

APOLLO

GUIDANCE, NAVIGATION AND CONTROL

Submitted by: Gerald M. Levine Date: 3 Apr '70
G. M. LEVINE, DIRECTOR, GUIDANCE ANALYSIS
APOLLO GUIDANCE AND NAVIGATION PROGRAM

Approved: R. H. Larson Date: 4-3-70
R. A. LARSON, LUMINARY PROJECT MANAGER
APOLLO GUIDANCE AND NAVIGATION PROGRAM

Approved: R. H. Battin Date: 4/3/70
R. H. BATTIN, DIRECTOR, MISSION DEVELOPMENT
APOLLO GUIDANCE AND NAVIGATION PROGRAM

Approved: D. G. Hoag Date: 3 Apr 70
D. G. HOAG, DIRECTOR
APOLLO GUIDANCE AND NAVIGATION PROGRAM

Approved: Ralph R. Ragan Date: 3 Apr 70
R. R. RAGAN, DEPUTY DIRECTOR
CHARLES STARK DRAPER LABORATORY

R-567

GUIDANCE SYSTEM OPERATIONS PLAN
FOR MANNED LM EARTH ORBITAL AND
LUNAR MISSIONS USING
PROGRAM LUMINARY 1C (LM131 REV. 1)

SECTION 5 GUIDANCE EQUATIONS
(Rev. 8)

APRIL 1970

MIT

**CHARLES STARK DRAPER
LABORATORY**

CAMBRIDGE MASSACHUSETTS 02139

ACKNOWLEDGEMENT

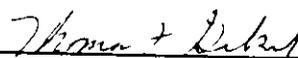
This report was prepared under DSR Project 55-23870, sponsored by the Manned Spacecraft Center of the National Aeronautics and Space Administration through Contract NAS 9-4065 with the Massachusetts Institute of Technology.

R-567

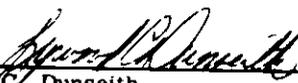
GUIDANCE SYSTEM OPERATIONS PLAN
FOR MANNED LM EARTH ORBITAL AND
LUNAR MISSIONS USING
PROGRAM LUMINARY 1C

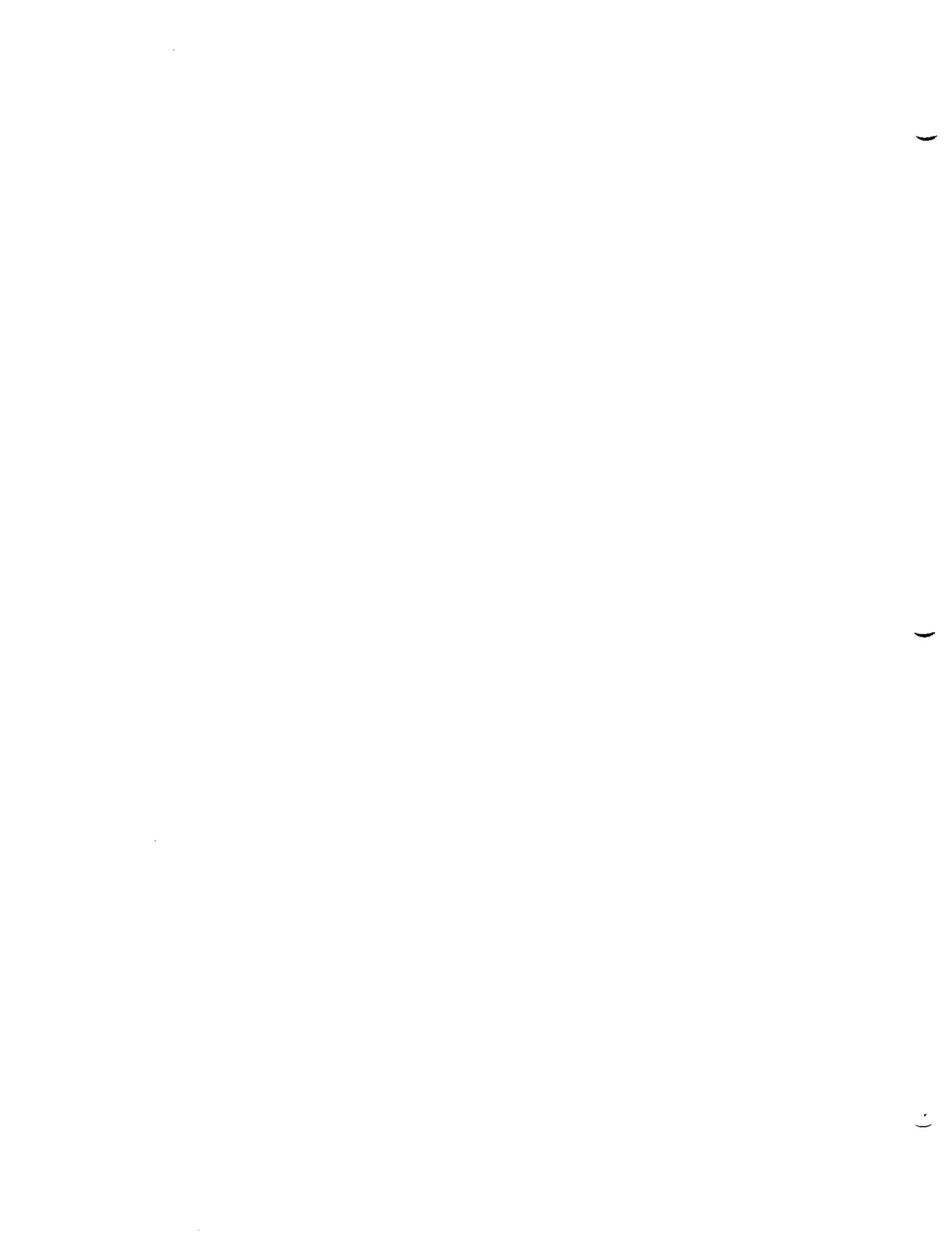
SECTION 5 GUIDANCE EQUATIONS

Signatures appearing on this page designate approval of this document by NASA/MSC.

Approved:  Date: 7/22/68
Thomas F. Gibson
Asst. Chief, Flight Software Branch
Manned Spacecraft Center, NASA

Approved:  Date: 7/29/68
James C. Stokes, Jr.
Chief, Flight Software Branch
Manned Spacecraft Center, NASA

Approved:  Date: 7-30-68
Lynwood C. Dunseith
Chief, Flight Support Division
Manned Spacecraft Center, NASA



Date: November 1969

REVISION INDEX COVER SHEET
GUIDANCE SYSTEM OPERATIONS PLAN

GSOP #R-567 Title: For Manned LM Earth Orbital and Lunar
Missions Using Program Luminary 1C (Rev 130)

Section #5 Title: Guidance Equations (Revision 7)

This revision of GSOP R-567 Section 5 describes the guidance equations used in Program Luminary 1C. All PCR's and PCN's approved by NASA for Luminary 1C as of the date of issue have been incorporated. In addition, many editorial changes have been included. PCR and PCN changes are indicated by denoting the applicable number at the bottom of the page and indicating the location of the change with a black line on the side of the page. Editorial corrections (not covered by PCR) are denoted by black dots .

Below is a list of PCR's and PCN's incorporated in this GSOP. In addition, for convenience, the PCR's and PCN's included in previous issues of the GSOP are listed.

Incorporated in Original Issue of GSOP (Rev. 0)

7	IMU Alignment Program Change
8	Mod. to ΔV Monitor
15	S-Band Antenna
30	DPS Trimming Time
37	Lambert Fix
39	Z Axis Track
*80.2	State Vector Synchronization

Incorporated in Rev. 1 of GSOP

13	Mode II Rendezvous Radar Designate
70	Changes to P12, P70, P71
88.3	Erasable Addition
95.2	Sign Change to Noun 72
97.2	RR Monitor Routine Change
105	Deletion of LM/CSM Separation Monitor
106	Deletion of Direct Transfer Ascent Targeting
118	Redefinition of R12 Proceed & Inhibit
121.2	New Engine Fail Routine
122.2	Terminate Ullage at Fixed Time after Engine on Signal
*124.2	Attitude Maneuver During RR Search Routine
133	LM Abort Insertion Targets
134.2	Pulse Torquing to Achieve IMU Realignment
137	R 12 Flag Change
138	Deletion of Predicted Launch Time Program
*144.2	Reduction of Ullage Duration of Delta V Threshold
*146.2	Emergency Termination of Integration Function
164.3	Mode II Attitude Error Display
*171.2	Targeting Interface with P40/41
*173.3	RMS Position and Velocity Error Display
*181.2	Reduction of Sun Occultation Cone
*191.2	Correct GSOP Reference to Range Low Scale Discrete

210	Increase DPS Throttle Recovery Limit
214	DSKY Light Utilization For LR
216	Initiation of LR Antenna Position Change
229	Addition of R77 to Luminary
233	Change to CDH Time Test
244	Delay use of LR Data
246	Implementation of One-Phase Descent Guidance Logic
248	LR Reasonability Test
252	Update GSOP Section 5
253	Landing Radar Read Initiation
400	Provide RR Downlink Data on Lunar Surface in P22
*409.2	Transfer of RR From Auto Track to LGC Mode
*410.2	Prevent Problems Arising from RR Control Mode Changes
*415.2	Prevent Display Conflicts When RR Goes Out of Auto Mode
*431.2	R24/ R61 Interface
*432.2	GSOP Section 4 R21 Repositioning Check
437	MIDTOAVE For P47, P12, P63
439.2	Downgrade the Authority of the Preferred Attitude Flag
*457.2	Correction to R22
468.2	Change R32 into Program P76
470	Addition of P68 Program
471	Revision of RR Remode Release Angles
472	Simplification of P71
*476	FINDCDUW - Gimbal Drive
478	Ascent Guidance Equation Compensation
489	Bypass R-54 and Noun 93 During Initial Alignment
*497	Do Not Delay P63 Throttle- Up Time
*499	R10 Computation Frequency
*507.2	Termination of Integration

527 Mark Verb for R59
537 Surface Navigation Flag Check in P20 and P22
541 Decrease Frequency of Marks in P22
549 RR Shaft/Trunnion Bias Mod. by Crew
*561 Reset RENDWFLG in P12
*564 Luminary GSOP Section 5 Update
*565 GSOP Section 5 Control Data Reference Update
568 Delete Use of R29 During P63
*573.3 Definition of A_x and A_y in Section 5 of GSOP
587.2 APS and DPS TAIL OFF Constants
*604 Provide Maximum Display for Pericenter and Apocenter in P30, P31, R30
*608 Simplification of Preferred Orientation Selection (P57)
609 Preferred IMU Orientation When Thrust is Along Local Vertical

Incorporated in Rev. 2 of GSOP

99.2 DSKY Display of RR Position in Mode II
254.2 Modification of CDH Time Computation Logic
258 Redefinition of Vertical Rise Velocity Cutoff
259 Omit Zone 1 from Descent Logic
*564 Luminary GSOP Section 5 Update
576.2 Removal of Backward Updating Constraint on State Vector
609 Preferred IMU Orientation When Thrust is Along Local Vertical
612.2 Raise Thresholds for Delta V Monitor
613 Automatic 4-jet Translation Capability in P12, P70, P71
630 Update of Ascent Guidance Engine Parameters
631 Reduce LGC Executive and Central Processor Load
632 Allow Astronaut to Continue Landing Display, When Radar Does Not Achieve Position #2
*634 Correct Design Flaw in R61, R65 for High LOS Rates
*635 FINDCDUW Gain Change for CSM-Docked Burns
636 Nov. 68 GSOP Section 5 Corrections in Landing Program
639 Altitude Reasonability Test Parameters in Erasable Memory
*640 Remove the Instabilities and Excessive Overshoots from the RR Designate Routine R21
652 Correction to Section 5 APS Minimum Impulse Burn Parameters
*750 Luminary GSOP Section 5 Update (Rev. 2)

Incorporated in Rev. 3 of GSOP

- *758 Allow 15 Seconds for an Integration Time Step in R-41
- 761.1 R-2 Lunar Potential Model

Incorporated in Rev. 4 of GSOP

- 254.2 Modification of CDH Time Computation Logic
- 260 Preferred Orientation During LM Aborts
- 268.2 Reduction of P34/35 Run Time
- 270 Placement of Desired Insertion Radial Velocity Component Into Erasable for P70/P71/P12
- *622 Correction to R-31 When LM is on the Lunar Surface
- 636 November, 1968 GSOP Section 5 Corrections in Landing Program
- 642 Provide "Wings Level", Heads Up, Fine Z Axis Tracking
- 646 Give Astronaut the Option to Confirm Mainlobe Lock-on after R-21 Acquisition
- 647 Replace Lambert with "A" Steer in P40, P41 and P42
- 654 Lessen Delays in R-31
- *659.2 Suppression of X-Modulo-ing by Kepler
- 696 V06N22 Display in P57
- 697 Limitation of LM Abort Orbit Insertion to $1/2^{\circ}$ Plane Change
- 698 Add LM Position Determination Capability to P57
- 699 Pad Load AOT Back Detent AZ and E1 Angles
- 702 Add COAS Calibration Option to R52
- 707.2 Change from 1968/1969 Ephemeris Data to 1969/1970 Ephemeris Data
- 708 Provide Continuously Variable Abort Orbit Insertion Targeting
- 709 Improve TGO Prediction for Short Burns in the BURNTIME Routine
- 716 Ascent Powered Flight RCS Control
- 719 Speed Up P21
- 720 Abort Coasting Integration When in Infinite Acceleration Overflow Loop
- 721 Time-Theta and Time-Radius Alarm Abort
- 722 Improve Performance of RR Designate Procedure on Lunar Surface
- 732 V67 Addition of Radar Bias Initialization and Deletion of $\sqrt{3}$ Factor
- 736 Add Source Code to Noun 49 in P20/P22
- 738 H, V, γ Display with P21
- *744 Change ϵ to 1.5 Seconds in R24
- 754 Provide IMU Orientation Selection Option Code in P57
- *755 Change IMU Gimbal Angles in Gravity Vector Determination

279 Variable Insertion Computation with Capability to
Abort at any Time

670 Simplification of Landing Programs

688 Guidance Frame Erection Check

695 CSI Apoapsis Computation Option

700A Improve the Rate of Descent Mode (P66) Performance

723 Two-Segment LR Altitude and Velocity Weighting
Functions

*731 Modify the Lunar Landing Guidance Equations to
Compensate for Computation, Throttle, FINDCDUW,
and Attitude Control Lags

737 Permit ATT HOLD Mode in P63, 64, 65

751 Make 1406 Alarm Non-Abortive

756 Guidance Frame Erection Trajectory Shaping
Factors

762 Delete V68

*765 New Propulsion System Constants

772.2 Libration Vector at Landing Time

773.2 Fix Constants for Planetary Inertial
Orientation Subroutine

775 Modify R12 to Permit LGC Compensation for
Doppler in LR Range Reading

776.2 Improved R2 Model Timing

780 Provide Pure RR Range, Range Rate, and Time Tag
during P20, P22 and P25.

801.2 Make BAILOUT Alarms Start with 3XXXX and
POODOO Alarms Start with 2XXXX.

817 Eliminate Undesirable LR Position Alarms from R12

818 Permit Rejection of Individual Measurement Incor-
porations in P20

820 Eliminate Lighting of ALT Light when Low Scale
Discrete is Absent

823 Delete P31 (Lambert Aim Point Guidance Program)

*830 Supplementary ASTEER Modifications

*831.2 Lambert Overflow Protection

832.2 Remove Restriction of Running R05 Only in POO

834 Descent Guidance Corrections

839 R12 and LR Re-position Routine Improvements

840 Reduce Oscillation in P64/P65

844 Deletion of P38/P78 and P39/P79

845 Do Not Turn On R29 During P70/P71

847 Eliminate Possible Lock-out of Pitch-over from
P12, P70, P71

848 Prevent RR ECDUs from Stealing LGC Memory
Cycles

Incorporated in Rev. 6 of GSOP

848 Prevent RR ECDUs from Stealing LGC Memory Cycles
854 Provide a Flexible Method for Crew to Modify RLS
855 Begin Reading LR Velocity as soon as Velocity Data Good Appears

Incorporated in this Issue of GSOP (Rev. 7)

285 Remove check of Auto Throttle Discrete
846 More Accurate Delta T Tail-Off for P70
863. 2 Make P76 Set NODO Flag
882 Replace VHORIZ with Something Better
893 Abort Targeting Flagbit
895 LR Reposition by V59E in P63
936. 2 Initialize V90 time to TIG
943 Velocity Reasonability Test
968 LPD Bias Correction
*971 Change Fixed Memory Constant (APS) Delta-T TAIL-OFF
972 Display Polarity of Sighting Angle Difference in R54

SECTION 5
GUIDANCE EQUATIONS

5.1 INTRODUCTION

5.1.1 GENERAL COMMENTS

The purpose of this section is to present the Guidance and Navigation Computation Routines associated with the LM Apollo Lunar Landing Mission. These Routines are utilized by the Programs outlined in Section 4 where astronaut and other subsystem interface and operational requirements are described. The guidance and navigation equations and procedures presented in Section 5 are primarily concerned with mission type programs representing the current LM PGNCS Computer (LGC) capability. A restricted number of LGC service type program operations which involve computation requirements are also included.

The LM PGNCS Computer (LGC) guidance and navigation equations for the lunar landing mission are presented in the following five categories:

- Section 5.2 Coasting Flight Navigation Routines
- Section 5.3 Powered Flight Navigation and
 Guidance Routines
- Section 5.4 Targeting Routines
- Section 5.5 Basic Subroutines
- Section 5.6 General Service Routines

Guidance equation parameters required for program initialization and operation are listed in Section 5.7. These selected parameters are stored in the LGC erasable memory. General constants used in the equations of this volume are presented in Section 5.8.

A complete table of contents for Section 5 is presented in the following Section 5.1.2.1. A cross-reference between the LGC programs and routines of Section 4 that are described in Section 5 is listed in Section 5.1.2.2. In the following Section 5 table of contents and text, missing section numbers correspond to LUMINARY programs that have been deleted from the previous Section 5 GSOPs by MSC direction resulting from the LGC Fixed Memory Storage Review Meeting of August 28, 1967 and subsequent PCN's and PCR's.

This volume constitutes a control document to govern the structure of the LGC Lunar Landing Mission, using LUMINARY including PGNCS interfaces with the flight crew and Mission Control Center.

Revisions to this plan which reflect changes in the above control items require NASA approval.

The Guidance System Operations Plan is published as six separate volumes (sections) as listed below:

Section 1	Pre-Launch
Section 2	Data Links
Section 3	Digital Auto-Pilots
Section 4	Operational Modes
Section 5	Guidance Equations
Section 6	Control Data

The principal authors contributing to this Section 5 were:

L. Berman	P. Philliou
D. Gustafson	J. M. Reber
A. Klumpp	D. Reinke
B. Kriegsman	W. Robertson
G. Levine	N. Sears
W. Marscher	W. Tempelman
R. Morth	R. White

5. 1. 2 TABLE OF CONTENTS

5. 1. 2. 1 Section 5 Table of Contents

<u>SECTION NO.</u>	<u>TITLE</u>	<u>PAGE</u>
<u>5. 1</u>	<u>Introduction</u>	
5. 1. 1	General Comments	5. 1-1
5. 1. 2	Table of Contents	5. 1-3
5. 1. 3	General Program Utilization	5. 1-13
5. 1. 4	Coordinate Systems	5. 1-22
5. 1. 5	General Definitions and Conventions	5. 1-26
<u>5. 2</u>	<u>Coasting Flight Navigation</u>	
5. 2. 1	General Comments	5. 2-1
5. 2. 2	Coasting Integration Routine	5. 2-11
5. 2. 2. 1	General Comments	5. 2-11
5. 2. 2. 2	Encke's Method	5. 2-11
5. 2. 2. 3	Disturbing Acceleration	5. 2-14
5. 2. 2. 4	Error Transition Matrix	5. 2-16
5. 2. 2. 5	Numerical Integration Method	5. 2-21
5. 2. 2. 6	Coasting Integration Logic	5. 2-22
5. 2. 3	Measurement Incorporation Routine	5. 2-28
5. 2. 4	Rendezvous Navigation Program	5. 2-35
5. 2. 4. 1	Target Acquisition Routine	5. 2-35
5. 2. 4. 2	Rendezvous Navigation Routine	5. 2-53
5. 2. 4. 3	RR Monitor Routine	5. 2-75
5. 2. 4. 4	Preferred Tracking Attitude Routines	5. 2-77

<u>SECTION NO.</u>	<u>TITLE</u>	<u>PAGE</u>
5. 2. 5	RR Lunar Surface Navigation Program	5. 2 -81
5. 2. 5. 1	General Comments	5. 2 -81
5. 2. 5. 2	CSM Orbital Plane Change Estimation Routine	5. 2 -82
5. 2. 5. 3	Target Acquisition Routine	5. 2 -86
5. 2. 5. 4	Lunar Surface Navigation Routine	5. 2 -89
<u>5. 3</u>	<u>Powered Flight Navigation and Guidance</u>	
5. 3. 1	General Comments	5. 3 -1
5. 3. 2	Powered Flight Navigation - Average G Routine	5. 3 -3
5. 3. 3	Powered Flight Guidance Using Cross Product Steering	5. 3 -8
5. 3. 3. 1	Introduction	5. 3 -8
5. 3. 3. 2	Powered Flight Guidance Computation Sequencing	5. 3 -11
5. 3. 3. 3	Pre-Thrust Computations	5. 3 -19
5. 3. 3. 4	Cross Product Steering Routine	5. 3 -33
5. 3. 3. 5	Velocity-to-be-Gained Routine	5. 3 -38
5. 3. 3. 6	ΔV Monitor Subroutine	5. 3 -44
5. 3. 4	Lunar Landing Guidance	5. 3 -47
5. 3. 4. 1	Introduction	5. 3 -47
5. 3. 4. 2	Lunar Landing Coordinate Systems	5. 3 -54
5. 3. 4. 3	State-Vector Update Routine	5. 3 -56
5. 3. 4. 4	LR Data Read Routine	5. 3 -68

<u>SECTION NO.</u>	<u>TITLE</u>	<u>PAGE</u>
5.3.4.5	LR Data Reasonableness Test Routine	5.3-81
5.3.4.6	Guidance-and-Control Routine	5.3-85
5.3.4.6.1	Nomenclature of the Guidance-and-Control Routine	5.3-104
5.3.4.7	Throttle-Command Routine	5.3-113
5.3.4.8	Landing Maneuver Display Computations	5.3-116
5.3.4.9	Landing Confirmation Routine and Post-Landing Sequences	5.3-121
5.3.5	Powered Ascent Guidance	5.3-125
5.3.5.1	Guidance Objective	5.3-125
5.3.5.2	Ascent Guidance Coordinate Systems	5.3-127
5.3.5.3	Input-Output	5.3-128
5.3.5.4	Powered Ascent Pre-Ignition Phase	5.3-132
5.3.5.5	Powered Ascent Vertical Rise Phase	5.3-137
5.3.5.6	Powered Ascent Guidance Phase	5.3-138
5.3.5.7	RCS Ascent Injection	5.3-139
5.3.5.8	Abort Guidance	5.3-140
5.3.5.9	Ascent Guidance Computations used by P12, P70 and P71	5.3-141

<u>SECTION NO.</u>	<u>TITLE</u>	<u>PAGE</u>
5.3.6	Thrust Monitor Program	5.3-155
5.3.7	FINDCDUW Routine	5.3-156
5.3.7.1	Introduction	5.3-156
5.3.7.2	Nomenclature	5.3-160
5.3.7.3	Detailed Flow Diagrams	5.3-162
5.3.8	MIDTOAVE Routine	5.3-174
<u>5.4</u>	<u>Targeting Routines</u>	
5.4.1	General Comments	5.4-1
5.4.2	Rendezvous Targeting	5.4-3
5.4.2.1	General	5.4-3
5.4.2.2	Pre-CSI Maneuver	5.4-6
5.4.2.3	Pre-CDH Maneuver	5.4-13
5.4.2.4	Pre-TPI Maneuver	5.4-22
5.4.2.5	Rendezvous Midcourse Maneuver	5.4-28
5.4.3	Abort Targeting	5.4-39
5.4.3.1	Aborts From Powered Landing	5.4-39

<u>SECTION NO.</u>	<u>TITLE</u>	<u>PAGE</u>
<u>5. 5</u>	<u>Basic Subroutines</u>	
5. 5. 1	General Comments	5. 5-1
5. 5. 1. 1	Solar System Subroutines	5. 5-1
5. 5. 1. 2	Conic Trajectory Subroutines	5. 5-2
5. 5. 2	Planetary Inertial Orientation Subroutine	5. 5-12
5. 5. 3	Latitude-Longitude Subroutine	5. 5-18
5. 5. 4	Lunar and Solar Ephemerides	5. 5-23
5. 5. 5	Kepler Subroutine	5. 5-25
5. 5. 6	Lambert Subroutine	5. 5-30
5. 5. 7	Time-Theta Subroutine	5. 5-34
5. 5. 8	Time-Radius Subroutine	5. 5-36
5. 5. 9	Apsides Subroutine	5. 5-39
5. 5. 10	Miscellaneous Subroutines	5. 5-41
5. 5. 11	Initial Velocity Subroutine	5. 5-50
5. 5. 13	LOSSEM Subroutine	5. 5-54
5. 5. 14	Pericenter-Apocenter (Periapo) Subroutine	5. 5-56
<u>5. 6</u>	<u>General Service Routines</u>	
5. 6. 1	General Comments	5. 6-1
5. 6. 2	IMU Alignment Modes	5. 6-2
5. 6. 2. 1	Orbital Alignment	5. 6-2
5. 6. 2. 2	Lunar Surface Alignment	5. 6-10
5. 6. 3	IMU Routines	5. 6-28
5. 6. 3. 1	AOT and COAS Transformations	5. 6-28
5. 6. 3. 2	IMU Transformations	5. 6-43
5. 6. 3. 3	Gravity Vector Determination Routine	5. 6-49
5. 6. 3. 4	REFSMMAT Transformations	5. 6-53
5. 6. 4	Star Selection Routine	5. 6-57

<u>SECTION NO.</u>	<u>TITLE</u>	<u>PAG</u>
5. 6. 5	Ground Track Routine	5. 6- 59
5. 6. 6	S-Band Antenna Routine	5. 6- 60
5. 6. 7	Additional Rendezvous Displays	5. 6- 64
5. 6. 7. 1	Range, Range Rate, Theta Display	5. 6- 64
5. 6. 7. 2	Final Attitude Display	5. 6- 65
5. 6. 7. 3	Out-of-Plane Rendezvous Display	5. 6- 65
5. 6. 8	AGS Initialization Routine	5. 6- 68
5. 6. 9	LGC Initialization	5. 6- 69
5. 6. 11	LGC Idling Program	5. 6- 72
5. 6. 12	FDAI-IMU Transformations	5. 6- 76
5. 6. 13	IMU Compensation	5. 6- 79
5. 6. 14	RR/LR Self Test Routine	5. 6- 81
5. 6. 15	RR Angle Transformations	5. 6- 84
5. 6. 16	Target ΔV Program	5. 6- 88
5. 6. 17	Orbit Parameter Display Routine	5. 6- 90
5. 6. 18	Powered Flight RR Designate Routine	5. 6- 98
5. 6. 19	RMS Position and Velocity Error Display	5. 6- 106
5. 6. 20	LR Spurious Test Routine	5. 6- 108
5. 6. 21	RR LOS Azimuth and Elevation Display	5. 6- 108
<u>5. 7</u>	<u>Erasable Memory Parameter List</u>	<u>5. 7-1</u>
<u>5. 8</u>	<u>Fixed Memory Constants</u>	<u>5. 8-1</u>
5. 8. 1	Fixed Constants	5. 8-1
5. 8. 2	References for Fixed Constants	5. 8-10
5. 8. 3	Comments on Fixed Constants	5. 8-13

5. 1. 2. 2 Sections 4 and 5 Cross Reference

<u>PROGRAM NUMBER</u>	<u>TITLE</u>	<u>PRINCIPAL SECTION 5 SUBSECTION NO.</u>	<u>PAGE</u>
P-00	LGC Idling Program	5. 6. 11	5. 6-72
P-12	Powered Ascent Guidance	5. 3. 5	5. 3-125
P-20	Rendezvous Navigation Program	5. 2. 4	5. 2-35
P-21	Ground Track Determination Program	5. 6. 5	5. 6-59
P-22	RR Lunar Surface Navigation Program	5. 2. 5	5. 2-81
P-25	Preferred Tracking Attitude Program	5. 2. 4. 4	5. 2-77
P-30	External ΔV Maneuver Guidance	5. 3. 3. 3. 1	5. 3-19
P-32	Coelliptic Sequence Initiation (CSI) Program	5. 4. 2. 2	5. 4-6
P-33	Constant Differential Altitude (CDH) Program	5. 4. 2. 3	5. 4-13
P-34	Transfer Phase Initiation (TPI) Program	5. 4. 2. 4	5. 4-22
P-35	Transfer Phase Midcourse (TPM) Program	5. 4. 2. 5	5. 4-28

<u>PROGRAM NUMBER</u>	<u>TITLE</u>	<u>PRINCIPAL SECTION 5 SUBSECTION NO.</u>	<u>PAGE</u>
P-40	DPS Thrust Program	5.3.3	5.3-8
P-41	RCS Thrust Program	5.3.3	5.3-8
P-42	APS Thrust Program	5.3.3	5.3-8
P-47	Thrust Monitor Program	5.3.6	5.3-155
P-51	IMU Orientation Determination	5.6.2.1.1	5.6-2
P-52	IMU Realignment Program	5.6.2.1.2	5.6-3
P-57	Lunar Surface Alignment Program	5.6.2.2	5.6-10
P-63	Braking Phase Guidance	} 5.3.4.6	5.3-85
P-64	Approach Phase Guidance		
P-66	Terminal Descent Phase Guidance		
P-68	Landing Confirmation Program	5.3.4.9	5.3-121
P-70	DPS Abort Guidance	} 5.4.3.1	5.4-39
P-71	APS Abort Guidance		
P-72	CSM CSI Targeting	5.4.2.2	5.4-6
P-73	CSM CDH Targeting	5.4.2.3	5.4-13
P-74	CSM TPI Targeting	5.4.2.4	5.4-22
P-75	CSM TPM Targeting	5.4.2.5	5.4-28
P-76	Target ΔV Program	5.6.16	5.6-88

5.1-10

<u>ROUTINES</u>	<u>TITLE</u>	<u>PRINCIPAL SECTION 5 SUBSECTION NO.</u>	<u>PAGE</u>
R-04	RR/LR Self Test Routine	5.6.14	5.6-81
R-05	S-Band Antenna Routine	5.6.6	5.6-60
R-10	Landing Analog Display Routine	5.3.4.8	5.3-116
R-12	Descent State Vector Update Routine	5.3.4.4	5.3-68
R-13	Landing Auto Modes Monitor Routine	5.3.4.6	5.3-88a
R-14	LR Position Command Routine	5.3.4.4	5.3-68
R-21	RR Designate Routine	5.2.4.1.1	5.2-39
R-22	RR Data Read Routine	5.2.4.2.1	5.2-53
R-23	RR Manual Acquisition Routine	5.2.4.1.2	5.2-47
R-24	RR Search Routine	5.2.4.1.3	5.2-48
R-25	RR Monitor Routine	5.2.4.3	5.2-75
R-26	Lunar Surface RR Predesignate Routine	5.2.5.3	5.2-86
R-29	Powered Flight RR Designate Routine	5.6.18	5.6-98
R-30	Orbit Parameter Display Routine	5.6.17	5.6-90
R-31	Rendezvous Parameter Display Routine	5.6.7.1	5.6-64
R-36	Out-of-Plane Rendezvous Display Routine	5.6.7.3	5.6-65
R-40	DPS/APS Thrust Fail Routine	5.3.3.6	5.3-44
R-41	MIDTOAVE Routine	5.3.8	5.3-174
R-47	AGS Initialization Routine	5.6.8	5.6-68
R-50	Coarse Alignment Routine	5.6.2.1.2	5.6-3
R-51	Inflight Fine Alignment	5.6.2.1.2	5.6-3
R-52	Auto Optics Positioning Routine	5.6.2.1.2	5.6-3

<u>ROUTINES</u>	<u>TITLE</u>	<u>PRINCIPAL SECTION 5 SUBSECTION NO.</u>	<u>PAGE</u>
R-53	Inflight Sighting Mark Routine	5.6.2.1 5.6.3.1.1	5.6-2 5.6-28
R-54	Sighting Data Display Routine	5.6.2.1.1	5.6-2
R-55	Gyro Torquing Routine	5.6.2.1	5.6-2
R-59	Lunar Surface Sighting Mark Routine	5.6.2.2 5.6.3.1.2	5.6-10 5.6-34
R-61	Preferred Tracking Attitude Routine	5.2.4.4	5.2-77
R-63	Rendezvous Final Attitude Routine	5.6.7.2	5.6-65
R-65	Fine Preferred Tracking Attitude Routine	5.2.4.4	5.2-77
R-77	Spurious Test Routine	5.6.20	5.6-108

5.1.3 GENERAL PROGRAM UTILIZATION

The following outline is a brief summary of the major LGC programs that would be used in the principal phases of a nominal lunar landing mission. This outline reflects the LGC capability for nominal and abort cases of such a mission.

