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GRUMMAN AIRCRAFT ENGINEERING CORPORATION
Bethpage, L. I., N. Y.
Code Ident. No. 26512



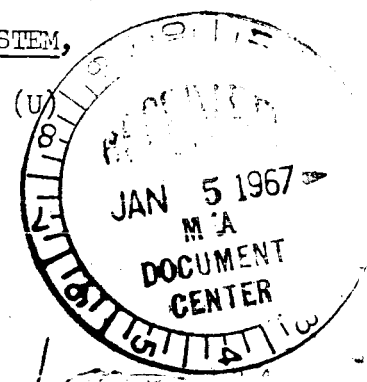
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Specification No. LSP-300-3B

Release Date ~~11-9-66~~
12-2-66

SPECIFICATION

ABORT GUIDANCE SECTION,
GUIDANCE, NAVIGATION AND CONTROL SUBSYSTEM,
DESIGN CONTROL SPECIFICATION FOR (U)



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ERRATA SHEET

LSP-300-3B

Dated 11-9-66

The paragraphs of this specification conforms in detail to the following documents:

<u>LSP-300-3B paragraph</u>	<u>Description</u>	<u>Conforms to</u>
1	Scope	LSP-300-3A
2	Applicable Document	LSP-300-3A modified by LVC-300-033-31 item 43. P & I specification para. 3.1
3.1.1, 3.1.2	General AGS Performance (Abort Guidance Section)	LSP-300-3A modified by LVC-300-033-47 and P&I specification para. 3.2.1; 3.2
3.1.3	AGS Functional Requirements	LSP-300-3A modified by LVC-300-033-31 and modified by P&I specification para. 3.2.2 to 3.2.2.10
3.1.4	Design Mission	P & I specification paras. 3.4.1.2 to 3.4.1.2.6
3.2.3	Mission Environment	LSP-300-3A and LVC-300-033-076A
3.3.1	Operations Performed	P & I specification paragraph 3.2.3
3.3.2.1	Inertial Reference Alignment Accuracy	LSP-300-3A modified by LVC-300-033-31, P&I Specifica- tion paragraph 3.2.4.1.1
3.3.2.1 (a), (b), (c)	Alignment Accuracy	LSP-300-3A modified by LVC-300-033-31, P&I Specification paragraph 3.2.4.1.1(a), (b), (c)
3.3.2.2 to 3.3.2.6	Error Signals	LSP-300-3A, P&I Specification paragraph 3.2.4.1.2

ERRATA SHEET (Continued)

<u>Paragraph</u>	<u>Description</u>	<u>Conforms to</u>
3.3.2.7	Navigation Accuracy	P&I Specification paragraph 3.2.4.1.4
3.3.2.8	Radar Utilization	LSP-300-3A, P&I Specification paragraph 3.2.4.1.8
3.3.2.9, 10,11,12	Orbit Insertion, etc.	P&I Specification, paragraphs 3.2.4.1.10, 11, 11.1
3.3.2.1.3	Engine on-off	LSP-300-3A and P&I Specification paragraph 3.4.2:3.2.1
3.3.14, 15, 16, 17, 18, 19 & 20	Calibration/Compensation	LSP-300-3A modified by LVC-300-033-31, TRW TWX 1330.4-6522; P&I Specification 3.2.4.1.12.1, .2, .3, .4, .5, .6 & .7
3.3.2.21	Combined Rates	LSP-300-3A Modified by LVC-300-033-31, Item 32. P&I Specification paragraph 3.2.4.1.13
3.3.2.22	Attitude Error	LSP-300-3A modified by LVC-300-033-31, Item 33. P&I Specification paragraph 3.2.4.1.14
3.3.2.23	Drift (AEA)	P&I specification paragraph 3.2.4.1.5 LSP-300-33C LVC-300-033-076(a)
3.3.2.24	Total Attitude Signal	P&I specification, paragraph 3.2.4.1.6
3.3.2.25	Acceleration Range	P&I specification paragraph 3.2.4.1.7 LSP-300-37D
3.3.2.26	Flight Director Attitude Indicator (FDAI)	LSP-300-3A modified by LVC-300-033-31, P&I specification paragraph 3.4.2.1.3

ERRATA SHEET (Continued)

<u>Paragraph</u>	<u>Description</u>	<u>Conforms to</u>
Section 4 except 4.4	Quality Assurance	LSP-300-3A
4.4	Acceptance Test	Change 20 hrs to 50 hrs and include DM of P&I specification paragraph 3.4.1.2
Table I	Functions	LSP-300-3A modified by LVC-300-033-47 and P&I specification paragraph 3.3.1
Table II	Mission Times	Table II of P&I Specification
Table III	LM Engine	Table IX of P&I Specification
Table IV	Dynamics	Table IV of P&I Specification
Table V	AGS Operations	Table V of P&I Specification
Table VI, VII	Trajectory Parameters	Tables VI, VII of P&I Specification
Table VIII	Weight History	Table VIII of P&I Specification
Figures 1A 1B 1C	Flight Plan Initial Conditions Initial Conditions	1A of P&I Specification 1B of P&I Specification 1C of P&I Specification
Figure 2A 2B 2C 2D	Inertia vs Weight Inertia vs Weight Center of Gravity vs Weight Center of Gravity vs Weight	Page 10.10.1 P&I Specification Page 10.10.2 of P&I Specification Page 10.10.3 Page 10.10.4
Figure 3	Logic	Figure 4, P&I Specification

All references to "P&I Specification" pertain to LSP-500-1 dated 25 July 1966 with negotiated errata enclosed with LMM 500-72 dated 19 August 1966 and with TRW TWX 13304 -6522 dated 11-7-66.

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REVISION DESCRIPTION

Spec No. LSP-300-3B

Rev.	Date	Description
A	11-19-65	Revised Specification to reflect changes negotiated with the vendor. Changes are indicated by marginal indicia.
B	11-9-66	Revised to incorporate the applicable requirements of LVC-300-033-031, LVC-300-033-047, LSP-500-1 and TRW-TWX 13304-6522 dated 11-7-66.

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SPECIFICATION

No. LSP-300-3BABORT GUIDANCE SECTION,GUIDANCE, NAVIGATION AND CONTROL SUBSYSTEM,DESIGN CONTROL SPECIFICATION FOR

1 SCOPE

1.1 Scope. - This specification establishes the general test and performance requirements for the Abort Guidance Section (AGS), which forms a part of the Guidance, Navigation and Control Subsystem (GN and C) for use in the Lunar Module (LM) of the Apollo Spacecraft.

1.2 Equipment Breakdown. - The AGS shall consist of the following assemblies:

1.2.1 Abort Sensor Assembly (ASA). - The ASA shall consist of three strapped-down pulse rebalanced rate integrating gyros and three strapped-down pulse rebalanced accelerometers. The ASA shall be capable of sensing accelerations along the vehicle axes and angular rates about the vehicle axes. The detailed requirements of the ASA are delineated in LSP-300-37.

