

LM AGS COMPUTER PROGRAM SPECIFICATION  
FLIGHT PROGRAM 6

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**TRW**  
SYSTEMS GROUP

ONE SPACE PARK • REDONDO BEACH, CALIFORNIA

LM AGS COMPUTER PROGRAM SPECIFICATION  
FLIGHT PROGRAM 6

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## 1. INTRODUCTION

### 1.1 PURPOSE

This document has been prepared for the Guidance and Control Division of the National Aeronautics and Space Administration, Manned Spacecraft Center, Houston, Texas, by TRW Systems, under Contract NAS 9-8166, ASPO Task 32E, "LM AGS Guidance Software." The purpose of this document is to establish the requirements for LM AGS Flight Program 6, a digital computer program to be produced for use in the Abort Electronics Assembly (AEA) of the Abort Guidance Section (AGS), a portion of the Stabilization and Control System of the Lunar Module (LM).

The AGS is a guidance system which provides attitude reference, navigation, monitoring, and abort guidance for the LM. Its primary purpose is to provide the means whereby the LM can be guided to a safe orbit for eventual rendezvous with the CSM from any point in the lunar landing phase of the Apollo mission in the event a malfunction has occurred in the Primary Guidance Navigation and Control Section. The flight program specified herein is that software which is required to enable the AGS to fulfill its primary purpose as well as such other functions which have been defined for it.

This specification defines the functional requirements for the flight program, defines the requirements for verification testing and specifies the format of the deliverable software package.

## 1.2 GENERAL REQUIREMENTS

### 1.2.1 Mission Requirements

Flight Program 6 shall be designed for use on lunar missions of the "G" type defined in References 1-1 and 1-2. It is required that the verification testing specified in Section 8 demonstrate that the program meets the requirements of this specification.

### 1.2.2 Functional Requirements

The functional requirements for Flight Program 6 are those which controlled the design of the previous baseline, LM AGS Flight Program 5 documented in Reference 1-3 as modified by the MSC approved changes listed below as Reference 1-4 through 1-5. These software changes represent a major change in the AGS computer program. The radar filter capability shall be expanded (SCP 43, reference 1-4) to appropriately process radar data at ranges up to 400 n. mi. ; detail requirements for the radar filter are in Subsection 5.4. An explicit CSI/CDH guidance capability shall be provided (SCP 50, reference 1-5) to solve for a CSI maneuver for a CSI to CDH central angle that is equivalent to one half or three halves of the LM orbital period; detail requirements are in Section 6.0. Continuous horizontal velocity targeting for orbit insertion shall be provided (SCP 51, reference 1-6) at the request of NASA/MSC (Reference 1-7) to be consistent with mission planning; detail requirements are in Section 6.0. Direction to implement the proposed changes was received by TRW in reference 1-8.

### 1.2.3 AEA Compatibility

An AEA contains 4096 memory locations of which 2048 are hardwired and a like number softwired (erasable). Flight Program 6 shall be designed to function using the hardwired memory program designated LEM AGS H03 which is defined by the wire list with TRW Drawing No. 222641-2 dated 1 June 1966. LEM AGS H03 will be contained in AEA's having serial numbers 6 and greater.

### 1.3 PROGRAM DELIVERY

The requirement for providing deliverable items is supplied to TRW by NASA/MSC in the Task Assignment (reference 1-9). These deliverable items fall into two categories, Deliverable Software and Supporting Documentation.

#### 1.3.1 Deliverable Software

The software delivery shall consist of an AEA software package (program listings, decks and tapes) as described in Section 9. This delivery shall be made several months prior to the defined launch date and shall have nominal values for the program constants. The software package shall be updated about one month prior to the defined launch date by incorporating a set of NASA approved constants.

#### 1.3.2 Documentation

The initial delivery of this program shall be accompanied by the necessary supporting documentation as defined in the Task Assignment (reference 1-9). This documentation consists of a Programmed Equation Document, and Operating Manual and a Summary Report on the verification testing. Statistical studies of the performance of the LM under AGS control shall be conducted and reported in a Performance Analysis Report.

#### 1.4 QUALITY CONTROL

This specification provides guidelines (Section 8.0) for verification of the deliverable software. The detailed testing shall be done in accordance with the NASA approved Program Verification Test Plan. This testing shall be devised to demonstrate that the flight program functions have been designed to fulfill the AGS program requirements as defined by this specification.

Each assembly of the flight program shall be given a unique identification and subsequent revisions will be made in a controlled manner with the previously delivered version of the same program as defined in Section 9.0.

To provide additional assurance for the quality of the flight program, a TRW Design Review Board of senior technical members shall review and approve all items.



## 1.5 REFERENCES

- 1-1 MSC Internal Note 68-FM-196, "Apollo Mission G Spacecraft Reference Trajectory," Vol. 1-5, 9 August 1968.
- 1-2 MSC Letter 68-FM64-325, "Currently Proposed Rendezvous Profile for Mission G (LLM)," E. C. Lineberry, 28 October 1968.
- 1-3 TRW Report 11176-6015-R0-00, "LM/AGS Programmed Equations Document, Flight Program 5," January 1969.
- 1-4 LM/AGS Guidance Software Change Proposal Number 43, "Expanded Capability Radar Filter," 28 June 1968.
- 1-5 LM/AGS Guidance Software Change Proposal Number 50, "CSI/CDH Equations for FPX," 14 November 1968.
- 1-6 LM/AGS Guidance Software Change Proposal Number 51, "Continuously Variable Orbit Insertion Targeting," 27 November 1968.
- 1-7 TRW Letter 68-FM64-1747 "MSC Request for AGS SCP on Automatic Continuous Horizontal Velocity Targeting as a Function of Time During Orbit Insertion," R. Braslau, 13 November 1968.
- 1-8 MSC TWX EG 43-69-40, "MSC Approval of FPX Configuration," R.G. Chilton, 31 January 1969.
- 1-9 MSC/TRW ASPO-32D Task Assignment for TRW Systems, "LM AGS Guidance Software," 1 August 1968.

## 2. AEA INTERFACES

### 2.1 INTRODUCTION

The LM AGS flight program contained in the AEA shall provide a capability for accepting and processing data and discrete inputs from the two other components of the AGS, the Abort Sensor Assembly (ASA) and the Data Entry and Display Assembly (DEDA). The program shall also be capable of processing and controlling the transfer of data from a restricted portion of the AEA software memory to the DEDA for display.

The flight program shall provide a capability for accepting and processing data transmitted from the LM Primary Guidance, Navigation, and Control Section (PGNCS) or from the ground support equipment (GSE) which may be either the Bench Test Maintenance Equipment (BTME) or the Automatic Checkout Equipment (ACE).

A capability shall be provided for the output of a preselected fixed data list by means of an output telemetry link to the Instrumentation Subsystem.

Attitude error signals and Engine ON/OFF commands shall be computed and output to the Control Electronics Section (CES) to provide a means for controlling the LM vehicle attitude through the operation of the Reaction Control System (RCS) and for determining the duration of the burns of the main engines, the Descent Propulsion System (DPS) and the Ascent Propulsion System (APS). Those discrettes furnished by the CES indicating main engine status and certain astronaut control manipulations shall be recognized and shall produce an appropriate response by means of the program logic.

Certain quantities shall be computed and output for display. Lateral velocity, altitude and altitude rate shall be furnished to the Flight Control Displays. Total attitude signals shall be furnished to the Flight Director Attitude Indicators along with the attitude error signals. A discrete signal shall be supplied to the Caution and Warning Displays in the event of the occurrence of certain AEA failures.

All of the input and output signals of the AEA related to flight program operation are listed in Table 2-I.



Table 2-I. AEA Interfaces

	<u>Input/Output</u>	<u>Source/Destination</u>	<u>Signal</u>	<u>Signal Characteristics</u>	<u>AEA Program Interface</u>	<u>Primary Function</u>
1	Input	ASA-Accelerometer	$\Delta V_X$ - Data	Pulse Train, 64 kpps max.	11-bit Ripple Counter	Accumulate Accelerometer Output Data
2	Input	ASA-Accelerometer	$\Delta V_Y$ - Data	Pulse Train, 64 kpps max.	11-bit Ripple Counter	Accumulate Accelerometer Output Data
3	Input	ASA-Accelerometer	$\Delta V_Z$ - Data	Pulse Train, 64 kpps max.	11-bit Ripple Counter	Accumulate Accelerometer Output Data
4	Input	ASA-Gyro	$\Delta \int P$ - Data	Pulse Train, 64 kpps max.	11-bit Ripple Counter	Accumulate Gyro Output Data
5	Input	ASA-Gyro	$\Delta \int Q$ - Data	Pulse Train, 64 kpps max.	11-bit Ripple Counter	Accumulate Gyro Output Data
6	Input	ASA-Gyro	$\Delta \int R$ - Data	Pulse Train, 64 kpps max.	11-bit Ripple Counter	Accumulate Gyro Output Data
7	Output	ASA	Clock	Pulse Train, 128 kpps	- - -	Timing Reference
8	Input	DEDA	DEDA Enter Discrete	Momentary Switch Closure	Discrete Word 2	Control Data Transfer Between AEA and DEDA
9	Input	DEDA	DEDA Readout Discrete	Momentary Switch Closure	Discrete Word 2	Control Data Transfer Between AEA and DEDA
10	Input	DEDA	DEDA Hold Discrete	Momentary Switch Closure	Discrete Word 2	Control Data Transfer Between AEA and DEDA
11	Input	DEDA	DEDA Clear Discrete	Momentary Switch Closure	Discrete Word 2	Control Data Transfer Between AEA and DEDA
12	Output	DEDA	Data	4-bit Pulse Burst, 64 kpps	4-bit DEDA Shift Register	Data for DEDA Display
13	Input	DEDA	Data	4-bit Pulse Burst, 64 kpps		Data for Program Control or Use
14	Output	DEDA	Clock	Pulse Train, 128 kpps	- - -	Timing Reference
15	Input	DEDA	DEDA Shift Clock	Pulse Burst, 4, 64 kpps	- - -	Implement Data Transfer into or out of 4-bit DEDA Shift Register
16	Output	DEDA	DEDA Shift In Discrete	Single Pulse	Discrete Output Pulse Trigger	Initiate Data Transfer into 4-bit DEDA Shift Register
17	Output	DEDA	DEDA Shift Out Discrete	Single Pulse	Discrete Output Pulse Trigger	Initiate Data Transfer out of 4-bit DEDA Shift Register
18	Input	PGNCS-IMU CDU	$\pm \Delta \theta$ - Data	Pulse Trains on Two Separate Lines, 6.4 kpps max.	15-bit Integrator Register	Accumulate IMU Gimbal Angle Data

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Table 2-I AEA Interfaces (Continued)

<u>Input/Output</u>	<u>Source/Destination</u>	<u>Signal</u>	<u>Signal Characteristics</u>	<u>AEA Program Interface</u>	<u>Function</u>	
19	Input	FGNCS - IMU CDU	+ $\Delta\psi$ - Data	Pulse Trains on Two Separate Lines, 6.4 kpps max.	15-bit Integrator Register	Accumulate IMU Gimbal Angle Data
20	Input	FGNCS - IMU CDU	+ $\Delta\phi$ - Data	Pulse Trains on Two Separate Lines, 6.4 kpps max.	15-bit Integrator Register	Accumulate IMU Gimbal Angle Data
21	Input	FGNCS	CDU Zero	Pulse Burst, 51.2 kpps 300 ms duration	- - -	Clears 15-bit Integrator Registers used to Accumulate IMU Gimbal Angle Data
22	Input	FGNCS Downlink Telemetry	Data	Pulse Burst, 40 bits, 51.2 kpps	18-bit Shift Register (Downlink Telemetry Register)	Store Navigation Initialization Data
23	Input	Ground Service Equipment	Data	Pulse Burst, 18 bits, 51.2 kpps		
24	Input	FGNCS Downlink Telemetry	Shift Clock	Pulse Burst, 40, 51.2 kpps	- - -	Implement Data Transfer into Downlink Telemetry Register
25	Input	FGNCS Downlink Telemetry	Stop Pulse Discrete	Pulse Train, 50 pps	Discrete Word 2	Sets Discrete Flip-Flop to indicate Completed Transmission of a Downlink Word
26	Output	Instrumentation Subsystem	Data	Pulse Burst, 24 bits, 51.2 kpps	24-bit Shift Register (Output Telemetry Register)	AGS Output Telemetry
27	Output	Ground Service Equipment	Data	Pulse Burst, 24 bits, 51.2 kpps		
28	Input	Instrumentation Subsystem	Shift Clock	Pulse Burst, 24, 51.2 kpps	- - -	Implement Data Transfer out of Output Telemetry Register
29	Input	Instrumentation Subsystem	Stop Pulse Discrete	Pulse Train, 50 pps	Discrete Word 1	Sets Discrete Flip-Flop to Indicate Completed Transmission of an Output Telemetry Word
30	Input	Ground Service Equipment	Shift Clock	Pulse Burst, 18, 51.2 kpps	- - -	Implement Data Transfer into Downlink Telemetry Register or out of Output Telemetry Register
31	Input	Ground Service Equipment	GSE Discrete 1	Switch Closure	Discrete Word 2	Indicates that the GSE has loaded a word into the Downlink Telemetry Register
32	Input	Ground Service Equipment	GSE Discrete 2	Switch Closure	Discrete Word 2	Indicates that the GSE has loaded the last word of the list in the Downlink Telemetry Register

Table 2-I. AEA Interfaces (Continued)