I Lunar Orbit Phase Prior to LM Descent Orbit Injection

A) Nominal

Voice Link LGC Initialization

P-27 LGC Update Program (LGC Initialization if required followed by lunar landing targeting and timing parameters)

R-47 AGS Initialization Routine

P - 20 Rendezvous Navigation Program
(Tracking mode only for Pre-DOI RR Check out)

B) Aborts to Return to Earth Trajectory (SPS Backup)

P - 47 Thrust Monitor Program
(Manual Re-docking)

P - 27 LGC Update Program (TEI targeting, state vector updating and cislunar MCC targeting as required)

P-30 External ΔV Maneuver Program
(TEI and cislunar MCC for SPS backup with RTCC targeting)

P-40 DPS Thrust Program

I Lunar Orbit Phase Prior to LM Descent Orbit
Injection (cont)

C) Service Programs for Nominal and Abort Cases

P - 52 IMU Realignment Program
R - 05 S-Band Antenna Routine
P - 21 Ground Track Program
R - 30 Orbit Parameter Display Routine
P - 00 LGC Idling Program

II Descent Orbit Injection (DOI) and LM Descent Coast
Phase

A) Nominal

P - 30 External ΔV Maneuver Program
 (DOI Maneuver Pre-Thrust)
P - 40 DPS Thrust Program (DOI)
P - 25 Preferred Tracking Attitude Program
 (Descent Monitoring)
R - 31 Rendezvous Parameter Display Routine
R - 30 Orbit Parameter Display Routine
P - 63 Landing Maneuver Braking Maneuver
 Program (Pre-Ignition Mode)
R - 47 AGS Initialization Routine

II Descent Orbit Injection (DOI) and LM Descent Coast
Phase (cont)

B) Aborts to Rendezvous Condition

1. LM Active Vehicle

P - 20 Rendezvous Navigation Program
 (LGC navigation)

P - 32 CSI Pre-Thrust Program

P - 33 CDH Pre-Thrust Program

P - 34 TPI Pre-Thrust Program

P - 35 TPM Pre-Thrust Program

P - 40 }
P - 41 } DPS, RCS or APS Thrust Programs
P - 42 }

(or)

P - 25 Preferred Tracking Attitude Program
 (CMC navigation)

(or)

P - 27 LGC Update Program (State vector
 update and rendezvous maneuver targeting).

P - 30 External ΔV Pre-Thrust Program
 (CMC or RTCC targeted)

P - 47 Thrust Monitor Program (Manual
 Terminal Rendezvous Maneuvers)

II

Descent Orbit Injection (DOI) and LM Descent Coast

Phase (cont)

B) Aborts to Rendezvous Condition

2. CSM Active Retrieval

P - 20 Rendezvous Navigation Program
(LGC navigation)

P - 76 Target ΔV Program

P - 72 CSM CSI Pre-Thrust Program

P - 73 CSM CDH Pre-Thrust Program

P - 74 CSM TPI Pre-Thrust Program

P - 75 CSM TPM Pre-Thrust Program

P - 25 Preferred Tracking Attitude Program
(CMC navigation)

C) Service Programs for Nominal and Abort Cases

P - 52 IMU Realignment Program

R - 30 Orbit Parameter Display Routine

R - 31 Rendezvous Parameter Display Routine

R - 36 Rendezvous Out-of-Plane Display Routine

R - 63 Rendezvous Final Attitude Routine

P - 21 Ground Track Program

R - 05 S-Band Antenna Routine

P - 00 LGC Idling Program

III Powered Landing Maneuver and Post Landing Phase

A) Nominal

- P-63 Landing Braking Phase Guidance
- P-64 Landing Approach Phase Guidance
- P-66 Terminal Descent
Phase Guidance

- P-68 Landing Confirmation Program
- P-57 Lunar Surface Alignment Program
- R-47 AGS Initialization Routine

B) Abort to Orbit

1. Aborts During Powered Landing Maneuver

- P-70 DPS Abort Guidance Program
- P-71 APS Abort Guidance Program

2. Aborts from the Lunar Surface (Anytime
Launch Case)

- P-27 LGC Update Program (CSM State vector update)
- P-57 Lunar Surface Alignment
(Fast Alignment Mode)
- P-12 Powered Ascent Guidance Program

5.1-17

IV

Lunar Pre-Launch Phase (Final 3 CSM Orbits before
LM Launch)

A) Nominal

- P - 27 LGC Update Program (state vector updates
and launch time)
- (or)
- P - 22 RR Lunar Surface Navigation Program
- R - 47 AGS Initialization Routine

B) Service Programs

- P - 57 Lunar Surface Alignment Program
- R - 05 S-Band Antenna Routine
- P - 00 LGC Idling Program

V LM Powered Ascent Phase

A) Nominal

 P - 12 Powered Ascent Guidance

 R - 29 Powered Flight RR Designate Routine

B) Aborts

 None

VI Rendezvous Phase

A) Nominal

 P - 20 Rendezvous Navigation Program

 P - 32 CSI Pre-Thrust Program

 P - 33 CDH Pre-Thrust Program

 P - 34 TPI Pre-Thrust Program

 P - 35 TPM Pre-Thrust Program

 P - 41 }
 P - 42 } RCS or APS Thrust Programs

 R - 47 AGS Initialization Routine

 P - 47 Thrust Monitor Program (Manual Terminal
 Rendezvous Maneuvers)

Rendezvous Phase (cont)B) Aborts to Rendezvous1. LM Active (RR Failure)

P - 27 LGC Update Program (state vector updates
and rendezvous targeting)

P - 25 Preferred Tracking Attitude Program
(CSM navigation)

P - 30 External ΔV Pre-Thrust Program
(CSM or RTCC targeted)

P - 41 }
P - 42 } RCS or APS Thrust Programs

P - 47 Thrust Monitor Program (Manual
Terminal Rendezvous Maneuvers)

2. CSM Active Retrieval

P - 20 Rendezvous Navigation Program
(LGC navigation)

P - 76 Target ΔV Program

P - 72 CSM CSI Pre-Thrust Program

P - 73 CSM CDH Pre-Thrust Program

P - 74 CSM TPI Pre-Thrust Program

P - 75 CSM TPM Pre-Thrust Program

P - 25 Preferred Tracking Attitude Program
(CMC navigation)

VI

Rendezvous Phase (cont)

C) Service Programs for Nominal and Abort Cases

- P - 52 IMU Realignment Program
- R - 05 S-Band Antenna Routine
- R - 30 Orbit Parameter Display Routine
- R - 31 Rendezvous Parameter Display
Routine
- R - 36 Rendezvous Out-of-Plane Display Routine
- R - 63 Rendezvous Final Attitude Routine
- P - 21 Ground Track Program
- P - 00 LGC Idling Program

5.1.4 COORDINATE SYSTEMS

There are six major coordinate systems used in the navigation and guidance programs. These six coordinate systems are defined individually in the following descriptions, and referenced to control specifications of Section 5.8.2 where applicable. Any other coordinate system used in any particular LGC program is defined in the individual section describing that program.

5.1.4.1 Basic Reference Coordinate System

The Basic Reference Coordinate System is an orthogonal inertial coordinate system whose origin is located at either the moon or the earth center of mass. The orientation of this coordinate system is defined by the line of intersection of the mean earth equatorial plane and the mean orbit of the earth (the ecliptic) at the beginning of the Besselian year which starts January 0.767239, 1970 et. The X-axis (\underline{u}_{XI}) is along this intersection with the positive sense in the direction of the ascending node of the ecliptic on the equator (the equinox), the Z-axis (\underline{u}_{ZI}) is along the mean earth north pole, and the Y-axis (\underline{u}_{YI}) completes the right-handed triad. In the lunar landing mission this coordinate system is located at the moon center of mass. During earth-orbital missions in which the LM is active near the earth, the Basic Reference Coordinate System will be earth-centered. All navigation stars and lunar-solar ephemerides are referenced to this coordinate system. All vehicle state vectors are referenced to this system during coasting or free fall phases of the mission.

The Basic Reference Coordinate System is presented in Ref. 1 of Section 5.8.2 as Standard Coordinate System 4, Geocentric Inertial.

5.1-22

Revised

LUMINARY

Added

GSOP # 567

PCR # 707.2 Rev. 7 Date 11-69

5.1.4.2 IMU Stable Member or Platform Coordinate System

The orthogonal inertial coordinate system defined by the PGNCS inertial measurement unit (IMU) is dependent upon the current IMU alignment. There are many possible alignments during a mission, but the primary IMU alignment orientations described in Section 5.6.3.4 are summarized below and designated by the subscript SM:

1. Preferred Alignment

$$\begin{aligned} \underline{u}_{XSM} &= \underline{u}_{TD} \\ \underline{u}_{YSM} &= \begin{cases} \text{UNIT}(\underline{u}_{XSM} \times \underline{r}) & \text{If } \underline{r} \text{ not parallel to } \underline{u}_{TD} \\ \text{UNIT}(\underline{u}_{XSM} \times \underline{v}) & \text{If } \underline{r} \text{ parallel to } \underline{u}_{TD} \end{cases} \\ \underline{u}_{ZSM} &= \underline{u}_{XSM} \times \underline{u}_{YSM} \end{aligned} \tag{5.1.1}$$

where:

$$\begin{aligned} \left. \begin{array}{l} \underline{u}_{XSM} \\ \underline{u}_{YSM} \\ \underline{u}_{ZSM} \end{array} \right\} & \text{IMU stable member coordinate unit} \\ & \text{vectors referenced to the Basic} \\ & \text{Reference Coordinate System} \\ \underline{u}_{TD} &= \text{unit vector in desired thrust direction} \\ & \text{at ignition} \\ \underline{r} &= \text{position vector at ignition} \\ \underline{v} &= \text{velocity vector at ignition} \end{aligned}$$

2. Nominal Alignment (Local Vertical)

$$\begin{aligned} \underline{u}_{XSM} &= \text{UNIT}(\underline{r}) \text{ at } t_{\text{align}} \\ \underline{u}_{YSM} &= \text{UNIT}(\underline{v} \times \underline{r}) \\ \underline{u}_{ZSM} &= \underline{u}_{XSM} \times \underline{u}_{YSM} \end{aligned} \tag{5.1.2}$$

where \underline{r} and \underline{v} represent the vehicle state vector at the alignment time, t_{align} .

3. Lunar Landing Alignment

$$\begin{aligned} \underline{u}_{XSM} &= \text{UNIT}(\underline{r}_{LS}) \text{ at } t_L \\ \underline{u}_{ZSM} &= \text{UNIT} \left[(\underline{r}_C \times \underline{v}_C) \times \underline{u}_{XSM} \right] \quad (5.1.3) \\ \underline{u}_{YSM} &= \underline{u}_{ZSM} \times \underline{u}_{XSM} \end{aligned}$$

where \underline{r}_{LS} is the lunar landing site vector at the predicted landing time, t_L , and \underline{r}_C and \underline{v}_C are the CSM position and velocity vectors, as maintained in the LGC.

4. Lunar Launch Alignment

The same as that defined in Eq. (5.1.3) except that \underline{r}_{LS} is the landing or launch site at the predicted launch time t_L .

The origin of the IMU Stable Member Coordinate System is nominally the center of the IMU stable member. In the following programs, however, the origin of the IMU or platform coordinate system is located at the moon center of mass:

Lunar landing programs P-63 to P-66, and P-68
Powered ascent guidance program P-12
Abort programs P-70 and P-71

5.1.4.3 Vehicle or Body Coordinate System

The Vehicle or Body Coordinate System is the orthogonal coordinate system used for the LM structural body. The origin of this coordinate system is 200 inches below the LM ascent stage base. The X-axis (\underline{u}_{XB}) lies along the longitudinal axis (centerline of the transfer tunnel) of the LM, positive in the nominal DPS-APS thrust direction. The Z-axis (\underline{u}_{ZB}) is parallel to the centerline of the exit hatch and directed forward from the design eye. The Y-axis (\underline{u}_{YB}) completes the right-handed triad. This coordinate system is defined in Ref. 1 of Section 5.8.2 as Standard Coordinate System 8d, LEM Structural Body Axis.

5.1-24

Revised LUMINARY
 Added GSOP # 567 PCR # 285 Rev. 7 Date 11 - 69

5.1.4.4 Earth-Fixed Coordinate System

The Earth-Fixed Coordinate system is an orthogonal rotating coordinate system whose origin is at the center of mass of the earth. This coordinate system is shown in Ref. 1 of Sec. 5.8.2 as the Standard Coordinate System 1, Geographic Polar. The Z-axis of this coordinate system is defined to be along the earth's true rotational or polar axis. The X-axis is defined to be along the intersection of the prime (Greenwich) meridian and the equatorial plane of the earth, and the Y-axis is in the equatorial plane and completes the right-handed triad.

5.1.4.5 Moon-Fixed Coordinate System

The Moon-Fixed Coordinate System is an orthogonal rotating coordinate system whose origin is at the center of mass of the moon. This coordinate system is shown in Ref. 1 of Sec. 5.8.2 as the Standard Coordinate System 2, Selenographic Polar. The Z-axis is defined to be along the true polar or rotation axis of the moon, the X-axis is through the mean center of the apparent disk or along the intersection of the meridian of 0° longitude and the equatorial plane of the moon, and the Y-axis is in the equatorial plane and completes the right-handed triad.

5.1.4.6 Navigation Base Coordinate System

The Navigation Base Coordinate System is an orthogonal coordinate system whose origin and axis orientation are defined by three mounting points between the PGNCS navigation base and the LM vehicle structure. These mounting point locations are defined in Section G-G of the GAEC-MIT Interface Control Document LID280-10004 (Ref. 13 of Sec. 5.8.2). The Y_{NB} axis is defined by the centers of the two upper mounting points along line F of Ref. 13, with the positive direction in the same general direction as the LM +Y vehicle axis. The X_{NB} axis is defined by a line through the center of the lower mounting point (point K of Ref. 13) and perpendicular to the Y_{NB} axis. The positive X_{NB} direction is in the same general sense as the LM +X vehicle axis. The $+Z_{NB}$ defined as $\underline{X}_{NB} \times \underline{Y}_{NB}$ to complete the right-handed triad. The Navigation Base Coordinate System is approximately parallel with the LM Vehicle Coordinate System.

5. 1. 5 GENERAL DEFINITIONS AND CONVENTIONS

In this section the definitions of and the conventions for using a selected number of parameters are given. Although virtually all of the information in this section appears elsewhere in this document, this section provides a summary of facts which are distributed among various other sections.

5. 1. 5. 1 Error Transition Matrix Maintenance

5. 1. 5. 1. 1 Definitions

The error transition matrix (W matrix) is defined in Section 5. 2. 2. 4 and is used in processing navigation measurement data. Control of the W matrix is maintained by means of the flag RENDWFLG (see Sections 5. 2. 4. 2. 2 and 5. 2. 5. 4). If RENDWFLG is equal to one, then the W matrix is valid for processing rendezvous navigation data; while this flag being equal to zero indicates that the W matrix is invalid.

5. 1. 5. 1. 2 W Matrix Control Flag

The W matrix control flag is maintained according to the following rules:

1. RENDWFLG is initially zero.
2. A CSM state vector update from the ground (RTCC) causes the flag to be zeroed.
3. A LM in-flight state vector update from the ground causes the flag to be zeroed. An update of the landing site vector when the LM is on the lunar surface does not cause the flag to be zeroed.

4. There exist special DSKY procedures by which the astronaut can zero the flag (verbs 67 and 93).
5. Selection of the lunar ascent program (P12) (or P70, P71 aborts) causes RENDWFLG to be zeroed.
6. Overflow of the W matrix during extrapolation causes RENDWFLG to be zeroed.
7. Initialization of the W matrix for rendezvous or lunar surface navigation causes RENDWFLG to be set to one.

With regard to the last item 7 above, there exist in erasable memory two sets of initialization parameters for the W matrix: one for rendezvous navigation and one for lunar surface navigation. Each of these sets contains two elements, a position element and a velocity element. In addition, the set for rendezvous contains initialization parameters for shaft and trunnion. At the time each set of navigation data is processed, RENDWFLG is tested. If the flag is found to be zero, then the W matrix is initialized consistent with the appropriate erasable parameters, and the flag is set to one. See sections 5.2.4.2.2 and 5.2.5.4 for more complete details of this initialization procedure.

5. 1-27

Revised
 Added

LUMINARY
 GSOP # 567

PCN # 561*
 PCR # 732

Rev. 7 Date 11-69

5.1.5.1.3 W Matrix Extrapolation

Extrapolation of the W matrix is described in Section 5.2.2.4. Required in this extrapolation is the specification of the appropriate vehicle's state vector with which the W matrix is extrapolated. This extrapolation occurs during programs P-00, P-20, and P-22; and at the conclusion of programs P-40, P-41, P-42, and P-47. The conventions under which the extrapolation occurs during each of these programs are as follows:

- P-00 : If RENDWFLG is equal to one, the W matrix is extrapolated with the LM state vector if the LM is in flight and with the CSM state vector if the LM is on the lunar surface. The W matrix is not extrapolated if the flag is equal to zero. (See Section 5.6.11).
- P-20 : The W matrix is extrapolated with the state vector that is being updated if RENDWFLG is equal to one, and not extrapolated if RENDWFLG is equal to zero. (See Section 5.2.4.2.2.)
- P-22 : The W matrix is extrapolated with the CSM state vector if RENDWFLG is equal to 1, and not extrapolated if RENDWFLG is equal to zero. (See Section 5.2.5.4.)
- P-40 }
P-41 } : The result of the maneuver will be a final
P-42 } state vector at the end-of-maneuver time
P-47 } t_F . The LM state vector that existed before
the maneuver program will still exist; and
cotemporal with it, there will also be the
CSM state vector and the W matrix. The
following steps are performed before the
program is terminated:

1. If the W matrix control flag is equal to one, the old LM state vector and the W matrix are extrapolated to time t_F .
2. The CSM state vector is extrapolated to time t_F .
3. The LM state vector is initialized to the end-of-maneuver state vector.



If a computation overflow occurs during any of the above W matrix extrapolations, a program alarm will result, the W Matrix control flag will be zeroed, and the extrapolation of the state vector will continue without the W matrix.

5. 1. 5. 2 Altitude Parameter Convention

In the following programs and routines the display parameter of the vehicle altitude or trajectory pericenter or apocenter altitude is measured with respect to the earth launch pad radius magnitude, r_{LP} , when in earth orbit or the lunar landing site radius magnitude, r_{LS} , when in lunar orbit. The earth launch pad radius parameter, r_{LP} , is stored in fixed memory, and the lunar landing site radius, r_{LS} , is the magnitude of the landing site vector, \underline{r}_{LS} , stored in erasable memory.

P-21	Ground Track Determination	Sec. 5. 6. 5
P-30	External ΔV Maneuver Guidance	Sec. 5. 3. 3. 3. 1
P-34, 74	TPI Pre-Thrust Program	Sec. 5. 4. 2. 4
R-30	Orbit Parameter Display Routine	Sec. 5. 6. 17

The above launch pad radii are also used in the following programs to compute pericenter altitudes prior to determining whether the pericenters are safe.

P-32, 72	Pre-CSI Program	Sec. 5. 4. 2. 2
----------	-----------------	-----------------

The following programs are operated in lunar orbit only, and the displayed vehicle altitude or trajectory pericenter or apocenter altitude is referenced to the lunar landing site radius magnitude, r_{LS} .

P-12	Powered Ascent Maneuver Guidance	Sec. 5. 3. 5
P-63	Braking Phase Guidance	Sec. 5. 3. 4. 6

P-64	Approach Phase Guidance	Sec. 5. 3. 4. 6
P-66	Terminal Descent Phase Guidance	Sec. 5. 3. 4. 6
P-70	DPS Abort Program	Sec. 5. 4. 3. 1
P-71	APS Abort Program	Sec. 5. 4. 3. 1

In the following programs a temporary r_{LS} can be displayed on the DSKY with respect to the original r_{LS} in erasable memory. The temporary altitude value is considered to be zero unless the astronaut keys in a change. In P52 the temporary value is used by the program but r_{LS} retains its original value in memory. In P57, when using alignment technique number 2, the temporary value replaces the old r_{LS} in memory.

P-52	IMU Realignment Program	Sec. 5. 6. 2. 1
P-57	Lunar Surface Alignment Program	Sec. 5. 6. 2. 2

Program P68 also displays altitude, with a base of the original r_{LS} site (or if there has been a restart, with a base of the final state vector).

5.1.5.3 Lunar Landing Site Definition

A lunar landing site vector, \underline{r}_{LS} , in the Moon Fixed Coordinate System (Section 5.1.4.5) is stored in an LGC erasable memory location at all times. This landing site vector is normally stored in erasable memory prior to earth launch, and is either modified or verified prior to CSM-LM separation in lunar orbit by the CMC Orbital Navigation Program P-22 and astronaut LGC initialization procedures, or by the RTCC uplink program P-27. After the LGC initialization in lunar orbit prior to the descent orbit injection maneuver, the landing site vector \underline{r}_{LS} is never changed until the landing confirmation program P68 replaces it with a value based on the actual spacecraft position vector. The landing site vector, modified via N69 or during P64 by astronaut redesignations, occupies separate erasables, is in platform coordinates, and is initialized from \underline{r}_{LS} at the beginning of P63. After lunar landing \underline{r}_{LS} can be changed by P57, (the Lunar Surface Alignment Program), or by Program P-27. The lunar landing and lunar launch LM IMU alignments are both referenced to the landing site vector stored in the LGC at the time of alignment.

5.2 COASTING FLIGHT NAVIGATION

5.2.1 GENERAL COMMENTS

The LGC Coasting Flight Navigation Routines which are presented in Sections 5.2.2 through 5.2.5 are used during non-thrusting phases of the Apollo mission. The basic objective of the navigation routines is to maintain estimates of the position and velocity vectors of both the CSM and the LM. Let \underline{r} and \underline{v} be the estimates of a vehicle's position and velocity vectors, respectively. Then, the six-dimensional state vector, \underline{x} , of the spacecraft is defined by

$$\underline{x} = \begin{pmatrix} \underline{r} \\ \underline{v} \end{pmatrix}$$

Coasting Flight Navigation is accomplished by extrapolating the state vector, \underline{x} , by means of the Coasting Integration Routine (Section 5.2.2), and updating or modifying this estimated state using Rendezvous Radar (RR) tracking data by the recursive method of navigation (Sections 5.2.3 - 5.2.5).

The Coasting Integration Routine (Section 5.2.2) is used by other navigation and targeting routines to extrapolate the following:

- 1) Present estimated LM state vector
- 2) Present estimated CSM state vector
- 3) An arbitrary specified state vector, such as the predicted result of a maneuver

State vector extrapolation is accomplished by means of Encke's method of differential accelerations. The motion of a spacecraft is dominated by the conic orbit which would result if the spacecraft were in a central force field. In Encke's method the differential equations for the deviations from conic motion are integrated numerically. This technique is in contrast to a numerical integration of the differential equations for the total motion, and it provides a more accurate orbit extrapolation. The numerical integration is accomplished by means of Nystrom's method which gives fourth-order accuracy while requiring only three computations of the derivatives per time step. The usual fourth-order Runge-Kutta integration methods require four derivative computations per time step.

Regardless of the accuracy of the state vector extrapolation, errors in the initial conditions will propagate and soon grow to intolerable size. Thus, it is necessary periodically to obtain additional data in the form of either new state vector estimates or modifications to the current state vector estimates. These state vector modifications are computed from navigation data obtained by means of navigation measurements.

The LM PGNCs uses RR tracking data to compute state vector changes. Navigation measurement data are used to update state vector estimates during rendezvous and lunar surface navigation procedures. These two navigation procedures will be used normally during all LM-CSM lunar-orbit rendezvous phases and the LM lunar surface prelaunch phase, respectively, in the lunar landing mission. However, in order to provide for alternate mission capability, the rendezvous navigation procedure can be used near the moon or the earth.

Although the state vector of the LM is six-dimensional, it is not necessary that the quantities estimated during a particular navigation procedure be the position and velocity vectors of the LM. A variety of "estimated state vectors", not necessarily of six-dimensions, are used.

In order to achieve desired rendezvous objectives, it is necessary to expand the rendezvous navigation procedure to nine dimensions, and to include in the estimation the constant RR angle biases. The estimated state vector that is used in rendezvous navigation is given by

$$\underline{x} = \begin{pmatrix} \underline{r} \\ \underline{v} \\ \underline{\text{bias}} \end{pmatrix}$$

where \underline{r} and \underline{v} are the estimated position and velocity vectors of either the LM or the CSM, and $\underline{\text{bias}}$ is a vector whose components are the estimates of the RR angle biases. Normally the LM state vector is updated, but the astronaut can select the CSM update mode. The selection of the vehicle update mode is based primarily upon

which vehicle's state vector is most accurately known initially, and which vehicle is controlling the rendezvous maneuvers.

In order to estimate the RR angle biases, it is necessary to restrict the LM attitude during RR tracking. This attitude restriction involves controlling the LM +Z-axis to be within 30° of the tracking line-of-sight and is described in Section 5.2.4.1. During normal RR tracking and update of the navigation equations with RR data the LM +Z-axis is actually directed along the line-of-sight to the CSM on a continuous basis as described in Sections 5.2.4.2.1 and 5.2.4.4.

During the LM lunar surface prelaunch phase of the lunar landing mission RR tracking is used for navigation. In this mode, however, the previously mentioned LM attitude restriction obviously cannot be met, and only the RR range and range rate data are used. Also, since it is assumed that the landing site is well known, the estimated state vector that is used in lunar surface navigation is the standard six-dimensional CSM state vector.

Navigation data is incorporated into the state vector estimates by means of the Measurement Incorporation Routine (Section 5.2.3) which has both six- and nine-dimensional modes. The Measurement Incorporation Routine is a subroutine of the following navigation routines:

- 1) Rendezvous Navigation Routine (Section 5.2.4.2)
- 2) Lunar Surface Navigation Routine (Section 5.2.5.4)

Simplified functional diagrams of the navigation programs which use these routines are given in Figs. 2.1-1 and 2.1-2, respectively.