1.2.2 Abort Electronics Assembly (AEA). - The AEA shall have a magnetic core memory with a total capacity of 4096 words and be capable of fulfilling all the computational requirements of the AGS. The detailed requirements of the AEA are delineated in LSP-300-33.

1.2.3 Data Entry and Display Assembly (DEDA). - The DEDA shall consist of a keyboard electroluminescent address, and data digital displays. The DEDA shall be capable of controlling AGS modes and submodes of operation, inserting data into the AEA memory and displaying data contained in an AEA memory location on the DEDA numerical display. The detailed requirements of the DEDA are delineated in LSP-300-39.

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2 APPLICABLE DOCUMENTS

2-1 Grumman Documents. - The following Grumman documents form a part of this specification to the extent specified herein.

SPECIFICATIONS

Number

Title

LSP-300-33

Abort Electronics Assembly, Guidance, Navigation and Control Subsystem, Design Control Specification for

LSP-300-37

Abort Sensor Assembly, Guidance, Navigation and Control Subsystem, Design Control Specification for

LSP-300-39

Data Entry and Display Assembly, Guidance, Navigation and Control Subsystem, Design Control Specification for

Drawings

LSC300-300

Abort Guidance Section, Guidance, Navigation and Control, Specification Control Drawing for

3 REQUIREMENTS

3.1 General AGS Performance. - The AGS shall be capable of performing the following during the LM mission:

3.1.1 Provide guidance functions which will result in a LM/CSM rendezvous from any point in the LM mission, for trajectory, vehicle, data availability, and environmental conditions and dispersions as specified in Section 3.2. The following requirements shall apply under the stated conditions:

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3.1.1.1 For aborts prior to powered descent, a direct intercept transfer or a concentric orbit sequence capability shall be provided, as described below.

3.1.1.2 For aborts subsequent to the start of powered descent and prior to the end of powered ascent, the AGS shall have the capability of inserting the LM initially into an elliptical orbit. After insertion into the elliptical orbit, a series of concentric rendezvous burns shall be performed as described below resulting in an intercept trajectory passing through the CSM. The AGS will perform midcourse corrections based upon Rendezvous Radar (RR) navigation update as specified in paragraph 3.3.2.8 and/or CSM/MSEFN update as specified in 3.2.11.2.

3.1.1.3 For aborts after the end of powered ascent, the AGS shall either complete the unfinished concentric rendezvous burns or perform a direct transfer intercept burn, as described below.

3.1.2 Guidance Programs. - The above abort missions shall be performed using the following four basic guidance programs.

3.1.2.1 Orbit Insertion. - The insertion targeting parameters will be fixed quantities specified before flight. These quantities will be burnout altitude, altitude rate, and horizontal velocity. The AGS shall steer the LM into the CSM orbital plane during the powered ascent to orbit. The powered ascent time shall be supplied from an external source. (i.e. CSM or MSEFN) and the maximum out-of-plane angle at launch shall be as specified in 3.2.20.

3.1.2.2 Coelliptic Rendezvous. - Subsequent to orbit insertion burnout, the astronaut will select the Coelliptic Rendezvous mode via the DEDA. At a specified AGS absolute time point (selected by the astronaut), a horizontal (thrust vector along local horizontal¹) Coelliptic Sequence Initiate (CSI) burn of flexible magnitude will be applied. The magnitude of this burn will be determined from the geometry shown in FIGURE 1A. A fixed inertial line (referenced to the selenocentric reference frame) will be chosen before flight. The parameters defining this line (parameters required for computing the CSI burn) will be Δ_0 (the angle between the projection of the LM-CSM LOS on the X-Z inertial plane and the LM local horizontal), and the absolute time of the LM crossing the inertial line. These parameters will be obtained from an external source (CSM or MSEFN) and input via the DEDA.

¹ Thrust constrained to point in forward direction

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An intermediate requirement of the coelliptic rendezvous program is that the IM reach this inertial line coincident with the CSM arrival at this line. Thus, the CSI burn is calculated to reach a desired apocynthion altitude such that if the IM were to perform a Constant Deltaⁿ Maneuver (CDH) at this apocynthion altitude, the subsequent time in the coelliptic orbit to the defined line plus the previous time from the CSI burn to apocynthion is equal to the CSM time from its position at CSI burn time to the same inertial line. Hence, the second maneuver in the coelliptic rendezvous program is the CDH (coelliptic) burn. This burn will be initiated at the predicated apocynthion time calculated by the AGS. This time is not to be updated, i.e., the CDH maneuver will be performed at whatever altitude is present at the predicated apocynthion time calculated by the AGS. Note that the coelliptic rendezvous sequence will be initiated only if the differential altitude Δr is within prescribed limits, ± 80 nautical miles (this parameter will be displayed via the DEDA prior to CSI) and if the Delta V constraint check (which is performed manually prior to CSI by displaying V_{Dx} , V_o , V_{po} via the DEDA, utilizing a chart for Delta V direct transfer and braking; and knowledge of the Delta V budget) is satisfied. The intercept transfer will be initiated on a relative IM/CSM line-of-sight angle with respect to the IM local horizontal. Midcourse corrections during the intercept trajectory will be performed by the AGS at fixed elapsed time from TPI. The braking maneuver will be performed manually (this is not an AGS requirement) according to a predetermined relative range, range-rate schedule.

3.1.2.3 Direct Intercept. - The AGS shall have the capability, upon DEDA command of calculating (and performing) a direct intercept transfer at a future specified absolute time point. This time point, and the transfer time of the direct intercept trajectory will be input via the elapsed time indicator.

3.1.2.4 External Delta V. - The AGS will be implemented to accept at any time during the IM mission, components of Delta V from an external source (e.g., MSTW, CSM) along the local radius vector (positive down), horizontal parallel to the CSM orbit plane (Positive forward), and normal to the CSM orbit plane. The positive direction of the third component is opposite to the angular momentum vector. The AGS must be

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3.1.2.4 (Continued)

implimented to perform this maneuver utilizing guidance steering, or manually, utilizing individual body axes jet firing to null out component of velocity to be gained along the body axes (while holding vehicle attitude).

NOTE: For the programs specified above, all burns resulting in a Delta V of less than 60 fps, shall be performed with the RCS. For velocity changes greater than 60 fps and less than 200 fps, either the main engine or the RCS shall be used at the crew option. For velocity changes greater than 200 fps, the main engine shall be used, and also for the CSI and CDH burns (specified above) the steering shall be such that thrusting is parallel to the CSM and the TFI burn steering shall remove all residual out-of-phase conditions.

3.1.2.5 Residual Trimming Capability. - At termination of AGS controlled maneuvers utilizing a main engine, the AGS shall be implemented (upon astronaut request) to null out individual body axes components of velocity to be gained.

3.1.3 AGS Functional Requirements. - The AGS is required to provide the following functions.

3.1.3.1 Navigation, Guidance, Steering. - The AGS is required to perform all navigation, explicit guidance, and steering functions consistent with the requirements of this specification.

