>	<u>CSM Event</u>	<u>Time from Insertion (min)</u>
		Cease tracking	96
Cease tracking	98		
CDH	108	CDH	108
V93 before first mark W = 2,000 ft, 2 ft/sec 5 mrad		V93 before first mark W = 2,000 ft, 2 ft/sec	
Initiate tracking	110		
		Initiate tracking	115
		Cease tracking	134
Cease tracking	137		
TPI	147	TPI	147
V93 before first mark W = 2,000 ft, 2 ft/sec, 5 mrad		V93 before first mark W = 2,000 ft, 2 ft/sec*	
Initiate tracking	149		
		Initiate tracking (SXT + VHF)	152
		Cease tracking	159
Cease tracking	160		
MCC-1	162	MCC-1	162

* If SXT or VHF only, no V93 between TPI and MCC-1

Table 3-2. Onboard Tracking Schedule for Mission H-2 (Reference 1) (Continued)

<u>LM Event</u>	<u>Time from Insertion (min)</u>	<u>CSM Event</u>	<u>Time from Insertion (min)</u>
V93 before first mark W = 2,000 ft, 2 ft/sec, 5 mrad		V93 before first mark W = 2,000 ft, 2 ft/sec*	
Initiate tracking	164	Initiate tracking	165
		Cease tracking	174
Cease tracking	175		
MCC-2	177	MCC-2	177

* If SXT or VHF only, V93 after 3 marks following MCC-1

Both the LM primary guidance, navigation, and control system (PGNCS) and abort guidance system (AGS) will independently be aligned on the lunar surface prior to lift-off. After insertion, a PGNCS P52 alignment will be performed without updating the AGS unless there is a real-time indication of degraded AGS performance during ascent. If the update is required, then the AGS state vectors will be initialized with the PGNCS state vectors, and the AGS will be aligned to the PGNCS. Nominally, the AGS will be used as an independent check on the PGNCS through the CSI maneuver. After CSI, the AGS will be periodically updated with the PGNCS and will be replaced by the onboard chart as a backup solution check for the CDH and TPI maneuvers (Section 4.3).

In the rendezvous tracking period preceding CSI, CDH, and TPI, the commander (CDR) will read RR range and range rate from the tape meter and transmit them to the CMP for comparison with range and range rate readout of the command module computer (CMC). The readouts are manually synchronized by voice communication to allow comparisons of the data for an evaluation of the CMC navigation accuracy.

3.3 CSM RENDEZVOUS NAVIGATION AND EVALUATION

Rendezvous navigation is conducted in the CSM using the Rendezvous Navigation Program (P20) and sextant (SXT) tracking and very high frequency (VHF) ranging data to update the LM state vector. Sextant and VHF ranging data are incorporated according to the tracking schedule (Table 3-2) and possibly during convenient intervals between CIRC and PDI. The rendezvous navigation conducted between CIRC and PDI is for the purpose of on-the-job training and will be performed at the CMP's discretion.

If valid VHF range data are available at ranges less than 327 nautical miles, the P20 Program should accept and allow the range data to correct the state vector. Since a range ambiguity exists at ranges greater than 327 nautical miles, the VHF data should be inhibited using V88 beyond this range. The W-matrix is manually reinitialized periodically (V93) as shown in the tracking schedule to reduce navigation errors. The W-matrix values are 2,000 feet in each position component and 2 feet per second in each velocity component except the reinitialization following orbit insertion. The values manually entered for the first reinitialization following orbit insertion are 10,000 feet and 10 feet per second.

In the rendezvous tracking periods preceding CSI, CDH, and TPI, the CMC's navigation capability is verified by evaluating the CMC input data. The CMP determines if the sextant data have been good (crew judgement) and compares navigated range and range rate from the CMC with RR data from the LM. If the CMC range and range rate agree with RR range and range rate and the sextant data have been good, the CMC solutions for CSI, CDH, and TPI will be used to verify the LGC solutions, and the CMC out-of-plane velocity will be used in the LM for plane corrections.

3.4 LM AND CSM STATE VECTOR CORRECTION LIMITS DURING RENDEZVOUS NAVIGATION

The LM and CSM differential state vector correction limits (RMAX and VMAX) in the LGC and CMC will be set to 2,000 feet and 2 feet per second. However, the RMAX and VMAX are expected to be exceeded for the first 5-10 marks after orbit insertion and the first 1 or 2 marks after CSI. Between orbit insertion and CSI, the crew will accept the state vector corrections if RMAX is less than 12,000 feet and VMAX is less than 12 feet per second but will verify a downward trend in the magnitude of the corrections as the relative state is continually improved. If the state vector corrections do not decrease, the crew will investigate the tracking data source to verify that the system is operating correctly before continuing to accept the corrections. If the corrections exceed 12,000 feet or 12 feet per second for the first mark after insertion, the crew will reject the mark but will accept the second mark if it agrees with the first or agrees with the correction obtained in the other vehicle. Again, a downward trend in the corrections should be verified as the relative state is improved.

After CSI, the differential correction acceptability limit will be reduced from 12,000 feet and 12 feet per second to 5,000 feet and 5 feet per second since the expected relative errors between the onboard state vector are larger before CSI than following CSI. It is possible that the corrections will exceed 5,000 feet or 5 feet per second for the first sextant mark after CSI. In this event, the CMP will again reject the first mark but will accept the second mark if it agrees with the first or agrees with the correction obtained in the other vehicle. Similarly, the crew should observe a downward trend in the magnitude of the corrections as

the relative state is improved. For RR marks after CSI, the corrections should not exceed the 5,000 feet or 5 feet per second limit and are not expected to exceed the 2,000 feet or 2 feet per second limit very often.

3.5 CSM ACTIVITIES DURING LM MANEUVERS

As a nominal procedure, the CSM will be mirror-image targeted to back up the CSI, CDH, and TPI maneuvers. The ΔV components will be approximately the same as those for the LM with the sign changed, and the CSM time of ignition (TIG) will be the same as the LM TIG. A 1-foot per second bias is added to the LM CSI solution for use in the CSM if the LM fails to perform the CSI maneuver. However, the LM and CSM solutions for CDH and TPI can be compared and used directly (with a sign reversal) without biasing.

Since the CSM will be targeted with the same TIG as the LM for CSI, CDH, and TPI, the maneuver will necessarily be executed slightly late by the CSM for a LM failure at (or very near) nominal TIG. However, this situation is considered very unlikely because these maneuvers are RCS burns. Obviously, if it is known beforehand that the LM will be unable to execute the maneuver, the CSM will perform the burn at the nominal TIG.

If it becomes necessary for the CSM to perform CSI or TPI, the SPS engine will be used to conserve RCS propellant. Since the CDH maneuver results in an SPS burn time of less than 0.5 second, RCS will be used to execute CDH (if required). The maneuver targeting to be used by the CSM in backing up a LM maneuver is further described in the nominal sequence of events (Section 4).

After each LM maneuver except ascent, the CMP will update the LM state vector in the CMC using the Target ΔV Program (P76), entering the LM maneuver V_g and TIG which is biased by one-half the nominal burn time. Following ascent, the LM state vector in the CMC will be updated by Mission Control Center - Houston (MCC-H) using the LM telemetry state vector.

4. DETAILED TECHNIQUES DISCUSSION

This section presents the detailed guidance and navigation (G&N) activities as related to the data requirements, real-time decisions, and techniques during the nominal lunar orbit activities.

4.1 DOCKED ACTIVITIES

This phase includes the lunar orbit activities from DOI to initiation of undocking. These activities are:

- Descent orbit insertion
- Prerest period LM partial activation and pseudo landing site observations
- State vector and REFSMMAT updates
- Docked IMU alignment and drift check
- CMC, LGC, and AEA clock synchronization
- AGS initialization and calibration

4.1.1 Descent Orbit Insertion

The DOI maneuver (212.9 feet per second/23.1-second SPS burn) is performed in the docked configuration two revolutions after the LOI maneuver. This burn locates the CSM/LM in an orbit whose parameters and orientation will, under lunar gravitational influence, converge to the proper conditions for PDI (nominally a 58.9- by 8.3-nautical mile orbit with the LM at pericyynthion for PDI).

During the DOI burn, the absence of physiological cues such as high rotational rates will be the indication that the GNCS has not incurred a catastrophic failure. Proper functioning of the GNCS will otherwise be established by onboard observations that the CSM is operating within the following limits:

- FDAI attitude excursions $< \pm 10$ degrees
- Altitude rates $< \pm 10$ degrees per second

If the GNCS is malfunctioning, the crew will switch to the SCS RATE CMD mode, damp rates, and immediately terminate the burn.

This burn will place the docked vehicles on a lunar impact trajectory with an overburn of 1 second (10 feet per second) or greater. Thus, a manual SPS shutdown will be executed by the crew at the nominal burn time + 1 second. This allows the GNCS a maximum opportunity to shut down the SPS automatically and, at the same time, prevents an overburn which cannot be safely aborted with a bailout burn scheduled at AOS + 12 minutes. The X- and Z-axis residuals will each be trimmed to within 1 foot per second. The CSM will be required to pitch over 180 degrees to trim out any +X residual greater than 2 feet per second due to a plume impingement constraint on the burn time of the +X-axis thrusters.

After the DOI maneuver, the crew will maneuver the CSM to the high-gain acquisition/bailout burn attitude. Due to the nature of the lunar terrain near the post-DOI pericyynthion and the mission requirement to maintain a clearance of the mountain peaks of 1.2 nautical miles, only a 5-foot per second overburn can be allowed. MCC-H will use the low-speed orbit determination program, using MSFN data from three sites to track the CSM/LM for 3 minutes after AOS to determine the height of perigee of the orbit. If the incoming doppler residuals (actual data minus predicted data) from the MSFN sites indicate less than 1.2-nautical mile closest approach (overburn greater than 5 feet per second), MCC-H will recommend NO STAY at AOS + 7 minutes.* If either the GNCS or entry monitoring system (EMS) also indicate an overburn, the crew will execute the canned bailout maneuver at DOI + 35 minutes (AOS + 12 minutes). Lack of MCC-H confirmation resulting from either MSFN data or a communication problem is also a NO STAY vote when the GNCS and EMS do not agree.

If the GNCS and EMS agree, where agreement is defined as an EMS ΔV counter reading of 11 feet per second or less in the overburn direction (6 feet per second are attributed to SPS tailoff and 5 feet per second to overburn), no MSFN vote is required and the crew will not execute the canned bailout maneuver. The canned bailout maneuver (if required) is a 100-foot per second burn pitched up 70 degrees above the local horizontal.

*This technique may not be applicable for easterly launch sites such as planned for mission H-3 (Apollo 14).

4.1.2 Prerest Period LM Partial Activation and Pseudo Landing-Site Observations

Following the DOI maneuver and prior to the first lunar orbit sleep period, the CMP performs a CSM inertial measurement unit (IMU) realign (P52) to the landing-site REFSMMAT uplinked earlier. In addition, the CMP performs pseudo landing-site observations to provide real-time training and familiarization with the terrain. The pseudo observations will be performed on landmark Pickering B using the Orbital Navigation Program (P22) in a manner similar to that discussed in Section 4.2.7.

During this period, the LMP and CDR enter the LM and perform general housekeeping chores and the initial LM activation. The PGNCS will not be powered up at this time. This 2-hour period of activity primarily provides the crew an opportunity to get equipment stowed and other housekeeping chores completed. It also provides the flexibility to add any additional activities which may be necessary at a later date.

4.1.3 State Vector and REFSMMAT Updates

After the first lunar orbit rest period, MCC-H will uplink to the CMC the CSM/LM state vector, which is time-tagged near the landmark sighting time, and the landing site REFSMMAT, which is computed for the nominal time of landing. Prior to undocking, MCC-H will uplink to the CMC and LGC the LM/CSM state vector, which is time-tagged near undocking, and to the LGC the landing-site REFSMMAT, which is computed for the nominal time of landing.

The LGC and CMC landing-site REFSMMAT's are identical and are computed from the a priori knowledge of the landing-site location. They will not be changed later to agree with an updated landing-site location based on optical observations. The use of a REFSMMAT that is not based on the updated landing-site location substantially simplifies procedures, but it will cause the FDAI angle readings to be slightly non-nominal during descent. However, this difference is expected to be less than the resolution of the flight director attitude indicator (FDAI).

4. 1. 4 Docked IMU Alignment and Drift Checks

After the CMC landing-site REFSMMAT update from MCC-H, the CSM IMU is realigned to REFSMMAT using the IMU Realign Program (P52). When P52 is completed, a third star check using the auto optics routine will be performed. Adequate alignment of the IMU is assured if the third star is centered in the sextant field of view. If the alignment is not satisfactory, the procedure will be repeated. Once a satisfactory IMU alignment has been obtained, the GDC will be aligned to the IMU.

After the rest period when the spacecraft enters the lunar umbra, the CSM IMU is again realigned to REFSMMAT using the IMU Realign Program (P52), and a third star check using the auto optics routine is performed. Once a satisfactory IMU alignment has been obtained, the gyro torquing angles will be used to assess the gyro drift from the previous alignment. If at any time the gyro torquing angles indicate a change in gyro drift greater than 1.5 degrees per hour (Reference 2), the GNCS IMU has failed.

Prior to undocking, a LM PGNCS alignment is performed. The LM IMU is coarse aligned to the CSM IMU using the docked alignment procedure (Reference 3) which is based on the docking ring angle (DKAN), known CSM/LM geometry, and the CSM inertial attitude. The CMP will read the IMU gimbal angles from the DSKY and pass them to the LM. The LMP will compute the LM gimbal angles using the following equations:

$$OGA_{LM} = DKAN - 60 \text{ deg} - OGA_{CM}$$

$$IGA_{LM} = IGA_{CM} + 180 \text{ deg}$$

$$MGA_{LM} = -MGA_{CM}$$

The LMP will coarse align the LM IMU using V41N20 with the computed angles and allow the LM IMU to go inertial. The CMP and LMP will then key V06N20 and stand by to key in "enter." The CMP will give a mark, and both the CMP and the LMP will enter "verb" to freeze the displayed gimbal angles. The IMU gimbal angles are voiced to MCC-H for calculation of gyro torquing angles. These angles, voiced to the LM crew,

are loaded into the LGC using fine align extended verb (V42) to torque the LM gyros to the desired angles.

While the spacecraft are docked, a LM IMU drift check will be performed by MCC-H noting changes in the difference between two sets of CSM and LM gimbal angles (obtained by the crew with V06N20). The first set of V06N20 gimbal angles are those used to fine align the LM IMU, and the second set are obtained just prior to undocking about 1-1/4 hours later. MCC-H will take into account the fine alignment (gyro torquing angles) that occurred between the two V06N20's. MCC-H will update the LM gyro compensation if any of the following angle limits are exceeded: $x > 0.23$ degree (0.145 degree/hour), $y > 0.52$ degree (0.376 degree/hour), and $z > 1.9$ degrees (1.5 degrees/hour). For this technique to give an accurate drift check, the CSM drift compensation must be maintained as accurately as possible and there can be no docking ring slippage between the two V06N20's. The indicated limits are based on the following criteria:

- x-axis misalignment at PDI which can result in a cross-range dispersion which might preclude landing within a 3,000-foot radius of the targeted landing site (about ± 0.2 degree)
- y-axis misalignment at PDI which can result in an unsafe descent abort condition (about ± 0.5 degree)
- z-axis gyro failure

If any gyro drift exceeds 1.5 degrees per hour, either the PGNCSS or GNCS has failed. Because the GNCS has been powered-up since launch, with approximately ten realignments and drift checks, it is much more probable that the PGNCSS has failed if the change in the difference between LM and CSM angles are excessive.

4.1.5 CSM and LGC Clock Synchronization

MCC-H and the crew will update the LGC clock and CMC clock as accurately as possible using ground elapsed time (g. e. t.) from lift-off as the time reference. The CMC clock will be updated by the ground, after which the LM crew will synchronize the LGC clock to the CMC clock. The ground will then compute the difference between ground time and LGC time

(ΔT) and voice it to the LM crew. Fine synchronization of the LGC clock will be accomplished by the ground or manually by the crew using the ΔT .

4.1.6 AGS Initialization and Calibration

Following the PGNCS alignment, the AGS state vectors are initialized from the PGNCS, AGS is aligned to the PGNCS, and the AGS accelerometers and gyros are calibrated. For the AGS alignment and calibration, the CSM is used to orient the LM to an AGS calibration attitude by displacing all three LM axes at least 11.5 degrees from zero or multiples of 45 degrees from the IMU principal axes. This technique will permit the AGS alignment and calibration to be performed without coupling data unit (CDU) transients.

The AGS calibration is performed after the navigation initialization. Gyro and accelerometer checks are made by the crew to determine if any of the accelerometer biases or gyro drift have exceeded the allowable limits (Reference 4). If the change in accelerometer bias since the earth prelaunch calibration exceeds 0.04 foot per second per second, the AGS is considered failed. If the change in gyro drift since the earth prelaunch calibration exceeds 2.0 degrees per hour, the AGS or PGNCS may have failed since the PGNCS is used as a reference for the AGS gyro drift test. If the change in gyro drift is less than 2.0 degrees per hour, the AGS is considered acceptable. If the change in gyro drift is less than 1.5 degrees per hour, both PGNCS and AGS are considered verified.