In rendezvous navigation, estimated LM and CSM position and velocity vectors are obtained at required times by means of the

5.2-5

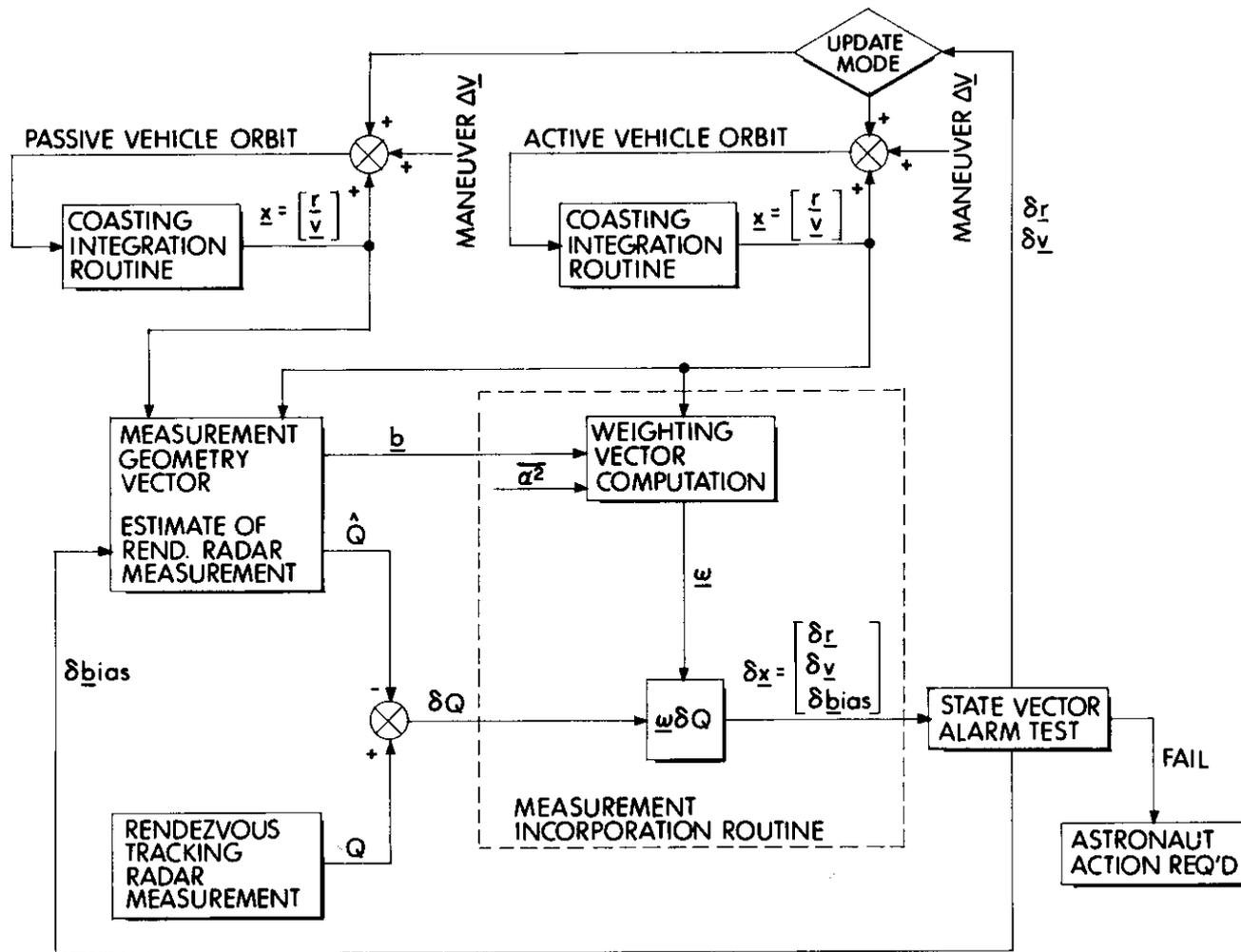


Figure 2.1-1 Simplified LGC Rendezvous Navigation Functional Diagram

5.2-6

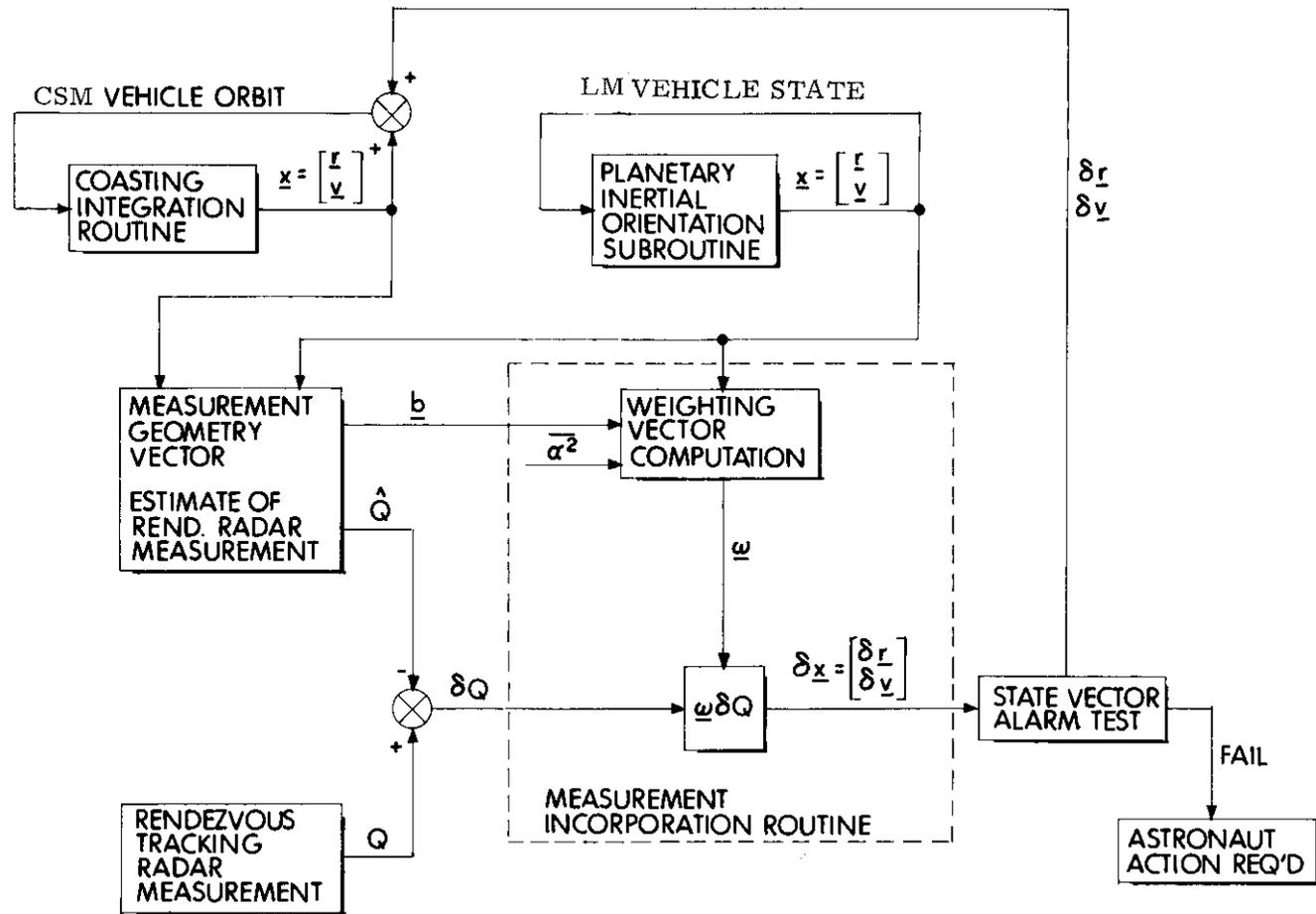


Figure 2.1-2 Simplified Lunar Surface Navigation Functional Diagram

Coasting Integration Routine (Section 5.2.2). The Measurement Incorporation Routine (Section 5.2.3) is used to incorporate the measurement data into the state vector estimates. In lunar surface navigation, the same process is performed except that the estimated LM state vector is obtained by means of the Planetary Inertial Orientation Subroutine (Section 5.5.2).

The navigation procedure, which is illustrated in simplified form in Figs. 2.1-1 and 2.1-2, involves computing an estimated tracking measurement, \hat{Q} , based on the current state vector estimates. This estimated measurement is then compared with the actual tracking measurement Q (RR tracking data in the LGC) to form a measured deviation δQ . A statistical weighting vector, $\underline{\omega}$, is computed from statistical knowledge of state vector uncertainties and tracking performance, $\underline{\alpha}^2$ plus a geometry vector, \underline{b} , determined by the type of measurement being made. The weighting vector, $\underline{\omega}$, is defined such that a statistically optimum linear estimate of the deviation, $\underline{\delta x}$, from the estimated state vector is obtained when the weighting vector is multiplied by the measured deviation δQ . The vectors $\underline{\omega}$, \underline{b} and $\underline{\delta x}$ are of six or nine dimensions depending upon the dimension of the state vector being estimated.

In an attempt to prevent unacceptably large incorrect state vector changes, certain validity tests have been included in the LGC navigation routines.

In the Rendezvous and Lunar Surface Navigation Routines (Sections 5.2.4.2 and 5.2.5.4) measurement data is processed periodically (once or twice per minute), and it is desirable that the CSM be tracked during the entire rendezvous phase up to the manual terminal maneuver. If the magnitudes of the changes in the estimated position and velocity vectors, δr and δv , respectively, are both less than preset tracking alarm levels, then the selected vehicle's state vector is automatically updated by the computed deviation, $\underline{\delta x}$, and no special display is presented, except that the tracking

measurement counter is incremented by one. If either δr or δv exceeds its alarm level, then the state vector is not updated, and the astronaut is alerted to this condition by a special display of δr and δv .

In this case the astronaut should place the RR under manual control and make the necessary radar operating and side lobe checks to verify main lobe lock-on and tracking conditions. After the tracking has been verified, and navigation data has again been acquired, the astronaut has the option of commanding a state vector update if the tracking alarm is again exceeded, or of repeating further RR checks before incorporating the measurement data. If the astronaut cannot verify the tracking, then he can terminate the program and try to achieve tracking conditions at a later time.

The tracking alarm criterion is incorporated in the navigation routines to alert the astronaut to the fact that the state vector update is larger than normally expected, and to prevent the estimated state vector from automatically being updated in such cases. The update occurs only by specific command of the astronaut. The tracking alarm level beyond which updating is suspended is primarily chosen to avoid false acquisition and tracking conditions. There is a low probability that the alarm level will be exceeded in the LGC if the estimated state vectors are essentially correct, since the RR Designation and Data Read Routines have partial internal checks for side-lobe acquisition before tracking data are incorporated in the navigation routines. The most probable condition for the state vector update alarm level being exceeded in the LGC is, therefore, initial acquisition and tracking in the case where a poor estimate of either the CSM or LM state vector exists. In this case the astronaut would have to command the initial state vector update, after which

the tracking alarm level would seldom be exceeded during the remainder of the navigation phase. It should be noted that this statement is true only if the estimated state vector of the active vehicle performing a powered rendezvous maneuver is updated by the Average-G Routine in the case of the LM being the active vehicle, or by a DSKY entry (P-76) of the maneuver ΔV if the CSM is the active vehicle (Section 5.6.16).

The displayed values of δr and δv which have not passed the tracking alarm test will depend upon the statistical parameters stored in the LGC and upon the following types of errors:

Type 1: Errors in the current state vector estimates

Type 2: Errors in alignment of the IMU

Type 3: Reasonable RR tracking performance errors

Type 4: A PGNCS or RR failure resulting in false acquisition

The existence of Type 1 errors is precisely the reason that the RR tracking is being done. It is the function of the navigation to decrease Type 1 errors in the presence of noise in the form of errors of Types 2 and 3. Since the RR tracking should not be performed unless the IMU is well aligned and the PGNCS and RR are functioning properly, it follows that the purpose of the state vector change validity check is to discover a Type 4 error. As previously mentioned, there is a low probability of this type of error occurring.

Based upon the last time that the state vector was updated and when the IMU last was realigned, very crude reasonable values for δr and δv can be generated by the astronaut. The LGC will provide no information to assist the astronaut in his estimates of reasonable values for δr and δv .

The parameters required to initialize the navigation routines (Sections 5.2.4.2 and 5.2.5.4) are the initial estimated CSM state vector, plus the initial estimated LM state vector for the Rendezvous Navigation Routine or the estimated landing site for the Lunar Surface Navigation Routine, initial state vector estimation error covariance matrices in the form of prestored diagonal error transition matrices (as defined in Section 5.2.2.4), and a priori measurement error variances. The basic input to the navigation routines is RR tracking data which is automatically acquired by the Data Read Routine. The primary results of the navigation routines are the estimated LM and CSM state vectors. The various guidance targeting modes outlined in Section 5.4 are based on the state vector estimates which result from these navigation routines.

5.2.2 COASTING INTEGRATION ROUTINE *

5.2.2.1 General Comments

During all coasting phase navigation procedures, an extrapolation of position and velocity by numerical integration of the equations of motion is required. The basic equation may be written in the form

$$\frac{d^2}{dt^2} \underline{r}(t) + \frac{\mu_P^{**}}{r^3} \underline{r}(t) = \underline{a}_d(t) \quad (2.2.1)$$

where μ_P is the gravitational constant of the primary body, and $\underline{a}_d(t)$ is the vector acceleration which prevents the motion of the vehicle (CSM or LM) from being precisely a conic with focus at the center of the primary body. The Coasting Integration Routine is a precision integration routine in which all significant perturbation effects are included. The form of the disturbing acceleration $\underline{a}_d(t)$ depends on the phase of the mission.

An approximate extrapolation of a vehicle state vector in which the disturbing acceleration, $\underline{a}_d(t)$ of Eq. (2.2.1), is set to zero may be accomplished by means of the Kepler subroutine (Section 5.5.5).

The LGC Coasting Integration Routine is restricted to earth or lunar orbit and is not to be used in cislunar-midcourse space. The routine does not contain the capability of computing the gravitational perturbations of the sun or the other body (moon or earth); and, therefore, cannot provide accurate midcourse integration.

5.2.2.2 Encke's Method

If \underline{a}_d is small compared with the central force field, direct integration of Eq. (2.2.1) is inefficient. Therefore, the extrapolation will be accomplished using the technique of differential accelerations attributed to Encke.

* This section does not reflect the entire LGC Coasting Integration Routine which is identical to that contained in the CMC. These two routines have been kept identical for the purpose of efficient flight program production. However, certain control constants are set in the LGC so that the midcourse perturbation calculations are locked out. Only those equations which can be executed are documented here.

** In the remainder of Section 5.2 the subscript P will denote primary body (earth or moon). When the body is known, then the subscripts E and M will be used for earth and moon, respectively. The vehicle will be indicated by the subscripts C for CSM and L for LM.

At time t_0 the position and velocity vectors, \underline{r}_0 and \underline{v}_0 , define an osculating conic orbit. The position and velocity vectors in the conic orbit, $\underline{r}_{\text{con}}(t)$ and $\underline{v}_{\text{con}}(t)$, respectively, will deviate by a small amount from the actual position and velocity vectors.

The conic position and velocity at time t are computed as shown in Section 5.5.5. Required in this calculation is the variable x which is the root of Kepler's equation. In order to minimize the number of iterations required in solving Kepler's equation, an estimate of the correct solution for x is obtained as follows:

Let

$$\tau = t - t_0 \quad (2.2.2)$$

During the previous computation cycle the values

$$\begin{aligned} \underline{r}' &= \underline{r}_{\text{con}} \left(\tau - \frac{\Delta t}{2} \right) \\ \underline{v}' &= \underline{v}_{\text{con}} \left(\tau - \frac{\Delta t}{2} \right) \end{aligned} \quad (2.2.3)$$

$$x' = x \left(\tau - \frac{\Delta t}{2} \right)$$

were computed. A trial value of $x(\tau)$ is obtained from

$$x_t = x' + s \left[1 - \gamma s (1 - 2 \gamma s) - \frac{1}{6} \left(\frac{1}{r'} - \alpha \right) s^2 \right] \quad (2.2.4)$$

where

$$\begin{aligned} s &= \frac{\sqrt{\mu_P}}{r'} \left(\frac{\Delta t}{2} \right) \\ \gamma &= \frac{\underline{r}' \cdot \underline{v}'}{2r' \sqrt{\mu_P}} \end{aligned} \quad (2.2.5)$$

$$\alpha = \frac{2}{r'} - \frac{v'^2}{\mu_P}$$

After specification of \underline{r}_0 , \underline{v}_0 , x_t and τ , the Kepler subroutine (Section 5.5.5) is used to compute $\underline{r}_{\text{con}}(\tau)$, $\underline{v}_{\text{con}}(\tau)$, and $x(\tau)$.

The true position and velocity vectors will deviate from the conic position and velocity since \underline{a}_d is not zero. Let

$$\underline{r}(t) = \underline{\delta}(t) + \underline{r}_{\text{con}}(t) \quad (2.2.6)$$

$$\underline{v}(t) = \underline{\nu}(t) + \underline{v}_{\text{con}}(t)$$

where $\underline{\delta}(t)$ and $\underline{\nu}(t)$ are the position and velocity deviations from the conic. The deviation vector $\underline{\delta}(t)$ satisfies the differential equation

$$\frac{d^2}{dt^2} \underline{\delta}(t) = - \frac{\mu_P}{r_{\text{con}}^3(t)} \left[f(q) \underline{r}(t) + \underline{\delta}(t) \right] + \underline{a}_d(t) \quad (2.2.7)$$

subject to the initial conditions

$$\underline{\delta}(t_0) = \underline{0}, \quad \underline{\nu}(t_0) = \underline{0} \quad (2.2.8)$$

where

$$q = \frac{(\underline{\delta} - 2\underline{r}) \cdot \underline{\delta}}{r^2} \quad (2.2.9)$$

$$f(q) = q \frac{3 + 3q + q^2}{1 + (1 + q)^{3/2}} \quad (2.2.10)$$

The first term on the right-hand side of Eq. (2.2.7) must remain small, i. e., of the same order as $\underline{a}_d(t)$, if the method is to be efficient. As the deviation vector $\underline{\delta}(t)$ grows in magnitude, this term will eventually increase in size. Therefore, in order to maintain the efficiency of the method, a new osculating conic orbit should be defined by the total position and velocity vectors $\underline{r}(t)$ and $\underline{v}(t)$. The process of selecting a new conic orbit from which to calculate deviations is called rectification. When rectification occurs, the initial conditions for the differential equation for $\underline{\delta}(t)$, as well as the variables τ and x , are again zero.

5.2.2.3 Disturbing Acceleration

The form of the disturbing acceleration $\underline{a}_d(t)$ that is used in Eq. (2.2.1) depends on the phase of the mission. In earth or lunar orbit, only the gravitational perturbations arising from the non-spherical shape of the primary body need be considered. Let \underline{a}_{dP} be the acceleration due to the non-spherical gravitational perturbations of the primary body. Then, for the earth

$$\underline{a}_{dE} = \frac{\mu_E}{r^2} \sum_{i=2}^4 J_{iE} \left(\frac{r_E}{r} \right)^i \left[P'_{i+1}(\cos \phi) \underline{u}_r - P'_i(\cos \phi) \underline{u}_z \right] \quad (2.2.11)$$

where

$$\begin{aligned}
P_2'(\cos \phi) &= 3 \cos \phi \\
P_3'(\cos \phi) &= \frac{1}{2} (15 \cos^2 \phi - 3) \\
P_4'(\cos \phi) &= \frac{1}{3} (7 \cos \phi P_3' - 4 P_2') \\
P_5'(\cos \phi) &= \frac{1}{4} (9 \cos \phi P_4' - 5 P_3')
\end{aligned}
\tag{2.2.12}$$

are the derivatives of Legendre polynomials,

$$\begin{aligned}
\cos \phi &= \underline{u}_r \cdot \underline{u}_z \\
\underline{u}_z &= \begin{pmatrix} 0 \\ 0 \\ 1 \end{pmatrix}
\end{aligned}
\tag{2.2.13}$$

and J_2, J_3, J_4 are the coefficients of the second, third, and fourth harmonics of the earth's potential function. The vectors \underline{u}_r and \underline{u}_z are unit vectors in the direction of \underline{r} and the polar axis of the earth, respectively, and r_E is the equatorial radius of the earth.

In the case of the moon,

$$\begin{aligned}
a_{dM} &= \frac{\mu_M}{r^2} \left\{ \sum_{i=2}^4 J_{iM} \left(\frac{r_M}{r} \right)^i \left[P_{i+1}'(\cos \phi) \underline{u}_r - P_i'(\cos \phi) \underline{u}_z \right] \right. \\
&\quad + 3J_{22} \left(\frac{r_M}{r} \right)^2 \left[\frac{-5(x_M^2 - y_M^2)}{r^2} \underline{u}_r + \frac{2x_M}{r} \underline{u}_x - \frac{2y_M}{r} \underline{u}_y \right] \\
&\quad + \frac{3}{2} C_{31} \left(\frac{r_M}{r} \right)^3 \left[\frac{5x_M}{r} (1 - 7 \cos^2 \phi) \underline{u}_r + (5 \cos^2 \phi - 1) \underline{u}_x \right. \\
&\quad \left. \left. + \frac{10x_M z_M}{r^2} \underline{u}_z \right] \right\}
\end{aligned}
\tag{2.2.14}$$

where in moon-fixed coordinates:

\underline{u}_r is the unit position vector in reference coordinates;

\underline{u}_x is planetary $\begin{pmatrix} 1 \\ 0 \\ 0 \end{pmatrix}$ transformed to reference coordinates;

\underline{u}_y is planetary $\begin{pmatrix} 0 \\ 1 \\ 0 \end{pmatrix}$ transformed to reference coordinates;
 \underline{u}_z is planetary $\begin{pmatrix} 0 \\ 0 \\ 1 \end{pmatrix}$ transformed to reference coordinates;

and x_M, y_M, z_M are the components of \underline{r} in planetary coordinates, which are computed by use of the Planetary Inertial Orientation Subroutine (Section 5.5.2). In addition, r_M is the mean lunar radius; and J_{22}, C_{31} are the coefficients of the terms describing the asymmetry about the pole of the moon's gravity; and the remaining symbols are defined as in Eq. (2.2.11).

5.2.2.4 Error Transition Matrix

The position and velocity vectors as maintained in the computer are only estimates of the true values. As part of the navigation technique it is necessary also to maintain statistical data in the computer to aid in the processing of navigation measurements.

If $\underline{\epsilon}(t)$ and $\underline{\eta}(t)$ are the errors in the estimates of the position and velocity vectors, respectively, then the six-dimensional correlation matrix $E(t)$ is defined by

$$E_6(t) = \begin{pmatrix} \overline{\underline{\epsilon}(t) \underline{\epsilon}(t)^T} & \overline{\underline{\epsilon}(t) \underline{\eta}(t)^T} \\ \overline{\underline{\eta}(t) \underline{\epsilon}(t)^T} & \overline{\underline{\eta}(t) \underline{\eta}(t)^T} \end{pmatrix} \quad (2.2.15)$$

In certain applications it becomes necessary to expand the state vector and the correlation matrix to more than six dimensions so as to include estimation of landmark locations in the CMC during orbit navigation, and rendezvous radar tracking biases in the LGC during the rendezvous navigation procedure. For this purpose a nine-dimensional correlation matrix is defined as follows :

$$E(t) = \begin{pmatrix} & & \overline{\underline{\epsilon}(t) \underline{\beta}^T} \\ & E_6(t) & \overline{\underline{\eta}(t) \underline{\beta}^T} \\ \overline{\underline{\beta} \underline{\epsilon}(t)^T} & \overline{\underline{\beta} \underline{\eta}(t)^T} & \overline{\underline{\beta} \underline{\beta}^T} \end{pmatrix} \quad (2.2.16)$$

where the components of the three-dimensional vector $\underline{\beta}$ are the errors in the estimates of three variables which are estimated in addition to the components of the spacecraft state vector.

In order to take full advantage of the operations provided by the interpreter in the computer, the correlation matrix will be restricted to either six or nine dimensions. If, in some navigation procedure, only one or two additional items are to be estimated, then a sufficient number of dummy variables will be added to the desired seven- or eight-dimensional state vector to make it nine-dimensional.

Rather than use the correlation matrix in the navigation procedure, it is more convenient to utilize a matrix $W(t)$, called the error transition matrix, and defined by

$$E(t) = W(t) W(t)^T \quad (2. 2. 17)$$

Extrapolation of the nine-dimensional matrix $W(t)$ is made by direct numerical integration of the differential equation

$$\frac{d}{dt} W(t) = \begin{pmatrix} O & I & O \\ G(t) & O & O \\ O & O & O \end{pmatrix} W(t) \quad (2. 2. 18)$$

where $G(t)$ is the three-dimensional gravity gradient matrix, and I and O are the three-dimensional identity and zero matrices, respectively. If the W matrix is partitioned as

$$W = \begin{pmatrix} \underline{w}_0 & \underline{w}_1 & \cdots & \underline{w}_8 \\ \underline{w}_9 & \underline{w}_{10} & \cdots & \underline{w}_{17} \\ \underline{w}_{18} & \underline{w}_{19} & \cdots & \underline{w}_{26} \end{pmatrix} \quad (2. 2. 19)$$

then,

$$\left. \begin{aligned} \frac{d}{dt} \underline{w}_i(t) &= \underline{w}_{i+9}(t) \\ \frac{d}{dt} \underline{w}_{i+9}(t) &= G(t) \underline{w}_i(t) \\ \frac{d}{dt} \underline{w}_{i+18}(t) &= \underline{0} \end{aligned} \right\} \quad i = 0, 1, \dots, 8 \quad (2.2.20)$$

The extrapolation may be accomplished by successively integrating the vector differential equations

$$\frac{d^2}{dt^2} \underline{w}_i(t) = G(t) \underline{w}_i(t) \quad i = 0, 1, \dots, 8 \quad (2.2.21)$$

The gravity gradient matrix $G(t)$ for earth or lunar orbit is given by

$$G(t) = \frac{\mu P}{r^5(t)} \left[3 \underline{r}(t) \underline{r}(t)^T - r^2(t) I \right] \quad (2.2.22)$$

5. 2. 2. 5 Numerical Integration Method

The extrapolation of navigational data requires the solution of a number of second-order vector differential equations, specifically Eqs. (2. 2. 7) and (2. 2. 23). These are all special cases of the form

$$\frac{d^2}{dt^2} \underline{y} = \underline{f}(\underline{y}, t) \quad (2. 2. 24)$$

Nystrom's method is particularly well suited to this form and gives an integration method of fourth-order accuracy. The second-order system is written

$$\frac{d}{dt} \underline{y} = \underline{z} \quad (2. 2. 25)$$

$$\frac{d}{dt} \underline{z} = \underline{f}(\underline{y}, t)$$

and the formulas are summarized below.

$$\begin{aligned} \underline{y}_{n+1} &= \underline{y}_n + \underline{\phi}(\underline{y}_n) \Delta t \\ \underline{z}_{n+1} &= \underline{z}_n + \underline{\psi}(\underline{y}_n) \Delta t \\ \underline{\phi}(\underline{y}_n) &= \underline{z}_n + \frac{1}{6} (\underline{k}_1 + 2\underline{k}_2) \Delta t \\ \underline{\psi}(\underline{y}_n) &= \frac{1}{6} (\underline{k}_1 + 4\underline{k}_2 + \underline{k}_3) \\ \underline{k}_1 &= \underline{f}(\underline{y}_n, t_n) \\ \underline{k}_2 &= \underline{f}\left(\underline{y}_n + \frac{1}{2} \underline{z}_n \Delta t + \frac{1}{8} \underline{k}_1 (\Delta t)^2, t_n + \frac{1}{2} \Delta t\right) \\ \underline{k}_3 &= \underline{f}\left(\underline{y}_n + \underline{z}_n \Delta t + \frac{1}{2} \underline{k}_2 (\Delta t)^2, t_n + \Delta t\right) \end{aligned} \quad (2. 2. 26)$$

For efficient use of computer storage as well as computing time the computations are performed in the following order:

- 1) Equation (2.2.7) is solved using the Nystrom formulas, Eqs. (2.2.26). It is necessary to preserve the values of the vector \underline{r} at times t_n , $t_n + \Delta t / 2$, $t_n + \Delta t$ for use in the solution of Eqs. (2.2.23).
- 2) Equations (2.2.23) are solved one-at-a-time using Eqs. (2.2.26) together with the values of \underline{r} which resulted from the first step.

The variable Δt is the integration time step and should not be confused with τ , the time since rectification. The maximum value for Δt which can be used for precision integration, Δt_{\max} , is computed from

$$\Delta t_{\max} = \text{minimum} \left(0.3 \frac{r_{\text{con}}^{3/2}}{\sqrt{u_P}}, 4000 \text{ sec} \right) \quad (2.2.27)$$

5.2 2.6 Coasting Integration Logic

Estimates of the state vectors of two vehicles (CSM and LM) will be maintained in the computer. In various phases of the mission it will be required to extrapolate a state vector either alone or with an associated W matrix of dimension six or nine.

To accomplish all of these possible procedures, as well as to solve the computer restart problem, three state vectors will be maintained in the computer. Let \underline{x}_C and \underline{x}_L be the estimated CSM and LM state vectors, respectively, and let \underline{x} be a temporary state vector. The state vector \underline{x} is a symbolic representation of the following set of variables:

- \underline{r}_0 = rectification position vector
- \underline{v}_0 = rectification velocity vector
- \underline{r}_{con} = conic position vector
- \underline{v}_{con} = conic velocity vector
- $\underline{\delta}$ = position deviation vector (2. 2. 28)
- \underline{v} = velocity deviation vector
- t = time associated with \underline{r}_{con} , \underline{v}_{con} , $\underline{\delta}$ and \underline{v}
- τ = time since rectification
- x = root of Kepler's equation

$$P = \text{primary body} = \begin{cases} 0 & \text{for earth} \\ 1 & \text{for moon} \end{cases}$$

The state vectors \underline{x}_C and \underline{x}_L represent an analogous set of variables.

The Coasting Integration Routine is controlled by the calling program by means of the two indicators D and V. The variable D indicates the dimension of the W matrix with

$$D = 0 \qquad (2. 2. 29)$$

denoting that the state vector only is to be extrapolated. The variable V indicates the appropriate vehicle as follows:



$$V = \begin{cases} 1 & \text{for CSM} \\ 0 & \text{for LM} \\ -1 & \text{for state vector specified by calling program} \end{cases} \quad (2.2.30)$$

In addition, the calling program must set the desired final time t_F ; and, for V equal to -1 , the desired state vector \underline{x} .

A simplified functional diagram of the Coasting Integration Routine is shown in Fig 2.2-1. In the figure the indicated state vector is being integrated to time t_F . The value of Δt for each time step is Δt_{\max} (Eq. (2.2.27)) or the total time-to-go whichever is smaller. The integration is terminated when the computed value of Δt is less than ϵ_t .

Figure 2.2-2 illustrates in more detail the logic flow of this routine. In this figure the following items, which have not been discussed fully in the text, are explicitly illustrated:

- 1) Saving of \underline{r} values for W matrix integration
- 2) Rectification procedure
- 3) Selection of disturbing acceleration

The logic flow shown in Fig. 2.2-2 is controlled by the switch F which is used to distinguish between state vector integration ($F = 1$) and W matrix integration ($F = 0$).