3.1.3.2 Attitude Reference. - Provide an inertial reference frame which can be used as an attitude reference for vehicle stabilization during the coasting phases of the LM mission and for stabilization and control during the powered phases of the LM mission, using the vehicle coordinate axes as defined by the three alignment submodes in 3.3.1.2.2.1.

3.1.3.3 Moding. - Provide mode and submode selection by utilizing DEDA inputs, Control Electronics Section inputs and AGS mode switch inputs.

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3.1.3.4 Time Initialization. - Absolute time shall be initialized through the DEDA, and shall be the difference in current mission time and a known time bias constant. Current mission time is denoted by G.E.I. (ground elapsed time from earth liftoff), which is the difference in Current Greenwich Mean Time (GMT) and the GMT of liftoff. Thus, the AGS absolute time is GET-K, where K is the time bias constant. Absolute time shall be inputted via the DEDA in seconds, with a quantization of 10 seconds.

3.1.3.5 Navigation Initialization. - Provide the necessary processing of the PGNCDS downlink data to extract LM and CSM position, velocity and their associate time of computation. This information will be used to initialize or update the AGS navigation computations upon command from the DEDA during non-powered flight phases of the mission. LM and CSM initialization or updating capability is also to be provided by means of inserting the data directly via the DEDA.

3.1.3.6 Telemetry. - Provide the capability of outputting digital data via the telemetry data output channel.

3.1.3.7 Monitoring. - Provide the capability of furnishing output analog signal to the instrumentation subsystem to verify AGS performance.

3.1.3.8 Acquisition. - Provide the capability of orienting the LM Z-axis toward the CSM and of accepting range and angle null data via the DEDA and using this data for midcourse correction computations. The radar data inputs will have accuracies which are consistent with the end point criteria specified in 3.2.1.

3.1.3.9 Sensor Calibration and Compensation. -

3.1.3.9.1 Calibration. - The AGS shall have the minimum capability to calibrate (measure) ASA gyro channel bias (non-acceleration sensitive) drift rate and accelerometer channel bias error under the following mission phases conditions:

- (a) Gyro channel bias drift rate
 - (1) During pre-launch checkout
 - (2) During coast flight (lunar orbit)

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3.1.3.9.1 (Continued)

- (a) (3) On the lunar surface
- (b) Accelerometer channel bias: during coast flight

3.1.3.9.2 Compensation. - The AGS shall have the minimum capability to compensate for the following ASA sensor errors during prelaunch checkout, coast flight and lunar stay;

- (a) Gyro channel non-acceleration sensitive drift rate
- (b) X-gyro channel spin axis mass unbalance drift rate
- (c) Gyro channel scale factor error
- (d) Accelerometer channel bias error
- (e) Accelerometer channel scale factor error

3.1.3.9.3 Error Values. - The AGS shall meet the requirements of this specification if the values of the errors in 3.1.3.9.2 (a), (c), (d), (e) are determined at the times specified below.

(a) Gyro channel non-acceleration sensitive (bias) drift rate determined during pre-launch checkout is no more than 8 days prior to LM/CSM docking.

(b) Gyro and accelerometer channel scale factor errors, and accelerometer channel bias error determined just prior to installation of the ASA in LM, no more than 120 days prior to LM/CSM docking.

3.1.3.10 In-Flight Self Test. - The self test shall provide a comprehensive in-flight check of the system logic and memory, the self test scheme shall also automatically display information identifying failures and their origin to the maximum extent possible. The self test shall have a minimum of 190 AEA instructions and shall consist of the following automatic continuous tests:

- (a) Instruction test

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3.1.3.10 (Continued)

(b) Memory Sum Check

(c) Results display (when requested by astronaut)

The test will stop and the AEA will provide a warning signal if a failure occurs. The astronaut may re-initiate the test by command through the DEDA.

3.1.4 Design Mission. - The AGS shall satisfy all requirement of this specification under the application of the design mission. The design mission shall consist of a restrictive set of initial conditions, operating modes and operating sequences as described below.

3.1.4.1 Initial Conditions. - Abort maneuvers shall start from anywhere in the nominal trajectory given in Tables VI and VII.

3.1.4.2 Alignment and Initialization Prior to Abort. - Alignment and initialization to the PGNCs shall take place at the time points given in 3.2.12(a) and 3.2.13, except for abort from Lunar Surface. For launch from the lunar surface a lunar align and MSFN initialization is assumed.

3.1.4.3 Calibration. - The Design Mission shall use only prelaunch and preflight calibration.

3.1.4.4 Navigation Initialization. - The sequence of initialization after abort shall be any of the following:

Sequence #1

20 min Prior to Launch MSFN update

Prior to CSI burn - no update

Prior to CDH burn - no update

10 min Prior to TPI burn - MSFN update

Prior to MC Correction RR update

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3.1.4.4 (Continued)

Sequence #2

20 min Prior to Launch MSFN update
Prior to CSI burn RR update
Prior to CDH burn RR update
10 min Prior to TPI burn RR update
Prior to MC Correction RR update

Sequence #3

20 min Prior to Launch MSFN update
Prior to CSI burn RR update
Prior to CDH burn RR update
10 min Prior to TPI burn MSFN update
Prior to MC Correction RR update

3.1.4.5 Guidance Sequence. - The following sequence of guidance maneuvers shall be used for the Design Mission.

Orbit Insertion

CSI

CDH

TPI

Two Mid Course Corrections

3.1.4.6 Targeting Parameters. - For the DM the nominal trajectory parameters to be used shall be as follows:

h = 60,000 ft. above the mean lunar radius

h = 0 fps

V_{ho} = 5510 fps

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3.1.4.6 (Continued)

$$\Delta_2 = 26.6^\circ$$

$$t_{igA} = 111.02$$

$$t_{igC} = 112.20$$

3.2 AGS Design Constraints. -

3.2.1 Trajectory Constraints. - The AGS shall be designed to meet the detailed requirements within the nominal ascent and descent trajectories as defined in Tables VI and VII and including the dispersions about the nominal trajectories as defined below. The end point conditions for abort trajectories shall be achieved within the injection ΔV constraints as listed in paragraph 3.2.17.