<u>Input/Output</u>	<u>Source/Destination</u>	<u>Signal</u>	<u>Signal Characteristics</u>	<u>AEA Program Interface</u>	<u>Function</u>	
33	Input	Ground Service Equipment	GSE Discrete 3	Switch Closure	Discrete Word 2	Indicates that the GSE has received a complete word from the Output Telemetry Register
34	Input	Ground Service Equipment	GSE In Use Discrete	Switch Closure	- - -	Inhibits receipt of signals from PGNC'S Downlink Telemetry when GSE is in use.
35	Output	Ground Service Equipment	GSE Discrete 4	Pulse Train, 128 kpps	Discrete Output Pulse Trigger	Notifies the GSE that a complete word has been loaded into the Output Telemetry Register
36	Output	Ground Service Equipment	GSE Discrete 5	Pulse Train, 128 kpps	Discrete Output Pulse Trigger	Notifies the GSE that a complete word has been transferred from the Downlink Telemetry Register into memory.
37	Output	Ground Service Equipment	GSE Discrete 6	Pulse Train, 128 kpps	Discrete Output Pulse Trigger	Notifies the GSE that the last word of the list has been loaded into the Output Telemetry Register
38	Output	Ground Service Equipment	Clock	Pulse Train, 50 pps	- - -	Timing Reference
39	Input	Control Electronics Section	Automatic Discrete	Switch Closure	Discrete Word 1	Indicates that the Attitude Mode Control Switch is in Automatic (AUTO) position
40	Input	Control Electronics	Abort Discrete	Switch Closure	Discrete Word 1	Indicates that the Abort Switch has been depressed.
41	Input	Control Electronics Section	Abort Stage Discrete	Switch Closure	Discrete Word 1	Indicates that the Abort Stage Switch has been depressed.
42	Input	Control Electronics Section	Follow-up Discrete	Switch Closure	Discrete Word 1	Indicates either that the Guidance Control Switch is in the PGNS position or, if the Guidance Control Switch is in the AGS position, that the Attitude Mode Control Switch is in the Attitude Hold position and an Attitude Controller is out of detent.
43	Input	Control Electronics Section	Descent Engine ON Discrete	Switch Closure	Discrete Word 1	Indicates that the Descent Engine is on.
44	Input	Control Electronics Section	Ascent Engine ON Discrete	Switch Closure	Discrete Word 1	Indicates that the Ascent Engine is on.
45	Output	Control Electronics Section	Engine OFF Discrete	Switch Closure	Discrete Output Flip-Flop	Commands Engine Off
46	Output	Control Electronics Section	Engine ON Discrete	Switch Closure	Discrete Output Flip-Flop	Commands Engine On

Table 2-I. AEA Interfaces (Continued)

<u>Input/Output</u>	<u>Source/Destination</u>	<u>Signal</u>	<u>Signal Characteristics</u>	<u>AEA Program Interface</u>	<u>Function</u>	
47	Output	Control Electronics Section and Flight Control Displays	$E_X$ - Control and Data	800 Hz Analog	10-bit Static Register and D/A Converter	Attitude Error Signal Transmitted to the Control Electronics Section for vehicle control and to the FDAI for display on two separate lines.
48	Output	Control Electronics Section and Flight Control Displays	$E_Y$ - Control and Data	800 Hz Analog	10-bit Static Register and D/A Converter	Attitude Error Signal transmitted to the Control Electronics Section for vehicle control and to the FDAI for display on two separate lines.
49	Output	Control Electronics Section and Flight Control Displays	$E_Z$ - Control and Data	800 Hz Analog	10-bit Static Register and D/A Converter	Attitude Error Signal transmitted to the Control Electronics Section for vehicle control and to the FDAI for display on two separate lines.
50	Output	Flight Control Displays	Sin $\alpha$ - Data	400 Hz Analog	10-bit Static Register and D/A Converter	FDAI Total Attitude Display Gimbal Drive Signal
51	Output	Flight Control Displays	Cos $\alpha$ - Data	400 Hz Analog	10-bit Static Register and D/A Converter	FDAI Total Attitude Display Gimbal Drive Signal
52	Output	Flight Control Displays	Sin $\beta$ - Data	400 Hz Analog	10-bit Static Register and D/A Converter	FDAI Total Attitude Display Gimbal Drive Signal
53	Output	Flight Control Displays	Cos $\beta$ - Data	400 Hz Analog	10-bit Static Register and D/A Converter	FDAI Total Attitude Display Gimbal Drive Signal
54	Output	Flight Control Displays	Sin $\gamma$ - Data	400 Hz Analog	10-bit Static Register and D/A Converter	FDAI Total Attitude Display Gimbal Drive Signal
55	Output	Flight Control Displays	Cos $\gamma$ - Data	400 Hz Analog	10-bit Static Register and D/A Converter	FDAI Total Attitude Display Gimbal Drive Signal
56	Output	Flight Control Displays	Lateral Velocity-Data	DC Analog	9-bit Static Register and D/A Converter	Lateral Velocity Display for X-Pointers
57	Output	Flight Control Displays	Altitude - Data	Pulse Burst, 15 bits, 64 kpps	15-bit Shift Register	Data for display on the Altitude/Range Indicator. Altitude and Altitude Rate are loaded alternately into the same register but are transmitted on separate lines.
58	Output	Flight Control Displays	Altitude Rate-Data	Pulse Burst, 15 bits, 64 kpps		
59	Output	Flight Control Displays	Shift Clock	Pulse Burst, 15, 64 kpps	Altitude Display Discrete	Implement Data Transfer into Altitude Display.
60	Output	Flight Control Displays	Shift Clock	Pulse Burst, 15, 64 kpps	Altitude Rate Display Discrete	Implement Data Transfer into Altitude Rate Display
61	Output	Caution and Warning Displays (AGS Failure Warning Light)	Test Mode Failure Discrete	Switch Closure	Discrete Output Flip-Flop	Indicate AEA Failure

## 2.2 ABORT SENSOR ASSEMBLY (ASA)

The ASA instrument package contains three single-degree-of-freedom pulse-rebalanced rate-integrating gyros, three pendulous pulse-rebalanced accelerometers, and their associated pulse torquing electronic circuitry in a strapdown configuration. The respective instrument input axes are constrained to be collinear, within specified tolerances, with a set of rectangular coordinates, X, Y, and Z, termed the ASA reference axes. The nominal ASA sensor triad configuration is shown in Figure 2-1. The tolerances on misalignment of the input axes of gyros and accelerometers with respect to the ASA reference axes are given in Paragraphs 3.1.2.1.2.2 and 3.1.2.1.3.2, respectively, of Reference 2-1.

For the purposes of this specification, the orientation of the ASA reference axes is assumed to be identical with that of the LM body axis coordinates when the ASA is properly mounted on the navigation base. No compensation is required in the flight program for misalignment of the ASA gyros and accelerometers with respect to the ASA reference axes.

### 2.2.1 LM Body Axis Coordinates

The LM body axis coordinate frame is a set of right-handed rectangular coordinate axes, X, Y and Z, fixed to the vehicle. The X axis lies along the longitudinal axis of the vehicle, positive in the nominal direction of positive main engine thrust acceleration. The Z axis is normal to the X axis, lies in the plane of symmetry of the LM cabin front view, and is positive in the direction of the front view. The Y axis completes the right-handed orthogonal triad. The LM body axis coordinate frame is illustrated in Figure 2-2.

### 2.2.2 ASA Outputs

The output of each of the six ASA instruments is a train of electrical pulses. These pulse trains are input to the AEA and accumulated in the six ASA input registers, 11-bit input counters. The registers are incremented by one count for each pulse received.

Each gyro pulse represents an incremental angular displacement about the corresponding body axis. Each accelerometer pulse represents an incremental change in velocity along the corresponding body axis.



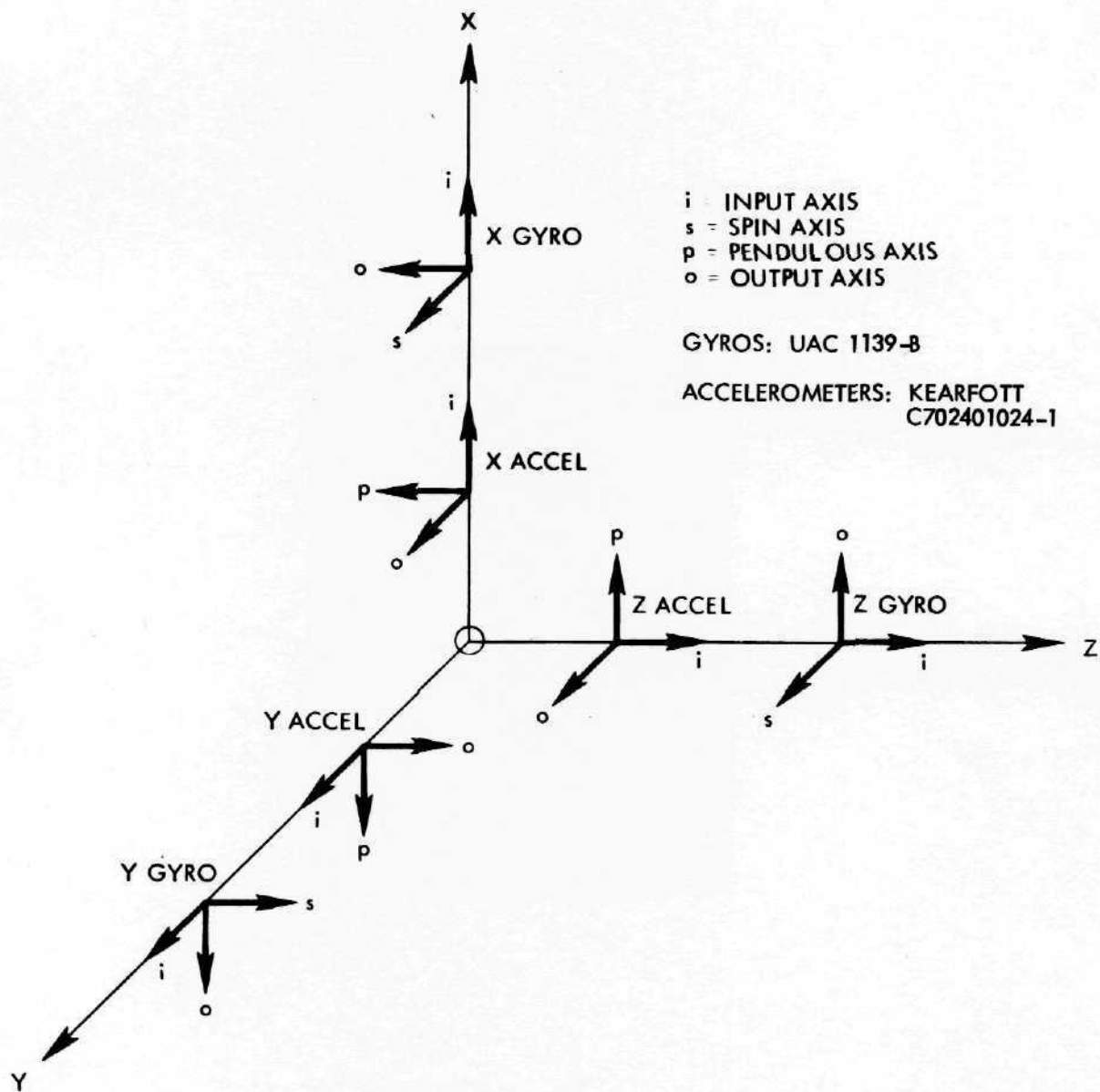


Figure 2-1. ASA Sensor Triad Configuration

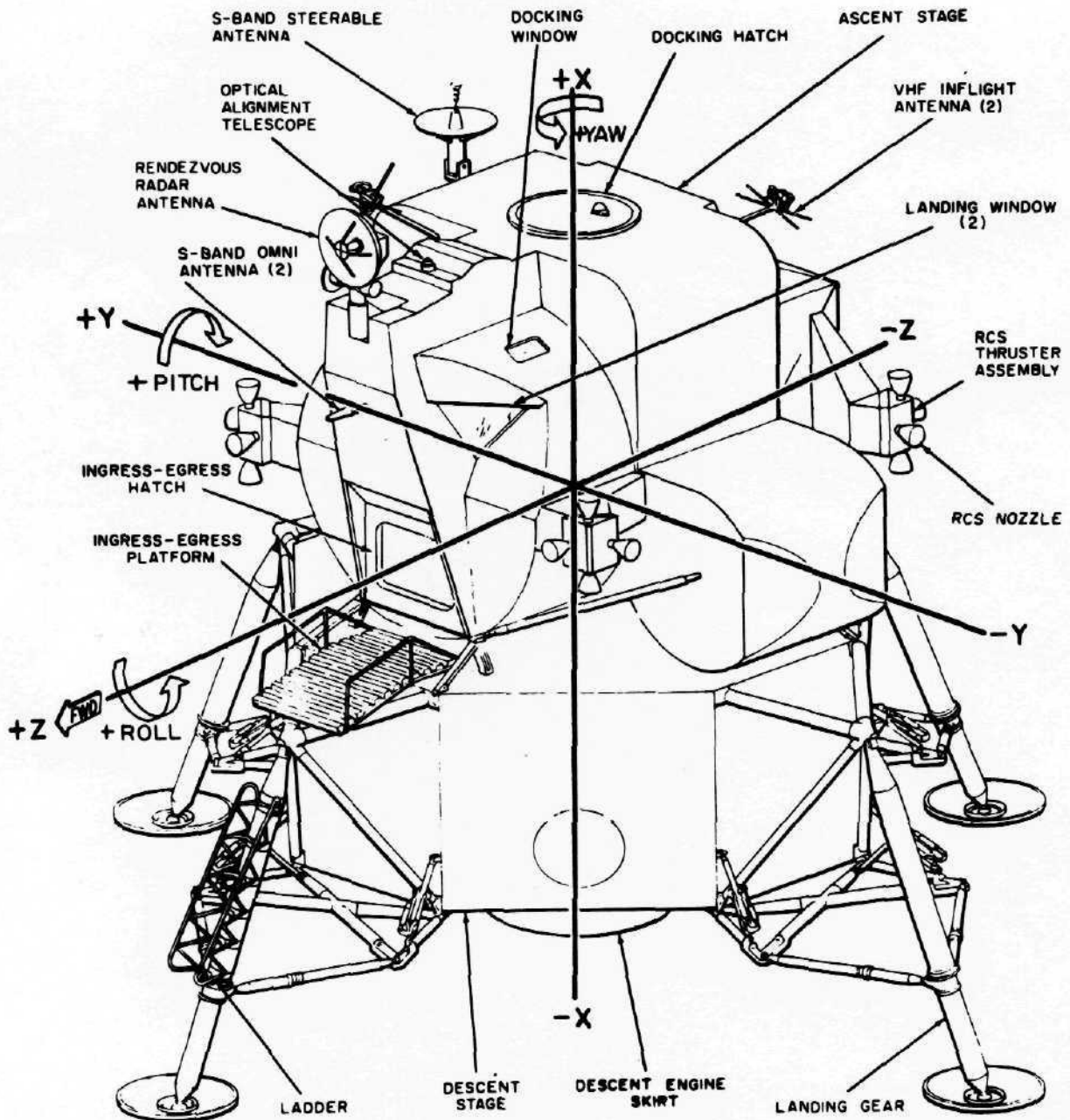


Figure 2-2. LM Body Axis Coordinate System

During any 1 ms period, the accumulation of 32 pulses from a precisely nominal instrument represents a zero signal; the accumulation of 61 pulses represents the maximum positive signal; the accumulation of 3 pulses represents the maximum negative signal.

A positive incremental angular signal from a given gyro represents a positive angular displacement, in the right-handed sense, about the corresponding body axis. The nominal weight of the incremental angular information is  $2^{-16}$  rad/pulse.