If the Coasting Integration Routine is requested to extrapolate the estimated LM state vector and the LM is on the lunar surface, then the routine will use the Planetary Inertial Orientation Subroutine (Section 5.5.2) to compute the desired LM position and velocity and the normal integration will not be performed. This procedure is not indicated in the figure.

There is a procedure for the emergency termination of the Coasting Integration Routine in order to permit correction of wrong erasable memory parameters. This emergency function is described in Section 5.6.11.

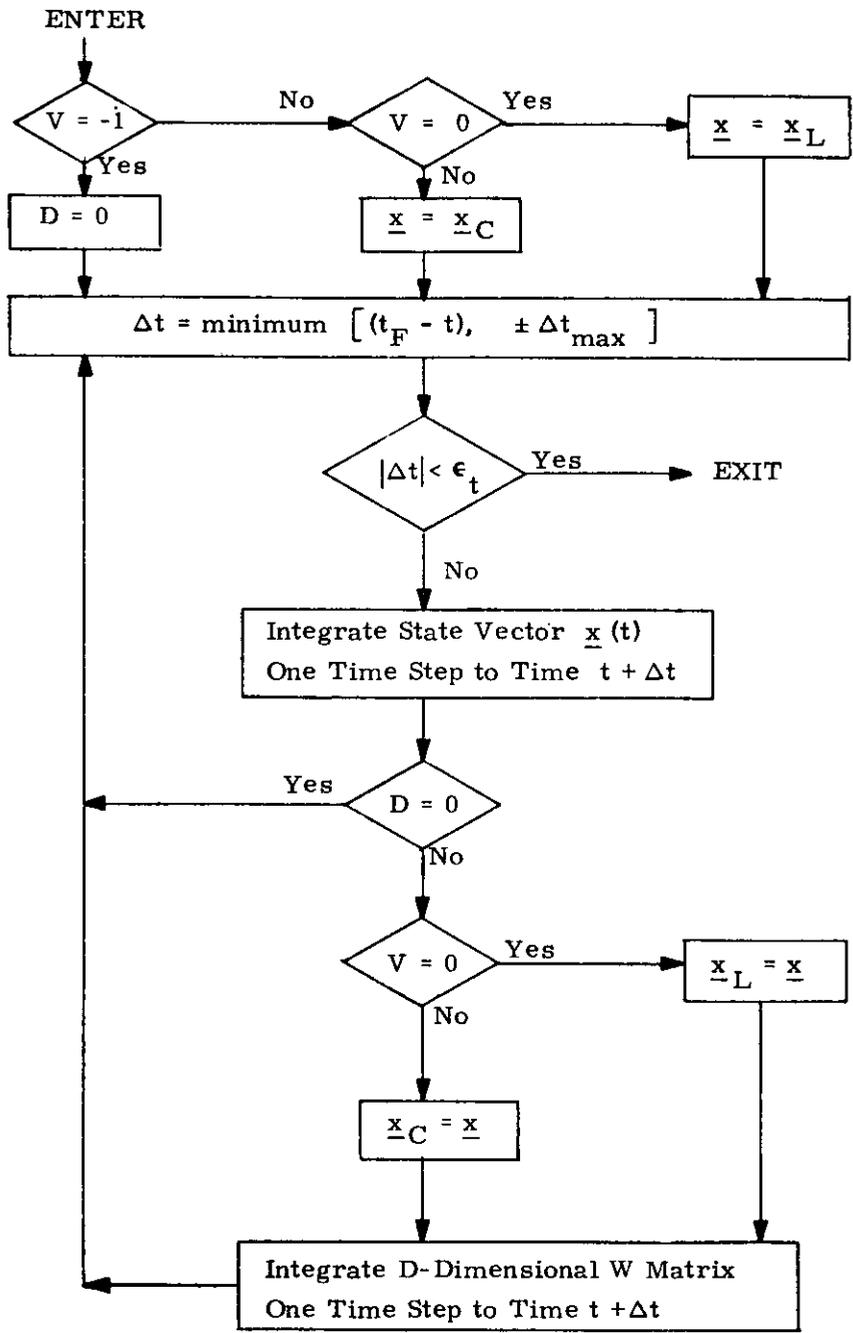


Figure 2.2-1 Simplified Coasting Integration Routine Logic Diagram

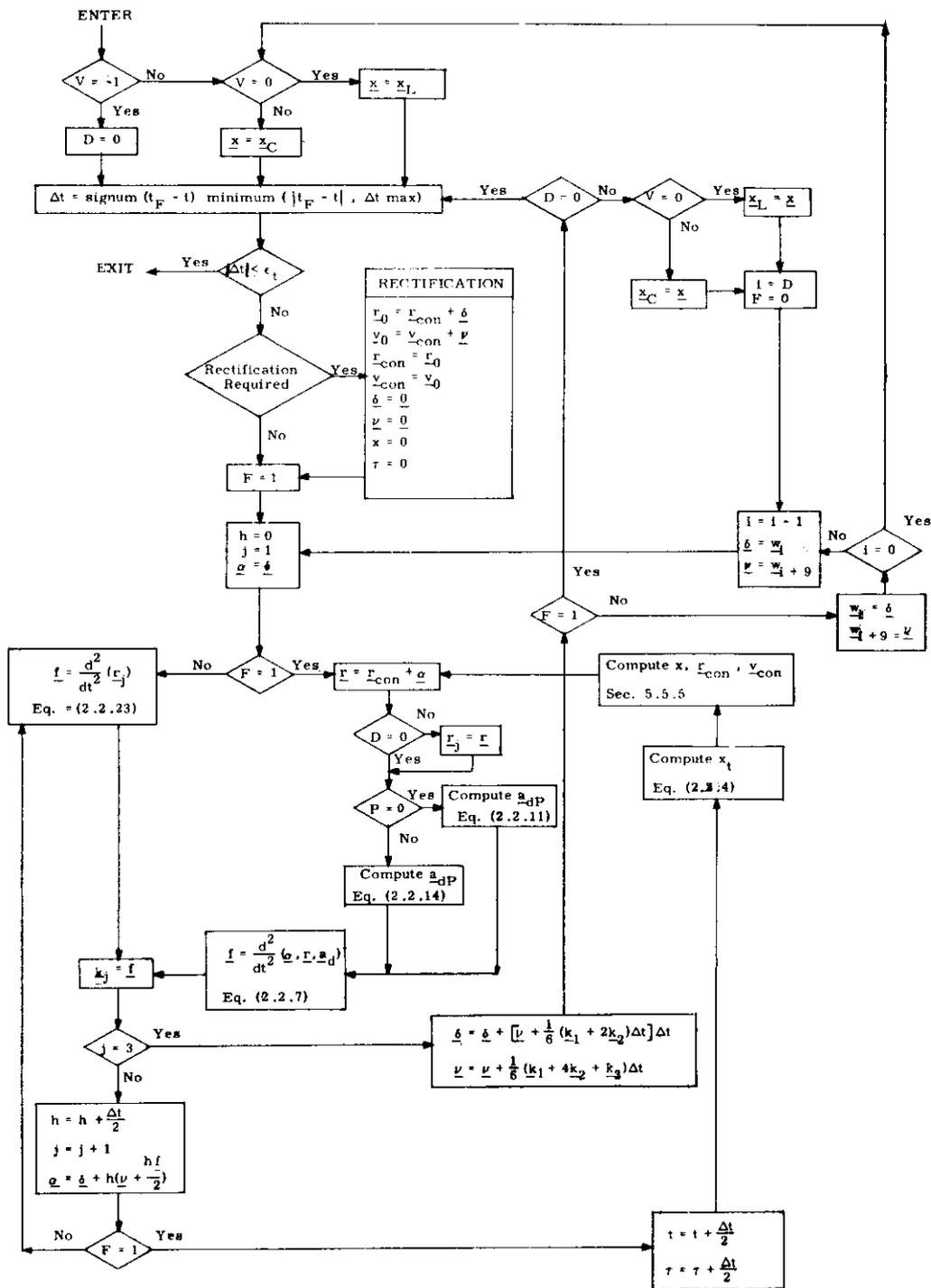


Figure 2.2-2 Coasting Integration Routine Logic Diagram

In addition to the general criterion discussed in Section 5.2.2.2, the requirements for rectification (which are not shown in Fig. 2.2-2) are functions of

- 1) the computer word length,
- 2) the fact that the computations are performed in fixed-point arithmetic,
- 3) the scale factors of the variables, and
- 4) the accuracy of the Kepler Subroutine (Section 5.5.5).

If

$$\frac{\delta}{r_{\text{con}}} > 0.01$$

or if

$$\delta > \begin{cases} 0.75 \times 2^{22} \text{ m for } P = 0 \\ 0.75 \times 2^{18} \text{ m for } P = 1 \end{cases}$$

or if

$$\nu > \begin{cases} 0.75 \times 2^3 \text{ m/csec for } P = 0 \\ 0.75 \times 2^{-1} \text{ m/csec for } P = 1 \end{cases}$$

then rectification occurs at the point indicated in Fig. 2.2-2. Also, if the calculation of the acceleration (Eq. (2.2.7)) results in overflow (i. e. any component is equal to or greater than 2^{-16} m/csec^2 for $P = 0$, or 2^{-20} m/csec^2 for $P = 1$), then the program is recycled to the beginning of the time step and rectification is performed, provided that δ is not identically zero (which may occur if an attempt is made to extrapolate a state vector below the surface). In this exceptional case, an abort occurs with alarm code 20430g.

5.2-27

Revised LUMINARY
 Added GSOP # 567 PCR # 801.2 Rev. 7 Date 11-69

Periodically it is necessary to update the estimated position and velocity vectors of the vehicle (CSM or LM) by means of navigation measurements. At the time a measurement is made, the best estimate of the state vector of the spacecraft is the extrapolated estimate denoted by \underline{x}' . The first six components of \underline{x}' are the components of the estimated position and velocity vectors. In certain situations it becomes necessary to estimate more than six quantities. Then, the state vector will be of nine dimensions. From this state vector estimate it is possible to determine an estimate of the quantity measured. When the predicted value of this measurement is compared with the actual measured quantity, the difference is used to update the indicated state vector as well as its associated error transition matrix as described in Section 5.2.1. The error transition matrix, W , is defined in Section 5.2.2.4.

This routine is used to compute deviations to be added to the components of the estimated state vector, and to update the estimated state vector by these deviations provided the deviations pass a state vector update validity test as described in Section 5.2.1.

Let D be the dimension (six or nine) of the estimated state vector. Associated with each measurement are the following parameters which are to be specified by the program calling this routine:

\underline{b} = Geometry vector of D dimensions

$\overline{\alpha^2}$ = A priori measurement error variance

δQ = Measured deviation, the difference between the quantity actually measured and the expected value based on the original value of the estimated state vector \underline{x}' .

The procedure for incorporating a measurement into the estimated state vector is as follows:

- 1 Compute a D-dimensional \underline{z} vector from

$$\underline{z} = W'^T \underline{b} \quad (2.3.1)$$

where W' is the error transition matrix associated with \underline{x}' .

- 2 Compute the D-dimensional weighting vector, $\underline{\omega}$, from

$$\underline{\omega}^T = \frac{1}{z^2 + \alpha^2} \underline{z}^T W'^T \quad (2.3.2)$$

- 3 Compute the state vector deviation estimates from

$$\delta \underline{x} = \underline{\omega} \delta Q \quad (2.3.3)$$

- 4 If the data pass the validity test, update the state vector and the W matrix by

$$\underline{x} = \underline{x}' + \delta \underline{x} \quad (2.3.4)$$

$$W = W' - \frac{\underline{\omega} \underline{z}^T}{1 + \frac{\alpha^2}{z^2 + \alpha^2}} \quad (2.3.5)$$

In order to take full advantage of the three-dimensional vector and matrix operations provided by the interpreter in the computer, the nine-dimensional W matrix will be stored sequentially in the computer as follows :

$$\underline{w}_0, \underline{w}_1, \dots, \underline{w}_{26}$$

Refer to Section 5. 2. 2. 4 for the definition of the W matrix. Define the three-dimensional matrices

$$W_0 = \begin{pmatrix} \underline{w}_0^T \\ \underline{w}_1^T \\ \underline{w}_2^T \end{pmatrix} \quad W_1 = \begin{pmatrix} \underline{w}_3^T \\ \underline{w}_4^T \\ \underline{w}_5^T \end{pmatrix} \quad \dots \quad W_8 = \begin{pmatrix} \underline{w}_{24}^T \\ \underline{w}_{25}^T \\ \underline{w}_{26}^T \end{pmatrix} \quad (2.3.6)$$

so that

$$W = \begin{pmatrix} W_0^T & W_1^T & W_2^T \\ W_3^T & W_4^T & W_5^T \\ W_6^T & W_7^T & W_8^T \end{pmatrix} \quad (2.3.7)$$

Let the nine-dimensional vectors $\underline{\delta x}$, \underline{b} , $\underline{\omega}$, and \underline{z} be partitioned as follows:

$$\underline{\delta x} = \begin{pmatrix} \underline{\delta x}_0 \\ \underline{\delta x}_1 \\ \underline{\delta x}_2 \end{pmatrix} \quad \underline{b} = \begin{pmatrix} \underline{b}_0 \\ \underline{b}_1 \\ \underline{b}_2 \end{pmatrix} \quad \underline{\omega} = \begin{pmatrix} \underline{\omega}_0 \\ \underline{\omega}_1 \\ \underline{\omega}_2 \end{pmatrix} \quad \underline{z} = \begin{pmatrix} z_0 \\ z_1 \\ \vdots \\ z_8 \end{pmatrix} = \begin{pmatrix} \underline{z}_0 \\ \underline{z}_1 \\ \underline{z}_2 \end{pmatrix} \quad (2.3.8)$$

Then, the computations shown in Eqs. (2.3.1) through (2.3.3) are performed as follows, using three-dimensional operations:

$$\underline{z}_i = \sum_{j=0}^{\frac{D}{3}-1} W'_{i+3j} \underline{b}_j$$

$$a = \sum_{j=0}^{\frac{D}{3}-1} \underline{z}_j \cdot \underline{z}_j + \alpha^2$$

(2.3.9)

$$\underline{\omega}_i^T = \frac{1}{a} \sum_{j=0}^{\frac{D}{3}-1} \underline{z}_j^T W'_{3i+j}$$

$$\delta x_i = \delta Q \underline{\omega}_i \quad \left(i = 0, 1, \dots, \frac{D}{3} - 1 \right)$$

Equation (2.3.5) is written

$$\gamma = \frac{1}{1 + \sqrt{\alpha^2/a}}$$

(2.3.10)

$$\underline{w}_{i+9j} = \underline{w}'_{i+9j} - \gamma \underline{z}_i \underline{\omega}_j \quad \left(\begin{array}{l} i = 0, 1, \dots, D-1 \\ j = 0, 1, \dots, \frac{D}{3}-1 \end{array} \right)$$

The Measurement Incorporation Routine is divided into two subroutines, INCORP1 and INCORP2. The subroutine INCORP1 consists of Eqs. (2.3.9), while INCORP2 is composed of Eqs. (2.3.4) and (2.3.10). The method of using these subroutines is illustrated in Fig. 2.3-1.

Since the estimated position and velocity vectors are maintained in two pieces, conic and deviation from the conic, Eq. (2.3.4) cannot be applied directly. The estimated position and velocity deviations resulting from the measurement, $\delta \underline{x}_0$ and $\delta \underline{x}_1$, are added to the vectors $\underline{\delta}$ and \underline{v} , the position and velocity deviations from the conics, respectively. Since $\underline{\delta}$ and \underline{v} are maintained to much higher accuracy than the conic position and velocity vectors, a possible computation overflow situation exists whenever Eq. (2.3.4) is applied. If overflow does occur, then it is necessary to reinitialize the Coasting Integration Routine (Section 5.2.2) by the process of rectification as described in Section 5.2.2.2. The logic flow of the subroutine INCORP2 is illustrated in detail in Fig. 2.3-2.

Overflow occurs when

or

$$\underline{\delta} \geq \begin{cases} 2^{22} \text{ m for } P = 0 \\ 2^{18} \text{ m for } P = 1 \end{cases}$$

$$\underline{v} \geq \begin{cases} 2^3 \text{ m/csec for } P = 0 \\ 2^{-1} \text{ m/csec for } P = 1 \end{cases}$$

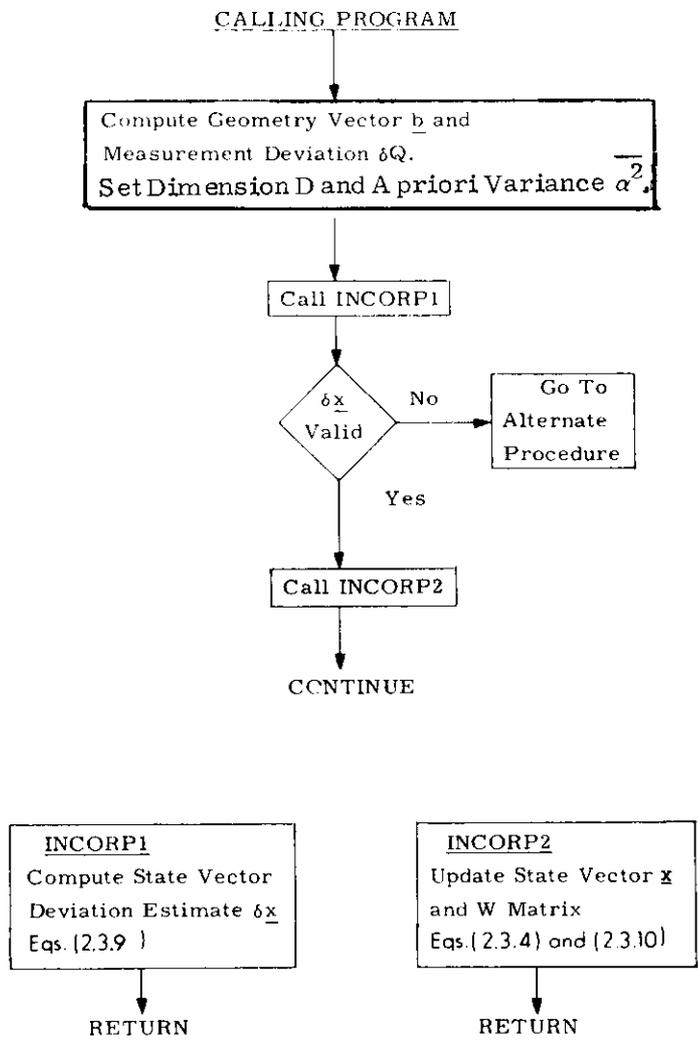


Fig. 2.3-1 Measurement Incorporation Procedure

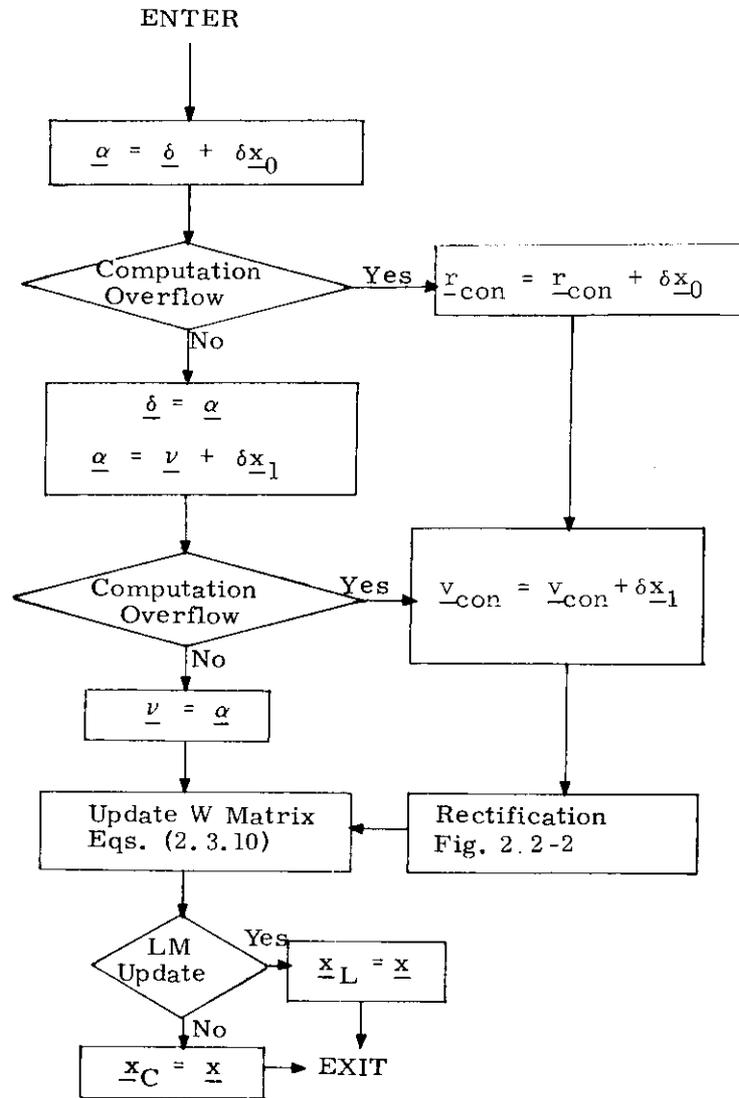


Fig. 2.3-2 INCORP2 Subroutine Logic Diagram

5.2.4 RENDEZVOUS NAVIGATION PROGRAM

5.2.4.1 Target Acquisition Routine

One of the first functions performed by the Rendezvous Navigation Program is to use the Target Acquisition Routine (Fig. 2.4-1) to establish lock-on of the LM Rendezvous Radar (RR) with the transponder on the CSM. Since the problem of acquiring the CSM with the RR is essentially the same for both the Rendezvous and Lunar Surface Navigation Programs (P-20 and P-22, respectively), the same Acquisition Routine is used for both except for certain differences in operation. For the moment, however, most of the operational details presented are for the case when the Rendezvous Navigation Program is being used. In either case it is assumed at the start of the routine that the RR is on and has been permitted to warm-up to operating conditions.

There are three modes (RR LGC, RR Manual, and RR Search) for controlling the RR in target acquisition. The RR LGC and RR Manual Modes can be selected by the astronaut at the beginning of program P-20 by a procedure described later. However, the RR Search Mode can be selected only after the RR LGC Mode has failed to acquire the target. The RR Manual Mode is not used in P-22.

Prior to using the Target Acquisition Routine, program P-20 sets the Rendezvous, Track, and Update flags and resets the LOSCM, R04, Manual Acquire, No Angle Monitor, and Search flags. The Rendezvous flag is set to denote that program P-20 or P-22 is being used. When this flag is reset, programs P-20 and P-22 are terminated. The purpose of most of the other flags is given in the following sections. In addition to initializing the above flags, the state vector update option is automatically set to the LM in program P-20.

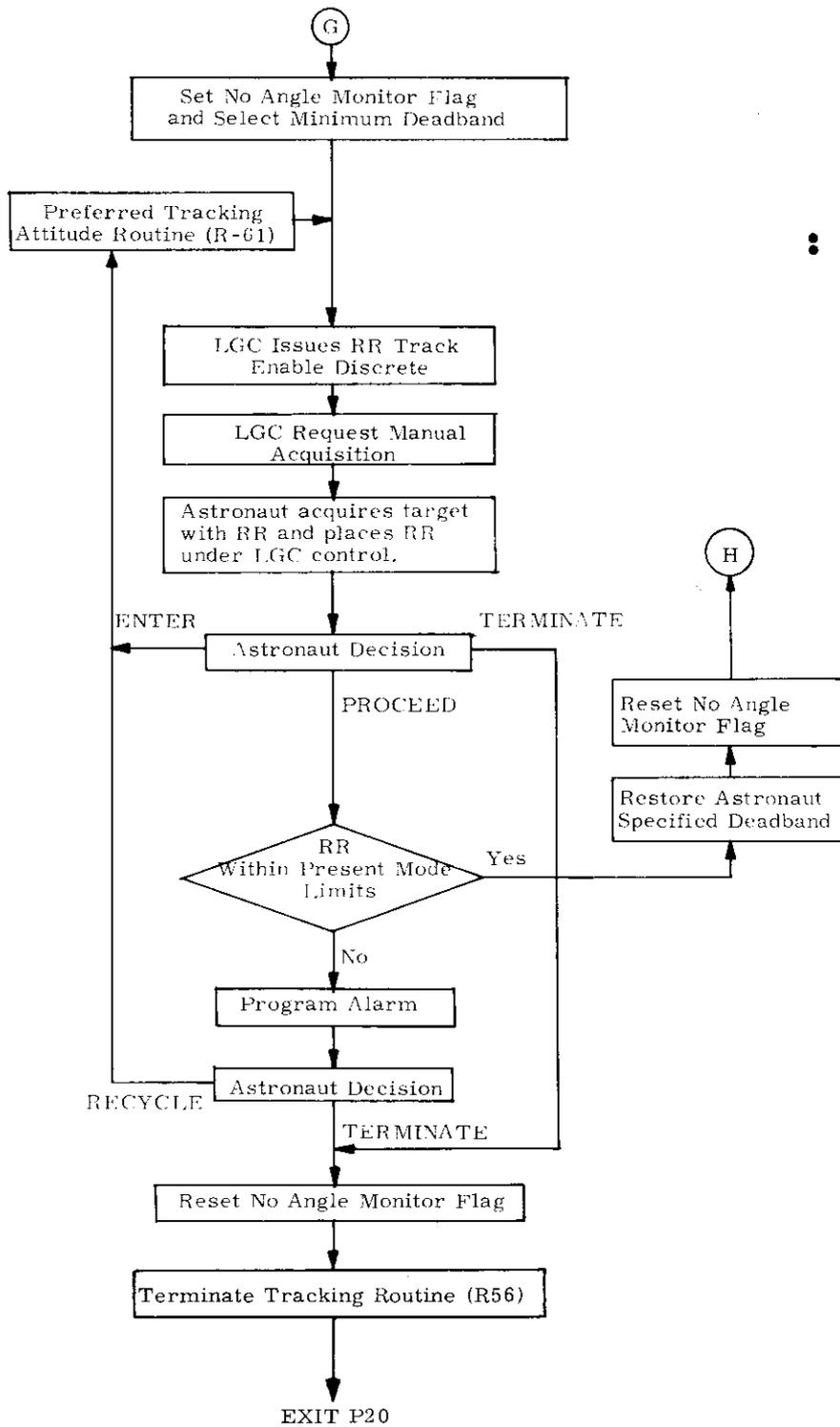


Figure 2. 4-1 Target Acquisition Routine
(page 2 of 2)

At the start of the Target Acquisition Routine in Fig. 2.4-1 the range is determined between the LM and CSM by taking the vector difference between the LM and CSM position vectors propagated to that time. If the range is greater than 400 nm, a program alarm is issued since the RR is unable to provide the correct range information to the LGC because of the ranging technique used in the radar.

After verifying that the range between the LM and CSM is less than 400 nm, the Preferred Tracking Attitude Routine (R-61 of Sections 4 and 5.2.4.4) is used to align the LM +Z-axis along the LOS to the CSM. This is done to insure that the RR antenna will be designated in the correct antenna angular coverage region (Mode 1 of Fig. 2.4-3) for operation with program P-20. In addition, this attitude permits the LM optical beacon to be seen by the CSM in case optical tracking is being performed. The LM optical beacon is centered with respect to the LM +Z-axis and has a beamwidth of approximately 60 degrees. In Sections 5.2.4.2.1 and 5.2.4.4 it is seen that the LM +Z-axis is continuously directed along the LOS to the CSM by use of the Fine Preferred Tracking Attitude Routine (R-65) when RR Data is being used to update the navigation equations in P-20. In addition it is seen in Section 5.2.4.2.1 that the RR data is not used to update the navigation equations if the RR is more than 30° from the LM +Z-axis. The above constraints on LM attitude not only insure that the LM optical beacon will be seen by the CSM but also insure a reliable estimate of the RR angle biases when processing RR data.

After using the Preferred Tracking Attitude Routine in Fig. 2.4-1 the LGC checks to see if the RR Auto Mode discrete is being received from the RR. This discrete signifies that the RR is on and has been placed under LGC control by a

mode control switch associated with the RR. In addition, reception of this discrete at this time automatically indicates that the astronaut wishes to use the RR LGC Mode of target acquisition. If the discrete is not present and P-20 is the only program in operation, a request is made to the astronaut to either select the RR Manual Mode by keying in "ENTER" or the RR LGC Mode by placing the RR mode control switch in the LGC position and keying in "PROCEED." If program P-20 is not the only program running, a program alarm is issued to the astronaut requiring the action indicated in Fig. 2.4-1. This latter step is taken to avoid conflict in DSKY displays between program P-20 and other programs.

5.2.4.1.1 Target Acquisition with the RR LGC Mode

If the check on the RR Auto Mode discrete in Fig. 2.4-1 indicates that the RR LGC Mode was selected, the routine next insures that the RR CDU's are not being zeroed. These CDU's are zeroed by the RR Monitor Routine (R-25) of Section 5.2.4.3 whenever the RR Auto Mode discrete is received from the RR. Afterwards, a check is made to see if either the Manual Acquire or Search flag is set, signifying that the RR Manual or RR Search Mode is being used. Since the present mode is assumed to be the RR LGC Mode, these flags should not be set. Just before entering the RR Designate Routine, a precision update is made on the CSM and LM permanent state, and the Rendezvous and Track flags are checked. The precision update is performed at this time in order to shorten the operation time of the Kepler Subroutine, which is used periodically in the RR Designate Routine to compute the LOS to the CSM. The operation time of the Kepler Subroutine is a function of the age of the permanent state vector. The purpose of the check on the Track flag just prior to using the RR Designate Routine is to insure that the RR is not designated by the LGC until this flag is set. The Track flag is reset by various LGC programs whenever there is no desire to designate or read data from the RR. Although the Track flag is set at the start of program P-20, it is possible for it to be temporarily reset during operation of program P-20.