(a) Dispersions during descent -

(1) Initial Conditions for AGS Aborts with the Descent Engine During Powered Descent

Time from Initiation of Braking Phase (Sec)	h (ft)	\dot{h} (fps)	R (ft)	\dot{R} (fps)	T (ft)	\dot{T} (fps)
0	+25,000	+100	+20,700	+16	+40,000	+39
100	+20,152 -15,788	+479	+20,800	+96	+40,000	+85
270	+16,579 -14,139	+123	+22,000	+44	+40,000	+215
350	+6778	+17	+28,000	+60	+46,500	+275
400	+2219	+79	+31,200	+145	+55,000	+200

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SPECIFICATION

No. LSP-300-3B

3.2.1 (Continued)

(a) (Continued)

(2) Initial Conditions for AGS Aborts with the Ascent Engine
During Powered Descent

Time (Min)	h (ft)	\dot{h} (fps)	R (ft)	\dot{R} (fps)	T (ft)	\dot{T} (fps)
0	+25,000	+110	+20,700	+16	+40,000	+39
100	+19,277 -15,085	+459	+20,800	+95	+40,000	+85
270	+14,949 -13,789	+107	+22,000	+44	+40,000	+215
350	+ 5,054	+12	+28,000	+60	+46,500	+275
400	+ 976	+33	+31,200	+145	+55,000	+200

(3) Abort Conditions During Hover-to-Touchdown - The initial vertical positions and velocities, corresponding to descent ascent engine aborts during the hover-to-touchdown maneuver, are presented in Figures 1B and 1C respectively. Table in 3.2.1(a)(4) presents the downrange and track position and velocity deviations from the nominal during the hover-to-touchdown maneuver. At the indicated times, these deviations may occur in any combination with the altitudes and altitude rates presented in Figures 1B and 1C.

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3.2.1 (Continued)

(a) (Continued)

(4) Initial Conditions in Downrange and Track Position and Velocity for AGS Aborts with the Ascent or Descent Engine During Hover-to-Touchdown

Time (Sec)	h (ft)	h (fps)	R (ft)	R (ft)	T (ft)	T (fps)
97.2833	See Fig. 1B & 1C	See Fig. 1B & 1C	+14,400	+120	+53,000	+145
515 Descent Engine	See Fig. 1B & 1C	See Fig. 1B & 1C	+14,896	+2.	+53,700	+2.
515 Descent Engine	See Fig. 1B & 1C	See Fig. 1B & 1C	+15,365	+2.	+54,530	+2.

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3.2.1 (Continued)

(b) Dispersions during Ascent

Time from Lift off (seconds)	h (ft)	\dot{h} (fps)	R (ft)	\dot{R} (ft)	T (ft)	\dot{T} (fps)
0	+ 0	+ 0	+ 9,180	+ 0	+ 4,260	+ 0
60	+ 320	+ 16	+ 9,200	+ 30	+ 4,400	+ 40
100	+ 1,200	+ 30	+ 9,370	+ 45	+ 5,110	+ 60
160	+ 3,750	+ 65	+ 11,000	+ 75	+ 9,150	+ 70
200	+ 6,300	+ 75	+ 13,250	+ 95	+ 12,750	+ 110
260	+ 11,650	+ 110	+ 18,800	+ 125	+ 18,500	+ 120
300	+ 12,000	+ 110	+ 19,300	+ 140	+ 24,300	+ 130
360	+ 12,000	+ 110	+ 32,300	+ 175	+ 32,200	+ 130
400						
430	(to be defined)					

Where h = altitude R = downrange (in the plane of CSM) T = track (normal to plane of CSM)

It shall be assumed that the LM fuel expended and the AGS navigation errors incurred in arriving at a dispersion from the nominal PGNCs guided trajectory shall be the same as that incurred for the corresponding time on the nominal trajectory.

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3.2.1 (Continued)

- (c) The targeting parameter scaling shall be constrained to the following ranges:

$$h = 0 \text{ to } 2,000,000 \text{ ft.}$$

$$\dot{h} = \pm 8000 \text{ fps}$$

$$V_{ho} = 0 \text{ to } 8000 \text{ fps}$$

$$\Delta_2 = \text{between } +20^\circ \text{ and } 90^\circ$$

$$t_{igA} = 0 \text{ to } (2^{18}-2)$$

$$t_{igC} = 0 \text{ to } (2^{18}-2)$$

3.2.2 CSM Orbit Characteristics. - The CSM shall have a nominal circular Orbit whose altitude above the mean lunar radius is between 10 n. mi. and 85 n. mi. The variation in the CSM circular orbit will be such that the eccentricity is less than .015. The orbital inclination will be within ± 10 degrees of the lunar equator.

3.2.3 Mission Environment. - The AGS shall meet all the requirements of this specification for vehicle environmental conditions (performance during) with duration and conditions of exposure as specified in Table I of LSP-300-33, LSP-300-37, and LSP-300-39. The mission profile shall be as per Table II.

3.2.4 Mass Properties. - The mass properties of LM shall be as follows:

- (a) Weight Breakdown - The weight breakdown of LM is shown in Table VIII.
- (b) Expendables Profile - Expendables as used here excludes main engine fuels and includes those items such as reaction control fuel, waste products, experiments, etc., which are jettisoned, expelled or left on the lunar surface during the LM mission. The decrease of LM mass due to the removal of expendables is insignificant and shall not be considered during the development of the AGS.

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3.2.5 Descent Engine Performance. - The characteristics of this engine, while operating in this condition, shall be as given in Table III.

3.2.6 Ascent Engine Performance. - The characteristics of this engine, while operating in this condition, shall be as given in Table III.

3.2.7 LM Center of Gravity. - The nominal position of the LM center of gravity shall be as shown by the curves presented in Figures 2C, 2D.

3.2.8 Staging. - The sequence of events occurring at staging of the LEM vehicle during AGS guidance shall nominally be as follows:

(a) Descent Engine Initially on

0 msec	Abort Stage Signal Received
30 msec	Descent Engine On Signal Removed
280 msec	Descent Engine Thrust Decay to 10%
420 msec	Ascent Engine Armed and On Signal Present
	Staging Signal Sent
460 msec	Blow Interface Bolts
	Open Ascent Dead Face
500 msec	Open Descent Dead Face
550 msec	Fire Cable Cutter Staging Complete
820 msec	Ascent Engine Thrust Build-up to 90%

(b) Descent Engine Off (Abort from Coasting Descent with Ascent Engine)

0 msec	Abort Stage Signal Received
420 msec	Ascent Engine Armed

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3.2.8 (Continued)

(b) (Continued)

Period of Vehicle Orientation and Ullage

0 msec.	AGS issues Engine On Command
50 msec.	Ascent Engine On Signal Received
	Staging Signal Sent
90 msec.	Blow Interface Bolts
	Open Ascent Dead Face
130 msec.	Open Descent Dead Face
180 msec.	Fire Cable Cutter
	Staging Complete
450 msec.	Ascent Engine Thrust Build-up to 90%

(c) Liftoff from Lunar Surface - Same as paragraph 3.2.8(a), except delete the following steps and remaining times apply:

30 msec.	Descent Engine On Signal Removed
280 msec.	Descent Engine Thrust Decay to 10%

3.2.9 LM Attitude Control Characteristics. - The LM attitude control characteristics from AGS command (attitude errors) to vehicle rates for the pitch, roll and yaw axes as a function of mission phase are shown below. Moment of inertia variations shall be in accordance with the curves presented in Figures 2A, 2B.