A positive incremental velocity signal from a given accelerometer represents a positive translational velocity change, due to thrust acceleration, along the corresponding body axis. The nominal weight of the incremental velocity information is 0.003125 fps/pulse.

Each ASA input register shall be sampled and cleared every 20 ms to prevent overflow. The nominal bias of 640 pulses per 20 ms shall be removed from the data as part of the data processing in the flight program.



## 2.3 DEDA

The DEDA is a general-purpose, manually operated, input-output device for use with the AEA. Its major external features are a keyboard and electroluminescent data display devices. The DEDA keyboard consists of ten decimal-digit pushbuttons, two sign pushbuttons, and four control pushbuttons. The Display Registers provide for a three octal-digit address, a sign plus five octal- or decimal-digit data word, and an operator error indication (see Figure 2-3). The display devices are activated either by use of the appropriate pushbuttons, by the extraction of data from the AEA, or, in the case of the operator error indicator, by certain improper procedures in the use of the keyboard.

Each of the four control pushbuttons, Clear, Enter, Readout, and Hold, when depressed, cause corresponding DEDA control discretes to be set in the Discrete Word 2 Register of the AEA. The DEDA control discretes are buffered switch closures to ground and remain set for only so long as the corresponding DEDA control pushbutton remains depressed.

The DEDA shall be used for manual insertion of data into the AEA and manual initiation of data readout from the AEA. In addition, selection of certain of the operating modes of the flight program and the AGS generally shall be made by appropriate inputs via DEDA. The flight program shall not be required to process DEDA operations at intervals of less than 0.6 seconds except when depressing clear to erase a previous operation; nor shall it be required to complete an enter or readout operation within 1.5 seconds of the time the DEDA display has gone blank following the depression of enter.

### 2.3.1 Data Transfer

The DEDA Register, a 4-bit, input-output shift register in the AEA, provides temporary storage for data which is being transferred between the AEA and DEDA under the control of the DEDA Processing Routine in the flight program. The DEDA Processing Routine shall initiate or terminate the data transfer in response to the recognition of the setting of the various DEDA control discretes.

When data is to be transferred either to or from the DEDA, either the DEDA Shift-Out Discrete or the DEDA Shift-In Discrete, respectively,

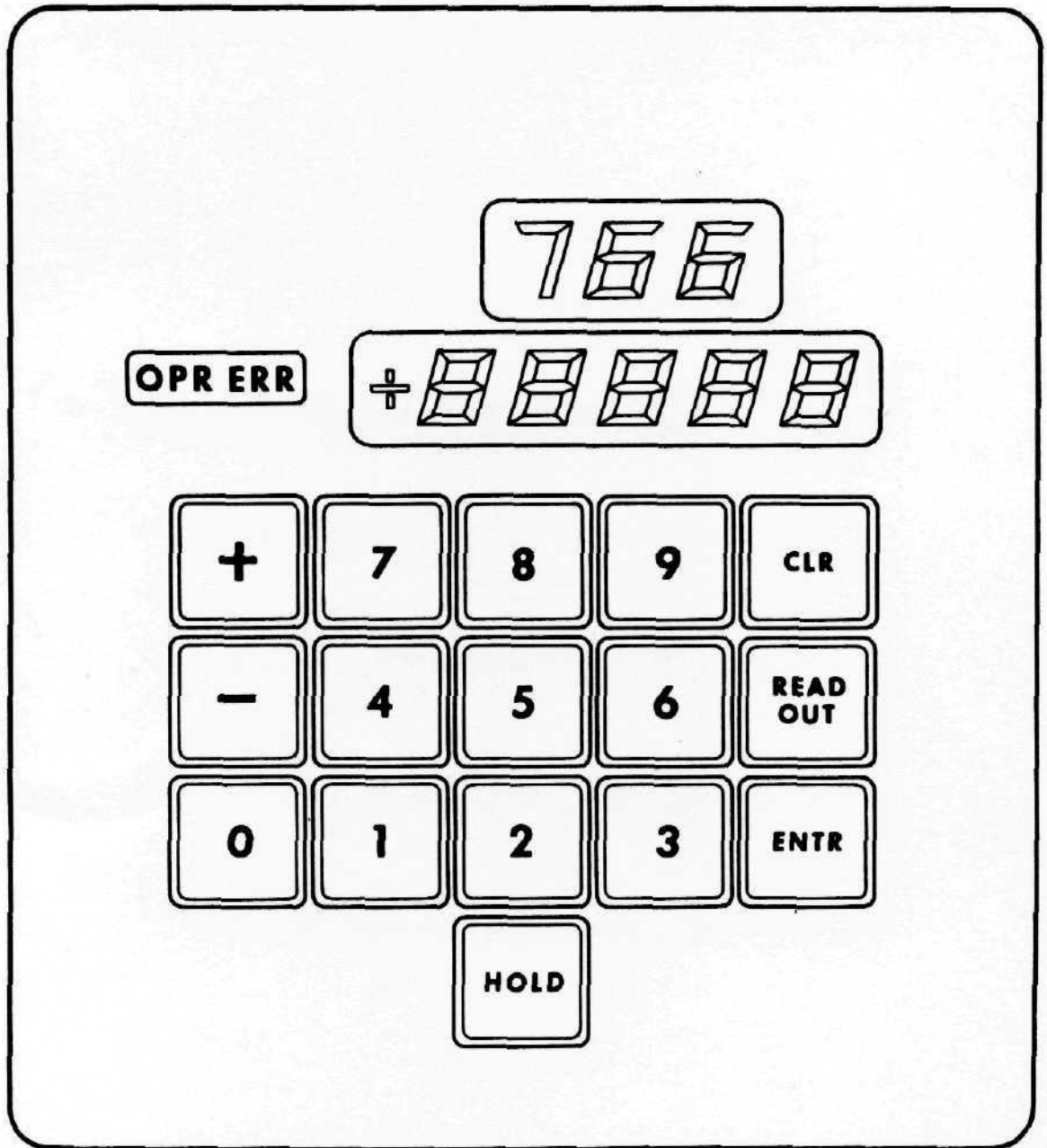


Figure 2-3. DEDA Front Panel

must be issued. Upon receipt of the resulting pulse, the DEDA will transmit a burst of four clock pulses to the DEDA Register. This will cause the shifting of 4 bits of data into or out of the register, as appropriate.

A minimum time interval of 80  $\mu$ s must be allowed between subsequent settings of either the DEDA Shift-In Discrete or DEDA Shift-Out Discrete to allow sufficient time for the data transfer.

### 2.3.2 DEDA Processing Routine Requirements

The maximum number of memory locations accessible for data input and output via the DEDA shall be 512. These memory locations shall be in a designated section of the software memory.

#### 2.3.2.1 DEDA Word Format

A complete DEDA word contains 36 bits consisting of nine 4-bit groups. Each 4-bit group is associated with one of the DEDA display devices. There are three 4-bit groups for the address, one 4-bit group for the data sign, and five 4-bit groups for the data.

#### DEDA Word Format

Total Bits = 36

0 X X X	0 X X X	0 X X X	0 0 0 X	* X X X	* X X X	* X X X	* X X X	* X X X
12 bits Address			4 bits Sign	20 bits Data				

#### Code

0 - Zero

X - Either One or Zero

\* { Either One or Zero for binary-coded decimal data  
Zero for octal data

In the 4-bit sign group, the three most significant bits are zeros. A zero in the least significant bit will indicate a positive sign, while a one in that location will indicate a negative sign.

### 2.3.2.2 Data Transmission and Processing

Data transmission either to or from the DEDA shall be in 4-bit groups beginning at the most significant bit of the 36-bit word. Each 4-bit group is transmitted serially, the most significant bit first.

The DEDA Processing Routine shall provide a capability for decoding address information and data which is transmitted to the AEA via DEDA. In decoding address information, the most significant bit of each 4-bit group shall be tested to determine if it is zero. If it is non-zero, an operator error shall be assumed. In decoding sign information, the three most significant bits of the 4-bit group shall be deleted. In processing the 20 bits of data, the DEDA Processing Routine shall provide for distinguishing between octal and binary-coded-decimal data based upon the particular address of the data word. For data entered in octal, the most significant bit of each 4-bit group shall be assumed to be zero. For data entered in decimal, the decoding operation shall provide an appropriately rescaled binary word for storage. Reasonableness or ambiguity checks on input data are not required.

The DEDA Processing Routine shall also provide a capability for coding and formatting the 36-bit DEDA word for output. Zeros shall be supplied in those bits requiring them and data from those memory locations designated for display of information in decimal shall be scaled according to specification prior to transmission in binary-coded decimal.

In the event that the operator error display has been activated, the DEDA will transmit all ones to the AEA when the Enter or Readout push-buttons are depressed. The DEDA Processing Routine shall recognize this error indication by detecting the presence of ones in locations normally constrained to contain zeros. In the event of this error detection, no input data shall be accepted and no data shall be transmitted until a new sequence is initiated by receipt of the Clear Discrete.

### 2.3.2.3 DEDA Input/Output Control

The DEDA Control Discretes are set for only so long as the corresponding DEDA pushbuttons are depressed. Therefore, the DEDA Processing Routine is required to sample the discretes at 40 ms intervals to ensure that transmission of a DEDA Control Discrete is recognized.



Each DEDA operation sequence shall be initiated by transmission of the Clear Discrete which shall place the DEDA Processing Routine in a clear mode. While in the clear mode, both the Readout and Enter Discretes shall be interrogate.

On receipt of the Enter Discrete, the DEDA Processing Routine shall cause nine consecutive DEDA Shift-In Discretes to be issued to the DEDA to transfer the 36-bit DEDA word into the AEA. The data word shall be decoded, rescaled if necessary, and stored in the address designated. The Clear Discrete must be received following such an entry before a subsequent entry or readout operation may be performed.

On receipt of a Readout Discrete, the DEDA Processing Routine shall enter the Readout Mode. Three consecutive DEDA Shift-In Discretes shall be issued to bring 12 bits of address information into the AEA. The DEDA Processing Routine shall extract the data from the address thus requested, code and scale the data as required, and transmit the full 36-bit DEDA word by issuing nine consecutive DEDA Shift-Out Discretes. The extraction of data from the requested address and transmission to the DEDA shall be repeated four times per two-second compute cycle until either a Hold or Clear Discrete is sensed.

Upon receipt of the Hold Discrete, the DEDA Processing Routine shall exit from the Readout Mode and suspend extraction and transmission of the data. Upon receipt of a subsequent Readout Discrete, the Readout Mode shall be reentered and data transmission from the same address shall be resumed.

Receipt of a Clear Discrete following either a Readout or Hold Discrete shall return the DEDA Processing Routine to the clear mode.

## 2.4 PGNCS

### 2.4.1 IMU Gimbal Angles

The PGNCS provides attitude information to the AGS by means of direct connections between the AEA and the three Coupling Data Units (CDU's) associated with the three gimbal angles of the Inertial Measurement Unit (IMU). The AEA contains three 15-bit integrator registers, each of which accumulates the pulses transmitted on a pair of lines from a corresponding CDU. Each pulse produces a one-count increment or decrement in the register, depending on whether the signal arrives on the plus or minus line.

The receipt of a single pulse nominally represents an angular change of  $360/2^{15}$  deg in the corresponding gimbal angle. Thus, the range of the data word in each register is 0 to 360 degrees.

The laboratory testing of the CDU's, reported in References 2-2, 2-3, and 2-4, was extensive and showed that large transient errors may occur in the CDU output. There is no requirement for the flight program to either detect the occurrence of CDU transients or to compensate for their effects. See Reference 2-5.

All three IMU gimbal angle registers are cleared whenever a CDU Zero signal is received from the PGNCS on another line. The corresponding registers in the PGNCS are cleared simultaneously by a similar signal. The transmission of the CDU Zero signal is controlled by the PGNCS.

After being cleared, the CDU's reload themselves and the IMU gimbal angle registers. This reloading requires no more than 20 seconds to complete. The result is that both guidance systems are initialized at this point to contain the same IMU gimbal angle information. Any accumulated errors due to previously occurring transients and drift in the CDU's are removed.

Under the control of the flight program, whenever the IMU gimbal data is required in order to perform an alignment of the AGS attitude reference matrix or an in-flight gyro calibration, the IMU gimbal angle registers shall be read and the data stored in memory every 20 ms.

The IMU gimballed angle data provides a measure of that set of Euler angles,  $\theta$ ,  $\psi$ , and  $\phi$ , which define a coordinate transformation, [B], from the rectangular coordinate frame of the IMU stable member, u, v, and w, to the reference axes of the IMU, U, V, and W, as mounted on the navigation base.

$$\begin{bmatrix} U \\ V \\ W \end{bmatrix} = [B] \begin{bmatrix} u \\ v \\ w \end{bmatrix}$$

The IMU reference axes, the ASA reference axes, and the LM body axes all have the same nominal orientation. The misalignment of the IMU reference axes, with respect to the ASA reference axes on the navigation base, will not exceed 5.2 arc-minutes. No compensation for this misalignment is required in the flight program.