To designate the RR along the LOS to the CSM, use is made of the RR Designate Routine of Fig. 2.4-2. Initially the RR Track Enable discrete is removed from the RR in order to ensure RR response to designate commands. Subsequently, the routine points the RR antenna along the reference direction of Mode 1 for P-20 (see Fig. 2.4-3). For P-22, a check is made to see if the antenna is in Mode 2, as defined in Section 5.2.5.3, and, if not, it is pointed along the reference direction for Mode 2. The LOS range vector (\underline{r}_{LOS}) and the velocity (\underline{v}_{LC}) of the CSM with respect to the LM in stable member coordinates are computed by using the CSM and LM position and velocity vectors ($\underline{r}_C, \underline{v}_C, \underline{r}_L, \underline{v}_L$). During P-20 the vectors $\underline{r}_C, \underline{v}_C, \underline{r}_L,$ and \underline{v}_L are obtained by using the Kepler Subroutine of section 5.5.5. During P-22 only \underline{r}_C and \underline{v}_C are obtained with the Kepler Subroutine -- \underline{r}_L and \underline{v}_L are obtained in the precision update prior to entering RR Designate Routine. The vectors \underline{r}_L and \underline{v}_L are not updated during operation of the routine for P-22, because the update is time consuming and, due to the slow rate of lunar motion, would result in very little improvement in the LOS accuracy. The specified time for use in the Kepler Subroutine is the present time plus an additional amount ϵ_1 . The quantity ϵ_1 is equal to 0.5 seconds and is a rough estimate of the LGC time required by the Kepler Subroutine to compute \underline{r}_C and \underline{v}_C during the operation of P-22. The main reason for using ϵ_1 is to ensure that the vector \underline{r}_{LOS} issued to the gyro command loop (see page 2 of Fig. 2.4-2) is not too stale in time during the relatively large LOS angular rates experienced in P-22. This time correction is used in P-20 also, to offset some of the time staleness. The correction is not as good for P-20, but the problem is not as acute because of the relatively low LOS angular rates experienced during free-fall.

After computing \underline{r}_{LOS} and \underline{v}_{LC} , a check is made (Fig. 2.4-2) to see if the LOS is within the angular tracking limits of either RR antenna mode shown in Fig. 2.4-3. This is accomplished by first computing the equivalent RR shaft and trunnion angles for \underline{r}_{LOS} with the method shown in Section 5.6.15.2 where

$$\underline{u}_D = [\text{SMNB}] \text{UNIT} (\underline{r}_{LOS})$$

and then comparing these angles with the electrical tracking mode limits. It should be noted that the tracking mode limits of Fig. 2.4-3 are not the true LOS limits of the RR antenna modes but are the LOS limits within which the RR should track satisfactorily. The polarity and magnitude of the shaft and trunnion LOS values given in Fig. 2.4-3 agree with the corresponding shaft and trunnion CDU indicated values except for shaft angles in Mode 2 where

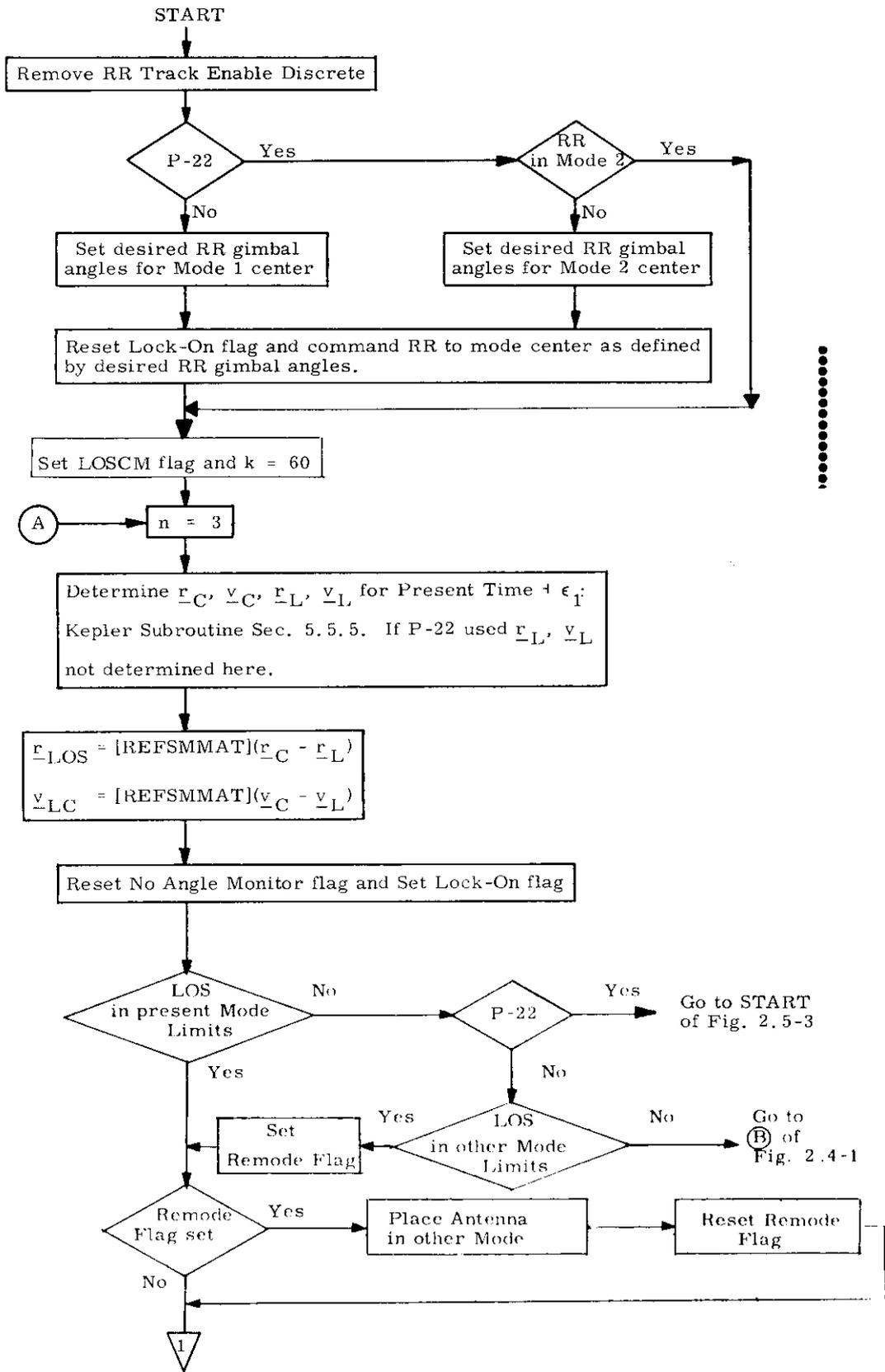


Figure 2.4-2 RR Designate Routine (Page 1 of 2)

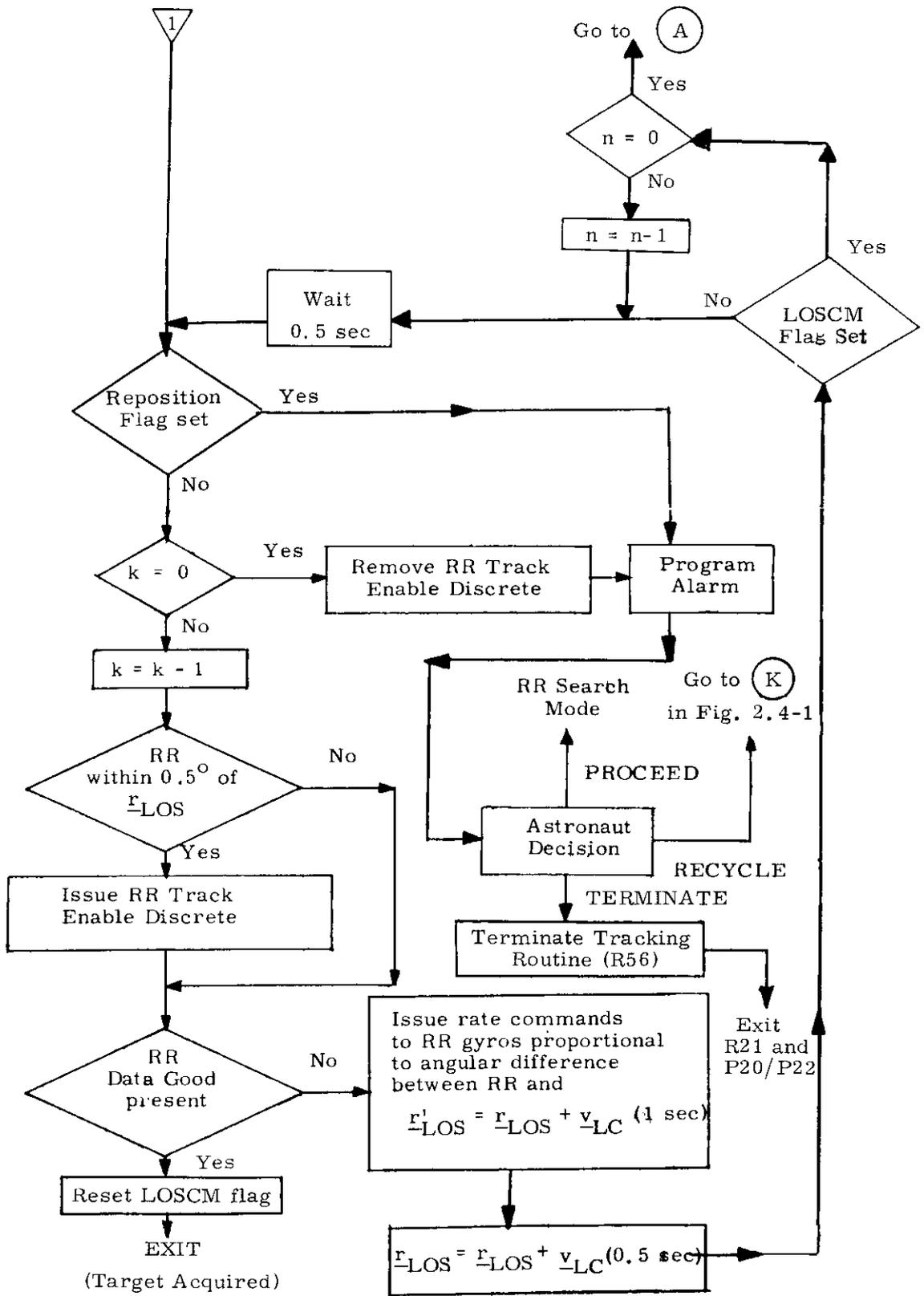


Figure 2.4-2 RR Designate Routine (Page 2 of 2)

$$S_{CDU} = S_{LOS} + 180^\circ$$

The RR Monitor Routine (R-25 of Section 5.2.4.3) continually updates the computer's knowledge of which mode the RR antenna is actually in, even though the antenna may not be within the tracking limits of that mode. Since the Preferred Tracking Attitude Routine was previously used in program P-20 to align the LM +Z-axis along the LOS, it is seen in Fig. 2.4-2 that steps are taken to insure that the RR antenna is in Mode 1 and the LOS is within the tracking limits of Mode 1 before designating the RR. Whenever a change in antenna mode is performed (i. e. a remode) the antenna is left pointing along the reference direction for the desired mode as shown in Fig. 2.4-3.

If the Reposition flag should be set by the RR Monitor Routine of Section 5.2.4.3 during the operation of the gyro command loop in the RR Designate Routine, a program alarm is issued as shown in Fig. 2.4-2. This flag when set denotes that the RR is being repositioned to the reference direction for the present RR antenna mode.

The RR is designated towards the CSM by issuing rate commands to the RR antenna gyros approximately every 0.5 seconds which are proportional to the angular difference between the indicated direction of the antenna and

$$\underline{r}'_{LOS} = \underline{r}_{LOS} + \underline{v}_{LC} (1 \text{ sec.})$$

where \underline{r}'_{LOS} is essentially \underline{r}_{LOS} advanced one second into the future and is obtained by adding to the present range vector (\underline{r}_{LOS}) the distance covered in a one second interval by \underline{v}_{LC} assuming a constant velocity. This correction to \underline{r}_{LOS} for RR designation essentially compensates for the lag error associated with the type of control system existing between the RR and the LGC. The method used to compute the gyro rate commands is given in Section 5.6.15.3.

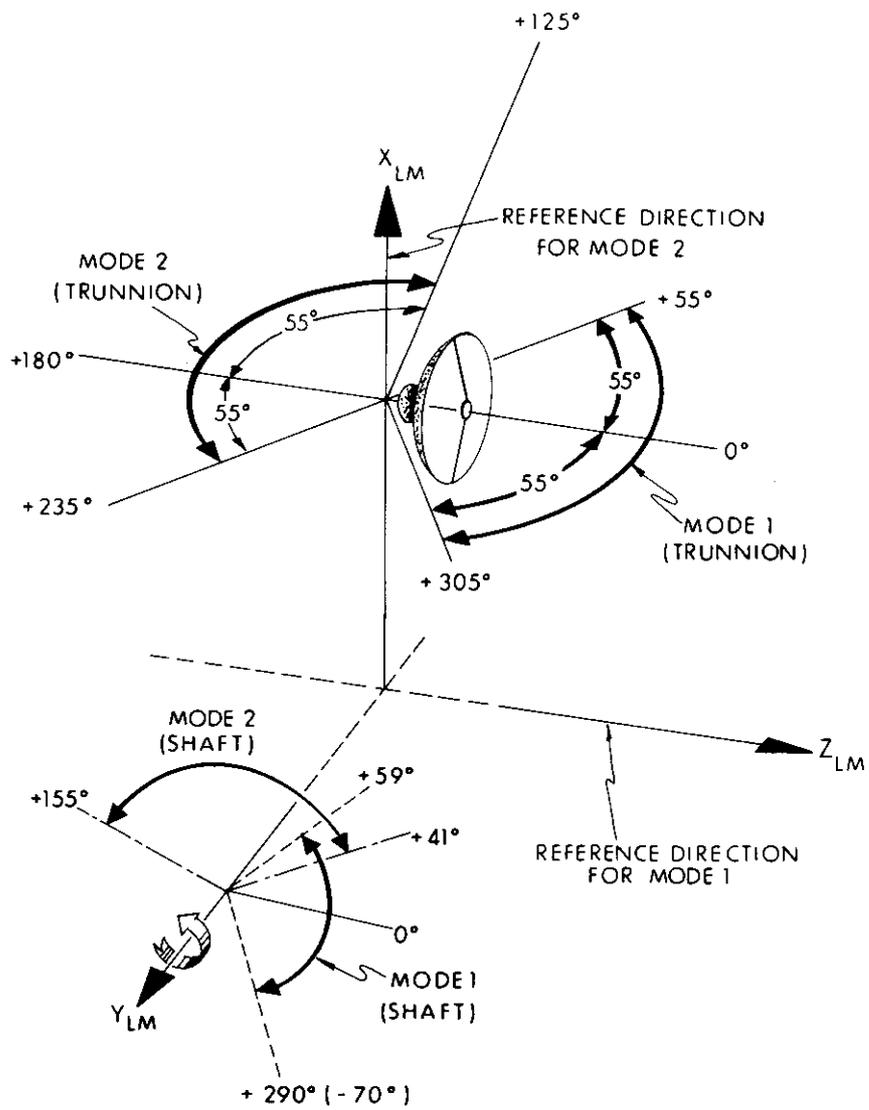


Figure 2.4-3 RR Antenna Shaft and Trunnion LOS Tracking Regions

In Fig. 2.4-2 it is seen that an approximate update for LOS motion is made to \underline{r}_{LOS} [i. e. $\underline{r}_{LOS} = \underline{r}_{LOS} + \underline{v}_{LC}$ (0.5 sec)] about every 0.5 seconds until the counter n has been decremented from 3 to 0. When n = 0 the routine returns to the top of Fig. 2.4-2 to compute new values of \underline{r}_{LOS} and \underline{v}_{LC} .

During the designation of the RR a check is made to see if the RR is within 0.5 degrees of the present LOS range vector \underline{r}_{LOS} . This is accomplished by using the method in Section 5.6.15.1 to obtain a unit vector \underline{u}_{RR} defining the direction of the RR in navigation base coordinates, which is then transformed to stable member coordinates as follows:

$$\underline{u}_{RR} = [\text{NBSM}] \underline{u}_{RR}$$

and compared with the present LOS range vector \underline{r}_{LOS} . When the angle between \underline{u}_{RR} and \underline{r}_{LOS} drops below 0.5 degrees, the RR Track Enable discrete is issued to the RR, enabling its angle tracking servos to track the target if its range rate tracking network has already acquired the target. Issuance of the RR Track Enable discrete also initiates the RR range tracker search.

Approximately every 0.5 seconds it is seen in Fig. 2.4-2 that a check is also made to see if the RR Data Good discrete is present. This discrete is sent to the LGC by the RR when lock-on has been achieved in range and range rate and the RR Track Enable discrete has been received from the LGC. In the event of failure to receive this discrete after issuing rate commands to the RR gyros 60 times, the RR Track Enable discrete is removed and a program alarm is issued to the astronaut whereupon he either repeats the designate process or goes to the RR Search Mode.

If the RR Data Good discrete is received during the designate process, the routine is terminated and it is seen in Fig. 2.4-1 that for P-20, the angles indicating the direction of the RR antenna are displayed to the astronaut so that he may confirm lock-on by the main radiation lobe before entering the RR Data Read Routine. This confirmation may be made through use of the Crew Optical Alignment Sight (COAS) or the RR signal-strength meter. The angles in the display are the same as those defined in Sections 5.6.14 and 5.6.21. If the astronaut should exercise manual control of RR to insure that it is tracking on the main lobe, he should not key in a PROCEED until after he has placed the RR mode control switch in the LGC position and the radar panel NO TRACK light is extinguished. This procedure is necessary for the reasons given in Section 5.2.4.1.2. After confirming main lobe lock-on, a 2.5 second delay is introduced in order to permit any transients in the RR angle tracking servos to settle out before RR data is taken. In addition, the status of the Rendezvous and Track flags is checked before entering the RR Data Read Routine of Fig. 2.4-6.

5.2.4.1.2 Target Acquisition With the RR Manual Mode

In Fig. 2. 4-1 it is seen that the RR Manual Mode of target acquisition can be obtained by keying in "ENTER" after the LGC discovers that the RR Auto Mode discrete is not present. Absence of this discrete at the beginning of program P-20 can be insured by not placing the mode control switch of the RR in the LGC position. The logic associated with the RR Manual Mode is shown on page 2 of Fig. 2. 4-1. Initially, the No Angle Monitor flag is set, the minimum deadband of the RCS DAP is selected, and the RR Track Enable discrete is issued to the RR. The No Angle Monitor flag is set during the RR Manual Mode so as to disable the angle monitor function of the RR Monitor Routine (see Section 5. 2. 4. 3). Selection of the minimum deadband permits the astronaut to manually designate the RR more accurately. If the Preferred Tracking Altitude Routine (R-61) is called by keying RECYCLE and an R-60 maneuver is called, the astronaut-specified deadband is restored at the conclusion of the maneuver. The RR Track Enable discrete, although it has no effect on manual control of the RR, is issued at this time so that there is no loss of RR angle tracking when the mode control switch of the RR is placed in the LGC position after manual target acquisition.

Afterwards, the LGC requests the astronaut to perform the manual acquisition. If manual acquisition is achieved, the astronaut places the RR under LGC control, waits until the radar panel NO TRACK light is extinguished, and keys in "PROCEED". This procedure is necessary in order to insure that the RR range tracking network has locked onto the target before entering the RR Data Read Routine. When the RR mode control switch is placed in the LGC position RR range tracking is interrupted by switching and a new range search is initiated. When the RR acquires the target in both range and range rate, the NO TRACK light is extinguished.

After the "PROCEED" the LGC checks to see if the RR is within the limits of the present coverage mode. If the RR is not within the coverage limits, a program alarm is issued and the astronaut either terminates the program or repeats the manual acquisition process as shown in Fig. 2. 4-1. Note that the Preferred Tracking Attitude Routine is used to re-align the LM +Z-axis with the target LOS whenever he elects to repeat the manual acquisition process.

Once target acquisition has been achieved and the RR is found to be within the present antenna mode limits, the astronaut specified deadband is restored, the No Angle Monitor flag is reset, and the Manual Acquire flag is set to denote manual acquisition. The remaining steps are indicated in Fig. 2. 4-1.

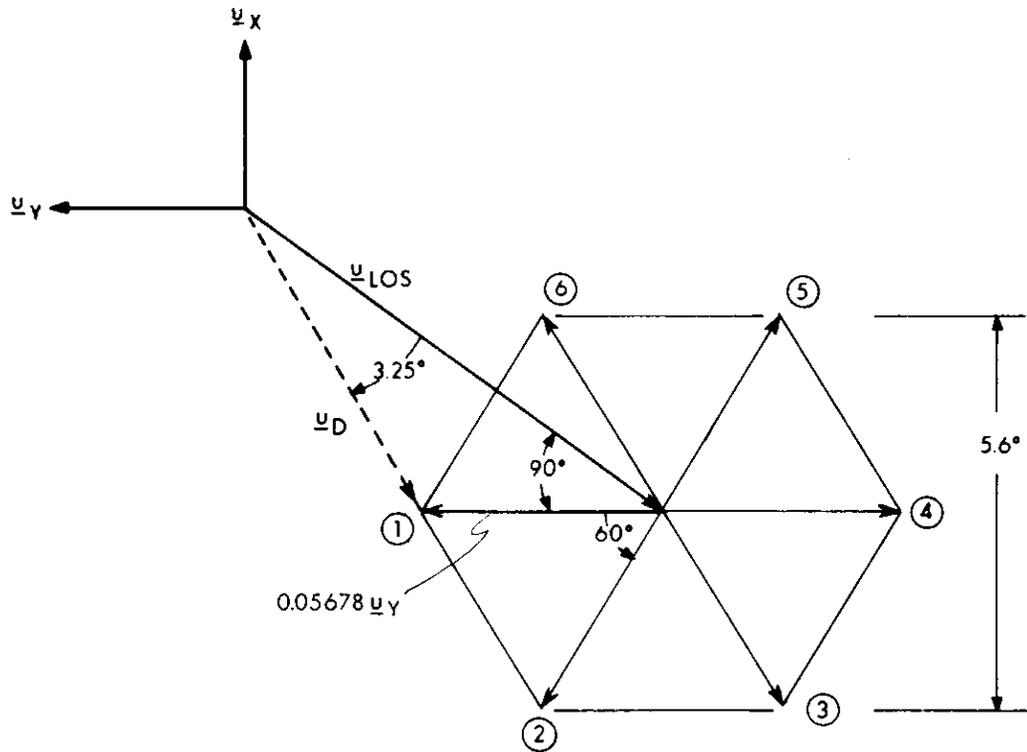
5.2.4.1.3 Target Acquisition With the RR Search Mode

In the RR Designate Routine (Fig. 2.4-2) it is seen that the astronaut may select the RR Search Mode if the RR Designate Routine fails to acquire the target. The RR Search Mode is obtained by using the RR Search Routine (R-24 of Section 4) to designate the RR antenna in a hexagonal search pattern about the estimated LOS. The search pattern is a box with six sides where the side to side dimensions are 5.6° as shown in Fig. 2.4-4. At the beginning of this mode the RR is designated for six seconds along the estimated LOS to the target defined by the unit vector \underline{u}_{LOS} in Fig. 2.4-4. Afterwards, the LGC sequentially designates the RR to each corner of the hexagon for a period of six seconds. Having completed the designate to each corner, the LGC repeats the above process starting with a designate along \underline{u}_{LOS} for six seconds. The time required to generate this search pattern is approximately 42 seconds.

The logic flow associated with the RR Search Routine is shown in Fig. 2.4-5. Initially, the Search flag is set to denote that this mode is being used and a display of certain RR search parameters is instigated to which the astronaut must respond later. The RR Track Enable discrete is then issued to the RR so that the RR may acquire the target during the search pattern generation. This discrete is also re-issued at each corner of the search pattern in case it has been removed by some source such as the RR Monitor Routine.

Approximately every six seconds the position and velocity vectors of the CSM and LM are used to compute the line-of-sight unit vector (\underline{u}_{LOS}) in basic reference coordinates and the relative velocity (\underline{v}_{LC}) in stable member coordinates. Note that \underline{u}_{LOS} and \underline{v}_{LC} are based on the CSM and LM position and velocity vectors computed for the present time plus ϵ_2 where ϵ_2 is equal to 1.5 seconds and is a rough estimate of the LGC time required to compute the CSM and LM state vectors during the operation of P-22. The reason for using ϵ_2 is to ensure

that the vector \underline{u}_{LOS} is not too stale in time during the relatively large LOS angular rates experienced in P-22. After computing \underline{u}_{LOS} and \underline{v}_{LC} , the routine computes the desired RR pointing direction (\underline{u}_D) which may be



Note: This is the search pattern as viewed from the CSM.

Fig. 2.4-4 RR Search Pattern

along \underline{u}_{LOS} or to one corner of the search pattern (see Figure 2.4-4). Afterwards, \underline{u}_D is transformed to stable member coordinates and a check is made on the direction of \underline{u}_D with respect to the angular coverage modes of the RR antenna, just as was done with \underline{r}_{LOS} in the RR Designate Routine in Section 5.2.4.1.1. If \underline{u}_D is not within the angular coverage limits of either mode during program P-20, the program alarm light is turned on, an alarm code is stored, and the search pattern is stopped. If the astronaut wishes to continue the search in program P-20, he must re-establish the preferred tracking attitude with the Preferred Tracking Attitude Routine (R-61) in the manner shown at the end of Fig. 2.4-5. Once \underline{u}_D is found to be within the coverage limits and the correct antenna mode has been established, the routine proceeds to designate the RR by issuing rate commands to the RR gyros about every 0.5 seconds with approximate corrections being made each time for lag error and target motion.

Note in Fig. 2.4-5 that the angle between the RR and the LM +Z-axis is periodically determined and displayed as one of the RR search parameters during the search operation. By observing the displayed angle the astronaut can determine during program P-20 when he should re-establish the preferred tracking attitude with the Preferred Tracking Attitude Routine (R-61) in the manner shown at the end of Fig. 2.4-5.

Near the end of Fig. 2.4-5 it is seen that a periodic check is made to see if the RR Data Good discrete is being received from the RR, signifying that the RR has acquired the target in range and range rate. If this discrete is present, the search pattern is stopped and the astronaut is notified. The astronaut then checks to see if acquisition was obtained with the main radiation lobe of the RR. By manually positioning the RR and observing the RR signal strength meter, he should be able to distinguish the main lobe from any side lobes. Having achieved and verified lock-on with the main radiation lobe, the astronaut places the RR mode control switch in the LGC position, waits until the radar panel NO TRACK light is **extinguished**, and keys in a "PROCEED". The remaining steps are indicated in Fig. 2.4-1.