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3.2.9 (Continued)

- (a) The following equivalent transfer function from AGS attitude error signal (E) to body rate (θ) will be used for the pitch, yaw and roll axes:

$$\frac{\dot{\theta}}{E} = \frac{1/K_r}{\frac{I}{KK_r} S + 1}$$

where

$$K_r = \begin{array}{l} 1.5 \text{ sec (Descent System)} \\ 0.4 \text{ sec (Ascent System)} \end{array}$$

$$K = \begin{array}{l} 6.7 \times 10^3 \text{ ft-lb/rad-(Descent System)} \\ 1.16 \times 10^4 \text{ ft-lb/rad-(Ascent System)} \end{array}$$

and where "I" is the vehicle inertia in slug-ft³ as referenced in curves of Figures 2A and 2B.

- (b) The Body Axes attitude Dynamics shall be as specified in Table IV.

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3.2.10 Accuracy of PGNC'S Alignment Data. - The accuracy of PGNC'S alignment data transferred to the AGS is determined by errors associated with the measurements and computations involved in defining the desired orientation, the process of fine IMU alignment, the orientation drift of the IMU since the last fine alignment, the angle readout associated with the CDU's and the errors associated with the mounting misalignment of the ASA with respect to the attachment points of the PGNC'S Navigation Base.

A limit on AGS alignment input data from the CDU, referred to the PGNC'S Navigation Base attachment points for the purposes of interface design is 2 milliradians plus 3 milliradians times the time in hours (3σ) since the last IMU fine alignment.

A limit on alignment of the ASA reference axes with respect to the PGNC'S Navigation Base is 0.6 milliradians (3σ).

3.2.11 Initialization Uncertainties. -

3.2.11.1 Initialization Via PGNC'S Downlink. -

3.2.11.1.1 Prior to Descent Orbit Insertion. - The LM and CSM state vector covariance matrix at the start of insertion.

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Matrix in Landing Site Inertial Coordinates

x	y		z		x		y		z	
	E 05	E 03	E 05	E 04	E 05	E 02	E 01	E 00	E 01	E 01
2.5810	6.2126	E 03	7.8080	E 05	-4.6841	E 02	1.0752	E 01	8.4849	E 01
6.2132	3.4745	E 05	-3.5151	E 04	1.8546	E 01	-9.8968	E 00	-1.7999	E 01
7.8080	-3.5152	E 04	3.6544	E 06	-2.2177	E 03	3.8407	E 01	8.2620	E 02
-4.6842	1.8546	E 01	-2.2177	E 03	1.4119	E 00	-2.2902	E-02	-5.1607	E-01
1.0752	-9.8968	E 00	3.8407	E 01	-2.2902	E-02	2.0520	E-01	6.8779	E-03
8.4849	-1.8000	E 01	8.2620	E 02	-5.1608	E-01	6.8779	E-03	2.5974	E-01

These numbers represent 1-sigma deviations in landing site inertial coordinates. The epoch of CSM and IEM data shall not be more than 10 minutes old. During ascent operations from the lunar surface, the above matrix shall apply for the CSM at its orbital position at the time of initialization. When initialized during ascent coast or in a parking orbit, the initialization uncertainty of IEM shall be described by the above matrix.

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3.2.11.1.2 Prior to Powered Descent. - The following uncertainties shall be used during initialization of the navigation ten minutes prior to the start of powered descent.

(a) LEM state vector covariance Matrix 10 minutes prior to the start of power descent:

	<u>X</u>	<u>Y</u>	<u>Z</u>	<u>\dot{X}</u>	<u>\dot{Y}</u>	<u>\dot{Z}</u>
X	2.3187 E06	-2.8489 E04	3.0386 E06	-1.9319 E03	-1.4087 E01	2.6618 E03
Y	-2.8493 E04	3.2322 E05	-1.0407 E04	2.9613 E01	-2.5314 E01	-1.3729 E01
Z	3.0385 E06	-1.0398 E04	5.9319 E06	-2.7513 E03	-1.4959 E01	5.2682 E03
\dot{X}	-1.9319 E03	2.9610 E01	-2.7514 E03	1.8266 E00	6.4785 E-03	-2.5664 E00
\dot{Y}	-1.4089 E01	-2.5314 E01	-1.4964 E01	6.4804 E-03	2.9290 E-01	-7.7541 E-03
\dot{Z}	2.6617 E03	-1.3721 E01	5.2681 E03	-2.5664 E00	-7.7493 E-03	4.7989 E00

(b) CSM State vector covariance matrix 10 minutes prior to the start of powered descent:

	<u>X</u>	<u>Y</u>	<u>Z</u>	<u>\dot{X}</u>	<u>\dot{Y}</u>	<u>\dot{Z}</u>
	E0	E0	E0	E0	E0	E0
X	1.896938+6	-3.129885+4	1.518988+6	-1.107371+3	-1.744878+1	1.463417+3
Y	-3.130225+4	3.342345+5	-6.074862+3	2.694324+1	2.444488+1	-1.441706+1
Z	1.51889+6	-6.066941+3	1.542828+6	-7.076788-2	-1.393094+1	1.302133+3
\dot{X}	-1.107369+3	2.694118+1	-7.077218+2	7.716263-3	6.887765-3	-8.166749-1
\dot{Y}	-1.745097+1	2.444488+1	-1.393605+1	6.889093-3	2.148093-1	-8.921451-3
\dot{Z}	1.463323+3	-1.440985+1	1.302121+3	-8.166330-1	-8.916801-3	1.223637+0

NOTE: The above two matrices should be modified by the time synchronization errors specified in the following paragraph.*

* Absolute Time Uncertainty.

The Absolute time uncertainty is \pm 400 milliseconds or less.

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3.2.11.1.3 On Lunar Surface. - The one-sigma site location uncertainties are:

- (a) In any direction in the local horizontal plane - 1500 feet
- (b) Along the local vertical - 1000 feet

3.2.11.1.4 Prior to CSI burn . - TBD

3.2.11.1.5 Prior to CDH burn . - TBD

3.2.11.1.6 Prior to TPI burn . - TBD

3.2.11.1.7 Prior to Mid-Course Corrections. - TBD

3.2.11.2 Initialization Via CSM Voice Link. -

3.2.11.2.1 On Lunar Surface. - The one-sigma site location uncertainties are:

- (a) In any direction in the local horizontal plane - 1500 feet
- (b) Along the local vertical - 1000 feet

3.2.11.2.2 Prior to CSI burn . - TBD

3.2.11.2.3 Prior to CDH burn . - TBD

3.2.11.2.4 Prior to TPI burn . - TBD

3.2.11.2.5 Prior to Mid-Course Corrections. - TBD

3.2.11.3 Initialization Via MSEN. -

3.2.11.3.1 Lunar Surface. - The one-sigma site location uncertainties are:

- (a) In any direction in the local horizontal plane - 1500 feet
- (b) Along the local vertical - 1000 feet

(c) Ten Minutes Prior to Launch. - The following covariance matrix represents the expected accuracy of the CSM. It is based on 98.4 minutes of tracking. The units are ft^2 and ft^2/sec^2 .