The order of rotation from the IMU stable member coordinate axes to the IMU reference axes is illustrated in Figure 2-4. This rotation defines the transformation matrix to be as follows:

$$[B] = \begin{bmatrix} C\psi C\theta & S\psi & -C\psi S\theta \\ (-C\phi S\psi C\theta + S\phi S\theta) & C\phi C\psi & (C\phi S\psi S\theta + S\phi C\theta) \\ (S\phi S\psi C\theta + C\phi S\theta) & -S\phi C\psi & (-S\phi S\psi S\theta + C\phi C\theta) \end{bmatrix}$$

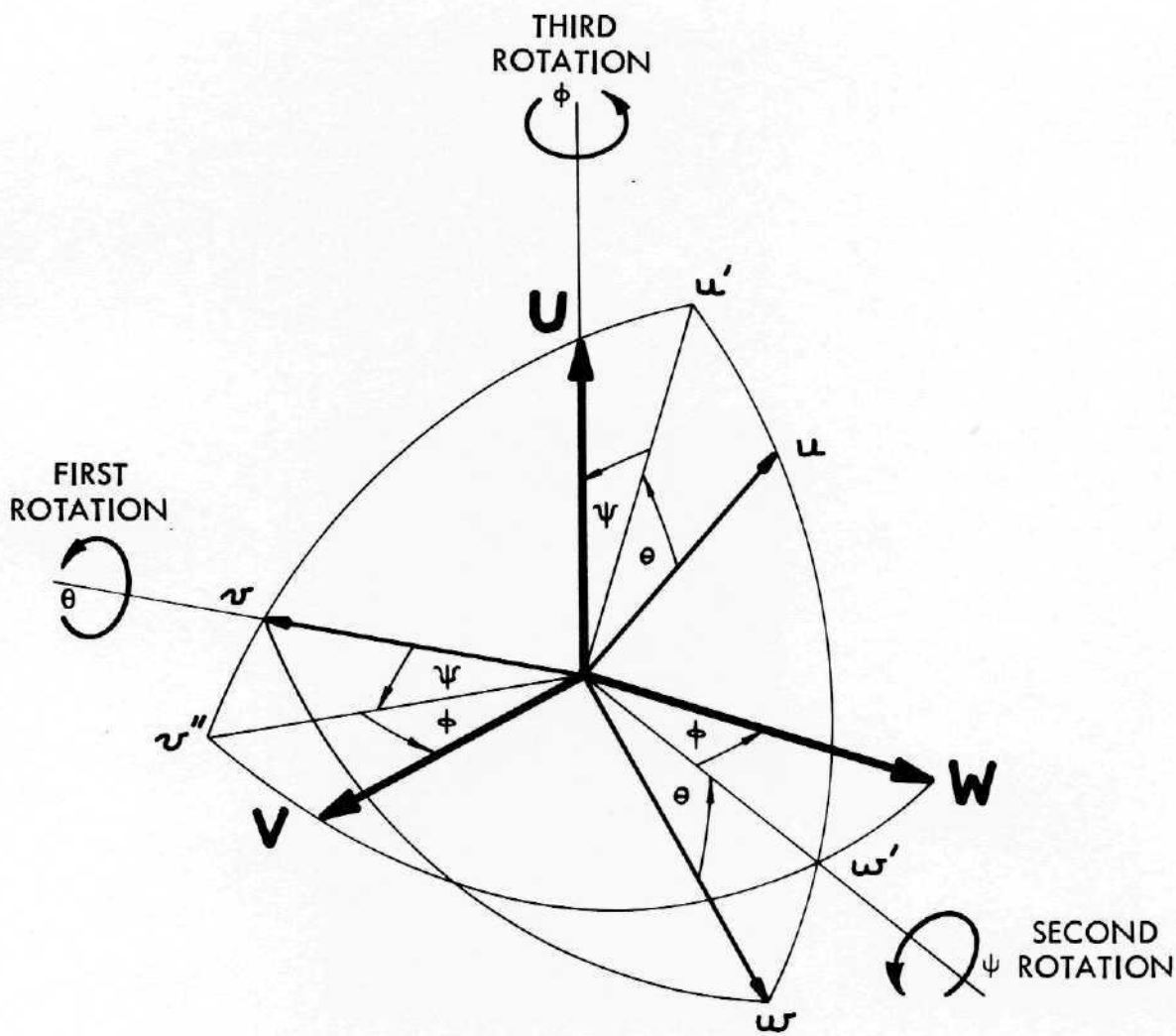
The PGNCS limits the range of the middle gimbal angle,  $\psi$ , to  $\pm 60$  degrees. The inner and outer gimbal angles,  $\theta$  and  $\phi$ , have a range of  $\pm 180$  degrees.

#### 2.4.2 PGNCS Downlink Telemetry

##### 2.4.2.1 PGNCS Downlink Telemetry Data Transmission

Data is transmitted serially over the PGNCS Downlink Telemetry line at a rate of fifty 40-bit words per sec. The Downlink Telemetry Register in the AEA, an 18-bit shift register, provides a means for accepting the PGNCS Downlink data under appropriate conditions.

The data is loaded into the Downlink Telemetry Register under the control of clock pulses supplied by the Instrumentation Subsystem at a rate of 51.2 kpps. A counter associated with the register permits



FIRST ROTATION,  $\theta$ , INNER GIMBAL ANGLE  
 SECOND ROTATION,  $\psi$ , MIDDLE GIMBAL ANGLE  
 THIRD ROTATION,  $\phi$ , OUTER GIMBAL ANGLE

Figure 2-4. IMU Euler Angles



the loading of only 18 bits of data from the downlink. Once the 18 bits have been loaded into the register, no additional data can be accepted until the register has been read. The counter is reset when the contents of the register have been read, in parallel, into the AEA memory under flight program control.

The Instrumentation Subsystem transmits a single PGNCS Downlink Stop Pulse to the AEA after the transmission of each 40-bit word. The Stop Pulse sets a discrete flip-flop in the Discrete Word 1 Register. The Downlink Stop Pulse Discrete provides a means for synchronizing the loading of data into the AEA memory to the downlink transmission. Each time the Downlink Telemetry Register is read, the Downlink Stop Pulse Discrete flip-flop is reset.

When PGNCS Downlink data is being accepted by the AEA, under flight program control, the Downlink Stop Pulse Discrete must be interrogated at least twice during each 20-ms period with a minimum separation of 1 ms between samplings to ensure synchronous loading of the Downlink data.

#### 2.4.2.2 PGNCS Downlink Telemetry Data Format

Upon appropriate command to the LM guidance computer (LGC) via the Display and Keyboard Assembly (DSKY), a special data list will be transmitted over the PGNCS Downlink Telemetry line. This list provides navigation initialization data for use by the AGS flight program.

The data list will consist of a block of 17 consecutive words consisting of:

- 1) An identification (ID) word
- 2) Six words defining a position and velocity for the LM in the rectangular coordinates of the IMU stable member
- 3) Two words giving the epoch of the LM position and velocity data in double precision
- 4) Six words defining a position and velocity for the CSM in the rectangular coordinates of the IMU stable member
- 5) Two words giving the epoch of the CSM position and velocity data in double precision

The sequence of words in the list is illustrated in Figure 2-5.

The complete list will be transmitted a total of ten times for each commanded transmission. During the period required for the ten transmissions, no LGC interrupts will occur.

The first, or most significant, bit of a PGNCS downlink word is the word-order-code bit. This bit will be a one if the word is a data word and a zero if the word is an ID word. Following the word order code bit, data words will consist of a sign bit followed by 14 value bits, the most significant bit first. For ID words, a 15-bit binary data index will follow the word order code bit. The binary data index for navigation initialization data is 111 111 111 111 110. For both ID and data words, the 2 least significant bits of the 18 bits loaded are not applicable and should be discarded in processing the word.

Data will be in two's complement binary form. A one in the most significant bit location for position data will have the value  $2^{22}$  ft and a one in the most significant bit location for velocity data will have the value  $2^{12}$  fps. The most significant bit of the epoch will have the value  $2^{17}$  sec.

#### 2.4.2.3 PGNCS Downlink Coordinate Frame

The position and velocity initialization data furnished to the AEA on the PGNCS downlink will be in the inertial reference coordinate frame defined by the orientation of the IMU stable member. Therefore, the data will be compatible with the IMU gimbal angle information described in Paragraph 2.4.1.

#### 2.4.2.4 PGNCS Downlink Reference Time

The epoch of the navigation initialization data provided by the PGNCS will be compatible with the current time base of the AGS clock, i. e., the AGS absolute time base. This compatibility will be achieved by appropriate astronaut procedures.

When the data is furnished on the downlink, the epoch shall be no more than one orbital period earlier than current time and shall be subsequent to the last time of powered flight for the corresponding vehicle. See Subsection 5.3 for additional assumptions which may be made with regard to the constraints on the epochs of navigation initialization data.

- 1) ID word
- 2) LM u position
- 3) LM v position
- 4) LM w position
- 5) LM epoch most significant word
- 6) LM u velocity
- 7) LM v velocity
- 8) LM w velocity
- 9) LM epoch least significant word
- 10) CSM u position
- 11) CSM v position
- 12) CSM w position
- 13) CSM epoch most significant word
- 14) CSM u velocity
- 15) CSM v velocity
- 16) CSM w velocity
- 17) CSM epoch least significant word

Figure 2-5. PGNCS Downlink Word Sequence

Bit No.	0	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	17
ID Word	0	1	1	1	1	1	1	1	1	1	1	1	1	1	1	0	Y	Y
Data Word	1	S	X	X	X	X	X	X	X	X	X	X	X	X	X	X	Y	Y

X = 0 or 1

Y = Not applicable

S = Sign bit (0 or 1)

Figure 2-6. PGNCS Downlink Word Format

#### 2.4.2.5 PGNCS Downlink Input Subroutine

PGNCS Downlink Telemetry data shall be accepted and stored in the AEA memory under flight program control only after an appropriate command is given the system via DEDA. When this command is given, the program shall interrogate the PGNCS Downlink Stop Pulse Discrete twice every 20 ms to determine if a word has been loaded into the downlink telemetry register. When the Stop Pulse Discrete is detected, the contents of the register shall be read into the AEA and the register cleared. Before any data can be stored in memory, a search must be conducted for the ID word which heads the block of navigation initialization data. When the correct ID word is found, the PGNCS Downlink Input subroutine shall read and store in memory the following 16 words of the data block. The data block shall be read and stored one time only.



## 2.5 AGS OUTPUT TELEMETRY

### 2.5.1 AGS Output Telemetry Transmission

The AGS shall supply data for telemetry to the Instrumentation Subsystem by means of the Output Telemetry Register, a 24-bit shift register. AGS Output Telemetry transmission shall be synchronized to a rate of 50 24-bit words per sec by means of control signals supplied by the Instrumentation Subsystem. The Instrumentation Subsystem will supply a burst of 24 AGS telemetry shift pulses, followed by an AGS Output Telemetry Stop Pulse once each 20 ms. The shift pulses will unload the Output Telemetry Register serially at 51.2 kpps, the most significant bit first. The stop pulse sets a discrete flip-flop in Discrete Word 1, the Output Telemetry Stop Discrete.

### 2.5.2 AGS Telemetry Subroutines

The flight program shall contain telemetry subroutines to format each telemetry word and load the Output Telemetry Register in synchronism with the control signals supplied by the Instrumentation Subsystem. One output telemetry word shall be loaded every 20 ms, provided that an AGS Output Telemetry Stop Discrete has been sensed. The program shall interrogate this discrete twice per 20 ms cycle, with at least 1-ms separation between interrogations. The Output Telemetry Register will be loaded only if the discrete is set to ensure that data will not be loaded into the register at the same time data is being shifted out. The AGS Output Telemetry Stop Discrete shall be reset each time a new word has been stored in the Output Telemetry Register.

The telemetry output data shall consist of a 50-word block of computer memory, transmission of which shall be repeated every second. At one second intervals, the program shall be initialized so that transmission of the block of memory is repeated, in sequence, starting with the first word of the block.

### 2.5.3 AGS Output Telemetry Data Format

The 24-bit telemetry output word assembled in the Output Telemetry Register shall consist of a 6-bit ID code and 18 bits of information from a particular AEA memory cell. The ID code shall range sequentially from octal 01 to octal 62 with the ID code octal 01 being used to designate

the first word of the 50-word block. The ID code shall occupy the first 6 most significant bits of the register. The remaining 18 bits, the computer word, shall represent either codes or numerical data. For numerical data, the 18 bits shall consist of one sign bit followed by 17 value bits, the most significant bit first. The sign bit shall be 0 for positive data and 1 for negative data. The value bits shall be in two's complement binary format.

## 2.6 CONTROL INPUTS

Six hardwired control discretets are furnished to the AEA by elements of the Control Electronics Section. These are:

- 1) Automatic Discrete
- 2) Abort Discrete
- 3) Abort Stage Discrete
- 4) Follow-Up Discrete
- 5) Descent Engine On Discrete
- 6) Ascent Engine On Discrete

The function of these discretets is discussed more fully in Section 7, Program Logic. In general terms, they are supplied to control the flight program functions in accordance with the operational decisions and control manipulations of the astronaut and the state of the main engines. The state of these discretets, all of which are included in Discrete Word 1, shall be examined continuously at a rate of once every 40 ms and the information conveyed by them shall be used to accomplish the objectives of the program.



## 2.7 CONTROL AND DISPLAY OUTPUTS

### 2.7.1 Engine Commands

The flight program shall generate either an Engine OFF or an Engine ON command at a rate of once every 40 ms by setting and resetting the appropriate discrete outputs. The rules for determining which command is to be issued are given in Section 7, Program Logic.

### 2.7.2 Attitude Errors

The attitude error signals are defined to be the errors in angular displacement about the LM body axes with respect to the desired orientation as determined by the program logic and steering equations. A positive value of attitude error shall require that a positive right-handed rotation be produced about the corresponding body axis in order to reduce the attitude error to zero.

Attitude error signals shall be computed for both control and display in accordance with the guidance requirements discussed in Section 6 and the program logic discussed in Section 7. The output data shall have the following characteristics:

- 1) Update rate: 25 times per second
- 2) Computation delay: The computation interval between the output of an attitude error signal and the most recent gyro input on which the computation is based shall not exceed 15 ms.
- 3) Word length: Sign plus 9 bits
- 4) Scaling: Least significant bit equals  $0.5113269 \times 10^{-3}$  rad.

The above word length and scaling correspond to a nominal excursion of  $\pm 15$  deg for attitude error.

### 2.7.3 Total Attitude Display Signals

The flight program shall compute and output six quantities suitable for driving the gimbals of the Flight Director Attitude Indicators (FDAI). These quantities are the sines and cosines of that set of Euler angles,  $\alpha$ ,  $\beta$ , and  $\gamma$ , which will cause the FDAI to display the total attitude of

the vehicle with respect to the inertial reference frame of the flight program. The relationship of these Euler angles to the elements of the transformation matrix maintained by the attitude reference routine, Section 4., is as follows:

$$[A] = \begin{bmatrix} (C_{\alpha} C_{\gamma} + S_{\alpha} S_{\beta} S_{\gamma}) & S_{\alpha} C_{\beta} & (-C_{\alpha} S_{\gamma} + S_{\alpha} S_{\beta} C_{\gamma}) \\ (-S_{\alpha} C_{\gamma} + C_{\alpha} S_{\beta} S_{\gamma}) & C_{\alpha} C_{\beta} & (S_{\alpha} S_{\gamma} + C_{\alpha} S_{\beta} C_{\gamma}) \\ C_{\beta} S_{\gamma} & -S_{\beta} & C_{\beta} C_{\gamma} \end{bmatrix}$$

The total attitude display quantities shall be computed and output with the following characteristics:

- 1) Update rate: 25 times per second
- 2) Word length: Sign plus 9 bits
- 3) Scale factor: Most significant value bit equals 0.5.

#### 2.7.4 Lateral Velocity Signals

The projection of the inertial velocity of the LM vehicle onto the angular momentum vector of the CSM orbit plane shall be computed and output to provide a signal for display on the X-Pointers. The output characteristics shall be as follows:

- 1) Update rate: 5 times per second
- 2) Word length: Sign plus 8 bits
- 3) Scaling: Most significant value bit equals 100 fps.