4.2 UNDOCKED ACTIVITIES

This phase includes the lunar orbit activities from undocking to PDI. These activities are:

- Undocking and separation of the CSM from the LM
- PIPA bias calibration
- AGS initialization and calibration
- IMU alignments and drift check
- RR and VHF verification
- CIRC maneuver

- Landing-site observations
- Descent targeting

4.2.1 Undocking and Separation of the CSM from the LM

A soft undocking with the LM above the CSM will take place approximately two and a quarter revolutions prior to PDI with the spacecraft oriented in the inertial undocking attitude (X-axis along the local vertical with the CSM below the LM). Simultaneous with the undocking is the CSM separation maneuver. The soft undocking involves catching the LM on the ends of the docking latches and using the CSM RCS to overcome an approximate 40-pound force holding the two vehicles together. Thrust monitor program (P47) is used to set up a separation rate by thrusting -X RCS 1 foot per second by the CSM. This will place the CSM in front and above the LM three-quarters of a revolution later at the time of the CIRC burn.

After the CSM undocks from the LM, the LM will yaw left 60 degrees and pitch up 90 degrees. This will place the CSM in the forward window of the LM. If there is some indication (such as a "barber pole" indication) that the landing gear is not extended, the LM will perform a rotation maneuver (yaw) of 1 degree per second for 360 degrees in order to do a complete inspection.

4.2.2 PIPA Bias Calibration

The CSM and LM PIPA biases are calibrated before or after undocking, depending on where the calibration can be accomplished conveniently in the timeline. The calibration can be performed completely by MCC-H from the PIPA outputs on the downlink during coasting navigation. If the uncompensated PIPA bias in the CMC is greater than 0.003 foot per second per second, the bias compensation will be updated. The PIPA bias compensation in the LGC will be updated by MCC-H if the first check after manning the LM exceeds 0.001 foot per second per second. Thereafter, the uncompensated PIPA bias will be updated if it exceeds 0.003 foot per second per second. The mission is discontinued if the change in PIPA bias (LM or CSM) is greater than 0.164 foot per second per second, the impending failure limit (compensation limit is 0.1025 foot per second per second) (Reference 2).

4. 2. 3 AGS Initialization and Calibration

Following the PGNCS alignment, the AGS state vectors are initialized from the PGNCS, AGS is aligned to the PGNCS, and the AGS accelerometers and gyros are again calibrated. To avoid the CDU transient problem for the AGS alignment and calibration, the LM attitude will be 22.5 degrees (OGA), 247.5 degrees (IGA), and 22.5 degrees (MGA). The purpose of the undocked calibration is to determine its feasibility and so will not be used to update the previous docked calibration.

4. 2. 4 IMU Alignments and Drift Checks

After undocking, the LM's IMU is aligned to REFSMMAT (P52), (gyro compensation has been updated, if necessary, using V06N20 gimbal angles). The third star check is used to calibrate the LPD. The LPD's position is keyed into the DSKY as a COAS sighting, 0 degree for the shaft angle and -40 degrees as the trunnion. If the preflight calibrations values, as determined by MCC-H from the crew observations, do not agree within $\pm 1/2$ degree in elevation or ± 2 degrees in azimuth, the MCC-H will update the LPD calibration values in the LGC.

Prior to PDI, the LM IMU is realigned to REFSMMAT (P52), and a third star check using the auto optics routine is performed. The gyro torquing angles are used to assess the gyro drift from the previous alignment. If the gyro torquing angles* exceed 0.37 degree in x, 0.83 degree in y, and 3.0 degrees in z, then PDI will be delayed one revolution. A real-time analysis will determine whether or not to attempt lunar landing and, if so, whether or not to update the gyro compensation.

The CSM IMU is aligned to REFSMMAT (P52) once prior to CIRC and again prior to PDI. The gyro torquing angles are used to assess the gyro drift. If the gyro drift prior to either the CIRC or PDI maneuver is greater than 1.5 degrees per hour in any axis, the maneuver will be delayed one revolution depending on a real-time analysis of the problem.

*The limits are based on the same criteria indicated in Section 4.1.4.

4.2.5 RR and VHF Verification

The RR is evaluated to ensure that it is operating properly before the LM is committed to PDI since this system is required for accurate rendezvous navigation. The VHF is evaluated at this time although the mission would be continued whether or not this system was operating properly (a CSM rescue of the LM could be accomplished using only sextant data for rendezvous navigation). The CDR and CMP call the rendezvous parameter display routine (R31) for the display of range and range rate which are then compared with the RR and VHF measurements. The range at the time of the check is approximately 5000 feet, and range rate is less than 5.6 feet per second. The allowable limits that have been determined are 1600 feet for the range measurements and 7 feet per second for range rate (Reference 5).

4.2.6 CSM CIRC Maneuver

After separation, when the low orbit landing-site observations have been completed, MCC-H will voice PAD data for the CIRC maneuver (Table 4-1) to the CMP, and the targeting data for the maneuver will be uplinked to the CMC (P27). The radial inward separation burn of 1-foot per second will result in the CSM being in front of and above the LM three-quarters of a revolution later at the time of the CSM CIRC burn. The CIRC maneuver is a 70.4-foot per second/3.9-second SPS burn which places the CSM in a 63.6-by 54.3-nautical mile orbit. Due to the short burn duration, no residual trimming will be required.

4.2.7 Landing-Site Observations

Subsequent to undocking during revolution 12, the CMP will perform landing-site observations while in the DOI orbit. The optical sightings will be performed in the pitch mode using the Orbital Navigation Program (P22 (P22)), and the data will be processed by the Real-Time Computer Complex (RTCC). A predetermined inertial attitude will be held until the spacecraft is at a 9.4-degree elevation angle as viewed from the landmark. At this time, a pitch rate of 2 degrees per second is initiated. The first mark will be taken when the spacecraft is at 26-degree elevation angle as viewed from the landmark. The local vertical attitude at the time of starting the pitch rate is 80, 0, 0 degrees roll, pitch, and yaw, respectively. MCC-H will supply the same information as listed below for the CIRC orbit. These

Table 4-1. CSM External ΔV (P30) PAD

P30 MANEUVER										
SET STARS							PURPOSE			
							PROP/GUID			
+							WT N47			
R ALIGN	0 0						P _{TRIM} N48			
P ALIGN	0 0						Y _{TRIM}			
Y ALIGN	+ 0 0						HRS GETI			
+							0 0 0			
+							0			
+							0			
							•			
ULLAGE								ΔV _X N81		
								•		
								ΔV _Y		
								•		
								ΔV _Z		
	X X X							R		
	X X X							P		
	X X X							Y		
+							•			
							•			
+							•			
							ΔVT			
HORIZON/WINDOW	X X X		•				BT			
	X						•			
	X X X X						SXTS			
+							0			
+							•			
							0 0			
							•			
X X X							BSS			
X X							SPA			
X X X							SXP			
•							•			
OTHER	0						LAT N61			
								•		
								LONG		
+							•			
+							RTGO EMS			
+							V10			
							•			
							•			
							GET 0.05G			

data will establish the feasibility of landmark tracking in a DOI orbit. If successful, an R_{LS} update may be made to the LGC after AOS revolution 13.

Subsequent to the CIRC maneuver at the beginning of revolution 13, the CMP will perform landing-site observations. Again the optical sightings will be performed in the pitch mode using the Orbital Navigation Program (P22), and the data will be processed by the RTCC. A predetermined inertial attitude will be held until the spacecraft is at a 35-degree elevation angle as viewed from the landmark. At this time, a pitch rate of at least 0.3 degree per second is initiated. The local vertical attitude at the time of starting the pitch rate is 0, -2.1, 0 degrees roll, pitch, and yaw, respectively. MCC-H will supply the following data to the crew in real time:

- Time at which the identification point is at an elevation angle of 35 degrees
- Time to start the pitch rate
- Inertial attitude to be held until the pitch rate is started
- Shaft and trunnion angles at the initiation of the pitch rate
- Whether the landmark will be north or south of groundtrack

The CMP will decide whether or not to slightly roll the spacecraft to avoid excessively high sextant shaft rates if the landmark is too close to the groundtrack. The constraint is that the sextant trunnion angle should never be less than 10 degrees. The spacecraft should be rolled if the landmark is too far from groundtrack.

The landing-site observation data will be relayed to the ground using low bit rate telemetry communication and the omni antennas. The DSKY display of the state vector velocity correction (ΔV) will be maintained by the crew until MCC-H verifies receipt of the sighting data.

These sightings should eliminate some of the relative uncertainty between the landing site and the CSM orbit, and therefore, improve the

accuracy of the descent targeting.* The sightings also provide a complete, independent check on the overall targeting scheme. The method whereby the landing-site observation data are used to improve the relative accuracy of the landing site with respect to the orbit is outlined in Section 4.2.8.

4.2.8 Descent Targeting

The purpose of descent targeting is to accurately determine the LM position relative to the desired landing site. To accomplish this task, both inertial and relative estimates of vehicle state vector and landing site position are used. The procedure consists of updating the LM inertial state vector with the best available information (tracking data) and adjusting the landing site position (\underline{R}_{LS}) to correct the relative position vector of the landing site with respect to the LM state vector.

4.2.8.1 State Vector and \underline{R}_{LS} Update

The CMC and LGC will receive updates (as applicable) of the CSM and LM state vectors during revolution 11 prior to undocking, during revolution 12 after undocking, after AOS during revolution 13, and after AOS during revolution 14 (PDI revolution). The time tags for the vectors are as follows:

Revolution 11 (Prior to undocking)

- CSM state vector (CMC) Undocking - 10 minutes
- LM state vector (LGC) Undocking - 10 minutes

Revolution 12 (After undocking)

- CSM state vector (LGC) Current time
- CSM state vector (CMC) CIRC - 10 minutes

Revolution 13

- CSM state vector (CMC and LGC) PDI - 10 minutes
- LM state vector (CMC and LGC) PDI - 10 minutes

*The planned landing site for the H-2 mission has been well surveyed from H-1 mission bootstrap photography. Thus, the H-2 mission landmark tracking data are not considered mandatory for landing.

Revolution 14 (Prior to PDI)

- LM state vector (LGC)

PDI - 10 minutes

The landing site position (\underline{R}_{LS}) will be updated in the LGC after AOS during revolutions 13 (\underline{R}_{LS1}) and 14 (\underline{R}_{LS2}) (Section 4.2.8.4). Crossrange and radial components of \underline{R}_{LS} will be biased with sextant data to adjust for expected propagation errors of the lunar potential model and to obtain a better relative position of the landing site with respect to the orbit (Section 4.2.8.3). Expected downrange propagation errors are corrected adjusting the LM and CSM state vector time tags if they exceed 10,000 feet. An adjustment to the downrange component of \underline{R}_{LS} (ΔR_{LS}) will be voiced to the crew as soon as possible after PDI for crew input (N69) at about PDI + 2 minutes (Section 4.2.8.5).

The PAD data that the ground provides for PDI₁ (Tables 4-2 and 4-3), for the No PDI₀ abort and for No PDI₁ + 12 minute abort are transmitted to the crew during revolution 13 subsequent to the CIRC maneuver. The No PDI₀ abort PAD contains the targeting for an abort initiated one-half period after CIRC whereas the No PDI₁ + 12 minute abort PAD contains the targeting for an abort initiated at 12 minutes after PDI time.

4.2.8.2 MSFN Consistency Check

An unconstrained plane single-pass state vector solution* is obtained using MSFN tracking from revolution 3 and the solution propagated forward to the revolution 4 landing site longitude. This solution is differenced with the revolution 4 unconstrained single-pass solution (at the landing-site longitude), and the difference is recorded. This procedure is repeated for MSFN tracking data on revolution 4 propagated to revolution 5, etc., up to revolution 10 propagated to revolution 11. This procedure provides several sets of differences with three components per set.

In order to obtain two-revolution propagation error offsets, an unconstrained plane single-pass state vector solution is obtained using

*The unconstrained plane single-pass solution should give the best determination of spacecraft position for that pass; expected 1σ errors are 500 feet radius and downtrack and 2,000 feet crosstrack.

Table 4-2. LM PDI1 PAD

PDI 1 PAD					
+ 0 0	+ 0 0 1 0 7	HRS	N33		
+ 0 0 0	+ 0 0 0 5 1	MIN	PDI		
+ 0 .	+ 0 3 0 0 0	SEC			
X X .		TGO	N61		
		CROSSRANGE			
X X X	X X X	R	FDAI		
X X X	X X X	P	AT TIG		
X X X	X X X	Y			
		DEDA 231 IF RQD			
(1 < PDI 1 ≤ 5) ABORT PAD					
LOG INSERTION GET= _____ : _____ : _____					
+ _____ 1 0 0 0 0					
BOOST GET= _____ : _____ : _____					
+ _____ 1 0 0 0 0					
HAM, GET= _____ : _____ : _____					
+ _____ 1 0 0 0 0					
CSI, GET= _____ : _____ : _____					
+ 0 0	+ 0 0 1 1 2	HRS	N37		
+ 0 0 0	+ 0 0 0 3 9	MIN	TPI		
+ 0 .	+ 0 1 4 0 0	SEC	(5 MIN)		
T1-1 (5 < PDI 1 ≤ 15) MIN ABORT PAD					
LOG INSERTION GET= _____ : _____ : _____					
+ _____ 5 5 0 0					
CSI TIG= _____ : _____ : _____					
+ 0 0	+ 0 0 1 1 0	HRS	N37		
+ 0 0 0	+ 0 0 0 4 0	MIN	TPI		
+ 0 .	+ 0 3 1 0 0	SEC	(15 MIN)		

Table 4-3. CSM Target ΔV (P76) PAD

P76 UPDATE PAD														
														PURPOSE
+	0	0						+	0	0				HR N33
+	0	0	0					+	0	0	0			MIN TIG
+	0			•				+	0			•		SEC
					•								•	ΔV_X N84
					•								•	ΔV_Y
					•								•	ΔV_Z
														PURPOSE
+	0	0						+	0	0				HR N33
+	0	0	0					+	0	0	0			MIN TIG
+	0			•				+	0			•		SEC
					•								•	ΔV_X N84
					•								•	ΔV_Y TIG
					•								•	ΔV_Z
														PURPOSE
+	0	0						+	0	0				HR N33
+	0	0	0					+	0	0	0			MIN TIG
+	0			•				+	0			•		SEC
					•								•	ΔV_X N84
					•								•	ΔV_Y
					•								•	ΔV_Z
REMARKS: GETI (CSI) GETI (TPI) N														

MSFN tracking from revolution 3 and the solution propagated forward to the revolution 5 landing-site longitude. This solution is differenced with the revolution 5 unconstrained single-pass solution (at the landing-site longitude), and the difference is recorded. This procedure is repeated for MSFN tracking data on revolution 4 propagated to revolution 6, etc., up to revolution 9 propagated to revolution 11. This procedure provides several sets of differences of single-pass fit for two-revolution propagation error offsets with three components per set.

The radial (ΔU), downrange (ΔV), and crossrange (ΔW) differences are plotted for each of the sets of data and compared for agreement. If all agree within reasonable limits, * MSFN consistency is verified, and the biases can then be computed to compensate for expected propagation errors.

4. 2. 8. 3 Propagation Bias Computation

The radial (ΔU), downrange (ΔV), and crossrange (ΔW) differences are computed by averaging the differences available for each component. If the ΔV bias exceeds 10,000 feet, an equivalent time tag shift is determined and used to adjust the time tags of the CSM and LM state vectors in the RTCC and uplinked to the CMC and LGC. By shifting the state vector time tag instead of R_{LS} for large downrange position biases, insertion flight-path angle errors are avoided for aborts from descent (the value of downrange bias is expected to be much less than 10,000 feet). **

4. 2. 8. 4 Adjustments to R_{LS} Using Sextant Data

The landing site position, determined by using revolution 12 (low orbit CSM tracking) single-pass fit constrained to the pass 12 LM vector

*The maximum dispersion observed on mission F was about 3,500 feet downrange, 1,500 feet crossrange, and 300 feet radius; while on mission G, the maximum dispersion was about 1,000 feet downrange, 9,000 feet crossrange, and 300 feet radius; and on mission H-1, the maximum dispersion was 1,119 feet downrange, 1,330 feet crossrange, and 61 feet radius.

**The mean value of the biases observed on mission F was about 337 feet downrange, 6,000 feet crossrange, and 1,029 feet radius; while on mission G, the mean value was 773 feet downrange, 3,000 feet crossrange and 1,964 feet radius; and on mission H-1, the mean value was 805 feet downrange, 163 feet crossrange, and 182 feet radius.

processed with the sextant data, is used to improve the relative accuracy of the orbit with respect to the landing site. This value of \underline{R}_{LS} is used only if it agrees with the premission value of \underline{R}_{LS} within the following limits, which are based on the expected 3σ error in the relative location of the CSM with respect to the landing site.