.690376E5	-.940972E5	.259044E5	.631467E2	.517066E2	.661812E3
	.634194E6	.282893E7	-.490897E3	.151523E2	-.328160E4
		.150280E8	-.217337E4	.411576E3	-.130120E5
			.380490E0	-.904297E-2	.255162E1
				.498366E-1	.154818E0
					.180870E2

1 m = 3.280833 ft

1 m² = 10.76387 ft²

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3.2.11.3.2 Prior to CSI burn . - TBD

3.2.11.3.3 Prior to TPI burn . - The LM State Vector
covariance matrix, 10 minutes prior to TPI burn:

.198490E06	-.749092E05	.730366E06	.273973E+03	-.392589E+02	.149117E 03
	.336705E05	-.575542E05	-.963142E+02	.139232E+02	.215871E 02
		.149486E08	.126712E+04	-.178509E+03	.546861E 04
			.390414E 00	-.570762E-01	.257122E 00
				.102501E-01	.100499E-01
					.473182E+01

3.2.11.3.4 Prior to Mid-Course Corrections . - TBD

3.2.12 Frequency of Initialization . -

(a) Automatic (LEM and CSM) . - During non-abort conditions, or an abort in which the PGCS is operating properly, the AGS shall be initialized using signals from the PGCS downlink at the following times:

- (1) 30 minutes or less prior to insertion into descent coast
- (2) 20 minutes or less prior to start of powered descent
- (3) 20 minutes or less prior to lunar lift off
- (4) 10 minutes or less prior to coelliptic sequence initiation (CSI)
- (5) 10 minutes or less prior to CDH burn
- (6) 10 minutes or less prior to terminal phase initiation (TPI)
- (7) 10 minutes or less prior to mid-course corrections

(b) Manual (LEM and CSM) . - During an abort situation in which the PGCS is not operating properly, no automatic initialization data shall be available from the PGCS. In this case initialization data will be available from MSFN or from the CSM and shall be inserted manually into the AGS using the DEDA. Manual initialization shall take place at the following times:

- (1) 20 minutes or less prior to lunar lift off
- (2) 10 minutes or less prior to coelliptic sequence initiation

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(b) (Continued)

(3) 10 minutes or less prior to CDH burn

(4) 10 minutes or less prior to terminal phase initiation

*(5) 10 minutes or less prior to mid-course corrections

(c) Manual (Absolute Time). - Initialization of AGS absolute time will take place twice during the LM mission in orbit (prior to insertion) and on the lunar surface. (See paragraph 3.2.11.1.2 Note).

3.2.13 Frequency of Alignment. -

(a) IMU Alignment. - During non-abort conditions or an abort in which the PGNCs is operating properly, the AGS shall be aligned using signals from the PGNCs at the following times:

(1) 15 minutes or less prior to insertion into coasting descent

(2) 5 minutes or less prior to start of powered descent

(3) Immediately after touchdown on the lunar surface (and again 25 to 30 minutes later after PGNCs, fine align)

(4) 4 minutes or less prior to lunar lift off

(5) 5 minutes or less prior to coelliptic sequence initiation

(6) 5 minutes or less prior to CDH burn

(7) 5 minutes or less prior to terminal phase initiation

* Should not be performed if RR data is used to update IM-AGS navigation.

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3.2.14 Frequency of Calibrations. - The AGS will be calibrated at the following times prior to, and during the lunar landing mission:

- (a) Preflight Calibration - The AGS will be calibrated immediately prior to installation in LM. This calibration will be performed a maximum of 120 days prior to completion of the lunar landing mission, i.e. docking. The following parameters will be measured and compensation will be inserted into the AEA:
 - (1) Gyro Channel Scale Factor
 - (2) Accelerometer Channel Scale Factor
 - (3) Accelerometer Bias
 - (4) Gyro Channel Bias Drift Rate
 - (5) Gyro Channel Acceleration sensitive drift (X gyro only)
- (b) Earth Prelaunch Calibration - The AGS automatic earth prelaunch calibration of gyro channel bias will be performed no more than 8 days prior to completion of the mission while the LM is in a stacked configuration.
- (c) Inflight Calibration - The AGS automatic inflight calibration of gyro channel non-g sensitive bias and accelerometer channel static bias will be performed during the period from LM/CSM separation to deboost into coasting descent.
- (d) Lunar Prelaunch Calibration - The AGS automatic lunar prelaunch calibration of gyro channel bias will be performed prior to liftoff from the lunar surface.

3.2.15 Time Range. - The range of absolute time to be employed by the AGS shall be from 0 to 72 hours. After absolute time initialization (specified in 3.1.3.4), there will be a known one-to-one correspondence between the AGS absolute time and the current mission time base.

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3.2.15.1 Compatibility with Epoch Times Obtained Via PGNCS Downlink. - The LM/CSM state vector epoch times will be referenced to GET, the PGNCS absolute time base. When initializing the AEA via the PGNCS downlink, the epoch times will be converted by the LGC to the AGS absolute time base, GET-K (as specified in 3.1.3.4); prior to placing it on the downlink.

3.2.16 Time to Rendezvous. - The maximum time allowed for any rendezvous conducted by the AGS shall be 10.5 hours including time spent on lunar surface waiting for proper CSM phasing.

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3.2.17 Delta V Constraints. - The ΔV used by the AGS for orbit insertion with the Ascent Engine shall be constrained by the following equation:

$$\Delta V_a \leq 6586 - \Delta V_2 - 30 - 50 - \Delta V_e - 25 \text{ (FPS)}$$

where: ΔV_a = ΔV allowed for orbit insertion with AGS and the conditions listed below:

6586 = Total guaranteed ΔV available from the full ascent tank plus the RCS subsystem.

ΔV_2 = The sum of the magnitude of ΔV impulses required in the CSI maneuver, the CDH maneuver, the TPI maneuver, and terminal braking maneuver, as computed at nominal orbit insertion (for the abort under consideration) by a perfect coelliptic rendezvous computation utilizing perfect navigation data. In the case of a direct intercept the ΔV_2 is equal to the sum of the direct transfer and the braking maneuver.

30 = Allotted ΔV for velocity uncertainty upon entering the 5 n. mi. sphere

50 = Allotted ΔV for midcourse corrections

25 = Allotted ΔV for manual docking

ΔV_e = Ascent Engine ΔV expended by the PGCS out of the 6586 fps, prior to the Abort initiation.

The conditions that apply to the above constraint equation are as follows:

- (1) Aborts take place from the nominal trajectory or from the lunar surface from points which are out of plane by an angle less than or equal to 1/2 degree.

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3.2.17 (Continued)

(2) The ΔV_a equation applies only for aborts from points in the Design Mission (See paragraph 3.1.4).

(3) $I_{SP} = 306.3$ secs.

LEM delta V Available -

(a) Descent Stage - 7332 ft/sec

(b) Ascent Stage - 6586 ft/sec

The Descent Propulsion Subsystem shall be used to available Delta V depletion.