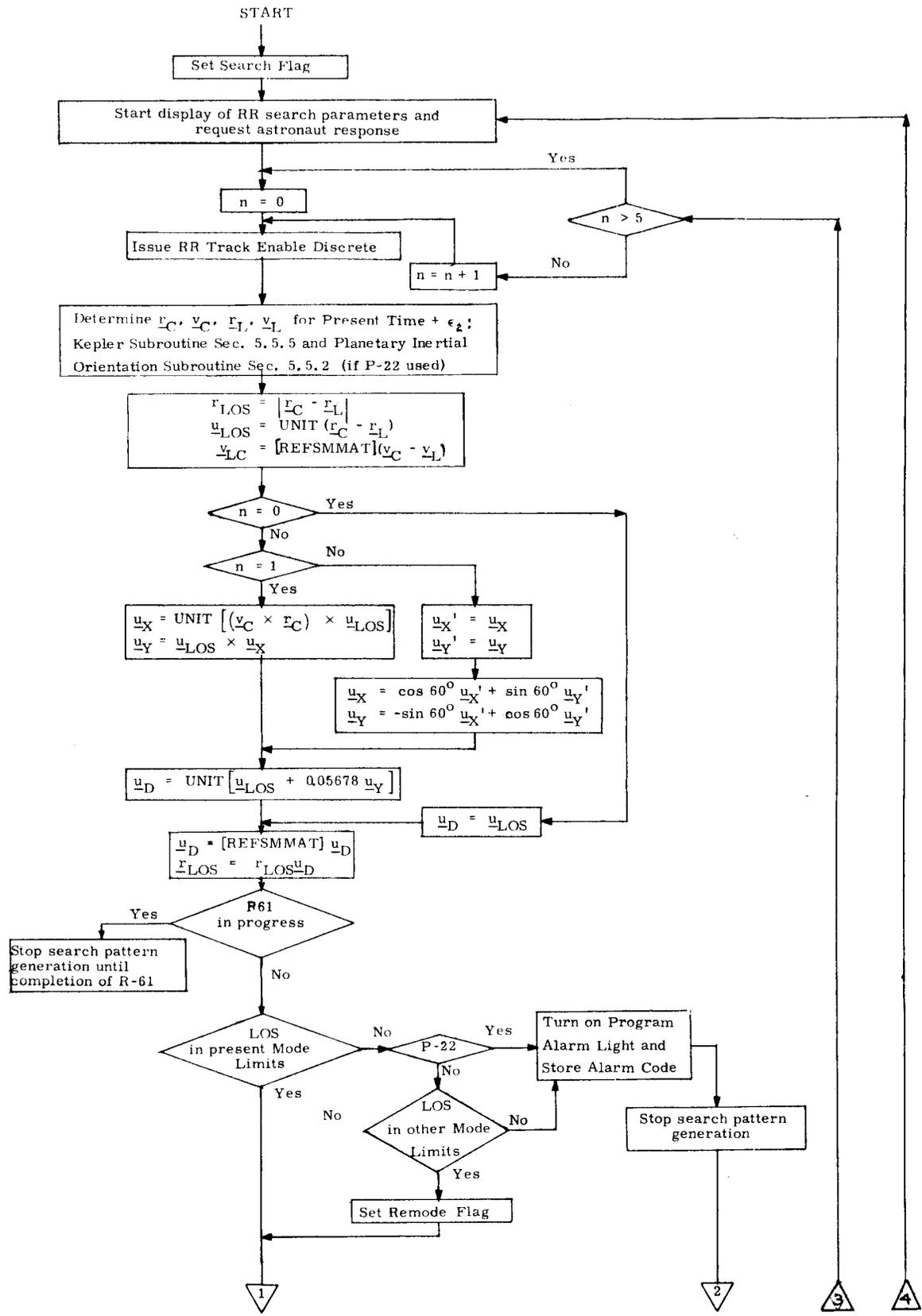


Figure 2.4-5 RR Search Routine (page 1 of 2)

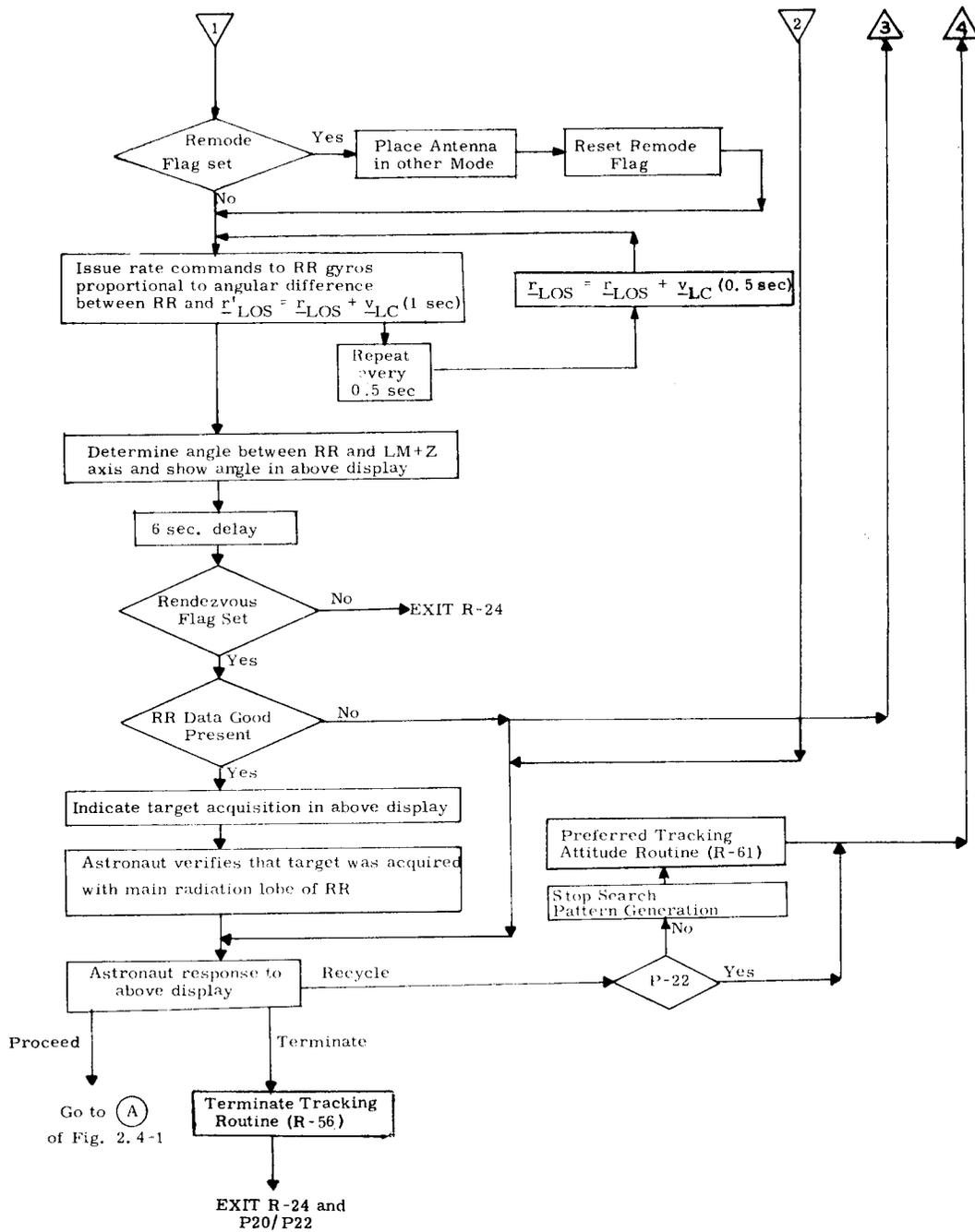


Figure 2.4-5 RR Search Routine
(page 2 of 2)

5. 2. 4. 2 Rendezvous Navigation Routine

5. 2. 4. 2. 1 RR Data Read Routine

During operation of the Rendezvous Navigation Routine use is made of the RR Data Read Routine (R-22 of Section 4) to obtain measurement data from the RR. The logic associated with the RR Data Read Routine is given in Fig. 2. 4-6. Like the Target Acquisition Routine, this routine is used by both the Rendezvous and Lunar Surface Navigation Programs, with different paths being taken at various points in the routine depending on which program (P-20 or P-22) is in operation. The RR Data Read Routine periodically obtains a complete set of data from the RR (range, range rate, shaft angle, and trunnion angle) for purposes of navigation, although only range and range rate data is used for updating during lunar surface navigation. When routine R-22 is used in program P-20, the Fine Preferred Tracking Attitude Routine (R-65 of Sections 4 and 5. 2. 4. 4) is called on a repetitive basis to obtain continuous or fine LM +Z-axis tracking of the CSM. Most of the details on continuous Z-axis tracking in program P-20 are given in Section 5. 2. 4. 4. During the operation of program P-20 the maximum frequency of update of the navigation equations with a complete set of RR data is about 64 seconds. This is based upon a rough estimate given in Section 5. 2. 4. 4.

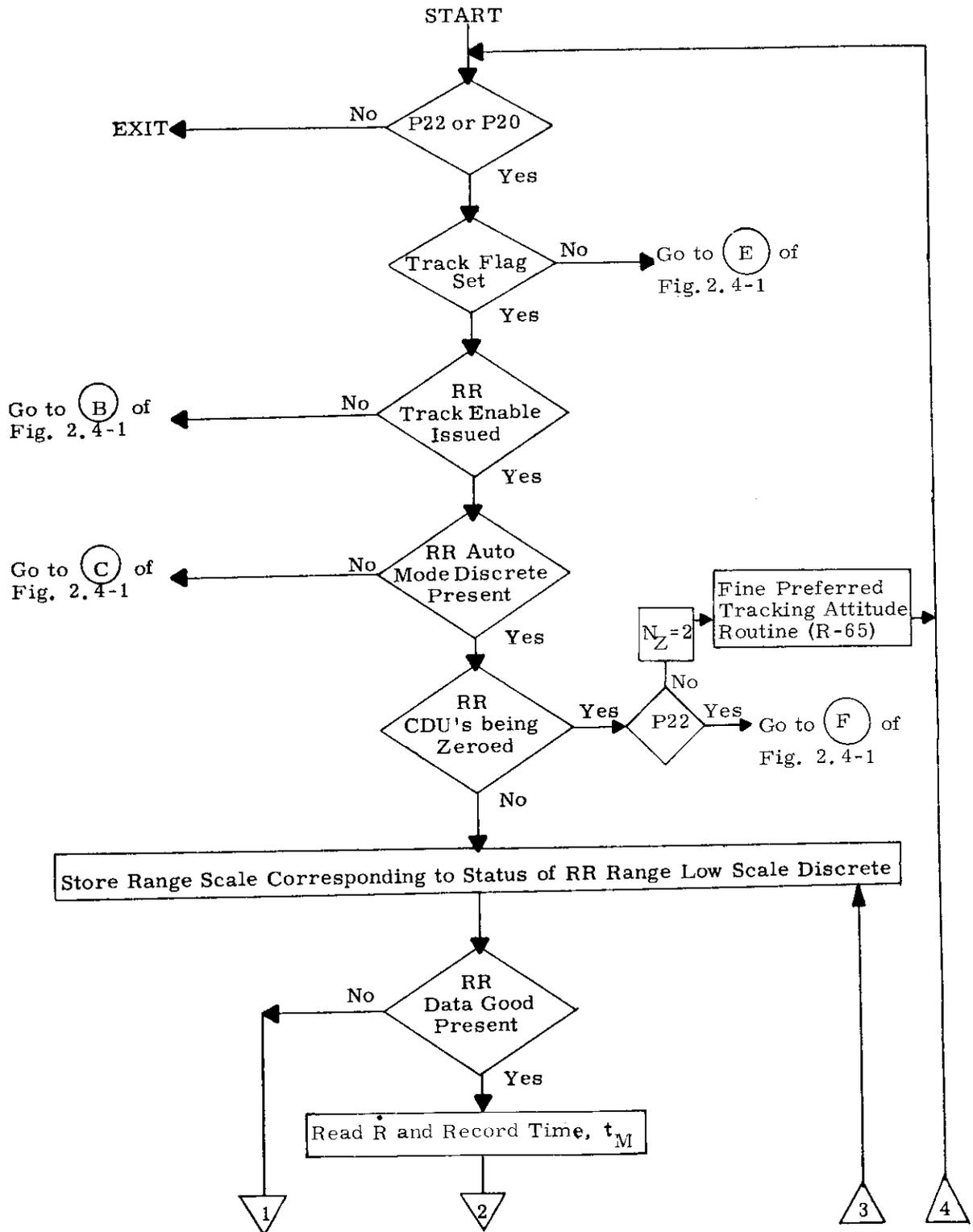


Figure 2.4-6 RR Data Read Routine (Page 1 of 3)

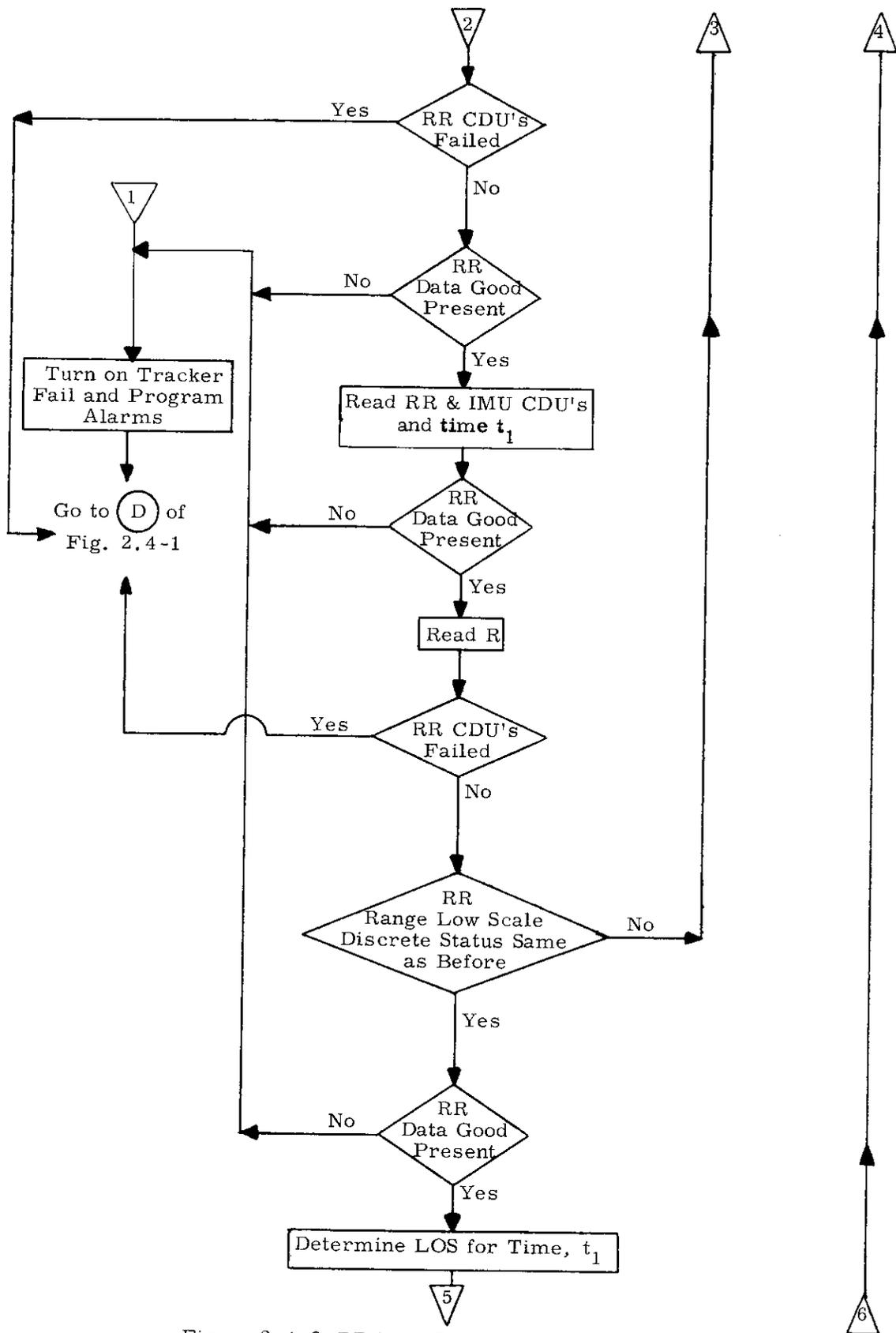


Figure 2.4-6 RR Data Read Routine (Page 2 of 3)

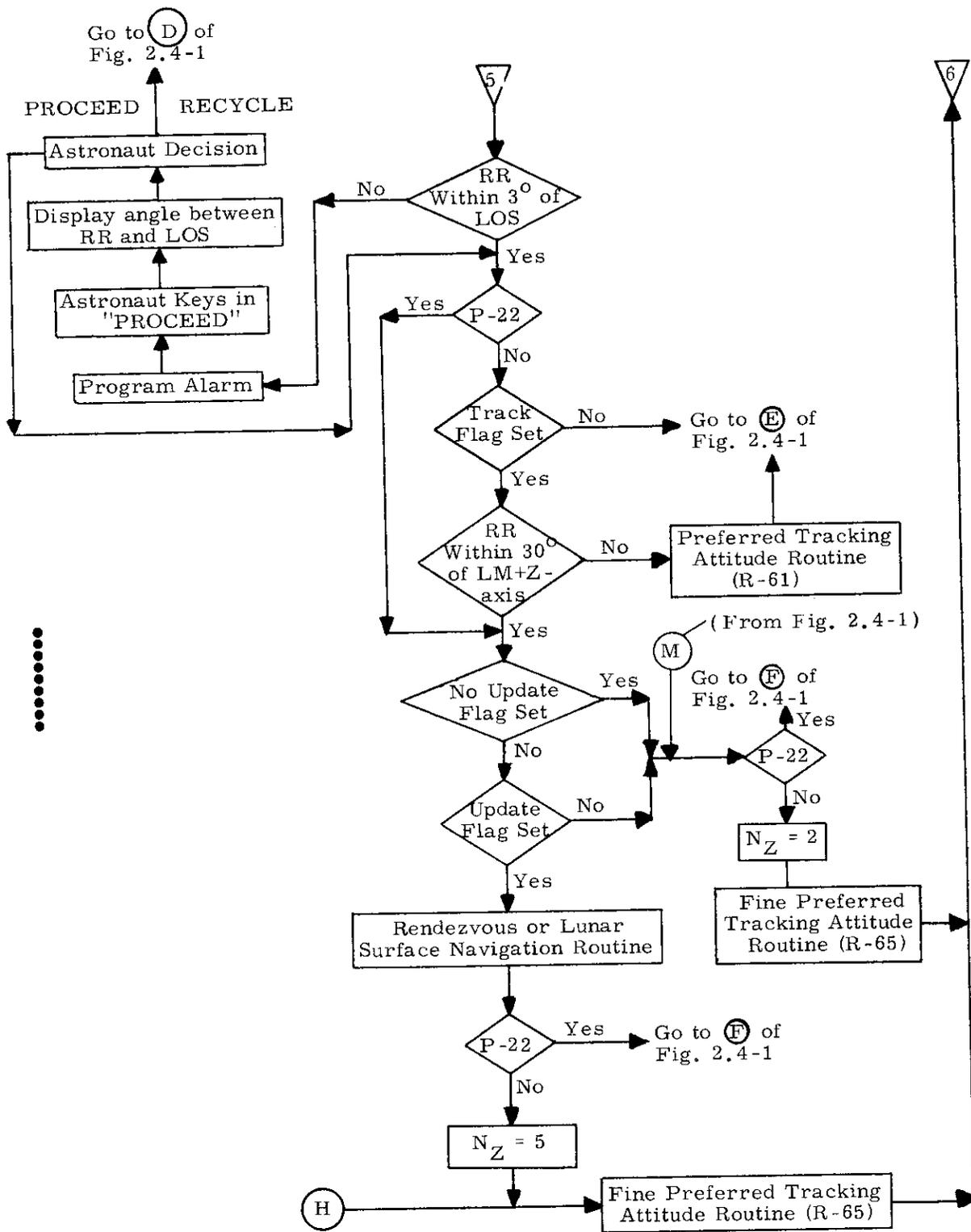


Figure 2.4-6 RR Data Read Routine
(Page 3 of 3)

In this section an explanation will be given only for those logic steps in Fig. 2.4-6 associated with the use of the RR Data Read Routine during operation of the Rendezvous Navigation Program (P-20).

In Fig. 2.4-6, the first check made by the routine, after establishing that P-20 or P-22 is being used, is to see if the Track flag is present. This flag is removed during the preparation and execution of a LM ΔV maneuver when there is no desire to have the routine request RR data or call any other routine such as the RR Designate and Fine Preferred Tracking Attitude Routines.

If the Track flag is present, the routine next checks for issuance of the RR Track Enable discrete and reception of the RR Auto Mode discrete. The former discrete is removed and not re-issued by the RR Monitor Routine whenever the RR antenna angles exceed the limits in Fig. 2.4-3. Its absence therefore indicates a need to go back to input (B) of Fig. 2.4-1 to re-establish the preferred tracking attitude and re-designate the radar. Afterwards, a check is made to insure that RR data is not taken if the RR CDU's are being zeroed. If the RR CDU's are being zeroed and program P-20 is in operation, the Fine Preferred Tracking Attitude Routine (R-65) of Section 5.2.4.4 is called with a specified value of 2 for the quantity N_Z , which indicates how many times routine R-65 is to repeat itself before returning to routine R-22.

The sequence used by the routine for reading RR data is shown in Fig. 2.4-6 where frequent checks are made on the RR Data Good discrete to insure that no RR tracking interruptions have occurred during the read-out. If this discrete is missing, the Tracker Fail and Program Alarm lights are turned on before going to point (D) of Fig. 2.4-1 to re-designate the RR. Note in Fig. 2.4-1 that checks are also made to see if the RR CDU's have failed during the read-out and, if so, a return will also be made to point (D) of Fig. 2.4-1, except that the Tracker Fail and Program Alarm lights are not turned on by the Data Read Routine, but are turned on by the RR Monitor Routine of Section 5.2.4.3.

Prior to and after reading the data, the routine checks the status of the Range Low Scale discrete to insure that the proper scaling is applied to the range data. For ranges below $9.38 \times 32,767$ feet, the RR issues the Range Low Scale discrete indicating to the LGC that the low scale factor should be used. If the status of this discrete should change during the data read process, it is seen in Fig. 2.4-6 that a new data request is immediately made by the routine.

It should be noted that a single time tag t_M is used for a complete set of RR data (range, range rate, shaft angle, and trunnion angle) in the navigation computations. This time (t_M) is the time recorded at the middle of the 80 millisecond counting interval for the range rate measurement.

After the RR data is read, the LOS is determined for time t_1 and compared with that indicated by the RR CDU's. This is done to insure against RR side-lobe lock-on. If the difference is more than 3 degrees, an alarm is issued to the astronaut who must then decide to go to the RR Designate Routine or request the LGC to proceed with the data.

If the Track flag is still present after the side-lobe check, the routine checks to see if the angle between the RR antenna and the LM +Z-axis is within 30 degrees. As mentioned previously, this is required in order for the LGC to perform a satisfactory determination of RR angle biases, and to also enable the LM Optical Beacon to be seen by the CSM. If the antenna is not within 30 degrees of the LM +Z-axis, the Preferred Tracking Attitude Routine (R-61) is called before returning to (E) of Fig. 2.4-1.

Afterwards, the LGC checks the Update and No Update flags. The Update flag is reset by various programs and routines whenever it is desired to temporarily stop the update of the navigation equations with RR data. The No Update flag is set by the astronaut whenever he wishes to permanently stop the update of the navigation equations with RR data. This flag is reset whenever the astronaut indicates that he wishes to update either the LM or CSM state vector. It should be noted that it is only necessary for the Track flag to be present in order to monitor RR tracking and read RR data even though the data may not be used for navigation updates.

The range obtained from the RR by the RR Data Read Routine is that measured by the RR between the LM and the CSM. This data is sent to the LGC from the RR as a binary data word R_{RR} . In the LGC the range r_{RR} in feet is obtained as follows:

$$r_{RR} = \begin{cases} k_{R1} & R_{RR} \\ k_{R2} & R_{RR} \end{cases}$$

where k_{R1} and k_{R2} are the bit weights respectively for the long and short range scales in order to obtain r_{RR} in feet. When the Range Low Scale discrete is being received from the RR by the LGC, k_{R2} is used.

The range rate data obtained from the RR by the RR Data Read Routine is in the form of a binary data word S_{RR} which represents the count in the RR of a frequency comprising both the doppler frequency and a bias frequency (f_{BRR}) over a time interval τ_{RR} . At present, τ_{RR} is given as 80,000 milliseconds. To obtain the range rate (\dot{r}_{RR}) in feet per second, the following computation is made:

$$\dot{r}_{RR} = k_{RR} (S_{RR} - f_{BRR} \tau_{RR})$$

where k_{RR} is the scale factor required to obtain the range rate in feet per second and is of such a polarity as to make \dot{r}_{RR} positive in the above equation for increasing range.

A summary of the processing constants required by the LGC for RR operation is given as follows:

f_{BRR}	Range rate bias frequency
τ_{RR}	Counting interval in RR for range rate measurements
k_{RR}	Scale factor to convert the range rate count obtained from the RR to feet per second for the counting interval τ_{RR} . The scale factor polarity is such as to make the converted result positive for increasing range.
k_{R1}	Bit weight in feet for long range scale.
k_{R2}	Bit weight in feet for short range scale.

5.2.4.2.2 Rendezvous Navigation Computations

During rendezvous phases RR navigation data are obtained by means of automatic rendezvous radar tracking of the CSM from the LM. These data are used to update the estimated six-dimensional state vector of either the LM or the CSM. The option of which state vector is to be updated by the RR tracking data is controlled by the astronaut as described in Section 5.2.1 and illustrated in Fig. 2.1-1. This decision will be based upon which vehicle's state vector is most accurately known, and upon which vehicle is performing the rendezvous. This process requires that the constant RR tracking angle biases be compensated for by estimating these biases along with the selected vehicle's state vector such that subsequent RR tracking angle data can be modified as shown in simplified form in Fig. 2.1-1.

This routine is used to process the CSM-tracking RR measurement data, and is used normally during lunar-orbit rendezvous in the lunar landing mission. The routine also can be used in earth orbit during alternate missions.

After the preferred LM attitude is achieved and RR tracking acquisition and lock-on is established (Section 5.2.4.1), the following tracking data are automatically acquired by the RR Data Read Routine (Section 5.2.4.2.1) at approximately one minute intervals:

Measured range, R_M

Measured range rate, \dot{R}_M

Measured shaft angle, β_M

Measured trunnion angle, θ_M

where the subscript M indicates the RR measured value. In addition to the above four measured quantities the time of the measurement and the three IMU gimbals angles are also recorded.

Although eight variables are estimated in the navigation procedure (six vehicle state-vector components and two RR angle biases), it is convenient to use the following nine-dimensional state vector:

$$\underline{x} = \begin{pmatrix} \underline{r} \\ \underline{v} \\ \delta\beta \\ \delta\theta \\ 0 \end{pmatrix} \quad (2.4.1)$$

where \underline{r} and \underline{v} are the estimated position and velocity vectors, respectively, of the selected vehicle (LM or CSM) which is being updated, $\delta\beta$ and $\delta\theta$ are the estimates of the biases in the RR shaft and trunnion angles, respectively, and the ninth coordinate is a dummy variable. This type of RR tracking angle bias is referred to as a boresight bias and is one of two types which may be defined. With the LM attitude restriction mentioned in Section 5.2.1, either type of angle bias (tilt or boresight) can be used, and the boresight type indicated in Eq. (2.4.1) is most convenient.

Let \underline{r}_L , \underline{v}_L , \underline{r}_C and \underline{v}_C be the estimated position and velocity vectors of the LM and CSM, respectively, at the time of the measurement. Then, the measurement error variances, $\overline{\alpha^2}$, the nine-dimensional geometry vectors, \underline{b} , and the measured deviations, δQ , for the range and range rate measurements are computed as follows:

Measured range, R_M

$$\underline{r}_{LC} = \underline{r}_C - \underline{r}_L$$

$$\underline{u}_{LC} = \text{UNIT}(\underline{r}_{LC})$$

$$\overline{\alpha^2} = \text{maximum}(r_{LC}^2 \text{var}_R, \text{var}_{Rmin})$$

(2.4.2)

$$\underline{b}_0 = \overline{\alpha} \underline{u}_{LC}$$

$$\underline{b}_1 = \underline{0}$$

$$\underline{b}_2 = \underline{0}$$

$$\delta Q = R_M - r_{LC}$$

where var_R is the RR range error variance corresponding to a percentage error, and var_{Rmin} is the minimum RR range error variance.

Measured range rate, \dot{R}_M

$$\underline{r}_{LC} = \underline{r}_C - \underline{r}_L$$

$$\underline{u}_{LC} = \text{UNIT}(\underline{r}_{LC})$$

$$\underline{v}_{LC} = \underline{v}_C - \underline{v}_L$$

(2.4.3)

$$\dot{r} = \underline{v}_{LC} \cdot \underline{u}_{LC}$$

$$\overline{\alpha^2} = r_{LC}^2 \text{maximum}(\dot{r}^2 \text{var}_V, \text{var}_{Vmin})$$

Measured range rate, \dot{R}_M (Continued)

$$\underline{b}_0 = \mp (\underline{u}_{LC} \times \underline{v}_{LC}) \times \underline{u}_{LC}$$

$$\underline{b}_1 = \mp \underline{r}_{LC}$$

(2.4.3 Continued)

$$\underline{b}_2 = \underline{0}$$

$$\delta Q = r_{LC} (\dot{R}_M - \dot{r})$$

where var_V is the RR range-rate error variance corresponding to a percentage error, and $\text{var}_{V\text{min}}$ is the minimum RR range-rate variance.

In Eqs. (2.4.2) and (2.4.3) the negative signs are selected if it is the LM state vector that is being updated, and the positive signs if it is the CSM state vector.

In order to process the RR angle data (β_M and θ_M), it is necessary to consider the relative orientations of the various coordinate systems. If \underline{u}_X , \underline{u}_Y and \underline{u}_Z are unit vectors along the X-, Y- and Z-axes, respectively, of the RR Measurement Coordinate System, then the measured shaft angle, β_M , and the measured trunnion angle, θ_M , are defined as shown in Fig. 2.4-7. In the figure, the vector \underline{r}_{XZ} is the projection of the measured LM-to-CSM line-of-sight vector on the XZ-plane.

The RR Measurement Coordinate System is coincident with the Navigation Base Coordinate System since all RR performance specifications are referenced to the PGNCS navigation base. The unit vectors \underline{u}_X , \underline{u}_Y and \underline{u}_Z are then given in the Basic Reference Coordinate System by

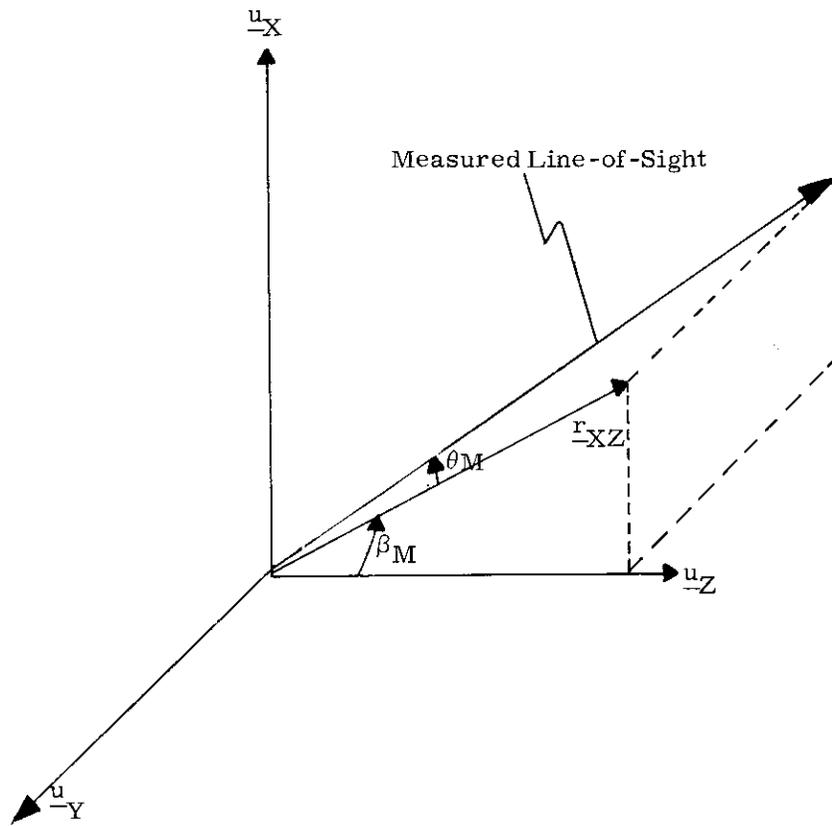


Figure 2.4-7 RR Measurement Coordinate System

$$\begin{pmatrix} \underline{u}_X^T \\ \underline{u}_Y^T \\ \underline{u}_Z^T \end{pmatrix} = [\text{SMNB}] [\text{REFSMMAT}] \quad (2.4.4)$$

where $[\text{SMNB}]$ and $[\text{REFSMMAT}]$ are transformation matrices as defined in Section 5.6.3 and the angles from which $[\text{SMNB}]$ is determined are the values of the IMU gimbal angles which were recorded at the measurement time.

The measurement error variances, $\overline{\alpha^2}$, the nine-dimensional geometry vectors, \underline{b} , and the measured deviations, δQ , for the shaft and trunnion angle measurements are computed as follows:

Measured shaft angle, β_M

$$\underline{r}_{LC} = \underline{r}_C - \underline{r}_L$$

$$\underline{u}_{LC} = \text{UNIT}(\underline{r}_{LC})$$

$$S = -\underline{u}_{LC} \cdot \underline{u}_Y$$

$$r_{XZ} = r_{LC} \sqrt{1 - S^2}$$

$$\overline{\alpha^2} = \text{var}_{\beta} + \text{var}_{\text{IMU}} \quad (2.4.5)$$

$$\underline{b}_0 = \frac{1}{r_{XZ}} \text{UNIT}(\underline{u}_Y \times \underline{u}_{LC})$$

$$\underline{b}_1 = \underline{0}$$

$$\underline{b}_2 = \begin{pmatrix} 1 \\ 0 \\ 0 \end{pmatrix}$$

$$\delta Q = \beta_M - \tan^{-1} \left(\frac{\underline{u}_X \cdot \underline{u}_{LC}}{\underline{u}_Z \cdot \underline{u}_{LC}} \right) - \delta\beta$$

where var_β is the RR shaft-angle error variance and var_{IMU} is the IMU angular error variance per IMU axis.