- $\Delta\lambda$ (longitude) \leq 2000 feet (0.02 degree)
- $\Delta\phi$ (latitude) \leq 6000 feet (0.06 degree)
- Δr (radius) \leq 2000 feet

If the \underline{R}_{LS} differences are not within the above limits, the mission will be continued using the premission values of \underline{R}_{LS} . The \underline{R}_{LS} update sent to the LGC at revolution 13 AOS will be based on the sextant \underline{R}_{LS} plus two-revolution propagation error biases in the crosstrack and radial components. In summary, the \underline{R}_{LS1} update with all adjustments uplinked at AOS (revolution 13) is as follows:

\underline{R}_{LS1}

- Longitude: $\lambda = \lambda_{SXT}^*$
- Latitude: $\phi = \phi_{SXT}^* - \phi_2^{**}$
- Radius: $R = R_{SXT}^* - \Delta U_2^{**}$

There will be a second attempt to obtain an acceptable \underline{R}_{LS} update from the CSM 60-nautical mile orbit during revolution 13; but mission H-2 will continue if no acceptable \underline{R}_{LS} update is obtained (from either revolution 12 or 13) since the planned landing site has been well surveyed from H-1 mission bootstrap photography. If acceptable \underline{R}_{LS} updates are obtained from both revolution 12 and 13, the update from revolution 13 will be used since there is a higher confidence in this value due to the longer tracking time.

* If the λ_{SXT} , ϕ_{SXT} , and R_{SXT} do not agree with the premission values within the above limits, the premission value will be used.

** Two-revolution propagation error offset.

The landing-site position will be determined by constraining the plane of the revolution 13 CSM single-pass solution to the revolution 13 LM single-pass unconstrained plane vector processed with the sextant data. The \underline{R}_{LS} update sent to the LGC at revolution 14 AOS will be based on the landing site position plus one revolution propagation error biases in the crosstrack and radial components. In summary, the \underline{R}_{LS2} update with all adjustments is as follows:

\underline{R}_{LS2}

- Longitude: $\lambda = \lambda_{SXT}^*$
- Latitude: $\phi = \phi_{SXT}^* - \Delta\phi_1^{**}$
- Radius: $R = R_{SXT}^* - \Delta U_1^{**}$

4. 2. 8. 5 Adjustments to \underline{R}_{LS} After PDI

Orbital mechanics yield a relation between downrange position error and PFP-PGNCS radial velocity residual prior to PDI (in the ratio of about 1000 feet to 1 foot per second) which is used to further update the LM state vector or \underline{R}_{LS} . It is not possible, however, to perform a state vector uplink during powered descent. Therefore, in order to use the PFP information, as well as the latest MSFN range data, to improve the LGC targeting, the crew will receive a PAD of downrange errors, *** stated in terms of a correction to the LGC \underline{R}_{LS} ($\Delta\underline{R}_{LS}$), **** which will be entered in the LGC with V21N69 at TIG + 2 minutes. (Noun 69 can also be used to input a crossrange and altitude correction; however, no techniques for determining these errors are available for the H-2 mission.)

* If the λ_{SXT} , ϕ_{SXT} , and R_{SXT} do not agree with the premission values within the stated limits, either the landing-site position obtained from revolution 12 tracking or the premission value will be used.

** Single-revolution propagation error offset.

*** Precise confidence limits for the real-time validity of $\Delta\underline{R}_{LS}$ have not been determined. At present, if the downrange error is less than 2,000 feet or greater than 35,000 feet, it will not be sent to the crew.

**** The LGC targeting equations use the vector LAND as the location of the landing site. LAND is computed as the sum of the preloaded landing site vector, \underline{R}_{LS} , and the N69-loaded correction to the landing site vector, $\Delta\underline{R}_{LS}$.

4.3 POSTORBIT INSERTION ACTIVITIES

This phase includes the lunar orbit activities from LM orbit insertion to docking.* These activities are

- Pre-CSI activities
- CSI maneuver
- Plane change maneuver
- CDH maneuver
- TPI maneuver
- TPF

4.3.1 Pre-CSI Activities

Following orbit insertion, MCC-H will uplink to the CMC the LM telemetry state vector. Both the LM and CSM perform an IMU realign to REFSMMAT (P52) after entering darkness to remove any alignment errors than may affect subsequent rendezvous navigation. If there is a real-time indication of degraded AGS performance during ascent, then following the PGNCS alignment, the AGS state vectors are initialized from the PGNCS, and the AGS is aligned to the PGNCS.

Rendezvous navigation is performed in both spacecraft (LM update option in both the LGC and CMC), and an evaluation is made of the CMC's navigation capability (Section 3.3). If the CMC is acceptable, the CMP reads out the LM out-of-plane velocity for LM TIG (CSI) using the LM rendezvous out-of-plane display routine (R36). The CSI TIG was voiced to the CMP prior to lunar ascent. The CMC estimate of \dot{y} is used instead of the LGC estimate because rendezvous navigation analysis (Reference 6) shows that prior to TPI the CMC estimate of \dot{y} based on sextant tracking is much more accurate than the LGC estimate based on RR angular errors. The \dot{y} is voiced to the LM if it exceeds 2 feet per second for incorporation into the LM CSI Targeting Program (P32).

* Beginning with mission H-3 (Apollo 14), the CSI/CDH/TPI/TPF rendezvous sequence may be replaced by a direct rendezvous technique in which TPI is initiated about 38 minutes after insertion. This would require 2 hours less between insertion and docking.

4.3.2 CSI Maneuver

The LM CSI targeting is performed in the LM using P32 in the LGC, AGS (DEDA address 410 + 10000) and using RR data with the RR backup charts. The P32 requires inputs of TIG (CSI), TIG (TPI), the elevation angle (E) at TPI, and N (the number of the apsidal crossing at which CDH occurs). The TIG (CSI and TIG (TPI) were included in the PAD data that were voiced to the LM and CSM prior to LM ascent. For the nominal rendezvous, E = 26.6 degrees, TIG (TPI) is about the midpoint of darkness for the CSM orbit (23 minutes prior to CSM sunrise), TIG (CSI) is insertion + 50 minutes, and N = 1.

The CSM also targets for CSI using P32 of the CMC. The CMC P32 requires inputs of TIG (CSI), TIG (TPI), E, and N. TIG (CSI), TIG (TPI), and N are the same for the CSM as for the LM but E = 208.3 degrees. Since the CSM P32 solution is calculated for a CSM maneuver, the solution cannot be used directly. A fixed bias of 1 foot per second must be subtracted from the magnitude of the CMC solution to obtain a value for LM use. Also, the sign of the CMC solution must be reversed.

The LM crew will compare CSI Vgx solutions from the PGNCS, GNCS, and AGS. If the LGC agrees with either the CMC or AGS (if RR is good) to within 3 feet per second, the LGC solution will be burned in the LM. If the LGC does not agree with either the CMC or AGS, the CMC solution will be burned in the LM if it agrees with the AGS within 3 feet per second by inserting the Vgx into the LGC Targeting Program (P32). If all three solutions disagree, the CMC solution will be burned unless real-time data indicate otherwise. In the event of an AGS failure, the onboard chart solution can be substituted for the AGS solution in the above comparison.

If the plane correction required at CSI is greater than 2 feet per second, the CMC R36 \dot{y} is inserted into the LGC CSI Targeting Program and burned in conjunction with the CSI maneuver. The LM uses the RCS Thrust Program (P41) to accomplish CSI with the +Z RCS jets to maintain RR lock and conserve RCS propellant (reorientation for the burn not required).

4.3.3 Plane Change Maneuver

After CSI, rendezvous navigation is performed in both spacecraft. The CMP obtains the LM out-of-plane velocity, \dot{y} , for LM CSI time + 29 minutes from the CMC using the out-of-plane rendezvous display routine (R36). The \dot{y} is voiced to the LM for the plane change maneuver. If \dot{y} is less than 2 feet per second, the maneuver will not be performed. If \dot{y} exceeds 2 feet per second, the plane change is accomplished using the External ΔV Program (P30) and RCS Thrust Program (P41). This maneuver is performed using Y RCS thrusters to avoid gimbal lock and with the +Z axis boresighted on the CSM to prevent loss of RR lock. The CSM will not be targeted to back up the plane change maneuver unless the LM is known to be inactive.

4.3.4 CDH Maneuver

After the plane change maneuver, rendezvous navigation is performed in both spacecraft, and another evaluation is made of the CMC's navigation capability. If the CMC is acceptable, the CMP obtains the LM out-of-plane velocity for LM TIG (CDH) using R36 and voices it to the LM for incorporation into the CDH Targeting Program (P33).

LM CDH targeting is performed in the LM using P33 of the LGC, AGS (DEDA address 410 + 20000) and RR data with the RR backup charts. Inputs to P33 are TIG (TPI) inserted before CSI and TIG (CDH) which is computed by the LGC in the CSI Targeting Program (P32).

The CSM also targets for CDH using P33 of the CMC. Inputs to this program are the same as for the LGC. P33 is used in preference to P73 since it is possible to make solution comparisons with the LM P33 and at the same time simplify the number of operations required by the CMP in preparing to back up the CDH maneuver. The CSM solution for CDH is voiced to the LM for comparison with the LGC solution. The sign of the V_g must be reversed for comparison.

The LM crew will compare the LGC P33 V_g solution with both the CMC P33 solution and the onboard chart solution. If the LGC agrees with either the CMC or the chart ($\Delta V_{gx} < 2$ feet per second and the $\Delta V_{gz} < 6$ feet per second), the LGC solution will be burned in the LM. If the LGC

does not agree with either the CMC or the chart, the chart solution will be burned in the LM if it agrees with the CMC ($\Delta V_{gx} < 2$ feet per second and $\Delta V_{gz} < 6$ feet per second). If all three solutions disagree, the CMC solution will be burned in the LM by inserting the V_g into the LGC Targeting Program (P33). If the LGC solution is verified, the AGS state vectors are initialized from the PGNCS, and the AGS is aligned to the PGNCS.

If a plane correction is required at CDH ($\dot{y} > 2$ feet per second), the CMC R36 \dot{y} is inserted into the LGC CDH Targeting Program and accomplished in conjunction with the CDH maneuver. The CDH maneuver is executed by separately nulling individual components of ΔV while maintaining RR lock. The RCS Thrust Program (P41) is used.

4.3.5 TPI Maneuver

Following the CDH maneuver, rendezvous navigation is performed in both the LM and CSM. In the CSM, another evaluation is made of the CMC navigation capability as described in Section 3.3. LM TPI targeting is performed in the LM using P34 of the LGC, AGS (DEDA address 410 + 30000) and the RR backup charts. In P34, the elevation angle option ($E = 26.6$ degrees) is used to calculate a V_g and TIG (TPI). If the P34 solution for TIG (TPI) is less than nominal TIG (TPI) - 8 minutes, the LM crew will recycle P34 using the time option with a TIG equal to nominal TIG (TPI) - 8 minutes. The TPI will not be executed earlier than nominal TIG (TPI) - 8 minutes in order to maintain adequate dark time between CDH and TPI for CSM rendezvous navigation. Since there is no hard TPF lighting constraint, TPI can occur late by any reasonable amount.

LM TPI targeting is also performed in the CSM using P34 of the CMC. The elevation angle option ($E = 208.3$ degrees) is used to calculate a V_g and TIG (TPI) which is voiced to the LM for comparison with the LGC computed V_g and TIG (TPI). The signs of the V_g components will be reversed for comparison.

The LM crew will compare the LGC P34 V_g solution with the CMC P34 solution (local vertical coordinates) and the onboard chart solution (line-of-sight coordinates). If the LGC agrees with either CMC or the chart ($\Delta V_{gx} < 2$ feet per second, $\Delta V_{gy} < 5$ feet per second, and $\Delta V_{gz} < 6$ feet per second), the LGC solution will be burned in the LM. If the LGC

does not agree with either the CMC or the chart, the chart solution will be burned in the LM if it agrees with the CMC ($\Delta V_{gx} < 2$ feet per second, $\Delta V_{gy} < 6$ feet per second). If all three solutions disagree, the CMC solution will be burned in the LM by inserting the V_g into the LGC Targeting Program (P34). If the LGC solution is verified, the AGS state vectors are initialized from the PGNCS, and the AGS is aligned to the PGNCS.

The TPI maneuver is executed along and normal to the line of sight to the CSM, burning each component individually while maintaining RR lock. The RCS Thrust Program (P41) is used.

4.3.6 Terminal Phase Midcourse Maneuvers

Following TPI, rendezvous navigation and terminal phase midcourse (TPM) targeting is performed in both spacecraft. TPM targeting is performed in the LM using P35 of the LGC. The CSM is targeted for TPM using P35 of the CMC. The CSM does not compute a LM TPM solution for comparison with the LGC solution. It is assumed that, if the LGC has successfully navigated through TPI, it is capable of calculating a TPM solution. The CSM will execute the TPM maneuvers if the LM is unable to do so.

4.3.7 TPF

The TPF braking is performed in the LM according to the planned braking gates. Rendezvous radar range and range rate data are used to determine the required ΔV , and the Thrust Monitor Program (P47) is used to measure the applied V_g . Line-of-sight rates are controlled using RCS X- and Y-translation.

In the CSM, the CMP monitors rendezvous parameter display routine (R31) range, range rate, and VHF range. If the LM fails to acquire the CSM visually (and the CSM acquires the LM) or if the DPS has not been staged, the CMP calls the Thrust Monitor Program (P47) and performs the LOS control. The LM performs the braking if possible.

4.4 POSTRENDEZVOUS ACTIVITIES

This phase includes the activities from docking to TEI. These activities are

- LM jettison and deorbit
- CSM plane change
- Photography and landmark tracking

4.4.1 LM Jettison and Deorbit

Subsequent to the completion of the CSM active docking, the CDR and LMP perform housekeeping duties (decontamination and equipment storage) in preparation for the intervehicle transfer (IVT) to the CM. Before IVT is accomplished, they will configure the LM for jettison and subsequent deorbit and the lunar surface impact near Fra Mauro. The LM will be targeted for an APS impulse burn, but the thrusting will be accomplished utilizing RCS jet thrust (ullage). During revolution 34, the CMP will call up the Thrust Monitor Program (P47) and jettison the LM radially outward (above the CSM) and out of plane toward the south. Total relative ΔV imparted immediately at jettison is approximately 0.45 foot per second with the unpressurized tunnel. Approximately 5.5 minutes after LM jettison, the CSM will perform a pitch maneuver, enter V49, call up P47, and execute a 1-foot per second horizontal retrograde burn to position itself ahead of the LM. The LM deorbit burn of 190 feet per second will be initiated by ground command at 144:32 in order to impact the moon at 144:59. Prior to impact, the LM will be photographed through the CSM sextant, after which the crew will enter an 8-hour rest period (four revolutions).

4.4.2 Plane Change Two

After the termination of the rest period on revolution 39, the crew will perform P52 IMU alignments and configure the CSM for the plane change two maneuver. The 824.6-foot per second maneuver will be executed with SPS thrusting (P40) at 154:13 (revolution 40). The resultant CSM orbit will allow landmark tracking and photography of candidate exploration sites.

4.4.3 Photography and Landmark Tracking

After the plane change maneuver, the crew will use the data acquisition camera (DAC), the electric Hasselblad (EH) camera, and the lunar topographic (TOPO) camera to photograph candidate exploration sites and dim-light astronomical phenomena.