3.2.18 Lunar Characteristics. - The reference lunar gravita-

tional mode shall be tri-axial with the following parameter values:

$$GM_m = \mu_M = 1.73139971 \times 10^{14} \text{ (international ft)}^3 / \text{sec}^2$$

Principal moments of inertia (rotational):

$$A = 0.8878179834 \times 10^{35} \text{ Kg meter}^2$$

$$B = 0.8880019542 \times 10^{35} \text{ Kg meter}^2$$

$$C = 0.8883697818 \times 10^{35} \text{ Kg meter}^2$$

Where A is about an axis directed toward the mean libration point (earth), C is about the lunar rotational axis, and B about the remaining orthogonal axis.

Mean lunar radius = 5.702395×10^6 international feet (6076.11549 international feet = 1 nautical mile.)

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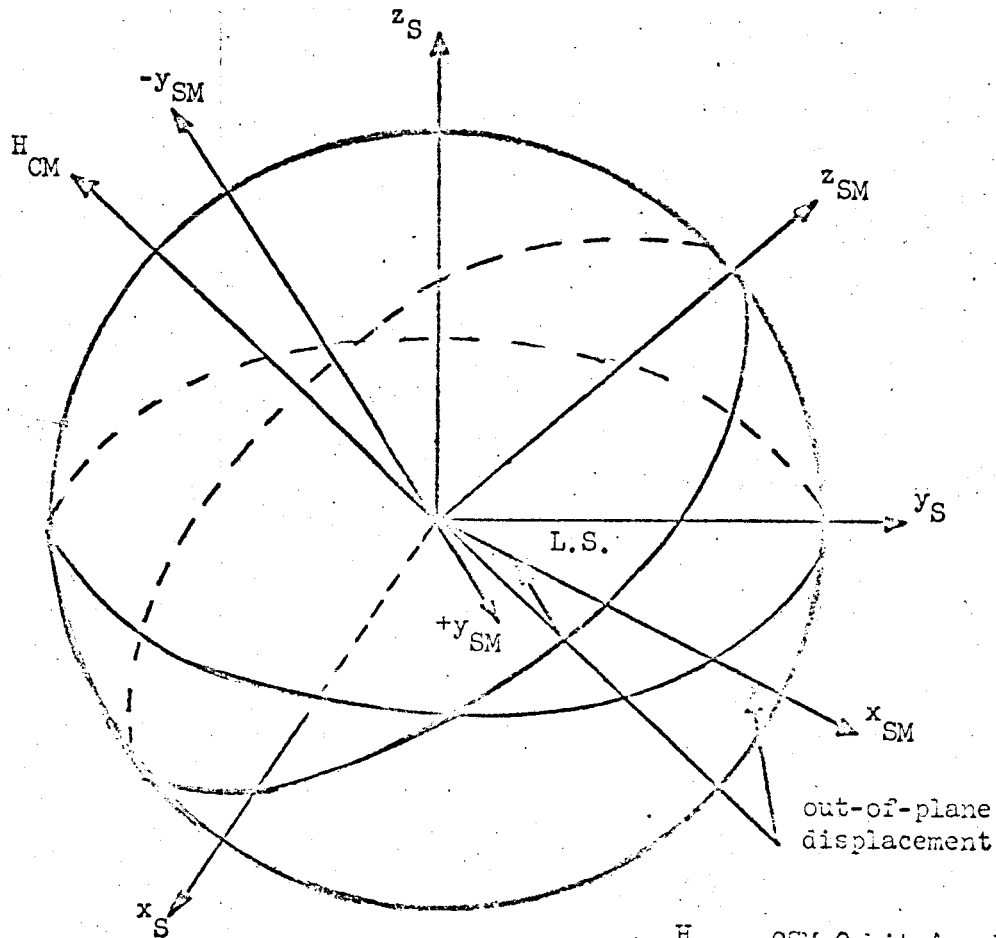
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3.2.19 Coordinate Systems. - The AGS shall utilize the following coordinate systems:

- (a) Stable Member Axis System (Inertial Frame): The Stable Member Axis System is inertially fixed in space with its origin at the moon's center. Its orientation is different for the descent and ascent phases. For the descent phase, the x_{SM} - axis passes through the intended landing-site at the nominal time of landing, positive outward from the lunar center. The z_{SM} -axis is normal to x_{SM} axis, and parallel to the CSM orbital plane. The y_{SM} -axis is normal to the x_{SM} - z_{SM} plane in a right hand sense. For the ascent phase, the x_{SM} -axis passes through the intended launch-site.

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at the nominal time of launch, positive outward from the lunar center. The z_{SM} -axis is normal to x_{SM} and parallel to the CSM orbit plane. The y_{SM} -axis is normal to the $x_{SM} z_{SM}$ plane in a right hand sense.



H_{CM} = CSM Orbit Angular
Momentum

L.S. = Landing or Launch Site

Stable Member Axis System

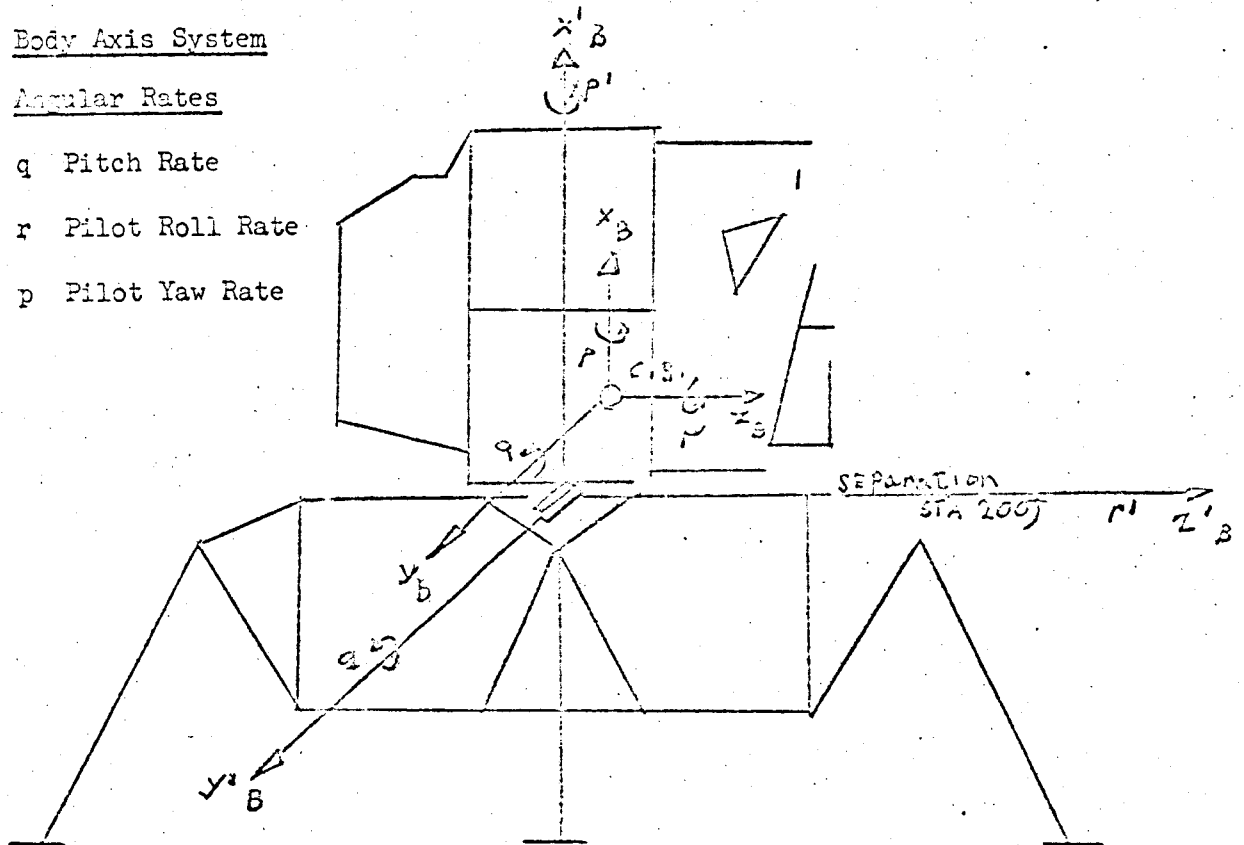
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(b) Body Axis System - The figure below defines the reference IM body axis system. (x'_B, y'_B, z'_B) . The origin of this frame is fixed at IM vehicle station $X = 200, Y = 0,$ and $Z = 0$. The z axis (z'_B) is normal to x'_B and passes through the intersection of the vehicle separation station ($X = 200$) and the plane of symmetry of the IM cabin front view. The y -axis (y'_B) is normal to the $x'_B z'_B$ plane in a right handed sense. Another body axis system (x_B, y_B, z_B) whose coordinates are parallel to reference body axis system but centered at the instantaneous vehicle center of gravity is necessary when discussing vehicle motion.