Measured trunnion angle, θ_M

$$\underline{r}_{\text{LC}} = \underline{r}_{\text{C}} - \underline{r}_{\text{L}}$$

$$\underline{u}_{\text{LC}} = \text{UNIT}(\underline{r}_{\text{LC}})$$

$$S = -\underline{u}_{\text{LC}} \cdot \underline{u}_{\text{Y}}$$

$$r_{\text{XZ}} = r_{\text{LC}} \sqrt{1 - S^2}$$

$$\overline{\alpha^2} = \text{var}_\theta + \text{var}_{\text{IMU}}$$

$$\underline{b}_0 = \mp \frac{1}{r_{\text{XZ}}} (\underline{u}_{\text{Y}} \times \underline{u}_{\text{LC}}) \times \underline{u}_{\text{LC}}$$

(2.4.6)

$$\underline{b}_1 = \underline{0}$$

$$\underline{b}_2 = \begin{pmatrix} 0 \\ 1 \\ 0 \end{pmatrix}$$

$$\delta Q = \theta_M - \sin^{-1}(S) - \delta\theta$$

where var_θ is the RR trunnion-angle error variance.

In Eqs. (2.4.5) and (2.4.6) the negative signs are used if it is the LM state vector that is being updated, and the positive signs if it is the CSM state vector.

The data are incorporated into the state vector estimates by means of four calls to the Measurement Incorporation Routine (Section 5.2.3). The updated components of the nine-dimensional state vector resulting from each incorporation are used as initial conditions for the next update in the sequence.

Included in each use of the Measurement Incorporation Routine is the state vector update validity check, as described in Section 5.2.1 and illustrated in Fig. 2.4-8. Note that if δr and δv are displayed, a Source Code is also displayed indicating which of the four RR measurement parameters is responsible for the δr and δv . The Source Code is set to 1, 2, 3 and 4, respectively, for RR range, range rate, shaft angle and trunnion angle. In some cases, the identity of the RR measurement source is useful in indicating the type of corrective action required to remedy the situation.

The results of the processing of the RR measurement data are updated values of the estimated position and velocity vectors of the CSM or the LM and estimates of the RR angle biases. The two estimated vehicle state vectors are used to compute required rendezvous targeting parameters as described in Section 5.4.4

For convenience of calculation in the LGC, Eqs. (2.4.5) and (2.4.6) are reformulated and regrouped as follows:

Preliminary Radar Angle Calculation

$$\begin{aligned}
 \underline{r}_{LC} &= \underline{r}_C - \underline{r}_L \\
 \underline{u}_{LC} &= \text{UNIT}(\underline{r}_{LC}) \\
 S &= -\underline{u}_{LC} \cdot \underline{u}_Y \\
 r_{XZ} &= r_{LC} \sqrt{1 - S^2}
 \end{aligned}
 \tag{2.4.7}$$

Measured shaft angle, β_M

$$\begin{aligned}
 \overline{\alpha^2} &= r_{XZ}^2 (\text{var}_\beta + \text{var}_{\text{IMU}}) \\
 \underline{b}_0 &= \mp \text{UNIT}(\underline{u}_Y \times \underline{u}_{LC}) \\
 \underline{b}_1 &= \underline{0} \\
 \underline{b}_2 &= \begin{pmatrix} r_{XZ} \\ 0 \\ 0 \end{pmatrix} \\
 \delta Q &= r_{XZ} \left[\beta_M - \tan^{-1} \left(\frac{\underline{u}_X \cdot \underline{u}_{LC}}{\underline{u}_Z \cdot \underline{u}_{LC}} \right) - \delta\beta \right]
 \end{aligned}
 \tag{2.4.8}$$

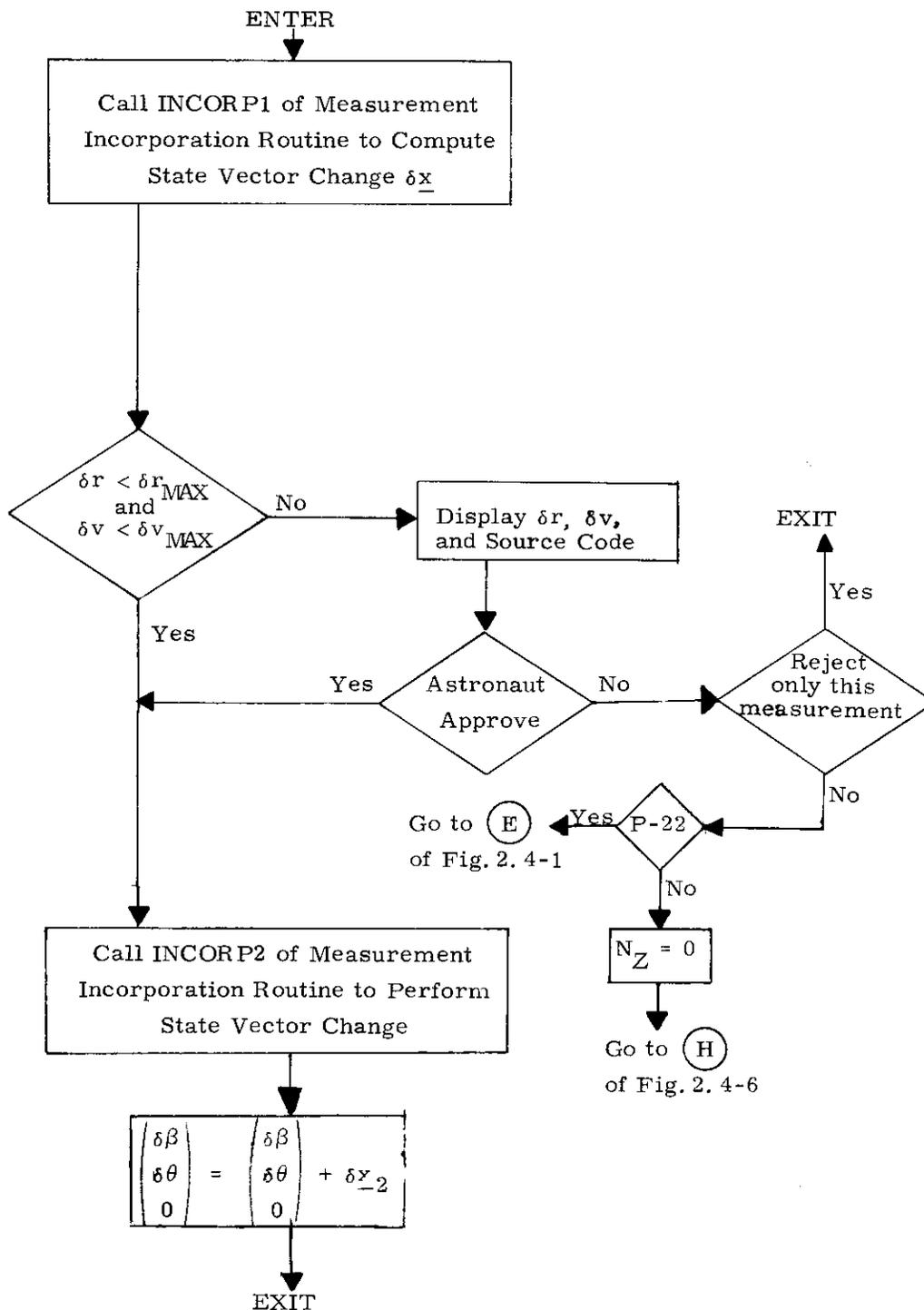


Figure 2.4-8 Rendezvous Navigation Measurement Incorporation Procedure

Measured trunnion angle, θ_M

$$\begin{aligned} \overline{\alpha^2} &= r_{XZ}^2 (\text{var}_{\theta} + \text{var}_{\text{IMU}}) \\ \underline{b}_0 &= \overline{\mp} (\underline{u}_Y \times \underline{u}_{\text{LC}}) \times \underline{u}_{\text{LC}} \\ \underline{b}_1 &= \underline{0} \\ \underline{b}_2 &= \begin{pmatrix} 0 \\ r_{XZ} \\ 0 \end{pmatrix} \\ \delta Q &= r_{XZ} \begin{bmatrix} \theta_M - \sin^{-1}(S) & -\delta\theta \end{bmatrix} \end{aligned} \tag{2.4.9}$$

The procedure for performing the rendezvous navigation computations is illustrated in Figs. 2.4-9 and 2.4-10. It is assumed that the following items are stored in erasable memory at the start of the computation shown in Fig. 2.4-9:

- \underline{x}_C = Estimated CSM state vector as defined in Section 5.2.2.6.
- \underline{x}_L = Estimated LM state vector.
- W = Six-dimensional error transition matrix associated with \underline{x}_C or \underline{x}_L as defined in Section 5.2.2.4.

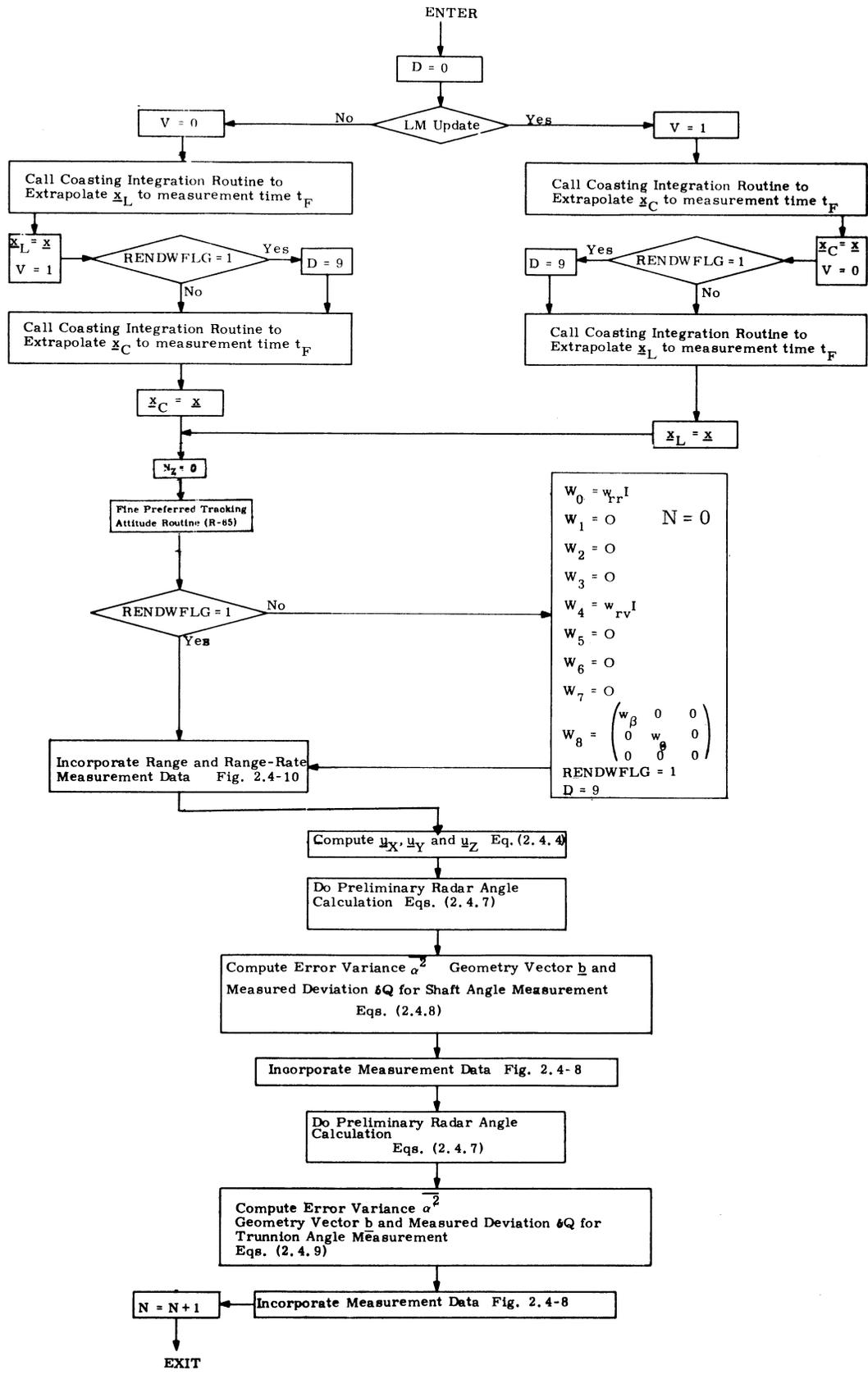


Figure 2.4-9 Rendezvous Navigation Computation Logic Diagram

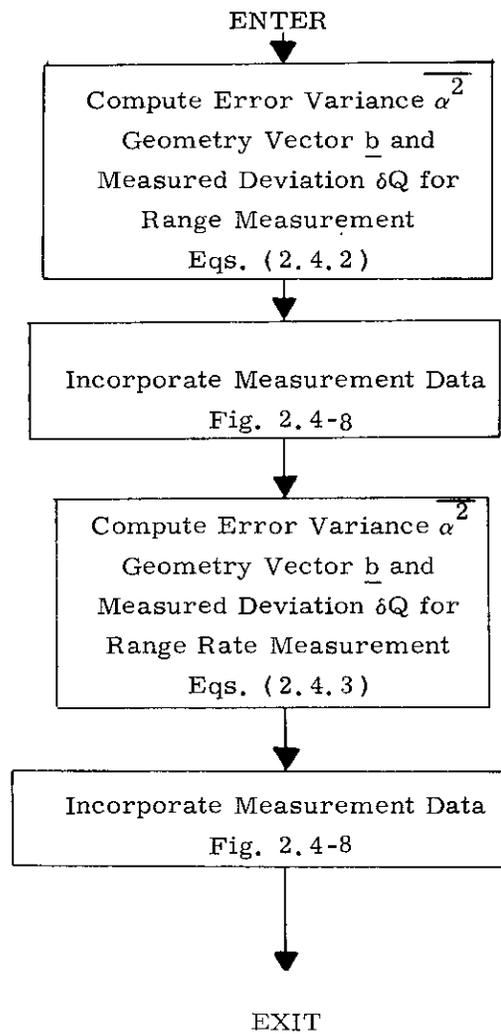


Fig. 2.4-10 RR Measured Range and Range Rate Incorporation

RENDWFLG = $\begin{cases} 1 & \text{for valid W-matrix} \\ 0 & \text{for invalid W-matrix} \end{cases}$

This flag or switch is maintained by programs external to the Rendezvous Navigation Routine. It indicates whether or not the existing W-matrix is valid for use in processing RR tracking data. The flag is set to zero after each of the following procedures:

- 1) State vector update from ground
- 2) Astronaut command
- 3) Overflow of W-matrix integration
- 4) When new W-matrix initialization values loaded by V67
- 5) Lunar ascent

[REFSMMAT] = Transformation Matrix: Basic Reference Coordinate System to IMU Stable Member Coordinate System.

N = Number of sets of rendezvous navigation data already processed since the last maneuver, (set to 0 when W-matrix is initialized).

t_F = Measurement time.

$R_M, \dot{R}_M, \beta_M, \theta_M$ = RR measurement data

$w_{RR}, w_{RV}, w_\beta, w_\theta$ = Preselected W-matrix initial diagonal elements

5.2-73

Revised
 Added

LUMINARY

GSOP # 567

PCR # 561*

Rev. 7

Date 11-69

Three IMU gimbal angles

Vehicle update mode

SURFFLAG = $\left\{ \begin{array}{l} 1 \text{ indicates RR angle measurements} \\ \quad (\beta_M \text{ and } \theta_M) \text{ are not to be used} \\ \\ 0 \text{ indicates RR angle measurements} \\ \quad (\beta_M \text{ and } \theta_M) \text{ are to be used} \end{array} \right.$

The term "SURFFLAG" is an abbreviated notation for the lunar surface flag; it is set to 1 by P-68 just after lunar landing and remains this value until launch. This flag indicates when the LM is on the lunar surface and is checked by a number of programs and routines.

The variables D and V are indicators which control the Coasting Integration Routine (Section 5.2.2) as described in Section 5.2.2.6, and I and O are the three-dimensional identity and zero matrices, respectively.

At the beginning of a new rendezvous sequence, the estimates of the RR angle biases, $\delta\beta$ and $\delta\theta$, may be initialized by the astronaut by direct addressing of erasable memory, and the W matrix is initialized by the program. During the remainder of the rendezvous sequence, $\delta\beta$ and $\delta\theta$ should not be re-initialized by the astronaut even though there may be additional W matrix initializations.

The RR-measurement-data incorporation procedure outlined above is repeated at approximately one-minute intervals throughout the rendezvous phase except during powered maneuvers. If the LM is the passive vehicle, the CSM rendezvous maneuvers are voice-linked to the LM as an ignition time and three velocity components in a CSM local vertical coordinate system, and then entered as updates to the estimated CSM state vector in the LGC. Upon receipt of these data, RR tracking and data processing should be suspended until after the maneuver. The update is accomplished by means of the Target ΔV Program, P-76 (Section 5.6.16). If the LM is the active vehicle, the estimated LM state vector is updated by means of the Average-G Routine (Section 5.3.2) during the maneuver.

5. 2. 4. 3 RR Monitor Routine

The logic associated with the RR Monitor Routine (R-25) is given in Fig. 2. 4-11. This routine is initiated every 0.48 seconds by an automatic program interrupt and monitors various items such as the RR Auto Mode discrete, the RR CDU Fail discrettes, and the angular excursions of the RR antenna.

Whenever the RR Auto Mode discrete changes status this routine resets various flags to insure proper initiation or termination of various radar control functions within the LGC. If the RR Auto Mode discrete has just been received from the RR as the result of placing the RR mode control switch in the LGC position, a turn-on sequence is initiated which zeroes the RR CDU's, determines the present RR antenna mode, and updates the Tracker Fail Light. The criterion used to determine the present RR antenna mode is the following:

$$\text{Mode 1: } 270^{\circ} \leq T \leq 90^{\circ}$$

$$\text{Mode 2: } 270^{\circ} > T > 90^{\circ}$$

where T is the RR antenna trunnion angle defined in Figs. 6. 15-1 and 2. 4-3. Note in Fig. 2. 4-11 that the resulting antenna mode determination is indicated by the RR Antenna Mode Flag. When the RR Auto Mode discrete is removed by the RR, routine R-25 removes the RR Error Counter Enable from the RR CDU's to insure that no commands are being sent to the RR gyros.

If there is an RR CDU failure while the RR Auto Mode discrete is present, it is seen in Fig. 2. 4-11 that the Tracker Fail Light is turned on. If P-20 or P-22 is in operation (i. e. the Rendezvous Flag is set), the Program Alarm Light is also turned on and an alarm code is stored.

After checking for RR CDU failure, it is seen in Fig. 2. 4-11 that a number of conditions must be met before the routine will check to see if the RR antenna angles are within the tracking limits of the present mode (see Fig. 2. 4-3). If all the conditions are met and the RR antenna is not within the present mode tracking limits, the routine will remove the RR Track Enable discrete from the RR, and cause the RR to be repositioned to the reference direction (see Fig. 2. 4-3) for the present mode. During the repositioning of the RR this routine sets the Reposition flag to indicate to other programs and routines that this is taking place.

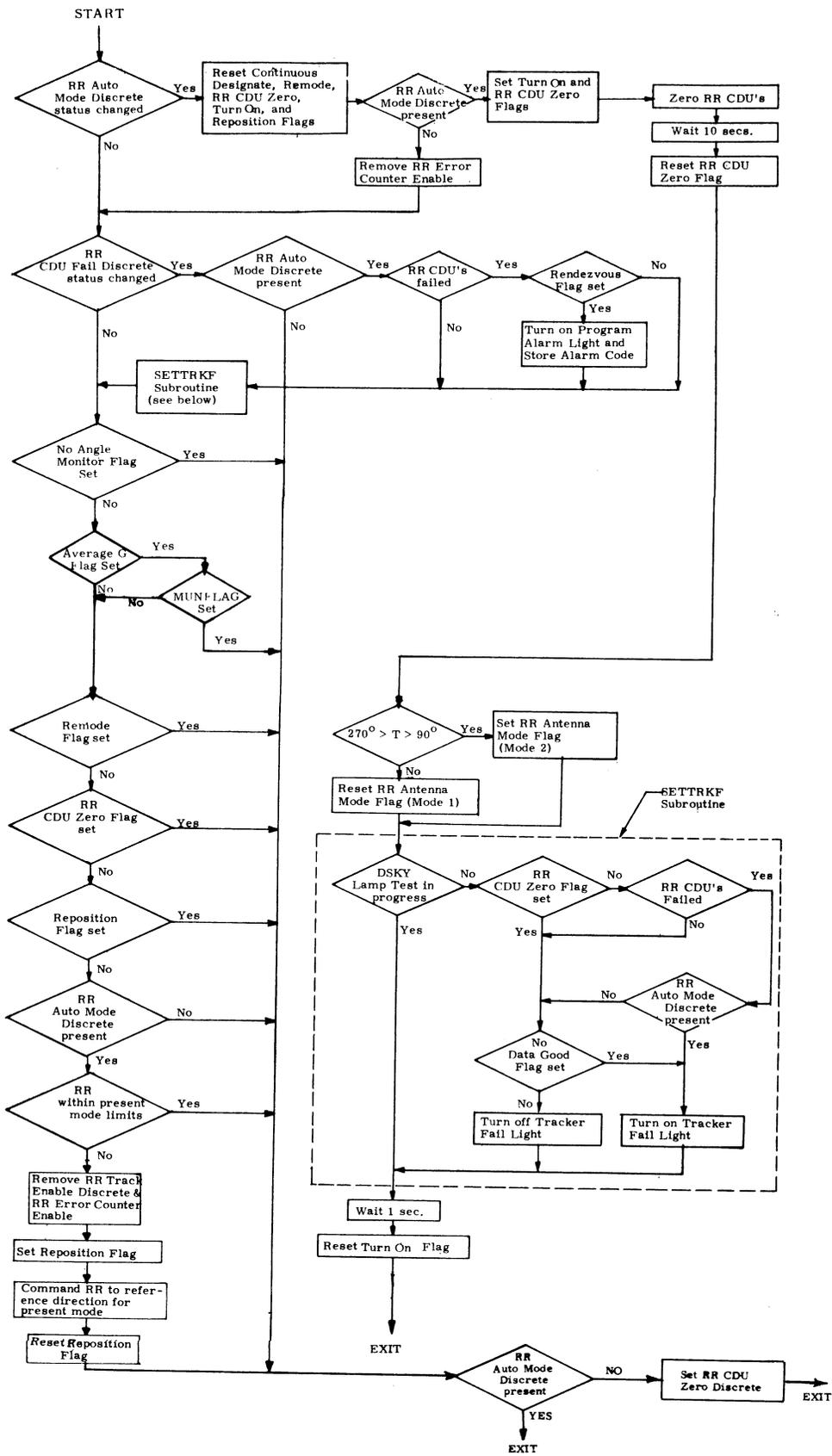


Figure 2.4-11 RR Monitor Routine
5. 2-76

5.2.4.4 Preferred Tracking Attitude Routines

In the Rendezvous Navigation Program (P-20) use is made of two routines for aligning the LM +Z-axis with the LOS to the CSM. These routines are denoted as the Preferred Tracking Attitude Routine (R-61) and the Fine Preferred Tracking Attitude Routine (R-65). Both of these routines perform the alignment in the same manner except that routine R-65 is capable of repeating the alignment a specified number of times before returning to the program or routine which called it. Whenever continuous or fine Z-axis tracking is desired in program P-20, routine R-65 is called on a repetitive basis in order to stay within the minimum impulse limit cycle of the autopilot.

The logic associated with routines R-61 and R-65 is shown in Fig. 2.4-12 where it is seen that a check is made to insure that the Track flag is set before computing the unit vector \underline{u}_{LOS} defining the LOS to the CSM in stable member coordinates and the angle ϕ between \underline{u}_{LOS} in navigation base coordinates and the unit vector \underline{z}_{NB} defining the +Z-axis of the Navigation Base Coordinate System. Using \underline{u}_{LOS} and \underline{z}_{NB} the Vecpoint Routine computes the new desired IMU gimbal angles (using the present desired IMU gimbal angles so as to prevent roll about desired vector). If the Attitude Control Switch is not in the Auto position, the desired IMU gimbal angles are converted to angular readings for display on the FDAI Ball by the method given in Section 5.6.12. If the Attitude Control Switch is in the Auto Position, it is seen in Fig. 2.4-12 that a check is made on the magnitude of ϕ with respect to 15 degrees to determine whether the vehicle attitude should be corrected by issuing the desired IMU gimbal angles to the RCS DAP or by using the Attitude Maneuver Routine (R-60 of Section 4).

5.2-77

<input checked="" type="checkbox"/>	Revised	LUMINARY					
<input type="checkbox"/>	Added	GSOP #	<u>567</u>	PCR #	<u>642</u>	Rev.	<u>7</u> Date <u>11-69</u>

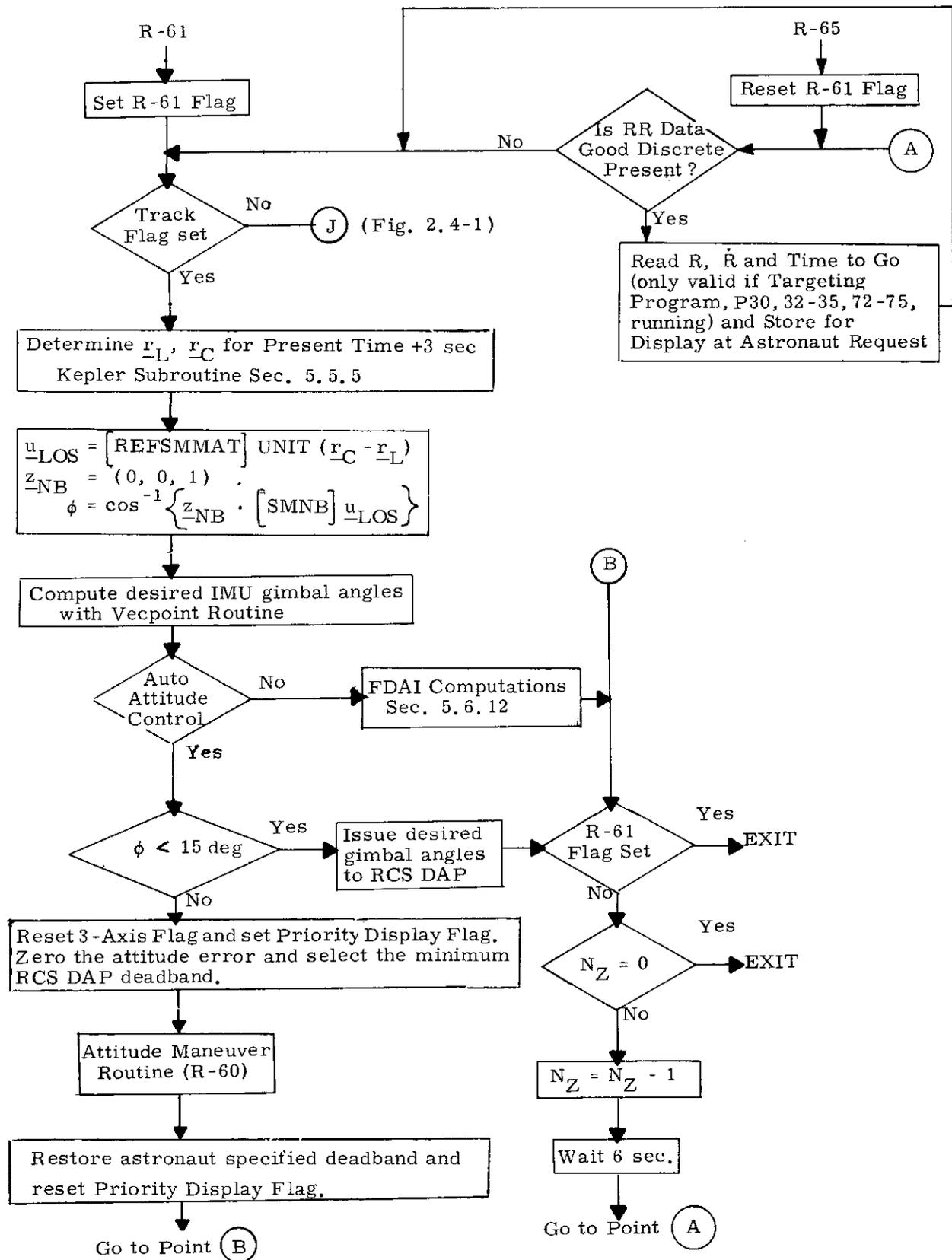


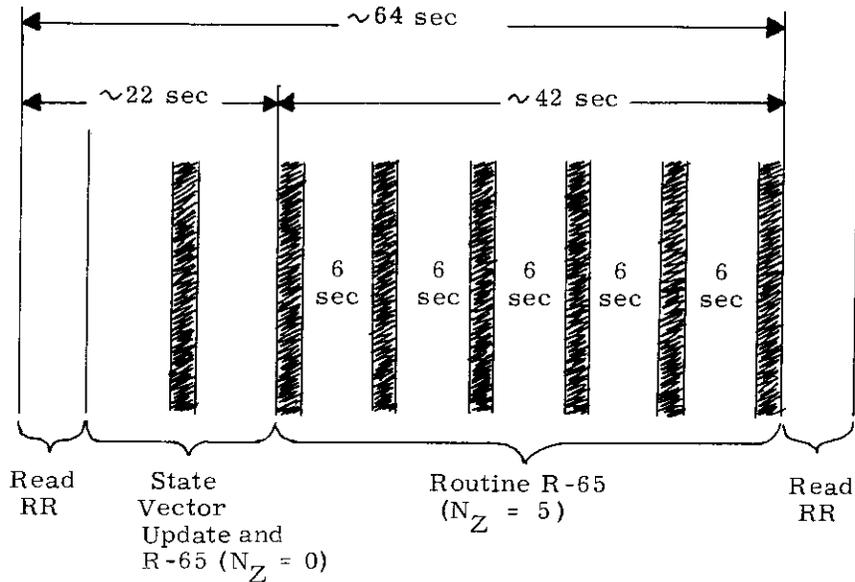
Fig. 2.4-12 Preferred Tracking Attitude Routines

5.2-78

In Fig. 2.4-12 it is seen that only one alignment is performed when routine R-61 is called. However, when routine R-65 is called, it is possible to have the alignment repeated N_Z times before returning to the program or routine which called routine R-65. When N_Z is greater than zero, the frequency of repetition of routine R-65 is approximately once every 8 seconds which is based upon a rough estimate of 2 seconds to perform the LOS and Vecpoint computations, and a 6 second wait period. The value of N_Z must be specified whenever routine R-65 is called.