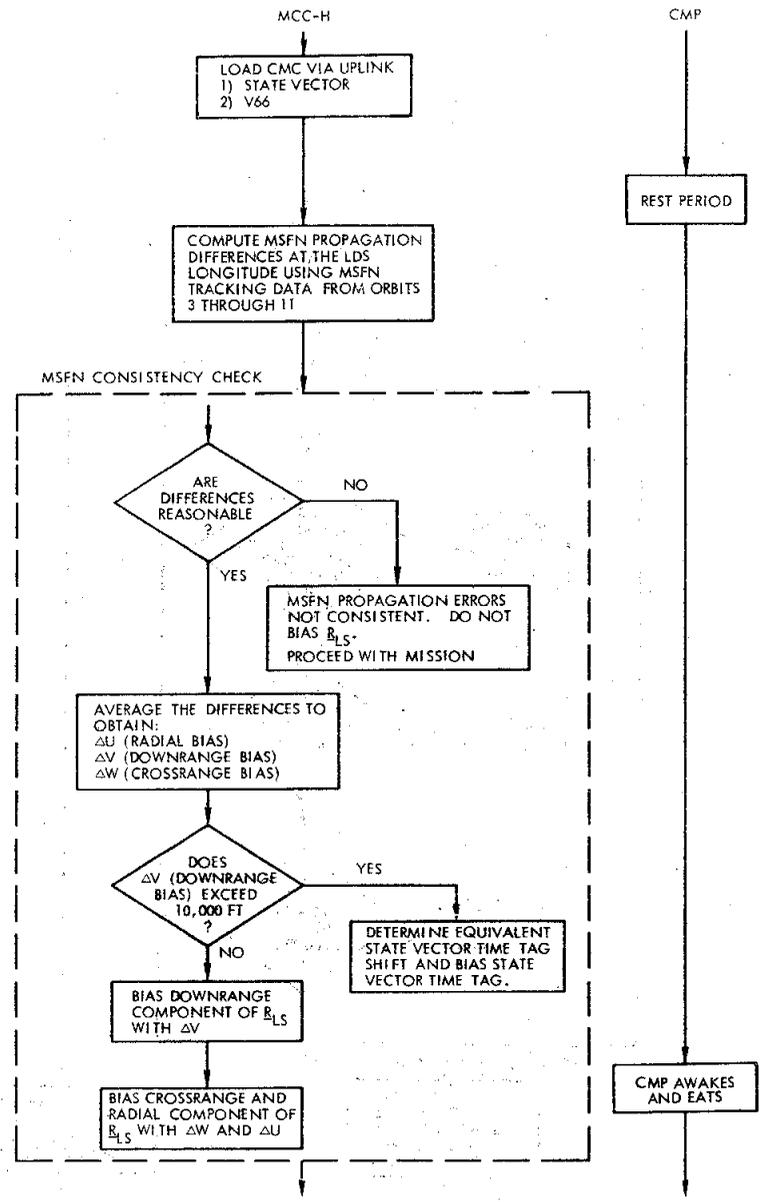
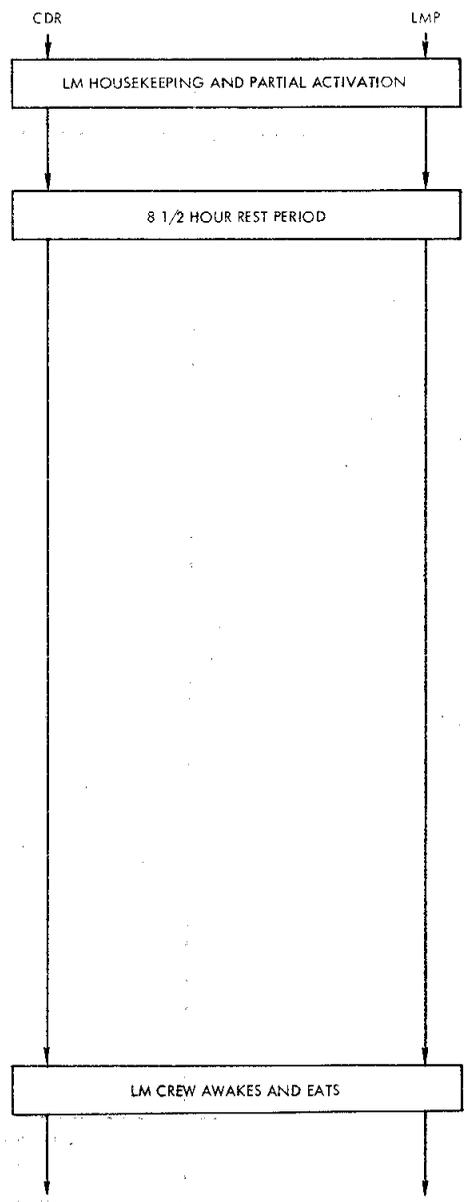
On revolution 39, the crew will set up the SXT/DAC for time-correlation photos and the EH camera in the right-hand rendezvous window for vertical stereoscopic strip photography. Vertical strip photography will commence during revolution 40 as the CSM leaves the lunar umbra and will end approximately 180 degrees later when the CSM enters darkness. To provide stereoscopic data, the photographs will be taken with a 60 percent overlap, and the results will be used to update the map location of the candidate exploration sites. During two successive revolutions (revolutions 41 and 42, the TOPO camera will be mounted in the hatch window and used to obtain high-resolution vertical and 30-degree forward-looking oblique photographs of Censorinus and Davy Rille. These high-resolution photographs will be used to evaluate touchdown hazards and redesignation ΔV requirements and provide information for crew training. On revolutions 43 and 44, the DAC will be used to obtain landmark tracking photographs through the SXT. These data will be used in conjunction with other photographs to provide improved accuracy in defining the locations of the Censorinus and Davy Rille landing sites with respect to known landmarks.

Dim-light photography will be performed with the DAC and Hasselblad mounted in the right-hand rendezvous window during periods of darkness on revolution 40 and subsequent revolutions. On revolution 40, the DAC will be used to obtain zodiacal light photographs at both lunar sunset and sunrise using the moon's horizon as an occultating edge. Zodiacal light results from the illumination of interplanetary particles, and the photographs will be used to provide detailed information as to the spatial distribution of these particles. Photographs of the solar corona (revolution 43) taken with the Hasselblad during lunar sunrise and sunset will be used to determine any large-scale coronal motions. The DAC will be used to photograph the LIMB brightening effect (the bright rim that appears

above the lunar horizon just after lunar sunset) on revolution 44 in an effort to determine the cause of this phenomena. Other dim-light photography will consist of photographs of the gegenschein (unknown light source) taken in total lunar darkness with the DAC. Hasselblad photographs of a selected star field during lunar darkness and lunar sunlight will be used to study the extent of the contamination cloud surrounding the CSM.

5. LUNAR ORBIT ACTIVITIES FLOW CHARTS

The following flow charts present the approved procedures to be followed for the guidance, navigation, and control sequence of events for the mission H-2 lunar orbit activities.



AOS REV 10

CDR

LMP

MCC-H

CMP

LM CREW DONS SUITS

LOAD CMC VIA UPLINK
1) STATE VECTORS
2) LANDING SITE
REFSMMAT

PERFORM IMU REALIGN
TO PREFERRED (P52).
VERIFY ALIGNMENT
WITH AUTO OPTICS 70
THIRD STAR

NOTE B

- CURRENT VALUE BASED ON ALLOWABLE DIFFERENCE BETWEEN ACTUAL AND INDICATED ANGLES BETWEEN THE TWO STARS
- RESPONSIBILITY
- DATE AVAILABLE 9-26-68
- REFERENCE 7

NOTE C

- CURRENT VALUE BASED ON CONVENIENCE OF SEXTANT FIELD OF VIEW
- RESPONSIBILITY MIT
- DATE AVAILABLE 9-26-68
- REFERENCE 7

NOTE D

- CURRENT VALUE BASED ON FAILURE LIMIT
- RESPONSIBILITY MIT
- DATE AVAILABLE 6-27-69
- REFERENCE 2

NOTE E

- CURRENT VALUE BASED ON 3 - MEASUREMENT ACCURACY OF 2 SXT ALIGNMENTS
- RESPONSIBILITY MPAD/MIT
- DATE AVAILABLE 6-27-69
- REFERENCE 2

LOS REV 10

CSM IMU ALIGN AND DRIFT CHECK

WAS STAR DATA CHECK OK? (< 0.05 DEG)
NOTE B

NO → REPEAT P52

YES →

WAS THIRD STAR IN SXT FOV? (> 0.9 DEG)
NOTE C

NO → REPEAT P52 AND ASSESS FAILURE

YES →

DOES ANY GYRO TORQ ANGLE INDICATE GNCS FAIL? (> 1.5 DEG/HR)
NOTE D

YES → GNCS FAIL GO TO ALTERNATE MISSION

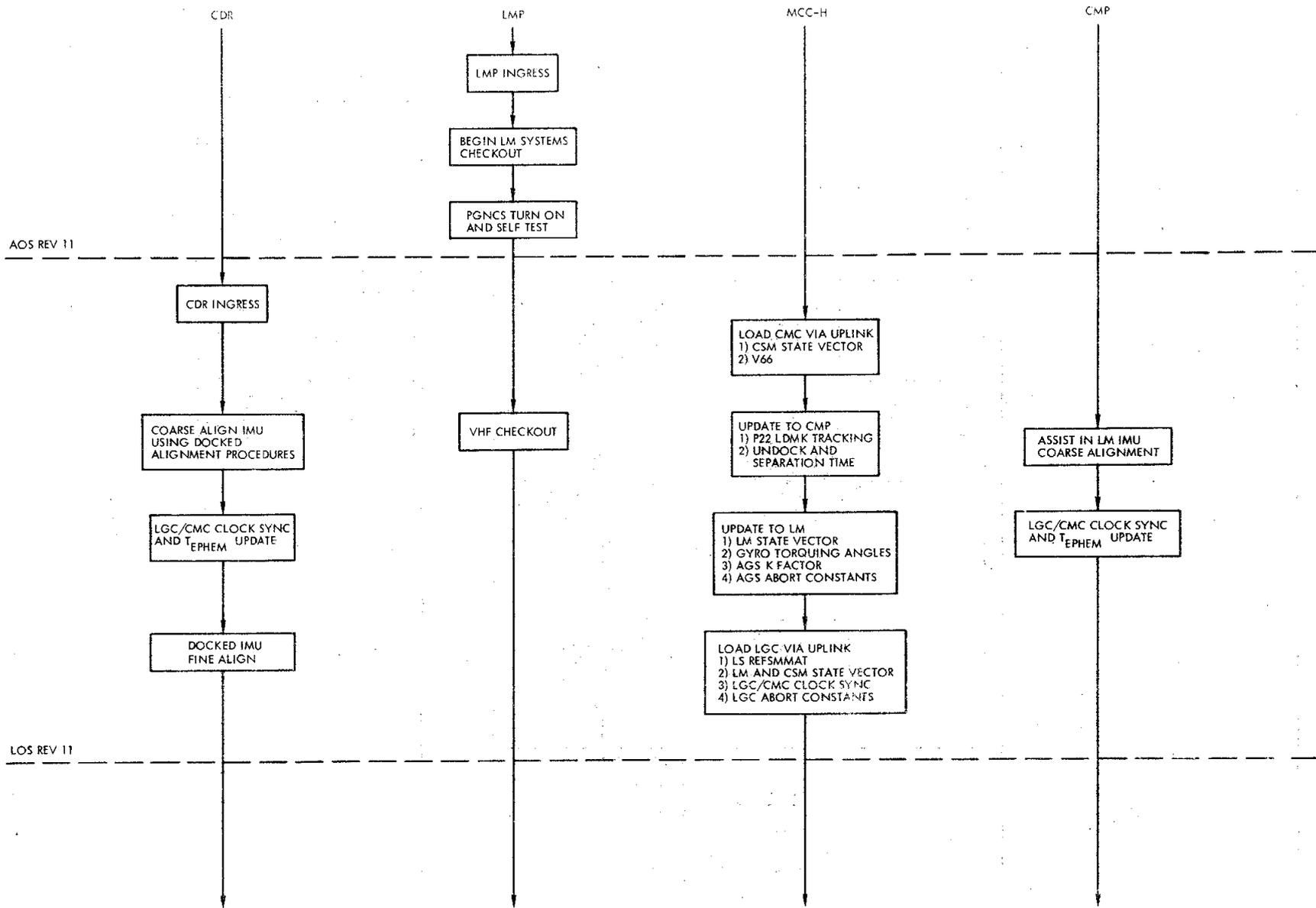
NO →

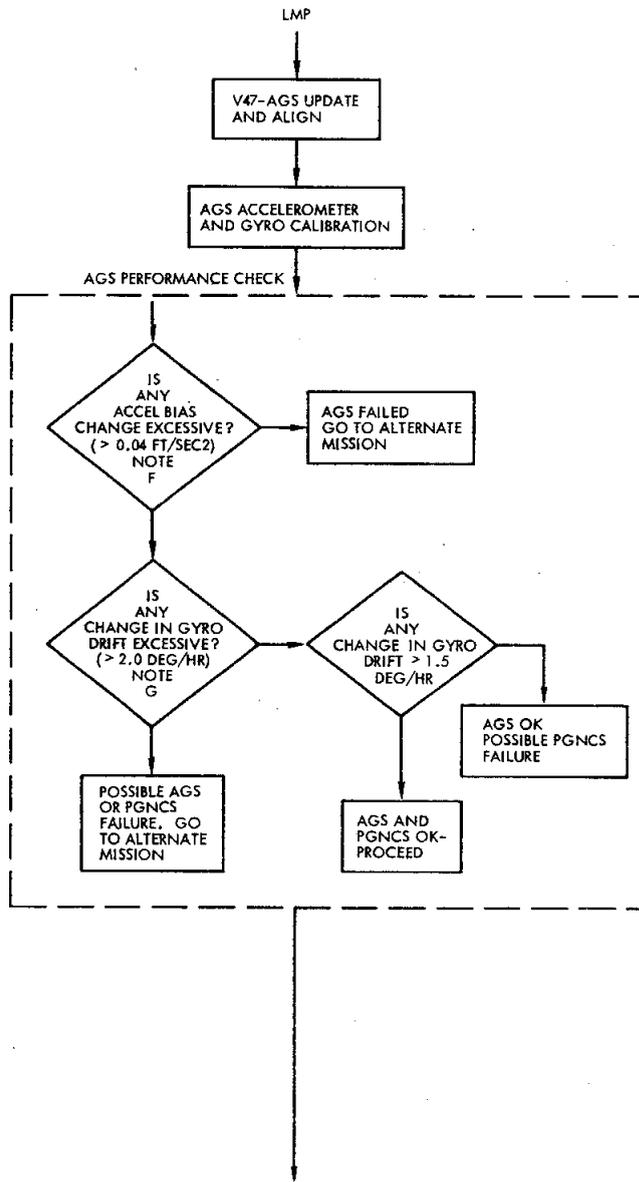
DOES ANY UNCOMPENSATED GYRO DRIFT EXCEED 0.045 DEG/HR?
NOTE E

YES → UPDATE NBD (MCC-H) USING GYRO TORQ ANGLES PROCEED

NO →

PROCEED





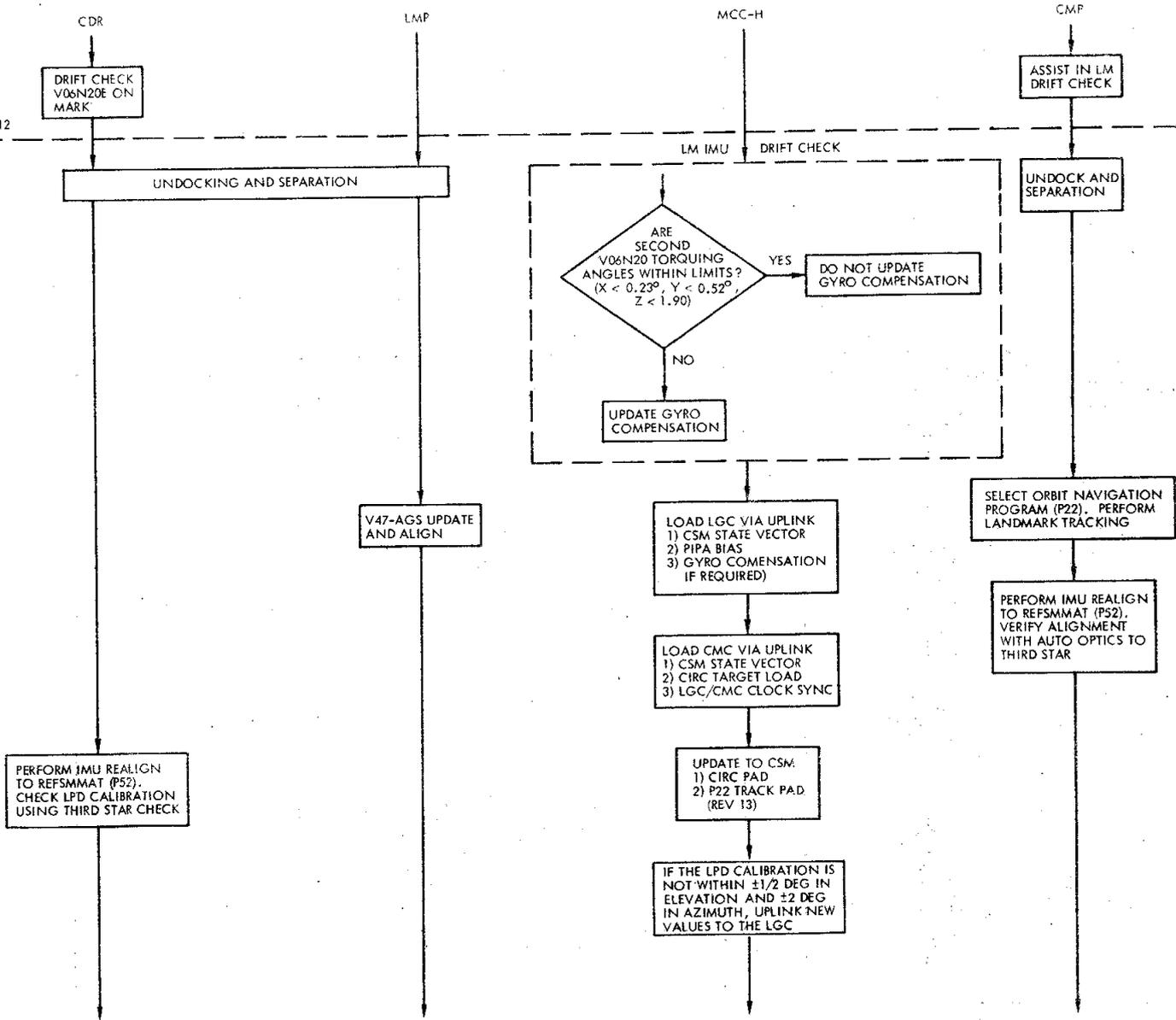
NOTE F

- CURRENT VALUE BASED ON ESTIMATED IMPENDING FAILURE LIMIT
- RESPONSIBILITY G*CD/TRW
- DATE AVAILABLE 6-27-69
- REFERENCE 2