#### 2.7.5 Altitude and Altitude Rate Signals

The altitude of the LM vehicle above the nominal landing site radius and the altitude rate shall be computed and output for display on the Altitude/Range Indicator. Since both quantities are loaded into the same output register, successive loadings of the register must be spaced by a minimum of 270  $\mu$ s to allow time for the previously loaded data to be shifted out. Each output of altitude shall follow the corresponding altitude rate output within no more than 1.5 ms. The update rate for both quantities shall be 5 times per second.

Other output characteristics shall be as follows:

Altitude

Word length: 15 bits

Scaling: Least significant bit equals 2.345 ft

Format: Magnitude

Altitude rate

Word length: Sign plus 14 bits

Scaling: Least significant bit equals 0.5 fps.

Format: Most significant bit is sign bit, one for positive value and zero for negative value.

2.7.6 Test Mode Failure Discrete

The Test Mode Failure Discrete shall be set to activate the AGS Failure Warning Light and the Master Alarm Light to indicate the occurrence of a failure in the AEA or ASA. The discrete shall be set in the event that any one of the following three conditions occur:

- 1) The detection by ASA hardware sensors of out-of-tolerance temperature or power supply voltage, i. e., the following four monitored parameters
  - a) ASA block temperature
  - b) 28 V DC ASA power supply
  - c) 12 V DC ASA power supply
  - d) 29 V, 400 Hz ASA power supply
- 2) The detection of asynchronous program operation as indicated by the fact that an instruction other than a Delay instruction is being executed when the 20 ms timing pulse is generated
- 3) The detection by the Inflight Self-Test Routine of a failure in the AEA memory or circuitry.

A capability for resetting the Test Mode Failure Discrete by means of a DEDA entry shall be provided. This entry shall be made into the inflight self-test status indicator memory location specified in the following paragraph.

#### 2.7.6.1 Inflight Self-Test Routine

The flight program shall contain an Inflight Self-Test Routine which shall be used continuously to indicate that the computer logic circuits and memory are functioning properly. The routine shall provide for two independent tests, a logic test and a memory test. The logic test shall provide for execution of all computer instructions, except those associated with input/output operations. The memory test shall perform a checksum of all of the hardware memory plus most of that portion of the software memory which contains fixed data.

A DEDA accessible memory location shall be varied to indicate the status of the inflight self test. An indication of successful test completion, logic test failure, memory test failure, and a combination of both logic and memory test failure shall be provided. In the event that a failure is indicated, the operation of the Inflight Self Test Routine shall be inhibited. The test shall be reinitiated by resetting the inflight self test status indicator memory location via the DEDA.



## 2.8 GSE SERVICE ROUTINE

The hardwire portion of the AEA flight program shall contain a GSE Service Routine to provide the capability for loading the software (erasable) portion of the AEA memory with data supplied from ground support equipment, either BTME or ACE, through the Downlink Telemetry Register. The number of 18-bit words to be loaded can be variable and will be controlled by the ground support equipment.

The first word loaded shall not be part of the data to be stored. That word shall have the octal form 77xxxx, where xxxx is the starting address for the sequence of memory locations in which the subsequent data is to be stored.

Initiation of data transfer will be controlled by the ground support equipment. In order to control the transfer of data into the AEA memory, the GSE Service Routine shall recognize the receipt of two input discretes, GSE 1 and GSE 2, transmitted from the ground support equipment. In addition, the routine shall generate an output discrete, GSE 5. GSE 1 will be set to notify the AEA that a complete word has been loaded by the ground support equipment into the AEA Downlink Telemetry Register. The GSE Service Routine shall cause the AEA to acknowledge receipt and storage of that word by setting the GSE 5 discrete. Receipt of GSE 5 by the ground support equipment will cause the test equipment to reset GSE 1, and to repeat the input sequence for the next word to be loaded.

Following the transmission of the last data word to be loaded, an additional word will be transmitted, accompanied by the setting of both GSE 1 and GSE 2. This additional word will not be loaded into the AEA memory. Upon sensing GSE 2, the GSE Service Routine shall transfer program control to the starting address given in the first input word.

No provision need be made in the GSE Service Routine for checksum verification of the data transfer or any other indication of successful information transfer.



## 2.9 REFERENCES

- 2-1. GAEC Specification No. LSP-500-1A, "Abort Guidance Section Software, GFE Performance and Interface Specification (U)," 4 February 1969.
- 2-2 MIT/IL STG Memo 1104, 24 January 1968.
- 2-3 G. P. Edmonds and G. C. Cox, MIT/IL STG Memo 1072, "Potential Effect of the CDU Transient Problem on AGS Alignment," 8 November 1967.
- 2-4 G. P. Edmonds, MIT/IL STG Memo 055, "Results of STG Investigation of CDU Problem," 12 September 1967.
- 2-5 MSC Letter EG43-305-68-477, "IMU Fine Align Transient Problem," 29 April 1968.

### 3. ASA DATA PROCESSING

#### 3.1 ACCELEROMETER DATA PROCESSING

As discussed in Paragraph 2.2.2, the flight program is required to read and clear the three accelerometer input registers every 20 ms while the AGS is operating. The accelerometer data obtained in this manner shall be processed as follows:

- 1) The nominal bias of 640 counts per 20 ms interval shall be subtracted from each of the three accelerometer inputs
- 2) Each of the resulting differences shall be multiplied by a corresponding scale factor to convert counts to feet per second
- 3) An accelerometer static bias compensation constant shall be added to each result.

This processed accelerometer data shall be used in the navigation equations (see Subsection 5.1) to provide a measure of the incremental changes in velocity of the LM due to the thrust forces of descent engine, ascent engine, and reaction control system operation and any externally applied forces.

##### 3.1.1 Accelerometer Scale Factor Compensation

A separate scale factor shall be provided for the data from each accelerometer. The scale factors will be determined by calibration and incorporated in the program prior to launch. A capability shall exist for accommodating deviations up to  $\pm 24\%$  of the nominal accelerometer scale factor of 0.003125 fps/pulse. The accelerometer scale factors shall be located in the DEDA accessible portion of the AEA memory. A positive residual scale factor error will exist in the sensed incremental velocity data if the true scale factor exceeds the scale factor in use.

### 3.1.2 Accelerometer Static Bias Compensation

A positive accelerometer static bias exists when the reading from an accelerometer input register exceeds the nominal 640 counts in free fall. The sensed incremental velocity data shall be compensated for positive static bias by the addition of a negative accelerometer static bias compensation constant.

A separate accelerometer static bias compensation constant shall be provided for correcting the data from each accelerometer. Compensation constants will be determined by calibration and incorporated in the program prior to launch. In addition, the flight program shall provide a capability for updating the compensation constants during free fall periods of the mission. See Paragraph 3.1.2.1. Compensation of the incremental sensed velocity data shall be provided for at least  $\pm 2000 \mu\text{g}$  of equivalent accelerometer static bias. The compensation constants shall be located in the DEDA accessible portion of the AEA memory.

No compensation is required for changes in the apparent accelerometer bias resulting from the effects of vehicle dynamics on the instruments.

#### 3.1.2.1 In-Flight Accelerometer Calibration

The flight program shall provide a capability for measuring the equivalent accelerometer static bias during free fall periods of the mission. Upon entry of an appropriate command via the DEDA, the measurements will be taken and the accelerometer compensation constants will be updated within approximately 32 seconds. The calibration technique employed is not required to account for rotation, attitude limit-cycling, or any other dynamic characteristics of the vehicle in free fall.

### 3.2 GYRO DATA PROCESSING

As discussed in Paragraph 2.2.2, the flight program is required to read and clear the three gyro input registers every 20 ms while the AGS is operating. The gyro data obtained in this manner shall be processed as follows:

- 1) The nominal bias of 640 counts per 20 ms interval shall be subtracted from each of the three gyro inputs
- 2) Each of the resulting differences shall be multiplied by a corresponding scale factor to convert counts to radians
- 3) A fixed gyro drift compensation constant shall be added to each result
- 4) A term shall be added to the incremental attitude change indicated by the X gyro reading to compensate for drift due to the spin axis mass unbalance of the X gyro.

#### 3.2.1 Gyro Scale Factor Compensation

Provision shall be made in the flight program to modify the nominal gyro scale factor of  $2^{-16}$  rad/count for each gyro by a correction factor. The deviation from nominal of the scale factor for each gyro shall be determined by calibration and incorporated in the program prior to launch. A capability shall exist for accommodating deviations up to  $\pm 0.78$  percent of the nominal gyro scale factor. The measured deviations, termed the gyro scale factor compensation constants, shall be located in the DEDA accessible portion of the AEA memory.

A positive gyro scale factor deviation exists when the true scale factor for the gyro is greater than the nominal scale factor. A positive gyro scale factor deviation shall be compensated in the AEA by means of a positive gyro scale factor compensation constant.

#### 3.2.2 Fixed Gyro Drift Compensation

Positive fixed gyro drift exists if the reading from a gyro input register exceeds the nominal 640 counts when the attitude of the vehicle is fixed in inertial space. The incremental angular displacement data



obtained from the gyros shall be compensated for positive fixed gyro drift by the addition of a negative fixed gyro drift compensation constant.

A separate fixed gyro drift compensation constant shall be provided for correcting the data from each gyro. Compensation constants will be determined by calibration and incorporated in the program prior to launch. In addition, the flight program shall provide a capability for updating the compensation constants by means of an in-flight calibration technique during free fall periods of the mission and by means of a surface calibration technique when the LM is resting on the lunar surface. See Paragraphs 3.2.2.1 and 3.2.2.2. Capability for fixed gyro drift compensation shall be provided over a range of  $\pm 10$  deg/hr. The compensation constants shall be located in the DEDA accessible portion of the AEA memory.

#### 3.2.2.1 In-Flight Gyro Calibration

The flight program shall provide a capability for updating the fixed gyro drift compensation constants during free fall periods of the mission. The calibration shall be initiated upon entry of an appropriate command via the DEDA and shall be completed within approximately 5 minutes.

The in-flight gyro calibration technique shall use the PGNCS IMU gimbal angle data as described in Paragraph 2.4.1 as reference information. The program shall initially perform a PGNCS to AGS Alignment as described in Paragraph 4.2.1. The calibration procedure shall consist of modifying the fixed gyro drift compensation constants so as to approach a null in the apparent angular displacement of the ASA reference axes with respect to the IMU reference axes as defined by the IMU gimbal angle data. It shall be assumed that an in-flight gyro calibration will not be performed when the rotation rate about any vehicle axis exceeds 0.1 deg/sec.

#### 3.2.2.2 Lunar Surface Gyro Calibration

The flight program shall provide a capability for updating the fixed gyro drift compensation constants while the LM is resting on the lunar surface. The calibration shall be initiated upon entry of an appropriate command via the DEDA and shall be completed within 5 minutes.



The lunar surface calibration technique employed shall be based on the following presumptions:

- 1) Either a PGNCS to AGS Alignment as described in Paragraph 4. 2. 1 or a Lunar Alignment as described in Paragraph 4. 2. 2 has been implemented, and the resulting y inertial axis is within  $\pm 10$  degrees of the lunar angular momentum vector.
- 2) The inclination of the LM X-body axis with respect to the vertical will not exceed 40 degrees.

During lunar surface gyro calibration, the gyro input data shall be modified to remove the effects of lunar rotation. The fixed gyro drift compensation constants shall then be modified so as to produce an apparent null in the rotation rate of the vehicle-fixed coordinate frame with respect to the inertial frame as determined by the attitude reference routine described in Subsection 4. 1.

### 3. 2. 3 X Gyro Spin Axis Mass Unbalance Drift Compensation

For a positive gyro spin axis mass unbalance, a positive thrust acceleration in the direction of the gyro input axis (see Figure 2-1) will produce a negative gyro output unrelated to vehicle rotation, i. e., a negative gyro drift.

The flight program shall compensate for this effect in X gyro data. An X gyro spin axis mass unbalance compensation coefficient will be determined by calibration and incorporated into the program prior to launch. The coefficient shall be located in the DEDA accessible portion of the AEA memory. For a positive gyro spin axis mass unbalance, the compensation coefficient will be positive. Capability shall be provided for accommodating compensation coefficients over a range equivalent to  $\pm 10$  deg/hr/g.

The X gyro spin axis mass unbalance compensation coefficient shall be multiplied by the scaled and compensated incremental velocity data from the X axis accelerometer and the result will be added to the X gyro data.

No spin axis mass unbalance compensation is required for the data from the Y and Z axis gyros. No compensation is required for other sources of error in the gyro data.

## 4. ATTITUDE REFERENCE

### 4.1 ATTITUDE REFERENCE ROUTINE

The flight program shall maintain an inertial attitude reference in the form of a 3 x 3 orthonormal transformation matrix, [ A ] , to define the orientation of the LM body axes (ASA reference axes) with respect to an inertial coordinate frame. The transformation relation is given by:

$$\begin{bmatrix} X \\ Y \\ Z \end{bmatrix} = [A] \begin{bmatrix} x \\ y \\ z \end{bmatrix}$$

The LM body axis coordinate frame, X, Y, Z, is defined in Paragraph 2.2.1. The inertial coordinate frame, x, y, z, shall be established by alignment as specified in Subsection 4.2.

The measurements of vehicle rotation, the processed gyro data specified in Paragraph 3.2, shall be used by the attitude reference routine to update the direction cosines which are the elements of [ A ] . The contribution of the computation alone to the attitude reference drift rate shall not exceed 0.3 deg/hr at any vehicle attitude for vehicle rotation rates up to 5 deg/sec or 0.5 deg/hr for vehicle rotation rates up to 25 deg/sec.

The primary functions of the attitude reference routine are:

- 1) Provide a means to transform the processed accelerometer data specified in Subsection 3.1 into the inertial reference frame in which the navigation and guidance computations will be made
- 2) Provide a means for transforming the results of the guidance computations into steering commands in the form of attitude error signals as specified in Paragraph 2.7.2.
- 3) Provide a source of data for the Total Attitude Display Signals specified in Paragraph 2.7.3.

## 4.2 ATTITUDE REFERENCE ALIGNMENT

The flight program shall provide a capability for attitude reference alignment by initializing the direction cosines of the transformation matrix [ A ] upon command via the DEDA.

Three methods of alignment shall be provided:

- 1) PGNCS to AGS Alignment
- 2) Lunar Surface Alignment
- 3) Body Axis Alignment

Misalignment error exists when the inertial reference frame established by an alignment procedure and maintained by the attitude reference equations through the use of processed gyro data differs from the inertial reference frame which was intended to be established.