In the target acquisition phase of program P-20 (see Section 5.2.4.1) only routine R-61 is used to align the LM +Z-axis to the LOS. However, during the operation of the RR Data Read Routine, routine R-65 is used to obtain continuous or fine Z-axis tracking. This is accomplished by automatically calling routine R-65 with $N_Z = 0$ near the middle of the state vector update computations of the Rendezvous Navigation Routine (see Fig. 2.4-9) and at the end of the RR Data Read Routine (see Fig. 2.4-6) with N_Z either equal to 0 or 5. During regular navigation updates with RR data the value of N_Z used at the end of the RR Data Read Routine is 5. However, if a proposed state vector correction exceeds the limits δr_{MAX} and δv_{MAX} in Fig. 2.4-8 and the astronaut rejects the entire mark set, it is seen that N_Z is set to 0. A timeline depicting the operation of routine R-65 with N_Z set to 5 at the end of the RR Data Read Routine is shown in Fig. 2.4-13. Note that the frequency of reading the RR and updating the state vector is approximately once every 64 seconds. This frequency is based upon a rough estimate* of 22 seconds to read the RR, perform the state vector update computations, and use routine R-65 with $N_Z = 0$ in the Rendezvous Navigation Routine, and a rough estimate* of 42 seconds to use routine R-65 with $N_Z = 5$ at the end of the RR Data Read Routine.

*The time to complete these calculations is highly dependent on other computational priorities. No explicit control of this procedure is exercised.



Note: Shaded intervals represent operation of Routine R-65 and are approximately 2 seconds duration each.

Fig. 2.4-13 Timeline for Fine LM +Z-Axis Tracking in Program P-20

In addition to program P-20, the Fine Preferred Tracking Attitude Routine (R-65) is also used to obtain continuous or fine Z-axis tracking in the Preferred Tracking Attitude Program (P-25 of Section 4). Program P-25 is used in place of program P-20 to keep the LM +Z-axis along the LOS to the CSM so that the CSM can optically track the LM optical beacon when the RR is not being used for navigation. This beacon is centered with respect to the LM +Z-axis and has a beamwidth of approximately 60 degrees. At the beginning of program P-25 the Track flag is set and a check is made to insure that the Attitude Control Switch is in the Auto position. Afterwards, routine R-65 is called with a specified value of 7 for N_Z , which causes routine R-65 to periodically perform the Z-axis alignment 8 times (see Fig. 2.4-12) before returning to program P-25 where a check is made on the Track flag. If the Track flag has been reset by some other program, program P-25 will wait until the Track flag is restored before recalling routine R-65.

5. 2. 5 RR LUNAR SURFACE NAVIGATION PROGRAM

5. 2. 5. 1 General Comments

The primary purpose of the RR Lunar Surface Navigation Program P-22 is to allow the LGC to update the CSM state vector using rendezvous radar (RR) tracking data prior to lunar launch in those cases in which a CSM state vector update cannot be obtained from the CMC or RTCC. This program essentially provides the LGC with a self contained capability of computing the parameters required for IMU alignment (P-57).

The RR Lunar Surface Navigation Program is normally used during at least one CSM orbital overpass prior to the intended launch orbit. A required astronaut input to program P-22 is the approximate expected lunar launch time. In normal operations the CSM performs a plane change maneuver at least two orbits prior to LM launch such that the CSM orbital plane contains the launch site vector at launch. In the case of loss of communications when P-22 is used to update the CSM state vector, the astronaut has an option to assume such a plane change maneuver has or has not been made. This decision is normally based on recorded block data for this phase of the mission. The program then directs the RR for initial acquisition and tracking. After RR lock-on and tracking have been established the navigation program uses range and range rate radar data to update the LGC estimate of the CSM orbital state vector. The operations mentioned above are described in more detail in the following sections.

5. 2. 5. 2 CSM Orbital Plane Change Estimation Routine

As mentioned in Section 5. 2. 5. 1, one of the purposes of this routine is to check that the CSM lunar orbit plane change maneuver has been accounted for in the estimated CSM state vector. In order to estimate the actual CSM maneuver, the estimated CSM orbital plane is rotated to contain the landing site at the launch time before the first RR data is used to update the CSM state vector. This procedure can be bypassed if the astronaut determines that the rotation should not be performed, as described in Section 5. 2. 5. 1.

The geometry associated with the rotation is illustrated in Fig. 2. 5-1 and the computation logic and the definitions of all variables are given in Fig. 2. 5-2.

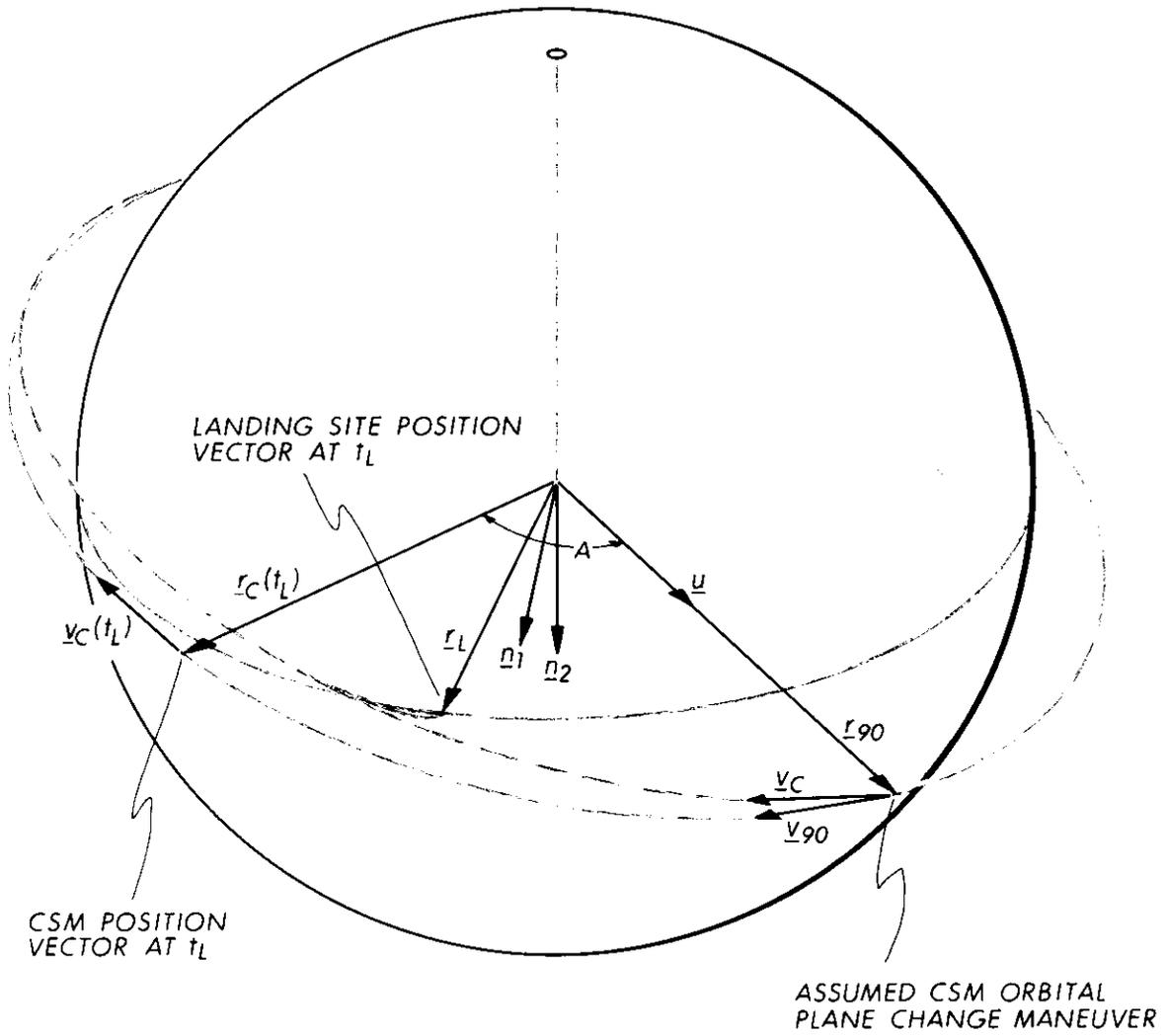


Fig. 2. 5-1 RR Lunar Surface Navigation CSM Orbital Plane Change Estimation

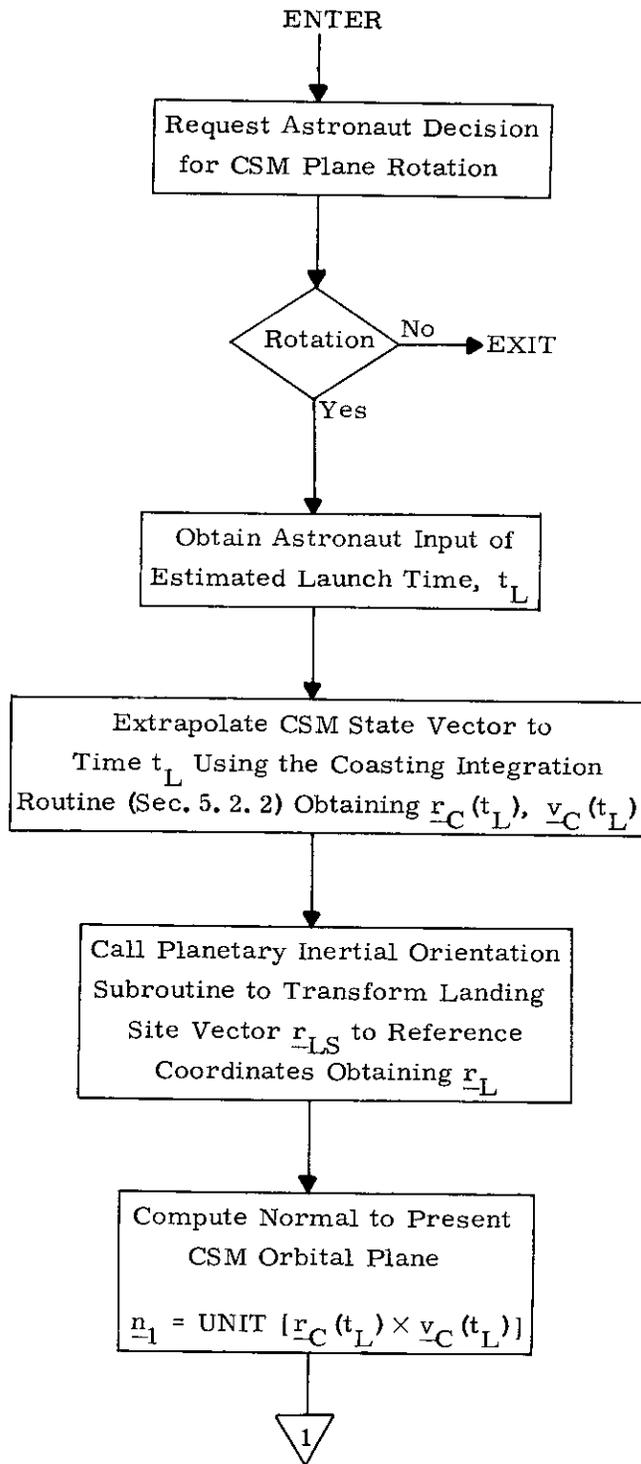


Figure 2.5-2 CSM Orbital Plane Change Estimation Routine Logic Diagram (page 1 of 2)

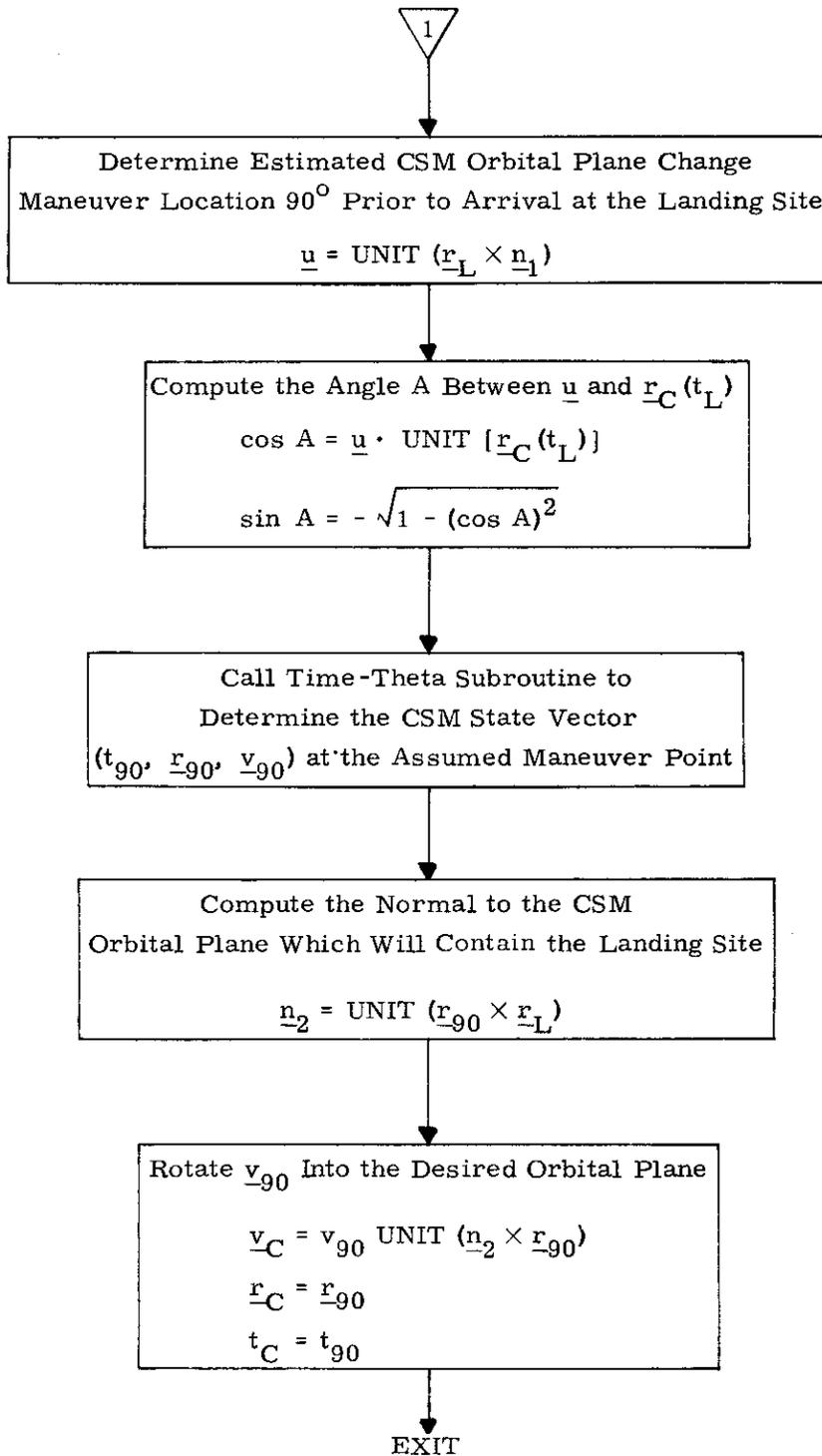


Figure 2.5-2 CSM Orbital Plane Change Estimation
Routine Logic Diagram (page 2 of 2)

5.2.5.3 Target Acquisition Routine

Since acquisition of the CSM with the LM Rendezvous Radar (RR) during lunar surface navigation is essentially the same as during rendezvous navigation, the same Acquisition Routine (see Fig. 2.4-1) is used for both except for slight differences in operation. Note in Fig. 2.4-1 that of the three modes (RR LGC, RR Manual, and RR Search) available for controlling the RR in target acquisition, only two (RR LGC and RR Search) are used in the Lunar Surface Navigation Program (P-22). In addition, the RR Search Mode can be selected only after the RR LGC Mode has failed to acquire the target.

The manner in which target acquisition is achieved with the RR LGC Mode is given in Figs. 2.4-2 and 2.5-3. Initially, steps are taken in Fig. 2.4-2 to insure that the RR antenna is in Mode 2. Afterward, the LOS to the CSM is computed and a check is made to see if it is within the angular tracking limits of Mode 2 (see Fig. 2.4-3). If the LOS is not within the Mode 2 tracking limits, the Lunar Surface RR Pre-Designate Routine (R-26) of Fig. 2.5-3 is used to determine whether the LOS will be inside the limits within approximately the next 10 minutes. This is accomplished by advancing the time t_{LOS} in 10-second increments, computing the LOS vector (\underline{r}_{LOS}) for each value of t_{LOS} , and checking to see if \underline{r}_{LOS} is within the tracking limits of the present antenna mode (i. e., Mode 2). When a successful value of \underline{r}_{LOS} is found, the routine advances t_{LOS} 10 more seconds, computes the corresponding \underline{r}_{LOS} , and starts designating the RR along \underline{r}_{LOS} until the present time equals t_{LOS} . Subsequently, the RR Designate Routine in Fig. 2.4-2 is re-entered via P-22 and starts designating the RR toward the CSM. When the RR is being designated to the CSM, the LOS is periodically updated by updating the CSM position and velocity vectors (\underline{r}_C and \underline{v}_C) with the Kepler Subroutine (Section 5.5.5). However, the LM position and velocity vectors (\underline{r}_L and \underline{v}_L) used in the computation of the LOS are the vectors computed just before entering the RR Designate Routine rather than updated ones. The reason for not updating \underline{r}_L and \underline{v}_L during the designation is that updating is time consuming and results in very little improvement in LOS accuracy because of the small amount of lunar rotation during the operation of the RR Designate Routine. On the lunar surface, the LM position and velocity vectors (\underline{r}_L and \underline{v}_L) are obtained by transforming the stored landing site position vector \underline{r}_{LS} and the vector (0, 0, 1) from moon-fixed to basic reference coordinates with the Planetary Inertial Orientation Subroutine (Section 5.5.2) where \underline{r}_L and \underline{u}_Z are the respective vectors in basic reference coordinates and \underline{v}_L is computed as follows:

$$\underline{v}_L = \omega_M \underline{u}_Z \times \underline{r}_L$$

where ω_M is the rotational rate of the moon with respect to inertial space.

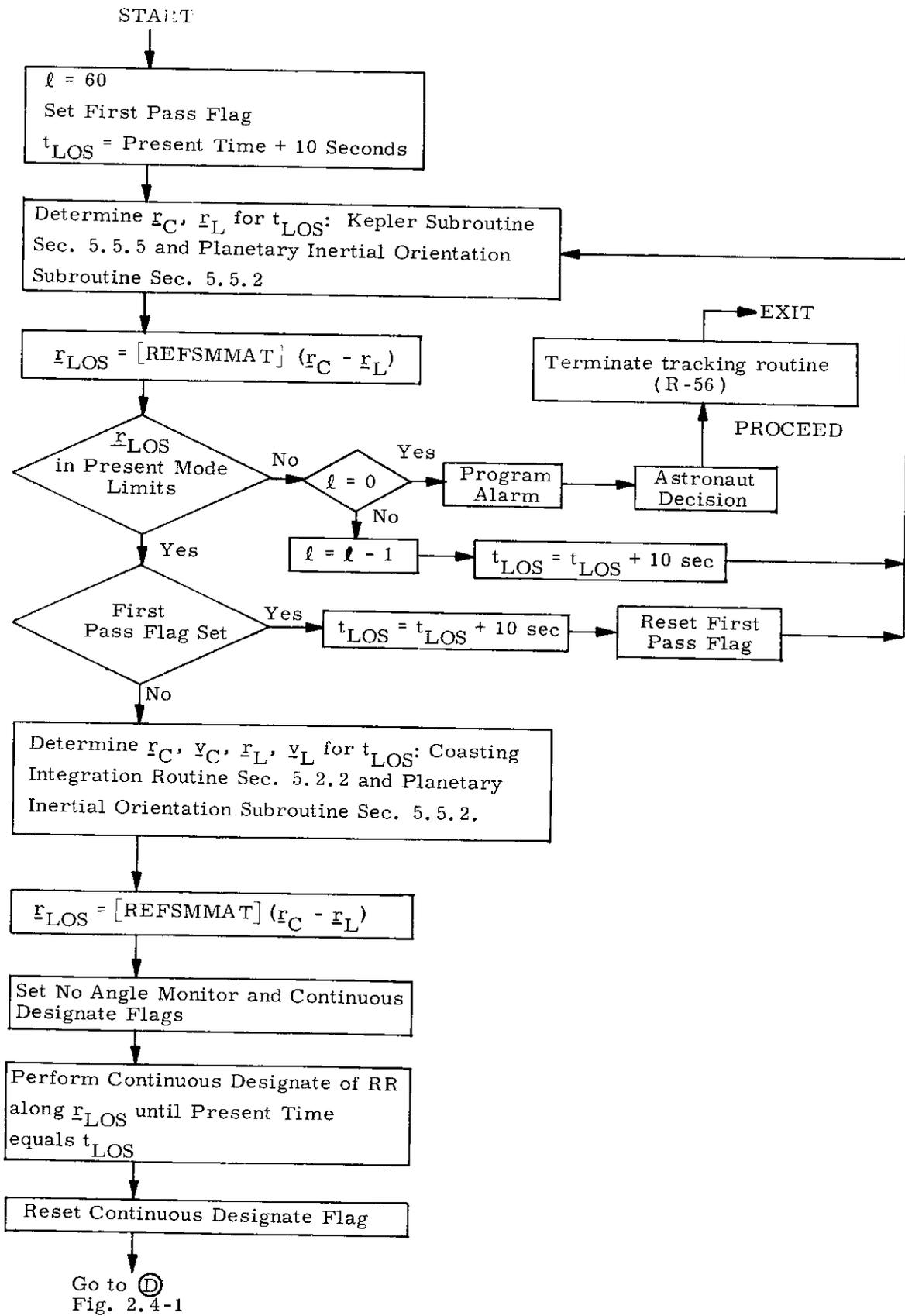


Fig. 2.5-3 Lunar Surface RR Pre-Designate Routine

If the RR Designate Routine is unable to acquire the target after designating the RR for a certain period of time, a program alarm is issued, whereupon the astronaut may either re-initiate the RR Designate Routine or proceed to the RR Search Mode as indicated in Fig. 2.4-2. In Figs. 2.4-1 and 2.4-5 it is seen that the logic associated with the RR Search Mode for program P-22 is essentially the same as for program P-20 except for the differences which have already been described for the RR Designate Routine such as the RR antenna mode used, the manner in which the LM position and velocity vectors are computed, and the nature of the test made on the LOS vector \underline{r}_{LOS} with respect to the antenna mode limits. In addition, it is seen in Fig. 2.4-5 that the Preferred Tracking Attitude Routine is not used during program P-22.

5. 2. 5. 4 Lunar Surface Navigation Routine

After target acquisition has been achieved with the Target Acquisition Routine (Section 5. 2. 5. 3), the RR data is read by the RR Data Read Routine of Section 5. 2. 4. 2. 1 and used by the Lunar Surface Navigation Routine to update the navigation equations. In Fig. 2. 4-6 it is seen that the operation of the RR Data Read Routine is essentially the same for both the Rendezvous and Lunar Surface Navigation Programs (P-20 and P-22) except that a 30° check is not made on the direction of the RR during program P-22. The maximum frequency of update of the navigation equations in program P-22 is about once every 13 seconds. This frequency is based upon a rough estimate* of 11 seconds to read the RR and perform the navigation computations and a 2 second wait period before repeating the process. During the no-update mode in P-22 (i. e. , when the No Update flag is set or the Update flag is not set), the frequency at which a complete set of RR data is read for downlink transmission is about 3.5 seconds -- 1.5 sec to read the RR data and perform the 3-deg test and a 2 sec wait period before repeating the process.

Although the RR Data Read Routine reads the RR angles along with the range and range rate, the angle data is not used for update purposes because of the uncertainties associated with the magnitude and nature of the RR angle biases which may be present during the large angular excursions of the RR with respect to the vehicle at this time. Thus, the estimated state vector in the Lunar Surface Navigation Routine is the six-dimensional CSM state vector, and only the range and range rate data (R_M and \dot{R}_M , respectively) are used in the navigation computations.

*The time to complete this calculation is highly dependent on other computational priorities. No explicit control of this procedure is exercised.

The computation logic for the Lunar Surface Navigation Routine is similar to the rendezvous navigation logic (Section 5.2.4.2) and is illustrated in Fig. 2.5-4. It is assumed that the following items are stored in erasable memory at the start of the procedure shown in the figure:

- \underline{x}_C = Estimated CSM state vector as defined in Section 5.2.2.6
- W = Six-dimensional error transition matrix associated with \underline{x}_C as defined in Section 5.2.2.4
- \underline{r}_{LS} = Estimated landing site or LM position vector on the surface of the moon in moon-fixed coordinates.
- N = Number of measurement data points already processed.
- w_{lr}, w_{lv} = Preselected W-matrix initial diagonal elements.
- RENDWFLG = $\begin{cases} 1 & \text{for valid W-matrix} \\ 0 & \text{for invalid W-matrix} \end{cases}$

This flag or switch is maintained by programs external to the Lunar Surface Navigation Routine. It indicates whether or not the W-matrix is valid for use in processing RR tracking data. The flag is set to zero after each of the following procedures:

1. State vector update from ground
2. Astronaut command
3. Overflow of W-matrix integration
4. New W-matrix initialization values are loaded via V67
5. Lunar ascent

The variables D and V are indicators which control the Coasting Integration Routine (Section 5.2.2) as described in Section 5.2.2.6, \underline{u}_Z is a unit vector along the rotational axis of the moon, and ω_M is the lunar angular velocity.

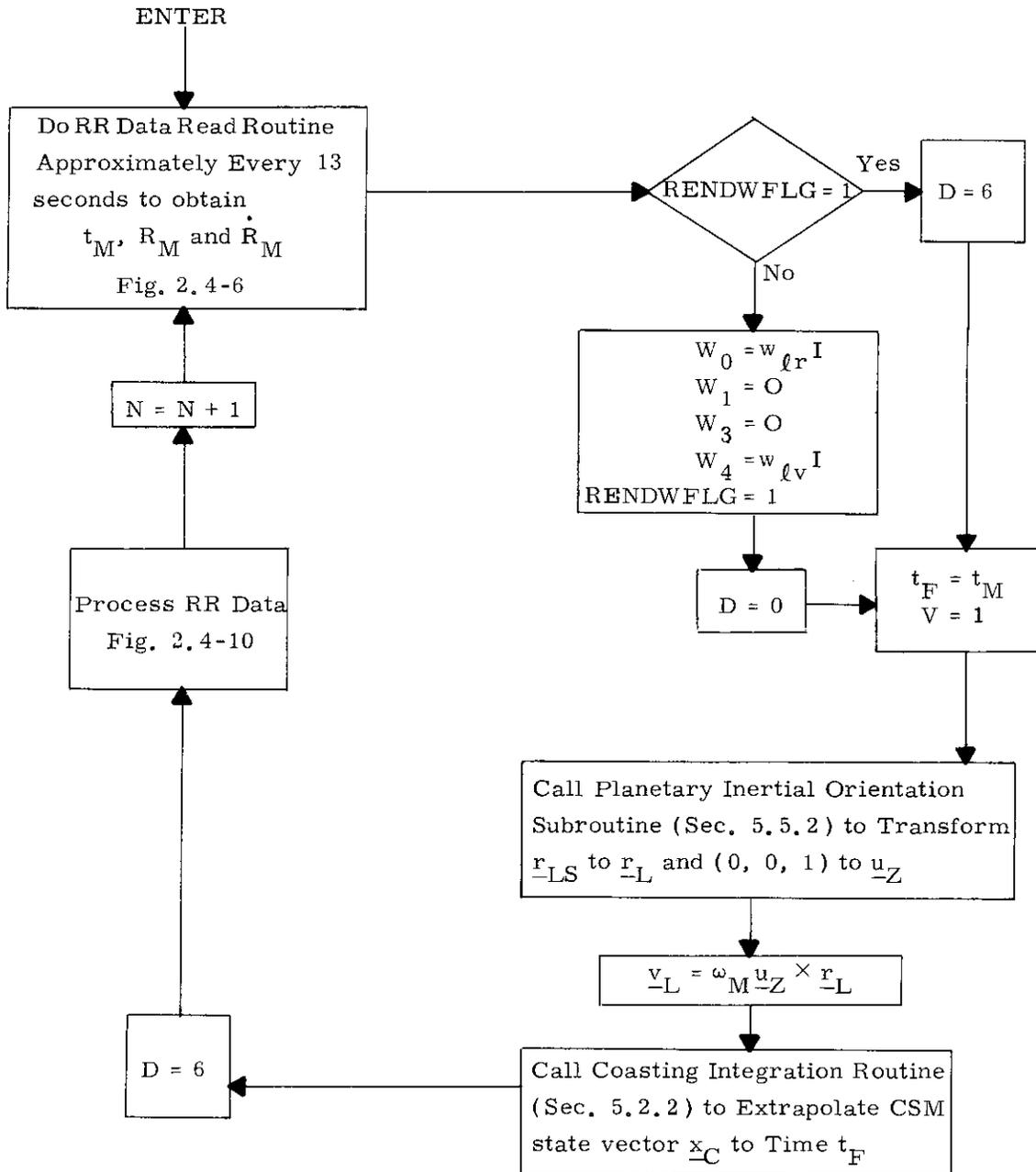


Fig. 2.5-4 Lunar Surface Navigation Routine Logic Diagram