NOTE G

- CURRENT VALUE BASED ON THE LARGER OF ESTIMATED IMPENDING FAILURE LIMIT FOR AGS AND PGNC'S GYROS
- RESPONSIBILITY G*CD
- DATE AVAILABLE 6-27-69
- REFERENCE 2

AOS REV 12



5-7

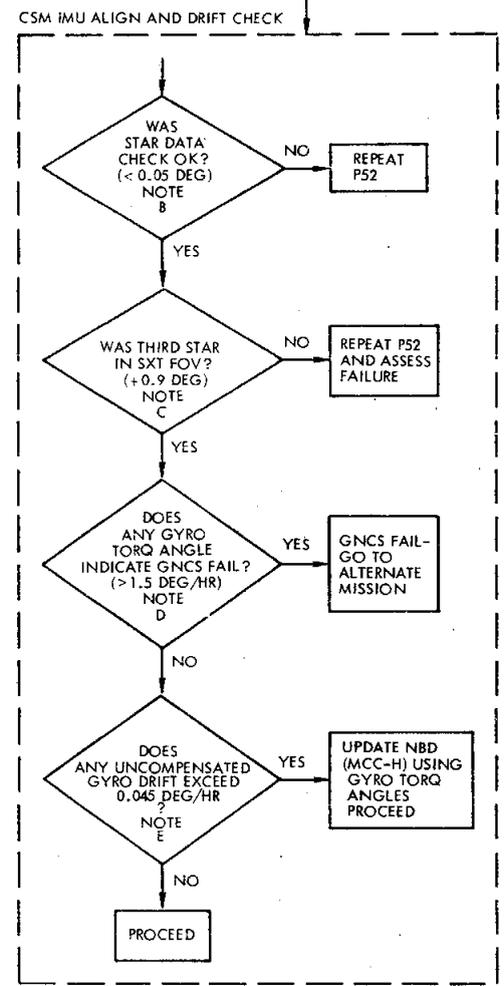
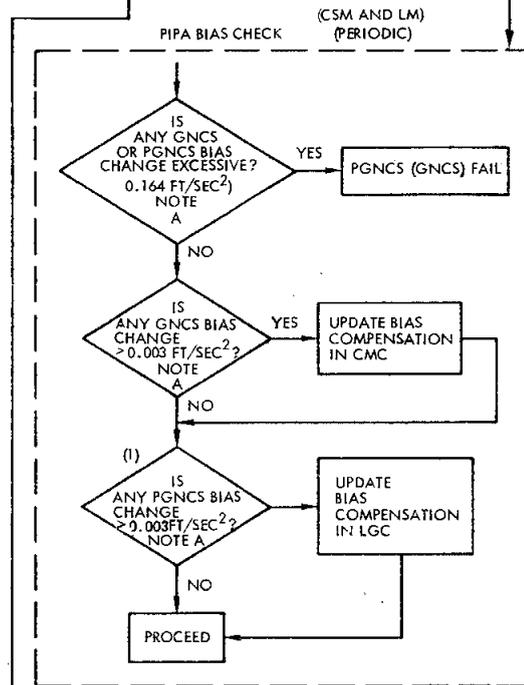
CDR

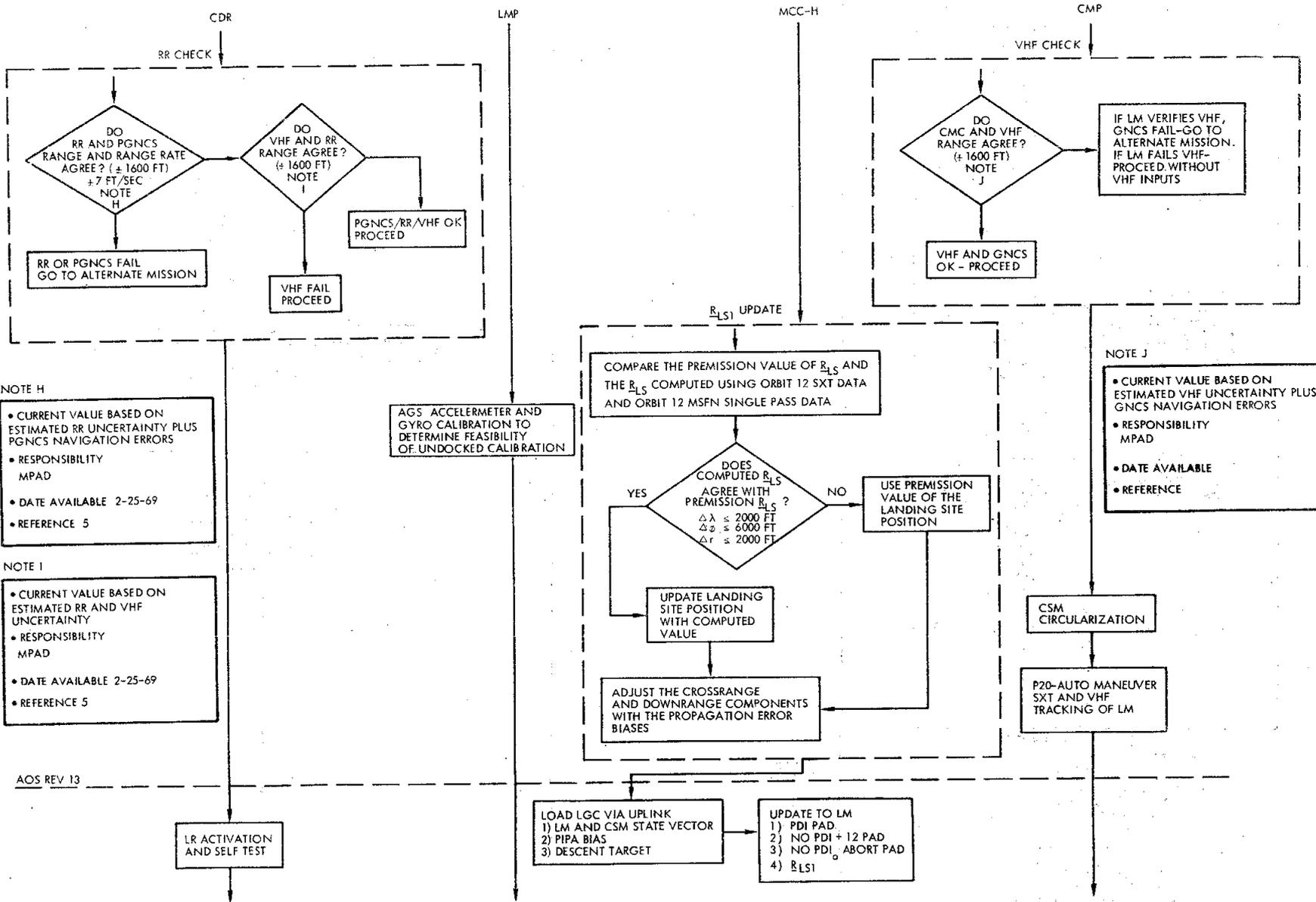
LMP

MCC-H

CMP

5-8





NOTE H

- CURRENT VALUE BASED ON ESTIMATED RR UNCERTAINTY PLUS PGNC'S NAVIGATION ERRORS
- RESPONSIBILITY MPAD
- DATE AVAILABLE 2-25-69
- REFERENCE 5

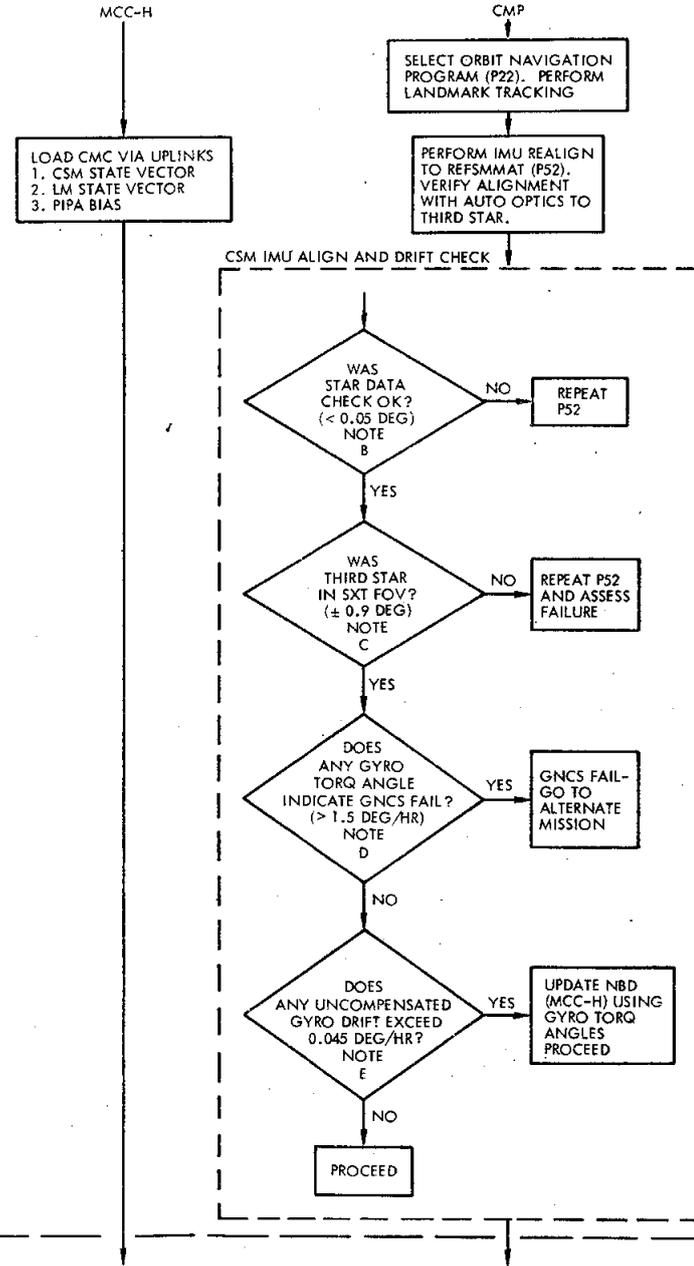
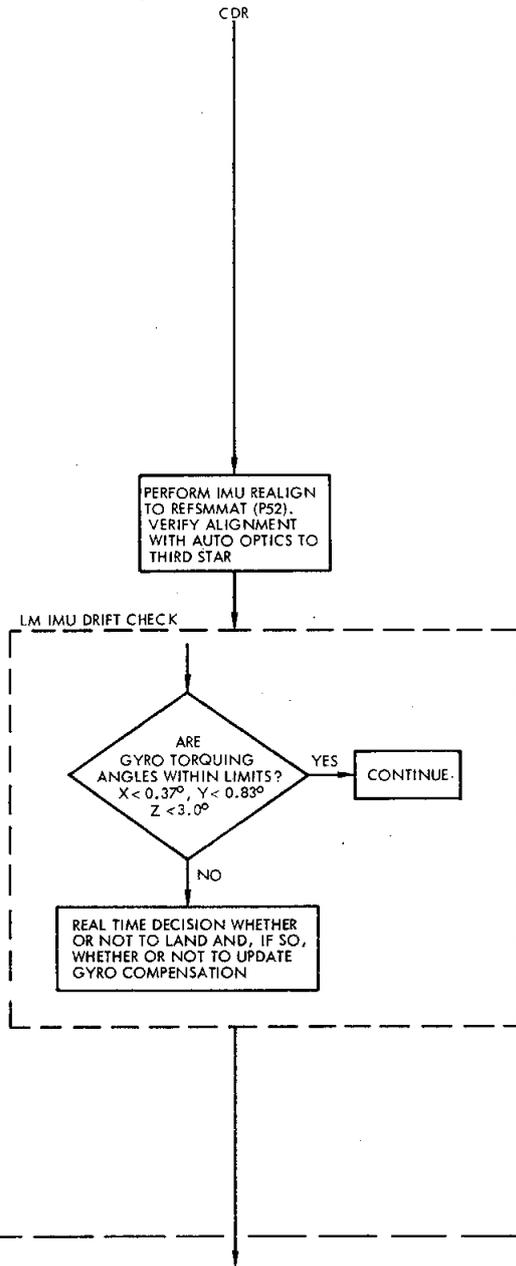
NOTE I

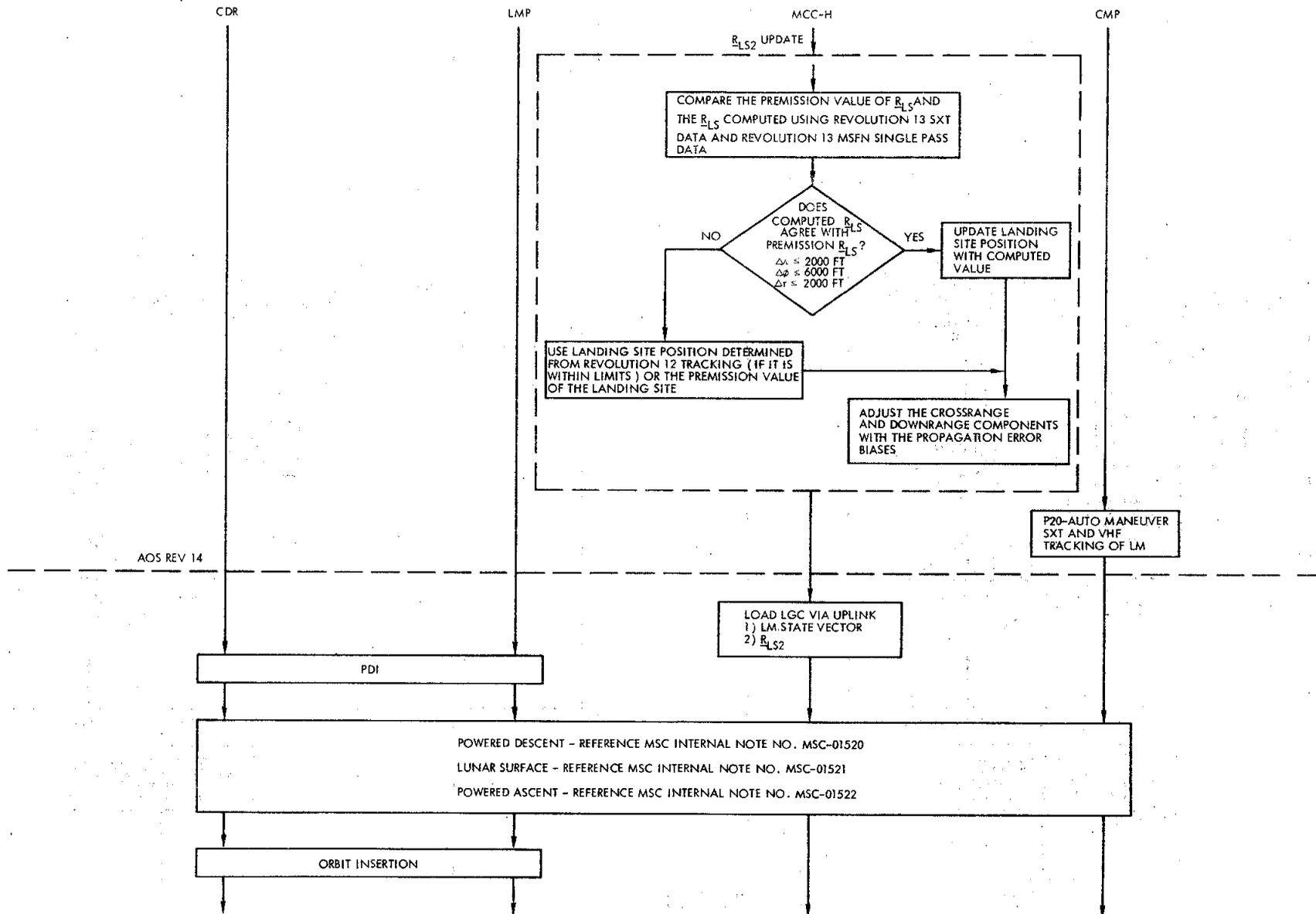
- CURRENT VALUE BASED ON ESTIMATED RR AND VHF UNCERTAINTY
- RESPONSIBILITY MPAD
- DATE AVAILABLE 2-25-69
- REFERENCE 5