### 4.2.1 PGNCS to AGS Alignment

When a PGNCS to AGS Alignment is commanded, the flight program shall input the data contained in the IMU gimbal angle registers as described in Paragraph 2.4.1. The data shall be used to compute the elements of an attitude reference transformation matrix according to the relations shown for the matrix [ B ] in that paragraph. The direction cosines of [ B ] shall then be used to initialize the corresponding elements of [ A ]. The time required to complete a PGNCS to AGS Alignment shall not exceed 5 seconds.

The inertial reference frame established in this manner will be nominally the same as the IMU stable member coordinate frame. It may be presumed that a PGNCS to AGS Alignment will never be commanded within 20 seconds of the time when the PGNCS has transmitted a CDU Zero Signal. See Paragraph 2.4.1.

#### 4. 2. 2 Lunar Surface Alignment

The purpose of a Lunar Surface Alignment is to establish the post-landing inertial reference frame of Paragraph 4. 3. 3 without using the IMU gimbal angle data.

The flight program shall provide a capability for storing the direction cosines of the pre-landing z inertial axis shortly after lunar touchdown upon an appropriate command via the DEDA. A capability shall also be provided for storing data, input via the DEDA, which will be furnished for the purpose of modifying the z inertial axis direction cosines to produce the post-landing inertial frame of Paragraph 4. 3. 3 by correcting for the effects of lunar rotation during the dwell period, changes in the CSM orbit plane, and any changes in the LM attitude due to settling and shifting.

When a Lunar Surface Alignment is commanded, the z inertial axis direction cosines shall be corrected by using the data entered for that purpose and the x inertial axis shall be established to be along the local vertical through the use of processed data from the ASA Y and Z accelerometers. See Subsection 3. 1. The time required for Lunar Surface Alignment shall not exceed 3 minutes.

For the purpose of designing and evaluating the Lunar Surface Alignment routine, the following assumptions shall be made.

- 1) An in-flight accelerometer calibration will be performed prior to landing
- 2) The inclination of the X body axis will not exceed 30 deg from the local vertical
- 3) The alignment will continue for at least 3 minutes and will be terminated within  $\pm 4$  minutes of the nominal lift-off time but prior to the actual lift-off.

#### 4. 2. 3 Body Axis Alignment

When a Body Axis Alignment is commanded, the flight program shall set and maintain the transformation matrix  $[A]$  to be an identity matrix until the alignment is terminated by another command via the DEDA. The inertial reference thus established will correspond to the

orientation of the ASA reference axes at the instant that Body Axis Alignment is terminated. The use of data sources external to the AGS and externally controlled maneuvers will be necessary to establish a specified inertial reference frame by the use of Body Axis Alignment.



### 4.3 INERTIAL REFERENCE FRAMES

There are many possible inertial reference frames which may be established during a mission. Since the preferred method of alignment is the PGNCS to AGS Alignment specified in Paragraph 4.2.1, the inertial reference frames established for given periods may be expected to correspond to the orientations established for the IMU stable member. These orientations are discussed in Paragraphs 5.1.4.2 and 5.6.3.4 of Reference 4-1 and are briefly described below. For all of these frames, the origin is at the center of the attracting body. The AGS shall be capable of operating with the described inertial frames; however when the rendezvous radar capability is being used it may be presumed that for the inertial frame established the x-z plane shall lie within 10 degrees of the CSM orbit plane. For other operating conditions it may be presumed that the x-z plane shall lie within 80 degrees of the CSM orbit plane.

#### 4.3.1 Preferred Alignment

For certain thrusting maneuvers, the x inertial axis will be established to be parallel to the desired thrusting direction at the instant of ignition as determined by the PGNCS. The y inertial axis will be determined by forming the vector product of the desired thrusting direction and the position vector provided these two vectors are not parallel. In the event the desired thrusting direction is parallel to the position vector, the y inertial axis will be determined by forming the vector product of the desired thrusting direction and the velocity error. The z inertial axis will complete the right-handed orthogonal triad.

#### 4.3.2 Nominal Alignment

The nominal alignment for orbital operations will establish the x inertial axis to be along the position vector at a given instant, positive outward from the center of the attracting body. The inertial y axis will lie along the LM orbit angular momentum vector, positive in the direction opposite to the momentum vector. The inertial z axis will complete the right-handed orthogonal triad.

#### 4.3.3 Lunar Landing and Launch Alignment

For the lunar landing mission, the x inertial axis will be established to be along the designated landing site position vector at the nominal

landing time, positive outward from the lunar center. The vector product of the CSM orbit angular momentum vector and the x inertial axis will define the z inertial axis. The y inertial axis will complete the right-hand orthogonal triad. It may be presumed that this resulting y inertial axis will be within  $\pm 10$  degrees of the lunar angular momentum vector for the purpose of Lunar Surface Gyro Calibration.

After landing and re-alignment, the inertial reference frame will be similar to that given above except that the x inertial axis will be defined to pass through the actual landing site at the nominal time of launch.

#### 4.4 REFERENCES

- 4.1. MIT/IL R-567, "Guidance System Operations Plan for Manned LM Earth Orbital and Lunar Missions using Program Luminary, Section 5, Guidance Equations (Revision 1)," November 1968.

## 5. NAVIGATION

### 5.1 LM NAVIGATION EQUATIONS

While the LM is in flight, the flight program shall maintain an estimate of the current position and velocity of the LM in the inertial coordinate system defined by the attitude reference routine specified in Subsection 4.1. This navigation function shall be accomplished by integrating the differential equations of motion. The force due to gravity may be assumed to be that of a spherical attracting body. The LM navigation equations shall accept the processed accelerometer readings specified in Subsection 3.1 and transform them from body axis coordinates to inertial coordinates. To reduce errors which may result from residual accelerometer bias and non-translational dynamic effects such as limit-cycling and vibration, the accelerometer data shall not be used in the integration unless the indicated thrust is greater than a prescribed threshold.

The LM navigation equations shall be initialized as specified in Subsection 5.3 and shall provide a capability for modifying the LM state vector estimates through the use of certain additional data entered via the DEDA. Processed Rendezvous Radar range and range-rate data, as specified in Subsection 5.4, shall be used to modify the current estimate of LM position and velocity relative to the current estimate of CSM position and velocity. Altitude data obtained from any suitable source external to the AGS, e.g., the Landing Radar, shall be used to modify the estimate of the magnitude of the LM inertial radius vector as specified in Subsection 5.5.

#### 5.1.1 LM Navigation Equations Accuracy

LM navigation computation errors are defined to be those errors resulting from non-commutativity in maintaining the inertial attitude reference, from the limitations of the integration and coordinate transformation techniques used, from the time lag due to data collection, processing, and computation, from round-off and truncation, and from the deviations in the actual gravity field from the central force field assumed. In the lunar landing mission, for the period from the initiation of powered

descent to the completion of orbit insertion after an abort from hover, the maximum navigation computation error accumulated in each component of LM inertial position and velocity shall be as follows:

- 1) 2500 ft in  $x$ ,  $y$ , and  $z$  (inertial position),
- 2) 4 fps in  $\dot{x}$ ,  $\dot{y}$ , and  $\dot{z}$  (inertial velocity).



## 5.2 ORBIT PREDICTION EQUATIONS

The flight program shall provide a capability for computing estimates of the position and velocity of a vehicle in a free fall orbit about a spherical attracting body using analytical elliptical orbit propagation equations. The position and velocity estimates shall be computed in the inertial reference frame. The equations shall be used to provide estimates of CSM position and velocity at whatever time these may be required by any function of the flight program. The orbit prediction equations shall also be used to initialize the LM navigation equations as specified in Subsection 5.3 and to provide estimates of LM position and velocity at times other than current time.

### 5.3 NAVIGATION INITIALIZATION

The flight program, upon receipt of an appropriate command via the DEDA, shall accept navigation initialization data either from the Downlink Telemetry Register (see Paragraph 2.4.2) or from those DEDA accessible memory locations where the data will have been previously stored for this purpose. For each vehicle, the LM and the CSM, a navigation initialization state vector will consist of three cartesian components of position, three cartesian components of velocity, and the epoch of this data in AGS absolute time (see Subsection 5.7).

The following assumptions may be made regarding the navigation initialization data furnished:

- 1) The components of position and velocity will be in that inertial reference frame which will be in use during the period for which the data was furnished.
- 2) The epoch of CSM navigation initialization data will be:
  - a) Earlier than current time
  - b) No more than 8191 sec earlier than current time
  - c) Subsequent to the most recent CSM powered maneuver
- 3) When furnished in flight, the epoch of LM navigation initialization data will be
  - a) Within +8191 sec of current time
  - b) Subsequent to the most recent LM powered maneuver
- 4) When furnished on the lunar surface, the epoch of the LM navigation initialization data will be:
  - a) No more than 8191 sec in advance of current time
  - b) Within 10 min of the actual time of lift-off
  - c) Presumed to be the nominal lift-off time.

The navigation initialization equations shall decode and scale the input data as required. The CSM state vector furnished shall be stored to serve as the initial point for use by the orbit prediction equations specified in Subsection 5.2. If the navigation initialization is performed in flight, the orbit prediction equations shall be used to propagate the LM initialization data to current time. The existing position and velocity components maintained by the LM navigation equations shall then be replaced by the new ones. A special sub-routine shall be provided for navigation initialization on the lunar surface. This sub-routine shall initialize the LM navigation equations at lift-off based on the assumption that, prior to lift-off, the LM is traveling in inertial space at a constant velocity equal to that furnished in the input data on a trajectory which passes through the position given in that data at the nominal lift-off time.

#### 5.4 RADAR DATA PROCESSING

The flight program shall provide a capability for accepting and processing data obtained from the Rendezvous Radar. This data shall be used to refine the estimate of the LM position and velocity state vector as maintained by the navigation equations specified in Subsection 5.1.

No direct connections exist between the Rendezvous Radar and the AEA. Radar data shall be obtained by appropriate astronaut procedures and entered into the AEA via the DEDA. The radar data will consist of the following:

- 1) Range between the LM and CSM entered in units of nautical miles quantized at 0.1 n.mi.
- 2) Range Rate between the LM and CSM entered in units of feet per second quantized at 0.1 fps. Range rate shall be positive for range increasing with respect to time.
- 3) A code entered at the instant an apparent null is achieved in the angular displacement between the Z body axis and the pointing direction of the Rendezvous Radar when the radar is locked onto the CSM.

It may be presumed that Range will be entered within 30 seconds of the time that the Z body axis null code is entered. Range Rate will be entered independently of the requirement to null the Z body axis/Rendezvous Radar angular displacement.

The flight program shall accept the radar input data without any editing or reasonableness checks. The data shall be processed to reduce the errors contained therein as defined in Paragraph 5.4.1. This processing shall be consistent with obtaining reasonable reductions in the errors in the estimates of LM position and velocity at ranges not exceeding 400 n.mi. There is no requirement to estimate residual misalignment errors, systematic errors, drifts, or biases in the radar data.

#### 5.4.1 Rendezvous Radar Error Model

- 1) Angle bias error: 15 mr ( $3\sigma$ )  
This angle bias error includes the total vector sum of the per axis misalignments between the ASA reference axes and the Rendezvous Radar.
- 2) Angle random error:
  - a) 8.1 mr to 6.9 mr ( $3\sigma$ ) linear variation from ranges 400 nmi to 200 nmi.
  - b) 6.9 mr ( $3\sigma$ ) from ranges 200 nmi to 5 nmi.
- 3) Range Rate bias error: 1 fps ( $3\sigma$ )
- 4) Range Rate random error: 1.3% of Range Rate or 1.3 fps ( $3\sigma$ ), whichever is larger from 400 nmi to 80 ft. Maximum Range Rate reading is  $\pm 700$  fps.
- 5) Range Rate display error:  $\pm 1/2$  fps (uniform distribution)
- 6) Range bias error:
  - a)  $\pm 120$  ft (uniform distribution) for range less than 50.8 nmi.
  - b)  $\pm 500$  ft (uniform distribution) for range greater than or equal to 50.8 nmi.
- 7) Range random error:
  - a) 0.083% or 300 ft ( $1\sigma$ ) whichever is larger for ranges from 400 nmi to 10 nmi.
  - b) 300 ft ( $1\sigma$ ) for ranges from 10 nmi to 5 nmi.
  - c) 0.33% or 80 ft ( $1\sigma$ ) whichever is larger for ranges from 5 nmi to 1000 ft.
  - d) 80 ft ( $1\sigma$ ) for ranges from 1000 ft to 80 ft.
- 8) Range display error:
  - a)  $\pm 1$  nmi (uniform distribution) for ranges from 400 nmi to 10 nmi.
  - b)  $\pm 50$  ft (uniform distribution) for ranges from 10 nmi to 1000 ft.
  - c)  $\pm 2.5$  ft (uniform distribution) for ranges from 1000 ft to 80 ft.



## 5.5 ALTITUDE UPDATE

The flight program shall provide a capability to update the estimate of the magnitude of the LM inertial radius vector through the use of altitude data entered via the DEDA. The altitude data will be obtained from any suitable source such as, for example, the Landing Radar.

## 5.6 SCALING OF NAVIGATION QUANTITIES

The flight program shall be scaled so as to provide the following ranges for the magnitude of navigation quantities:

Inertial position quantities	$2^{23}$ ft
Inertial velocity quantities	$2^{13}$ fps

Provision shall be made for the entry of navigation initialization data via the DEDA in decimal. The quantization of the least significant decimal digit entered shall be as follows:

Inertial position quantities	100 ft
Inertial velocity quantities	0.1 fps
Time quantities	0.1 min

## 5.7 AGS ABSOLUTE TIME

The flight program shall maintain a clock for use by the guidance and navigation computations. The time indicated by the clock is AGS absolute time. The range of AGS absolute time shall be 0 to 4369 minutes.

AGS absolute time shall be initialized by synchronizing the clock to an external reference through the use of the DEDA. The initial time entered shall be an exact multiple of 0.1 minute. The clock will be initialized each time the AGS is put into the OPERATE condition.

## 6. GUIDANCE ROUTINES

### 6.1 INTRODUCTION

The flight program shall, at all times, compute a guidance solution using one of the routines specified in this section. The particular guidance routine in use shall be determined by the manual setting of a six-state logic selector, termed the Guidance Routine Selector, located in the DEDA accessible portion of the AEA memory. The six selections shall correspond to the following:

<u>Guidance Routine Selector State</u>	<u>Guidance Routine</u>
0	Orbit Insertion
1	CSI
2	CDH
3	TPI - TPI Search Option
4	TPI - TPI Execute Option
5	External $\Delta V$

On the lunar mission, attitude error signals shall be computed to maintain the thrust axis along the vertical if either the LM altitude or altitude rate are less than prescribed threshold values when the Orbit Insertion, CSI, or CDH Routines have been selected.