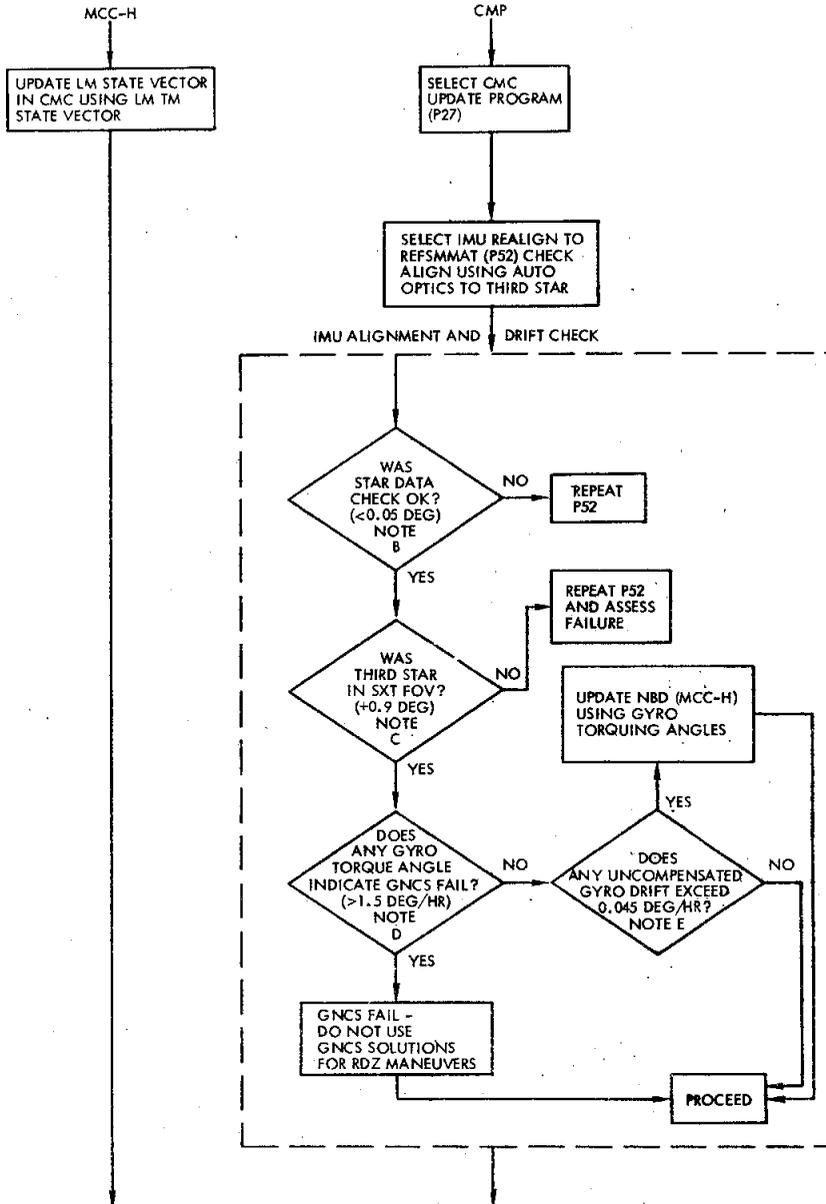
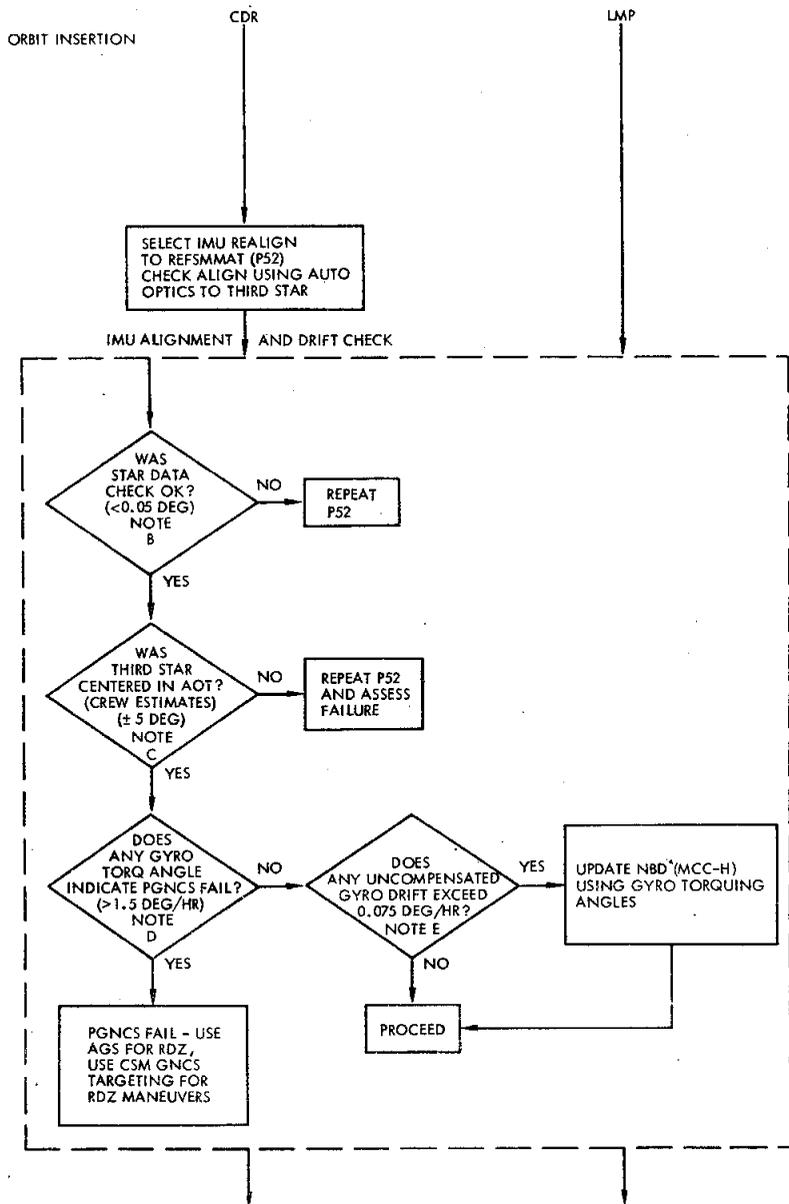
NOTE J

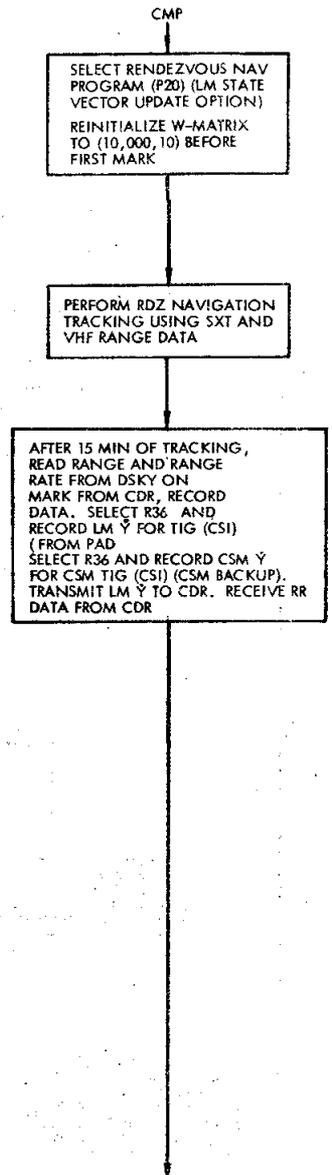
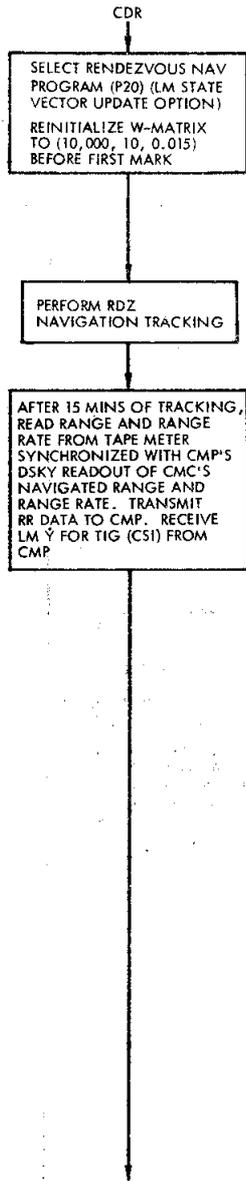
- CURRENT VALUE BASED ON ESTIMATED VHF UNCERTAINTY PLUS GNCS NAVIGATION ERRORS
- RESPONSIBILITY MPAD
- DATE AVAILABLE
- REFERENCE

AOS REV 13









CDR

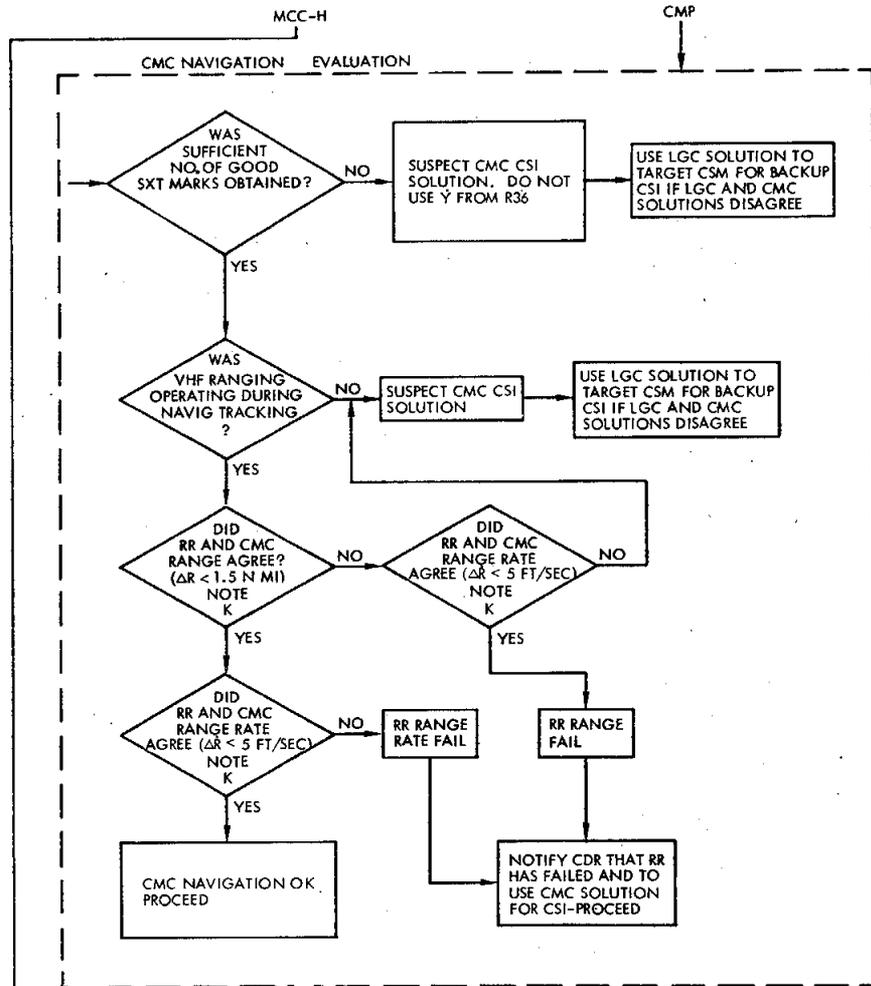
LMP

MCC-H

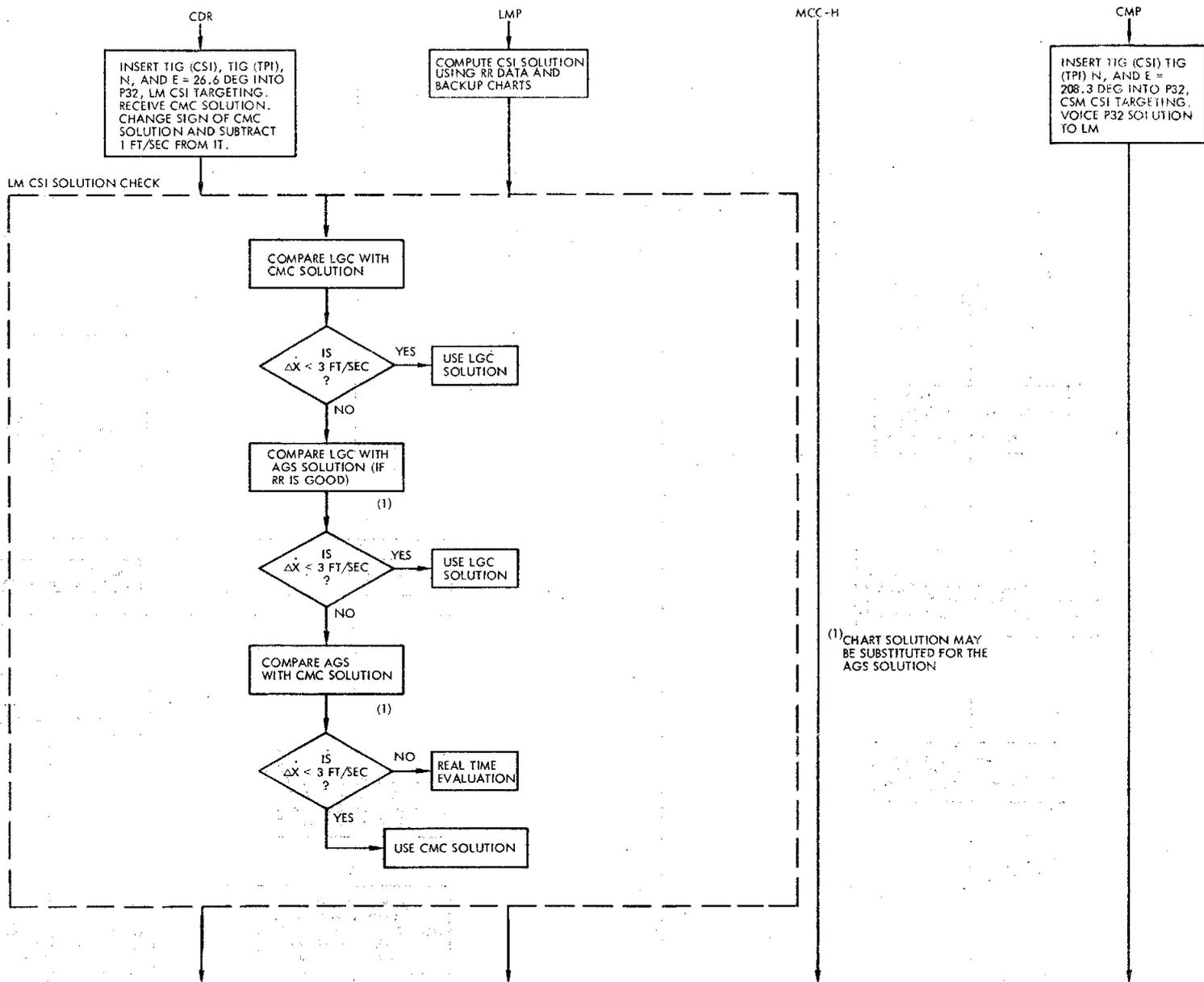
CMP

NOTE K

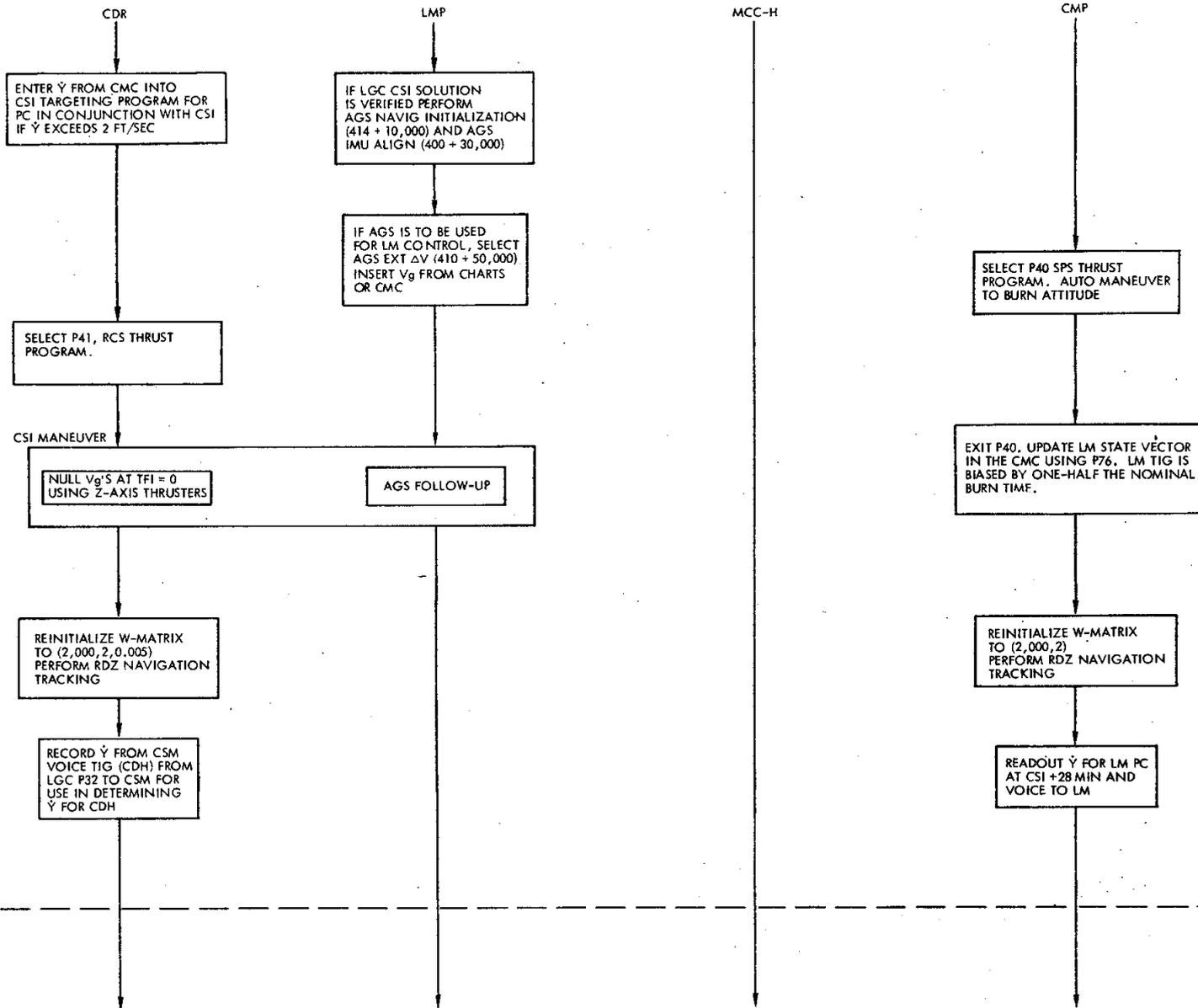
- CURRENT VALUE BASED ON ESTIMATED RR UNCERTAINTY PLUS GNCS NAVIGATION ERRORS
- RESPONSIBILITY MPAD
- DATE AVAILABLE
- REFERENCE

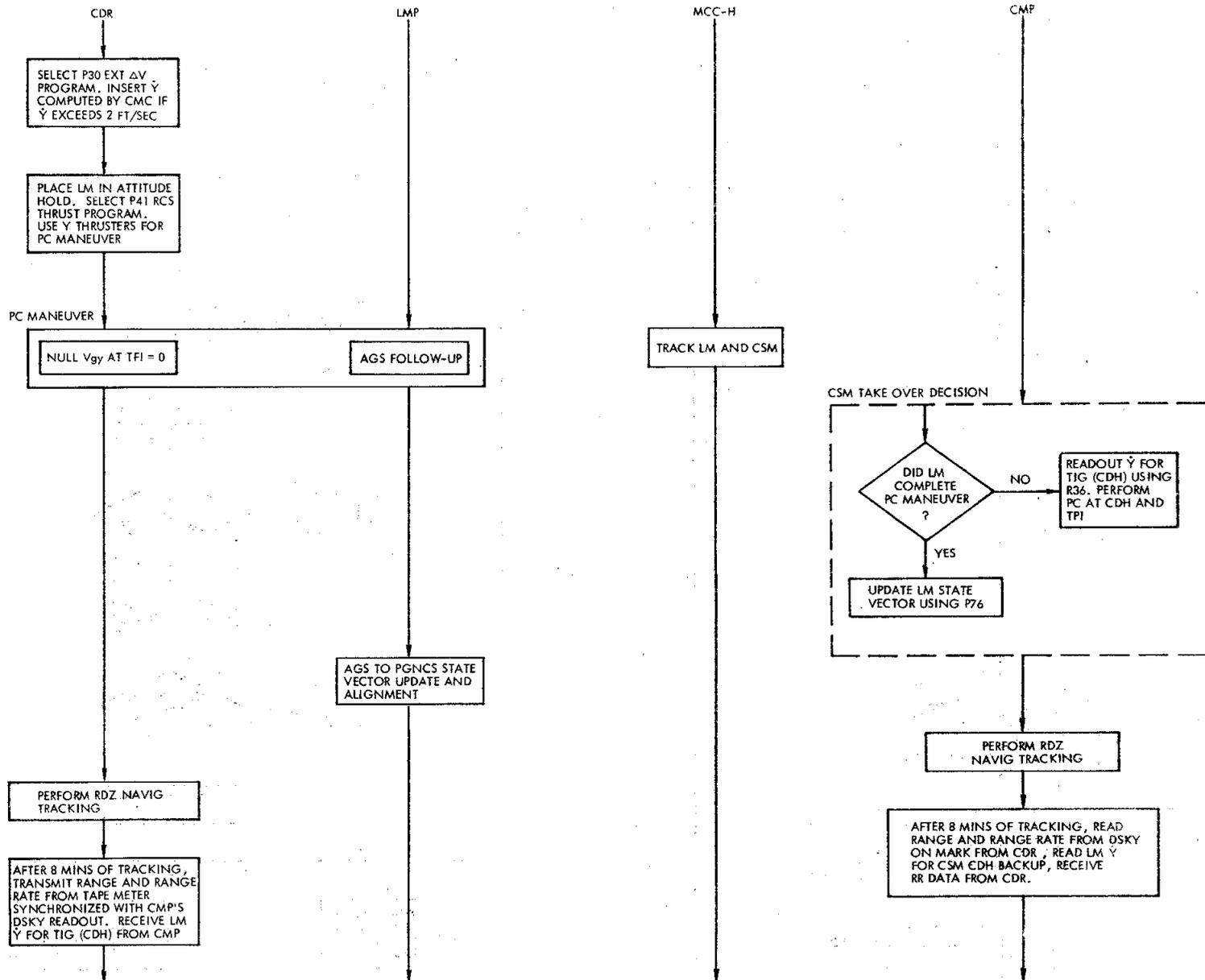


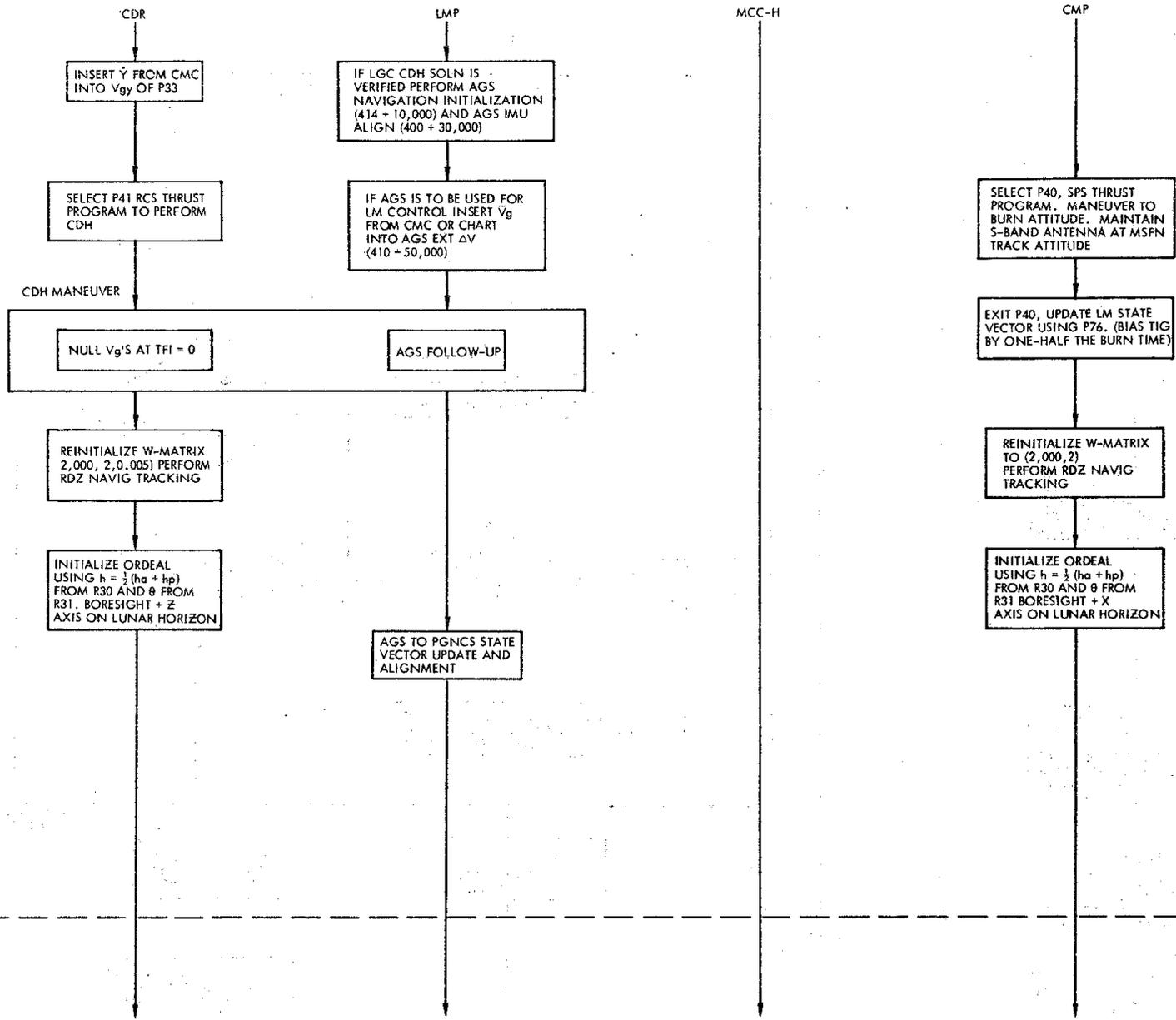
LOS REV 31



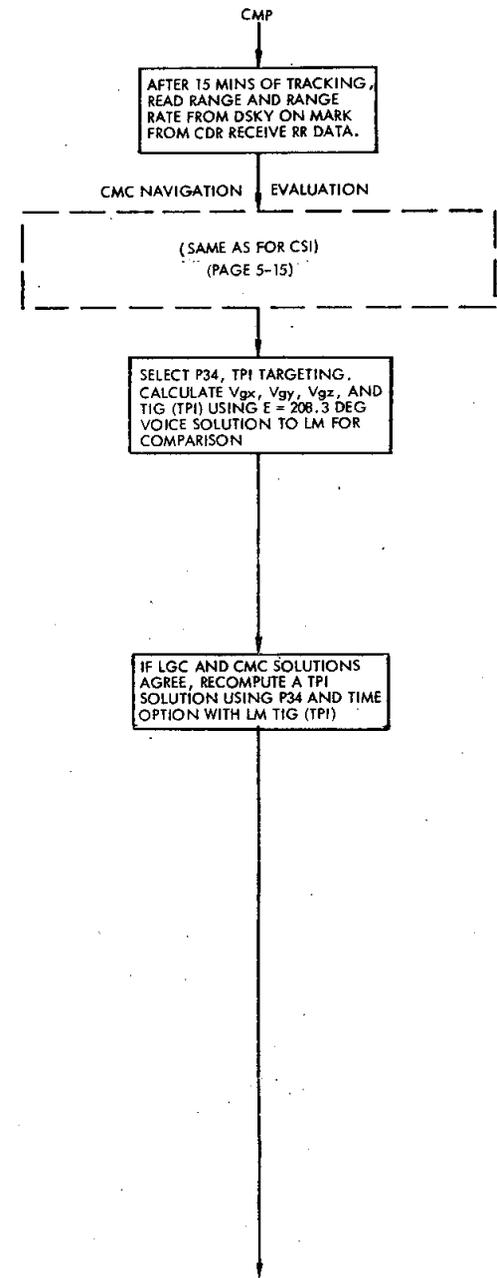
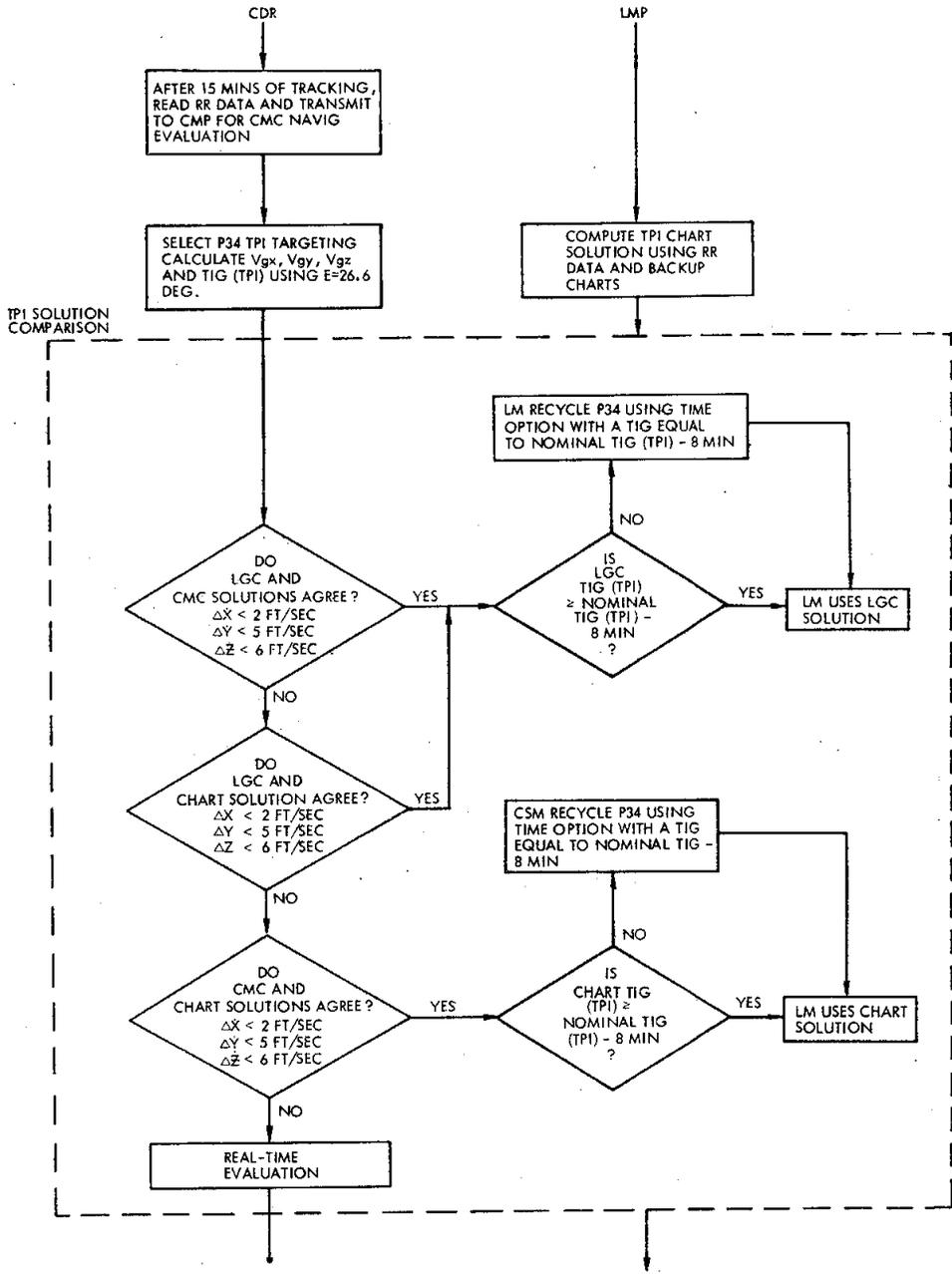
(1) CHART SOLUTION MAY BE SUBSTITUTED FOR THE AGS SOLUTION