## 6.2 ORBIT INSERTION

The Orbit Insertion Routine shall be designed to compute attitude error signals and that incremental velocity which will cause a powered flight maneuver to result in the achievement of desired orbital conditions at engine cutoff. The orbit insertion conditions to be targeted are as follows:

- 1) Injection Altitude
- 2) Lower Limit for injection altitude rate
- 3) The desired value of the semi major axis of the post injection coast ellipse when the central angle between the LM and CSM is zero
- 4) Upper and lower limits of the semi major axis of the post injection coast ellipse

The desired values of these target parameters shall be located in the DEDA accessible portion of AEA memory. They will be determined and incorporated into the program prior to launch.

The final horizontal velocity at engine cutoff shall be computed to provide an orbit which is a NASA/MSC specified function of the central angle between the LM and CSM. The Orbit Insertion Routine shall continuously alter the engine cutoff conditions to those that would produce the desired orbit condition. If the equations determine that these orbital conditions are not attainable, the logic shall compute targeting to place the LM on a rising trajectory at engine cutoff.

The Orbit Insertion Routine shall also provide for steering the LM toward the CSM orbit plane to reduce or eliminate wedge angle between the resulting LM and CSM orbit planes. The extent to which out-of-plane error is removed shall be determined by DEDA accessible constants. A capability for removing up to .5 degrees of wedge angle shall be provided for the case of abort from the lunar surface.



### 6.3 COELLIPTIC FLIGHT PLAN RENDEZVOUS GUIDANCE

Two orbits are defined to be coelliptic when their principal foci are coincident and when the products of the length of the semi-major axis and eccentricity of each orbit are equal. The definition also implies that when the two orbits are co-planar, they have coincident centers and vacant foci. For the purpose of designing the guidance routines, deviations from co-planarity may be assumed to be small. Coelliptic orbits will have an essentially constant differential altitude (constant delta-h).

The flight program shall contain a set of guidance routines designed to calculate the powered maneuvers required for the LM to execute a coelliptic flight plan for rendezvous with the CSM. This flight plan nominally calls for three powered maneuvers:

- 1) CSI (coelliptic sequence initiate) maneuver
- 2) CDH (constant delta-h) maneuver
- 3) TPI (terminal phase initiate) maneuver

The LM is placed in an orbit which is coelliptic with that of the CSM by first executing the CSI maneuver and then the CDH maneuver.

The final maneuver of the coelliptic flight plan sequence, the TPI maneuver, is nominally a burn which places the LM on a direct intercept trajectory with the CSM and is initiated when the angle of the LM-CSM line-of-sight with respect to the local horizontal is at a preferred value.

#### 6.3.1 CSI Routine

The CSI routine shall provide a capability for computing that velocity increment, parallel to the CSM orbit plane, which will be required to satisfy the following target conditions input via the DEDA.

- 1) Desired AGS absolute time of execution of the CSI maneuver
- 2) Desired AGS absolute time of execution of the TPI maneuver

- 3) Cotangent of the desired angle of the LM-CSM line-of-sight with respect to the local horizontal at the desired time of execution of the TPI maneuver
- 4) Codes designating the desired time of the CDH maneuver. The CDH maneuver will be designated to be performed at a time from CSI equal to one half or three halves of the LM orbital period.

The CSI routine shall compute that incremental velocity required from the CSI burn and from the CDH burn, that estimated differential altitude between the LM and CSM orbits at the time for the CDH maneuver, and that differential time from the CSI to the CDH burn which results in the desired LM to CSM line-of-sight angle at the desired time of the TPI maneuver.

The computed incremental velocities, time from CSI to CDH maneuver, and the predicted differential altitude between the LM and CSM orbits shall be made available for display via the DEDA along with such other parameters as may be useful in evaluating the result.

The computation of the required CSI maneuver velocity increment shall be based on the predicted positions and velocities of the CSM and LM at the targeted CSI maneuver execution time. The velocity increment computed shall be automatically furnished to the External  $\Delta V$  Routine. See Subsection 6.5.

The CSI Routine will not be in use during the burn.

#### 6.3.2 CDH Routine

The CDH Routine shall compute the incremental velocity required to make the LM orbit coelliptic with the CSM orbit. The maneuver execution time, determined explicitly upon entry to the CDH Routine, may be changed by substituting a value via the DEDA. The required velocity increment computed shall be based on the predicted positions and velocities of the LM and CSM at that time. The vector velocity increment computed shall be automatically furnished to the External  $\Delta V$  Routine. See Subsection 6.5.

The CDH Routine will not be in use during a burn.

### 6.3.3 TPI Routine

The TPI Routine shall calculate the velocity increment required to place the LM on a direct intercept transfer trajectory based upon a specified time interval for the transfer, input via the DEDA, and a designated time for initiation of the transfer. This routine shall be used in two alternative options as determined by appropriate setting of the Guidance Routine Selector, the TPI Search option and the TPI Execute Option.

In the TPI Search option, the routine shall continuously solve for that intercept trajectory which is initiated at a fixed time increment in advance of current time. The time increment will be input via the DEDA.

In the TPI Execute option, the designated transfer initiation time shall remain fixed. This fixed absolute time shall be either that which corresponded to the transfer initiation time at switchover from the TPI Search option to the TPI Execute option or a new time input subsequently via the DEDA.

A capability shall be provided for modifying the velocity increment computed by the routine so that the transfer orbit produced will intersect the CSM orbit at some selected distance from the CSM. This shall result from a DEDA input which displaces the estimate of CSM position. The input will be the time, in minutes to the nearest 0.01 minute, required for the CSM to traverse the desired displacement.

An additional capability shall be provided which will produce a maneuver similar to the nominal direct transfer. However, the direction of the burn and the plane of the transfer orbit shall be rotated so that the LM will intersect the CSM orbit plane at a specified time prior to the time at which rendezvous would have otherwise taken place. The specified time shall be entered via the DEDA in minutes to the nearest 0.01 minute. The altitude at which the LM will intersect the CSM orbit plane shall be the same as though the nominal maneuver were computed.

The required velocity increment computed by the TPI Routine shall be based on the predicted positions and velocities of the LM and CSM at the designated transfer initiation time. The vector velocity increment computed shall be automatically furnished to the External  $\Delta V$  Routine. See Subsection 6.5.

The TPI Routine will not be in use during a burn.

The TPI Routine shall not be required to compute transfer trajectories which result in intercept with the CSM more than  $2^{13}$  seconds in the future. Also, it may be presumed that the specified time interval for the transfer will not be chosen to produce solutions in which the central angle traversed by the LM between the transfer initiation time and the nominal rendezvous time is close to 0 degrees, 180 degrees, or any multiple of 180 degrees. The central angle traversed shall not lie within  $\pm 10$  degrees of these singularity regions. Subject to these constraints, the TPI Routine will be used to compute direct intercept maneuvers at any appropriate time including midcourse correction maneuvers subsequent to the TPI maneuver.

#### 6.4 PCI CAPABILITY

A capability shall exist to compute a PCI (plane change initiate) maneuver between orbit insertion and rendezvous which will satisfy the following target condition input via DEDA.

- 1) Desired AGS absolute time of execution of the PCI maneuver.

For this purpose the CSI Routine shall estimate the LM out-of-plane velocity, valid at a specified target time entered via DEDA (which if nulled would create a new nodal point  $90^\circ$  ahead of the target time). This computed velocity and the time to go to the PCI maneuver shall be made available for display via the DEDA.



## 6.5 EXTERNAL $\Delta V$ ROUTINE

The External  $\Delta V$  Routine shall provide a capability for controlling maneuvers for which the required incremental velocity is computed either by a source external to the AGS or by the CSI Routine, the CDH Routine, or the TPI Routine.

When the External  $\Delta V$  Routine is targeted from an external source via the DEDA, or by the CSI Routine as specified in Paragraph 6.3.1, or by the CDH Routine as specified in Paragraph 6.3.2, the three components of the desired incremental velocity furnished shall be defined in the following way:

- 1)  $\Delta V_1$  is in the local horizontal plane, positive in the direction of the motion and parallel to the CSM orbit plane
- 2)  $\Delta V_2$  is perpendicular to the plane in which  $\Delta V_1$  and  $\Delta V_3$  are defined, positive in the direction opposite to the CSM angular momentum vector
- 3)  $\Delta V_3$  is along the local vertical, positive downward toward the center of attraction.

The velocity components  $\Delta V_1$  and  $\Delta V_3$  shall be targeted by the CSI Routine and by the CDH Routine. Both the CSI maneuver and the CDH maneuver may be modified to include an out-of-plane component by an appropriate DEDA entry for  $\Delta V_2$ . Prior to the time the incremental velocity is fixed in inertial space, as described below, the External  $\Delta V$  Routine shall compute the desired incremental velocity vector along the vector resultant of the components in real time. The desired direction of the thrust axis shall be defined to lie along this incremental velocity vector and will be continually reoriented with respect to the inertial reference frame so that the definition of the incremental velocity vector as given above remains current. When the ullage counter is first incremented or a code is entered via the DEDA into an appropriate location provided for this purpose, the targeted incremental velocity vector shall be fixed in inertial space and attitude error signals shall be computed to maintain the thrust axis along this fixed vector.

When the External  $\Delta V$  Routine is targeted by the TPI Routine as specified in Paragraph 6.3.2, the required incremental velocity vector shall be held fixed in inertial space from the time the External  $\Delta V$  Routine is first entered. The routine shall compute the desired direction of the thrust axis to be along this fixed vector.

During a burn, the External  $\Delta V$  Routine shall continue to compute that value of the remaining vector velocity-to-be-gained so that the fixed required velocity increment is implemented.

## 7. PROGRAM LOGIC

### 7.1 PROGRAM MODES AND SUBMODES

The operating modes of the flight program shall be designated as follows:

- 1) Inertial Reference Mode
- 2) Alignment and Calibration Mode

When the flight program is in the Inertial Reference Mode, it shall be in one of the following submodes:

- 1) Attitude Hold Submode
- 2) Guidance Steering Submode
- 3) Z Body Axis Steering Submode

When the flight program is in the Alignment and Calibration Mode, it shall be in one of the following submodes:

- 1) PGNCS to AGS Alignment Submode
- 2) Lunar Alignment Submode
- 3) Body Axis Alignment Submode
- 4) Gyro and Accelerometer Calibration Submode
- 5) Accelerometer Calibration Only Submode

The particular submode in which the flight program will be operating shall be determined by the setting of a single eight-state logic selector, termed the AGS Submode Selector, which shall be located in the DEDA accessible portion of the AEA memory. The program shall execute the selection logic every 40 ms to determine the appropriate submode of operation.

#### 7.1.1 Inertial Reference Mode

##### 7.1.1.1 Attitude Hold Submode

When the flight program is in the Attitude Hold Submode, attitude error signals shall be computed which, if used to control the vehicle, shall maintain the attitude constant with respect to the inertial reference frame. The attitude maintained shall be that which existed at the time of entry into this submode.

#### 7.1.1.2 Guidance Steering Submode

When the flight program is in the Guidance Steering Submode, attitude error signals shall be computed which, if used to control the vehicle, shall orient the thrust axis in the direction of the desired incremental velocity computed by the particular guidance routine currently in use (in Orbit Insertion the desired thrust direction is computed by a boost guidance law). An operator option shall be provided to orient the LM Z body axis either parallel to the CSM orbit plane or parallel to an arbitrary plane specified by the components of a unit vector normal to that plane input via the DEDA.

For the purposes of this submode, the thrust axis is defined to be the X body axis when either the descent engine or the reaction control system is to be used to produce a powered maneuver. When the ascent engine is to be used, the thrust axis assumed for computing attitude error signals shall be offset from the X-body axis by a cant angle determined from program constants. A DEDA accessible logic switch shall be provided to control the use of the cant angle offset. Whether or not the switch is set manually, the cant angle offset shall be used automatically when the ascent Engine ON Discrete is sensed. The switch shall be automatically reset when Engine OFF is commanded at the completion of the maneuver.

#### 7.1.1.3 Z Body Axis Steering Submode

When the flight program is in the Z Body Axis Steering Submode, attitude error signals shall be computed in accordance with one of two options which shall be selected by a code entered via the DEDA. The attitude error signals, if used to control the vehicle, shall orient the Z body axis either along the estimated line-of-sight to the CSM or in the direction of the desired incremental velocity computed by the particular guidance routine currently in use. For this latter option, the Z body axis is defined to be the thrust axis. The X body axis shall be maintained parallel to the CSM orbit plane for either option.

### 7.1.2 Alignment and Calibration Mode

When the flight program is in any submode of the Alignment and Calibration Mode, the attitude error signals shall be set to zero and only Engine OFF commands shall be issued.

#### 7.1.2.1 PGNCS to AGS Alignment Submode

When the flight program is in the PGNCS to AGS Alignment Submode, attitude reference alignment shall be implemented as specified in Paragraph 4.2.1. When the alignment is completed, the program shall be automatically switched to the Attitude Hold Submode.

#### 7.1.2.2 Lunar Alignment Submode

When the flight program is in the Lunar Alignment Submode, attitude reference alignment shall be implemented as specified in Paragraph 4.2.2. The alignment shall be terminated via DEDA command.

#### 7.1.2.3 Body Axis Alignment Submode

When the flight program is the Body Axis Alignment Submode, attitude reference alignment shall be implemented as specified in Paragraph 4.2.3. The alignment shall be terminated via DEDA command.

#### 7.1.2.4 Gyro and Accelerometer Calibration Submode

When the flight program is in the Gyro and Accelerometer Calibration Submode, an in-flight gyro calibration as specified in Paragraph 3.2.2.1 and an in-flight accelerometer calibration as specified in Paragraph 3.1.2.1 shall be implemented provided the LM is not resting on the lunar surface. On the lunar surface, a lunar surface gyro calibration only shall be implemented as specified in Paragraph 3.2.2.2. When the calibration is completed, the program shall be automatically switched to the Attitude Hold Submode.

#### 7.1.2.5 Accelerometer Calibration Only Submode

When the flight program is in the Accelerometer Calibration Only Submode, an in-flight accelerometer calibration shall be implemented as specified in Paragraph 3.1.2.1. Approximately 300 seconds after this submode is entered, the program shall be automatically switched to the Attitude Hold Submode. The calibration shall be completed in 32 seconds, after which time the mode may be manually terminated.



## 7.2 SYSTEM OPERATING STATES

The flight program and the AGS generally shall be considered to have three operating states.

- 1) Follow-Up State
- 2) Attitude Control State
- 3) Complete Control State

The state in which the system is operating shall be determined by the six hardwired control discretes listed in Subsection 2.6.