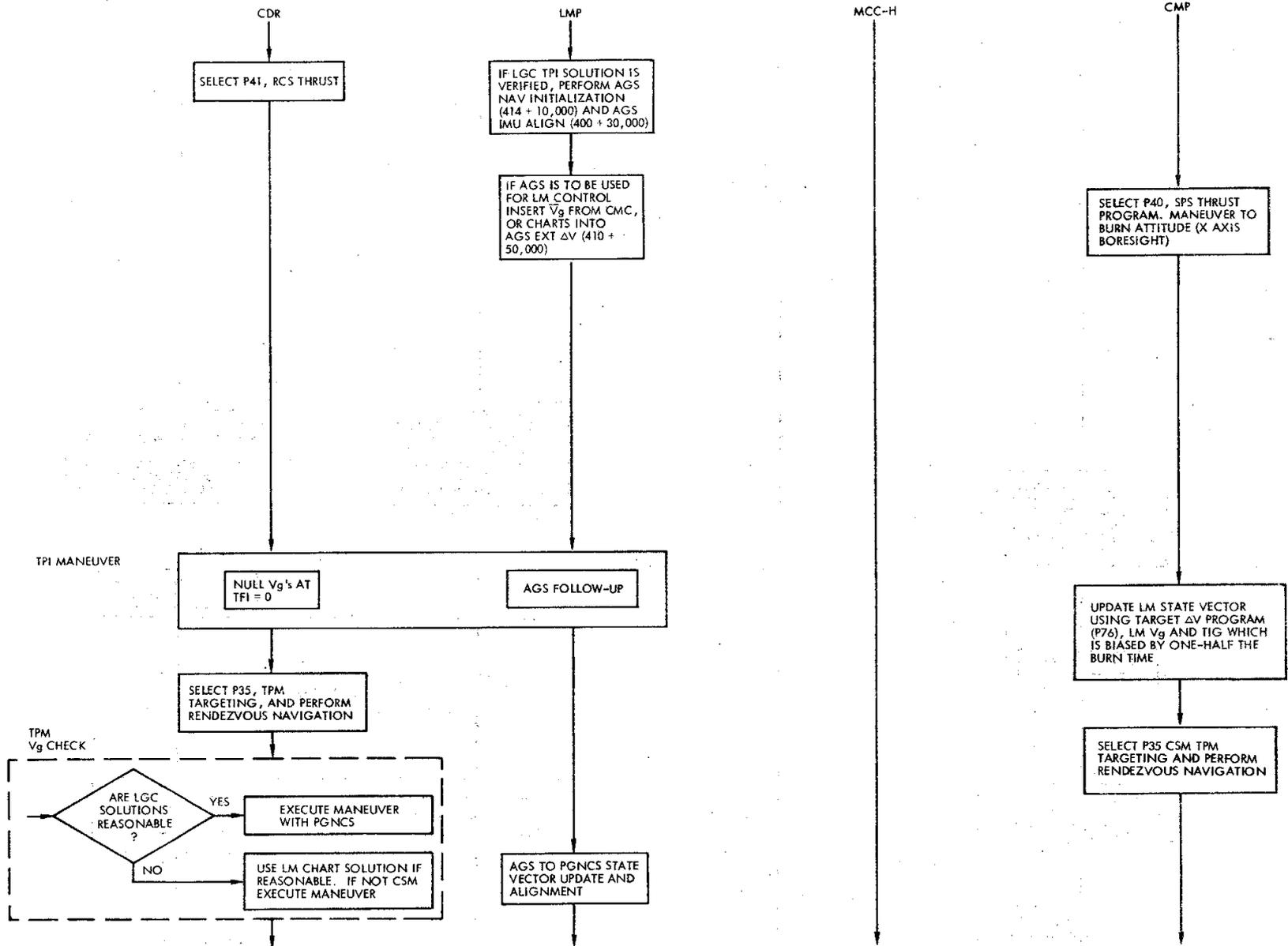


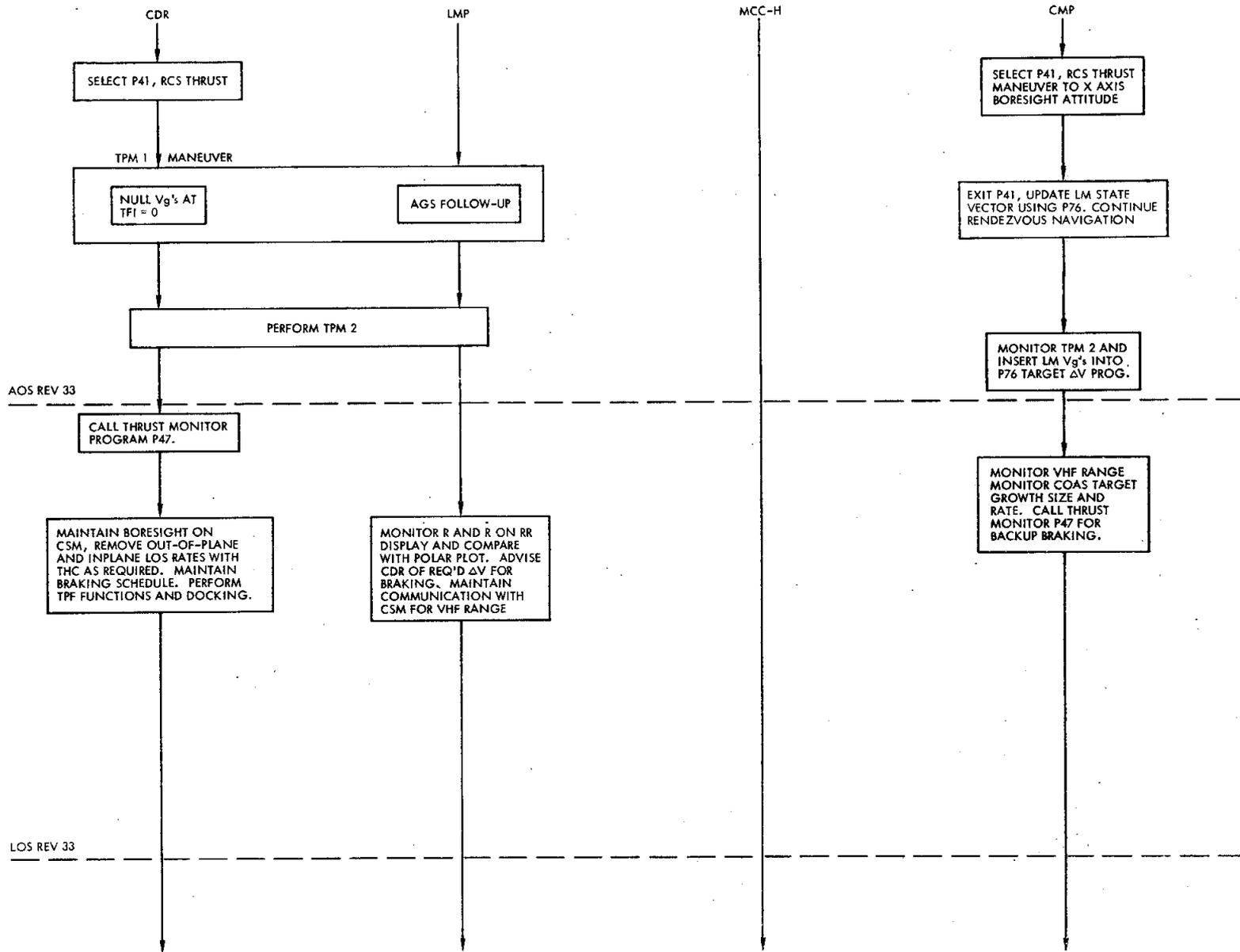


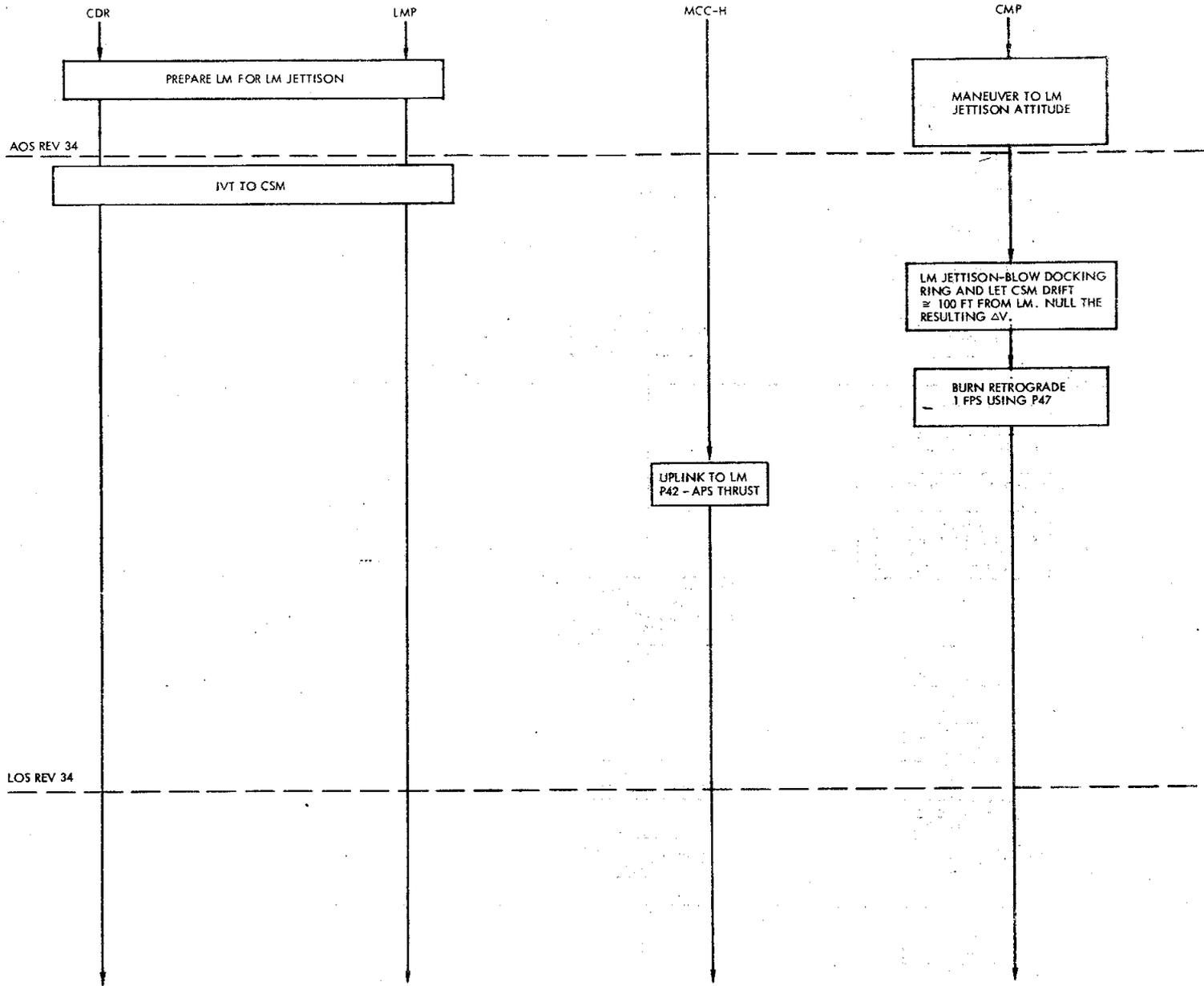


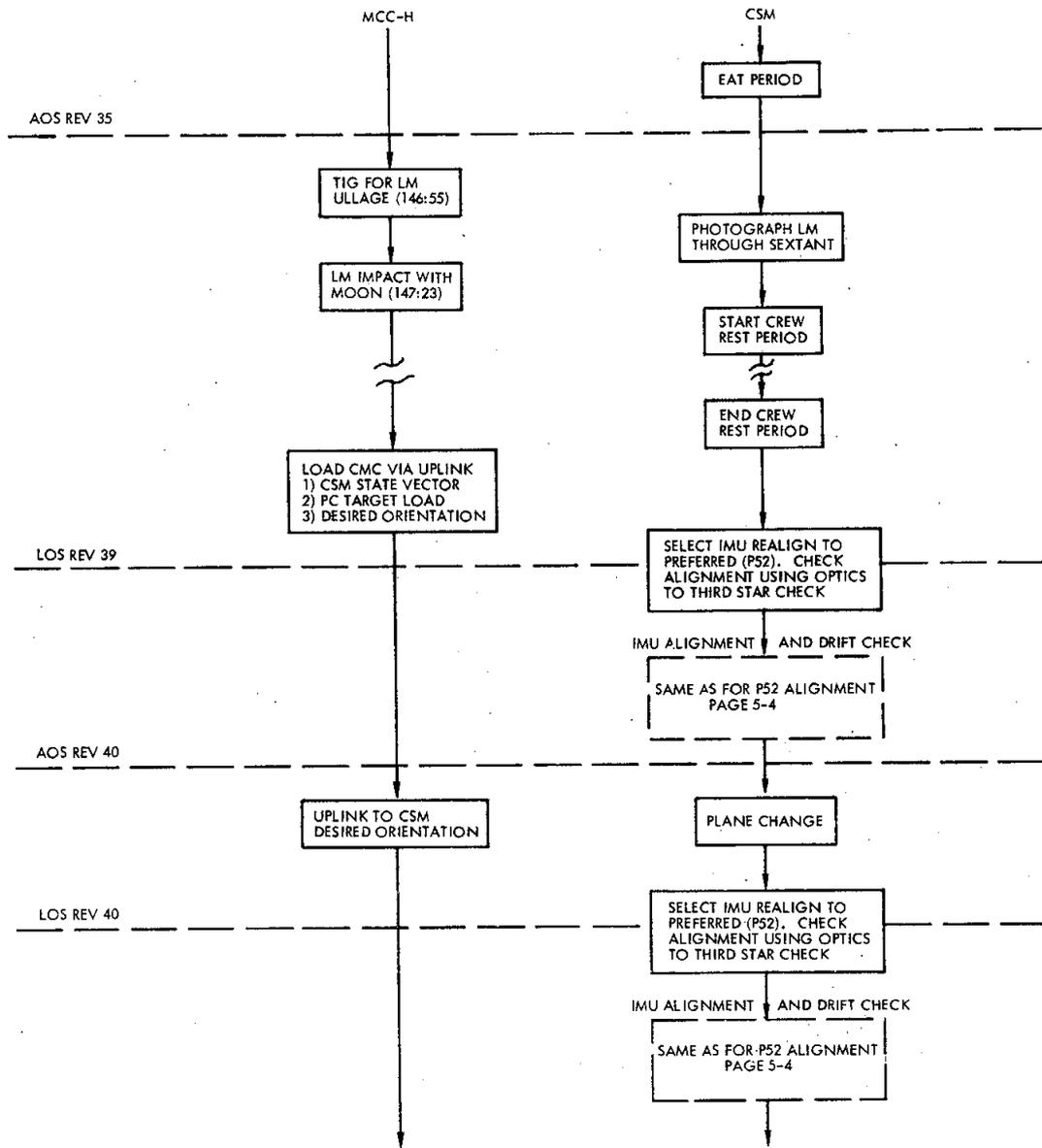
5-19











MCC-H

CMP

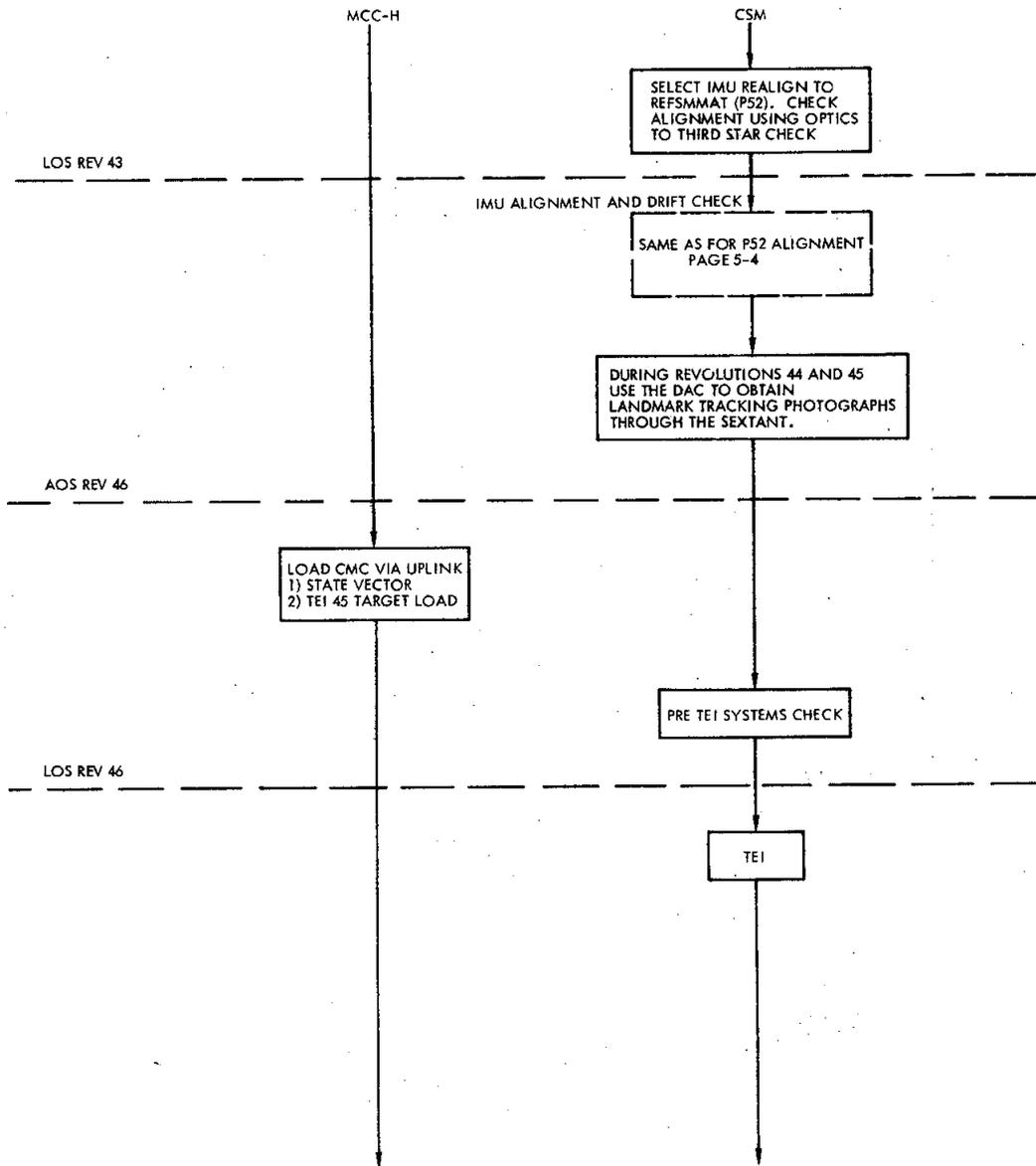
BEGIN STERO STRIP
PHOTOGRAPHY AT GROUND
TERMINATOR PLUS ONE MINUTE.

AOS REV 41

END STERO STRIP
PHOTOGRAPHY AT GROUND
TERMINATOR MINUS ONE MINUTE

LOS REV 41

DURING REVOLUTIONS 42 AND 43
USE THE TOPO CAMERA TO
OBTAIN HIGH RESOLUTION VERTICAL
AND 30 DEG FORWARD-LOOKING
OBLIQUE PHOTOGRAPHS OF CENSORINUS
AND DAVY RILLE.



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