### 7.2.1 Follow-up State

The flight program and the AGS shall be in the Follow-up State when the Follow-up Discrete is present. If the Automatic Discrete is present, the attitude error signals computed in accordance with the particular submode in which the program is operating will be output. If the Automatic Discrete is absent, attitude error signals shall be set to zero.

When the system is in the Follow-Up State, the program shall issue an Engine ON command every 40 ms provided the program is in the Inertial Reference Mode and either the Descent Engine On Discrete or Ascent Engine On Discrete is present. Otherwise, the flight program shall issue an Engine OFF command every 40 ms.

### 7.2.2 Attitude Control State

The flight program and the AGS shall be in the Attitude Control State when the Follow-up Discrete, the Abort Discrete, and the Abort Stage Discrete are all absent. In this state, attitude error signals shall be output which are computed in accordance with the particular submode in which the program is operating if the Automatic Discrete is present. If the Automatic Discrete is absent and if the program is in the Inertial Reference Mode, the program shall compute Attitude Hold error signals.

When the AGS is in the Attitude Control State and the flight program is in either the Guidance Steering Submode or the guidance steering option of the Z Body Axis Steering Submode and either the CSI, CDH or TPI Routine has been selected, the flight program shall command and hold the

desired attitude for the burn. After External  $\Delta V$  has been selected, orbital rate torquing shall result prior to the CSI and CDH burn but the desired inertial attitude shall be maintained prior to the TPI maneuver.

When the system is in the Attitude Control State, the program shall issue an Engine ON command every 40 ms provided the program is in the Inertial Reference Mode and either the Descent Engine On Discrete or Ascent Engine On Discrete is present. Otherwise, the flight program shall issue an Engine OFF command every 40 ms.

### 7.2.3 Complete Control State

The flight program and the AGS shall be in the Complete Control State when the Follow-Up Discrete is absent and either the Abort Discrete or the Abort Stage Discrete is present. In this state, attitude error signals computed in accordance with the particular submode in which the program is operating will be output. If the Automatic Discrete is absent and the program is in the Inertial Reference mode, the program shall compute Attitude Hold error signals. If the Automatic Discrete is present, any submode is permissible with two exceptions.

- 1) The program shall automatically be held in the Attitude Hold Submode when the Abort Stage Discrete is present from the time the Ascent Engine On Discrete is first received until a preselected time thereafter. This preselected time quantity shall have a range of 0 to 10 seconds.
- 2) At the completion of a maneuver when Engine OFF is commanded because the conditions for issuing Engine ON commands, as listed below, are no longer satisfied, the program shall automatically be returned to the Attitude Hold Submode, unless only the ullage condition is not satisfied. If only ullage conditions are lost (as discussed below) Engine OFF shall be issued but the Submode shall not automatically change.

When the system is in the Complete Control State, the program shall issue an Engine ON command every 40 ms when all of the following four conditions are simultaneously satisfied.

- 1) The Automatic Discrete is present
- 2) The AGS Submode Selector has been manually set to Guidance Steering via the DEDA

- 3) Either the ullage condition has been satisfied or the Ascent Engine has been ignited while the LM is on the lunar surface
- 4) Either the magnitude of velocity-to-be-gained exceeds a specified threshold or the projection of the velocity-to-be-gained on the X body axis is in the + X body axis direction and exceeds the nominal ascent engine cutoff velocity impulse.

Otherwise, the flight program shall issue an Engine OFF command every 40 ms.

The ullage condition will be satisfied when the accumulated sensed velocity increments along the X body axis have exceeded a specified threshold in each consecutive two-second computation cycle for a prescribed number of cycles. The ullage condition shall be reset and the count restarted if the accumulated sensed velocity increment along the X body axis falls below the specified threshold in any one two-second computation cycle.

## 8. PROGRAM VERIFICATION

### 8.1 INTRODUCTION

When the flight program has been designed and coded for the AEA, both the design and the coded program shall be verified in detail by formal studies and tests implemented on general purpose computers using simulation techniques. The various modes of testing to be employed are described generally below as guidelines. However, prior to the implementation of testing, a detailed Program Verification Test Plan document shall be prepared and submitted to NASA/MSC for approval and concurrence. The test plan shall describe the tests which will be conducted and their individual objectives. The tests shall be devised to demonstrate that the flight program functions have been designed to fulfill the requirements of the AGS in its mission aboard the LM vehicle as defined by this specification.

At the completion of the testing specified in the approved test plan, test results shall be documented in a Verification Test Results Summary which shall be submitted to NASA/MSC for approval. The Verification Test Results Summary shall contain sufficient data to indicate successful completion of the test plan. All computer outputs produced by the tests, whether included in the documentation or not, shall be retained and made available for examination on request.

Upon acceptance of the Verification Test Results Summary by NASA/ MSC, the verification of the flight program will be considered to be complete and the requirements of this specification fulfilled.

## 8.2 VERIFICATION TESTING

Three classes of tests shall be performed to verify that the flight program will fulfill its mission and that its design and coding are as free from inadequacies and errors as may reasonably be determined by the use of software simulation techniques only. The three classes of tests are characterized by the type of simulation used to implement them.

- 1) Scientific Flight Simulation Tests
- 2) Interpretive Computer Simulation Tests
- 3) Interpretive Computer Simulation/Flight Simulation Tests

### 8.2.1 Scientific Flight Simulation Tests

The scientific flight simulation (SFS) is a general purpose computer program in which the flight program equations and logic are programmed in a convenient compiler language suitable for scientific studies. The representation of the flight program is coupled to a flight simulation (FS) containing models of the vehicle and the environment along with the capability for integrating trajectories in a closed-loop configuration. See Reference 8-1. The FS is a 3 degree-of-freedom simulation.

The SFS shall be used to demonstrate equation performance and to generate trajectories and results for comparison with those generated by the ICS described in Paragraph 8.2.2 and the ICS/FS described in Paragraph 8.2.3.

### 8.2.2 Interpretive Computer Simulation Tests

The interpretive computer simulation (ICS) is a general purpose computer program which simulates the operation of the AEA precisely, bit-for-bit, including the operation of the interfaces.

The flight program, as coded for the AEA, shall be loaded into the ICS and exercised, using suitable driver routines, in an open-loop configuration. Sufficient testing shall be conducted to check out the program, that is, to demonstrate that the various flight program functions have been correctly coded in accordance with both the program and computer design.



### 8.2.3 Interpretive Computer Simulation/Flight Simulation Tests

The Interpretive Computer Simulation/Flight Simulation (ICS/FS) is a general purpose computer program which combines the ICS described in Paragraph 8.2.2 with the FS previously mentioned in Paragraph 8.2.1. See Reference 8-2.

The ICS/FS shall be used to demonstrate that the flight program, as coded for the AEA, functions correctly in a closed-loop configuration on all phases of the missions which have been defined by means of reference trajectories and mission profiles.

### 8.3 REFERENCES

- 8-1 TRW Report 05952-6176-T000, " User's Guide, Scientific Simulation, LM AGS Coelliptic Rendezvous Equations, " September 1967.
- 8-2 TRW Report 05952-6157-T000, " DMCP LM AGS Interpretive Computer Simulation/Flight Simulation (ICS/FS) Description, " March 1968.

## 9. PRODUCT FORMAT

### 9.1 SOFTWARE DELIVERABLES

For the purpose of what will be termed program delivery, TRW shall generate and furnish the following items to MSC, GAEC/Bethpage, and GAEC/KSC:

- 1) An AEA program listing
- 2) An AEA compressed deck
- 3) An AEA absolute deck
- 4) An L & V merged AEA absolute deck (deliverable to GAEC/Bethpage only)
- 5) A BTME tape and corresponding TTG (Test Tape Generator) listing

Final program delivery will ordinarily take place several months prior to launch to allow time for responsible organizations to familiarize themselves with the program and to conduct such additional verification as may be directed by MSC. Early delivery precludes use of instrument scale factors and compensation constants based on calibration data collected on the specific flight hardware. Neither can there be assurance, far in advance of launch, that the values of other hardware, vehicle, and mission dependent constants are firmly established. Therefore, as the term has been used and elsewhere throughout this specification, flight program delivery implies the use of either nominal values for certain constants or values based on such other data as may have been furnished in a timely fashion.

For loading the AEA software memory just prior to launch, TRW shall prepare and deliver an AGS Mission Constants Update Package as described in Paragraph 9.2.

#### 9.1.1 AEA Program Listing

AEA program listings are described in some detail in Reference 9-1. A listing is a printed record of the program which contains the program in symbolic machine language, the assembled AEA code in octal, and comments provided by the programmer for clarification of the coding.

### 9.1.2 AEA Compressed Deck

An AEA compressed deck is a deck of cards on which the symbolic form of the AEA program has been punched in a compressed binary format. This deck contains the complete program including both the hardware and software memory coding. In the course of developing a program for the AEA, modifications and corrections are initiated by punching new cards individually for a compressed deck. This modifying deck is then processed by the TRW LM Assembly Program (LEMAP) which produces an updated compressed deck, absolute deck, and program listing. For a description of LEMAP, see Reference 9-1 which also contains the basic information needed to code a program for the AEA.

### 9.1.3 AEA Absolute Deck

The AEA absolute deck produced by LEMAP is a deck of cards on which the assembled binary AEA code of the flight program, including both the hardware and software memory, is punched in column binary format as described below. The AEA absolute deck shall be the standard source of program information. It shall be used to load the ICS for all program verification testing and as the input source for those programs which generate the BTME tapes and AEA file tapes.

An L & V merged AEA absolute deck is one which contains an ACE Load and Verify Program in AEA memory addresses 0000 through 0205 along with the flight program in memory addresses 0206 through 7777. The ACE Load and Verify Program is described in Reference 9-3.

#### 9.1.3.1 AEA Absolute Deck Format

An AEA absolute deck is in column binary format. There are twelve bits in each column. Each successive group of three columns comprises an IBM 7094 word. Each card contains twenty-four 36-bit IBM 7094 words in card columns 1 through 72. The card format is as follows:

- 1) The first word on each card is always a control word. Labelling the bits 0-35 from top to bottom and left to right, Bit 0 is always 0; Bits 6-8 and 12-23 are the IBM 7094 octal address of the first word on the card (the first AEA word is always loaded into 7094 address 54000). Bits 9-11 are always 101; Bits 24-35 are a card checksum generated by adding the 12-bit columns with an end around carry of any overflow.

- 2) The second word on the first card always contains the contents of the first word of AEA memory (AEA address 0000, 7094 address 54000). The second word on all other cards will never be a word of all zeros. All zero words which might otherwise fall at the second word position of any card are omitted from the deck.
- 3) The AEA word is contained in the most significant 18 bits of the IBM 7094 word.
- 4) After the last AEA word in the absolute deck is a word in 7094 address 64000 which is a checksum of all previous AEA words in the deck. This word can have bits in the low order 18 bits, as the checksum is generated with an end around carry.
- 5) After the checksum word, eighty 7094 addresses are left blank after which a deck identifier and date appears starting at 7094 address 64121.
- 6) Each card of the deck contains a unique program identification number in card columns 73-76 and a sequence number in card columns 77-80.

#### 9.1.4 BTME Tape

A BTME tape is a punched mylar tape prepared for loading the software memory by way of the AGS Bench Test Maintenance Equipment. The format of the tape is defined in Reference 9-2.

A BTME tape is prepared by using the Test Tape Generator Program (TTG) to produce an intermediate magnetic tape and a TTG listing. The AEA absolute deck, along with certain necessary control cards, is used as the input to this program. The data on the intermediate magnetic tape is used to punch the BTME tape. The data on the BTME tape is then transferred to a second magnetic tape which is verified, bit-by-bit, against the original absolute deck and control cards using the TTG Verify Program.



## 9.2 AGS MISSION CONSTANTS UPDATE PACKAGE

Shortly prior to launch, TRW shall prepare and deliver an AGS Mission Constants Update Package. This will contain the same flight program furnished in accordance with Paragraph 9.1 but will use MSC approved equation constants which will have been determined to be appropriate for the specific flight hardware and mission profile. The package will contain the following items:

- 1) An AEA program listing
- 2) A constants dictionary listing
- 3) An updated AEA absolute deck
- 4) An AGS Mission Constants Tape (an AEA ACE file tape) and tape dump
- 5) An updated BTME tape and TTG listing

The AEA program listing furnished shall be identical to that which was furnished previously for final program delivery in accordance with Paragraph 9.1.

The constants dictionary listing and the updated AEA absolute deck will be produced using the TRW LM Dictionary Analysis Program (LMDAP). The inputs to LMDAP are the AEA absolute deck of the previously delivered version of the same flight program, a second card deck on which the new values of the updated constants will have been punched and a third deck of analysis cards defining the characteristics of all of the equation constants. The updated AEA absolute deck produced will contain the new values of the constants. The constants dictionary listing will define all of the equation constants in the updated flight program. The listing will also contain a table showing the address of every AEA memory cell for which the contents have been updated along with both the old and updated contents of that address in octal.

The updated AEA absolute deck will be used as input to the TTG to generate the updated BTME tape and TTG listing and as input to the TRW LM Magnetic Tape program (LMMT) to generate the AGS Program Tape with Updated Mission Constants and an AEA file tape. The AEA file

tape will be generated in accordance with the format specified in Reference 9-4. The AEA file tape will be verified bit-by-bit against the original data used in its generation by a separate computer run using an LMMT verify option.

### 9.3 SOFTWARE IDENTIFICATION

Each assembly of a flight program shall be given a unique numerical identification code and each released version shall be given a unique functional identification code.

The numerical identification, which shall be a four digit number, shall appear in the first line of all program listings, shall be punched on every card of all decks, and shall appear on the label of all tapes. All items with the same identification code shall be bit-for-bit identical in terms of AEA memory load data.

Each release of the program shall have a functional identifier of the form LM AGS FP6 S03Y. LM AGS is the system identifier. FP6 is the program identifier. S03 indicates a software program compatible with the H03 hardware memory. The Y is a revision letter. This revision letter shall be left blank for the initial release of a program and for subsequent releases shall contain A, B, etc. All items released shall be labelled with the functional identifier.

#### 9.4 REFERENCES

- 9-1. TRW Report 7322.2-17, "Abort Electronic Assembly Programming Reference," April 1966.
- 9-2. TRW Report LEM-MI, Revision D, "AGS/BTME Test Set Programming Manual," 5 May 1966.
- 9-3. GAEC Memorandum LMO-500-577, "Revisions to TRW Generated Flight Program Load and Verify," 15 May 1967.
- 9-4. Memorandum of understanding between Grumman Aircraft Engineering Corporation, NASA Contract NAS 9-1100 and TRW Systems NASA Contract NAS 9-4810 for Deliverable Abort Guidance System Software, 17 January 1969