Body Axis System

Angular Rates

- q Pitch Rate
- r Pilot Roll Rate
- p Pilot Yaw Rate



Body Axis Systems

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3.2.20 Out-of-Plane Requirement for Lunar Launch Phase. - Prior to IM ascent, a plane change will be made by the CSM to permit a nominal co-planar IM ascent. This plane change shall result in a maximum out-of-plane angle of 0.5° at IM launch time. This residual out-of-plane angle shall be removed by IM roll steering during orbit insertion. The AGS shall have the guidance capability of removing a residual out-of-plane angle of 2.0 degrees, (assuming sufficient ΔV is available).

3.2.21 Initial Attitude Constraint. - For analysis purposes, it shall be assumed that, at the initiation of abort during powered phases, the vehicle attitude shall be within $\pm 45^{\circ}$ of the nominal attitude which is specified in Table VII. Angular rate at abort initiation about each axis shall be less than $25^{\circ}/\text{sec}$.

3.2.22 Prelaunch Booster Sway Data. -

(a) Dynamic: Elliptical Motion

Frequency: 0.6 cps empty vehicle
 0.3 cps full vehicle

Amplitude: () indicates fueled vehicle

	<u>Design Goal</u>	<u>Requirement</u>
Perpendicular to wind:	14 in. (6 In)	7 in. (3 ins)
Parallel to wind:	2 in. (1 in.)	1 in. (0.5 in.)

(b) Static:

	<u>Design Goal</u>	<u>Requirement</u>
Wind Velocity (3 σ):	63 knots	38 knots
Lateral Displacement:	20 inches	10 inches

It shall be assumed that the sway data defines the total lateral and rotational motion (except in azimuth) and that the rotation in azimuth is negligible.

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CODE IDENT 26512SPECIFICATION NO. LSP-300-3B3.2.23 Radar Data Accuracy. -

3.2.23.1 Radar Range. - Radar range will be measured with no range averaging. There will be a bias error of +80 ft for ranges ≤ 50.8 nm and +500 ft for ranges > 50.8 nm. The random uncertainty on range will be independent between measurements and has the following characteristics:

Radar Range	Radar Random Error	Display Resolution
400 nm to 10 nm	0.083% (1σ) or 300 ft*	1 nm
10 nm to 5 nm	300 ft	50 ft
5 nm to 1000 ft	0.33% (1σ) or 80 ft*	50 ft
1000 ft to 80 ft	80 ft	2.5 ft

*whichever is greater

The input data word format will be 4 significant decimal digits. The data will be in units of nautical miles. The least significant digit will be 0.1 n.mi. The maximum limit range data will be 400 n. mi. Maximum variation in RR angle bias shall not exceed 0.5 mr in any 10 minute time period.

3.2.23.2 Radar Pointing Error Nulled. - When the radar pointing error is nulled, a code is entered via the DEBA. The error in nulling has the following characteristics.

The one-sigma bias error on the radar gimbal null angular measurement is 5 mr., vector sum, total.

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The random error has the one sigma values given below:

<u>Range</u>	<u>Random Error</u>	<u>Random Error Variation</u>
400 n. mi.	2.7 mr	Direct Variation with range
300 n. mi.	2.5 mr	
200 n. mi.	2.3 mr	
200 n. mi.	2.3 mr	Constant
5 n. mi.	2.3 mr	

Display resolution is $1/12^\circ$ (1σ)

(a) Bias error is the mean value of the error between the true value of the parameter and the indicated value averaged over any 10 min. interval.

(b) Random error is the variation of the error about the bias error. It is assumed that the astronaut generates error signals for the autopilot to null radar angle based on these measurements.

At least one radar data point will be available to the AGS in every 4 minute interval. Maximum variation in RR angle bias shall not exceed 0.5 mr in any 10 minute time period.

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3.3.1.1 Functions of AGS. - The functions of the AGS during various phases of the mission shall be as shown in Table I.

3.3.1.2 AGS Operational Requirements. - Selection of AGS operation shall be performed via the AGS mode switch, the DEDA, and discrettes from the CES. These AGS operations are described below.

3.3.1.2.1 AGS Operational Conditions. - The AGS shall be placed in any one of the following conditions:

3.3.1.2.1.1 Unpowered. - The AGS will be in this condition when AGS is unpowered as determined by circuit breakers on the circuit breaker panel. The circuit breaker panel is not part of the AGS. This mode is used in the Event of an AGS failure.

3.3.1.2.1.2 Off. - The ASA is in a temperature controlled condition.

3.3.1.2.1.3 Standby. - The ASA is fully powered. In addition, the capability shall be provided in the AEA to accept PGNCS alignment information twenty (20) seconds after entering the STANDBY condition. This information shall include the $\pm \theta$, $\pm \phi$ and $\pm \psi$ and the "CDU Zero" signals.

3.3.1.2.1.4 Operate. - In this condition the three assemblies (AEA, ASA, DEDA) must be functioning, inputting, and outputting such information as is needed to properly execute the Flight Programs under one of the submodes defined in 3.3.1.2.2 of this specification.

3.3.1.2.2 AGS Modes and Submodes. - The AGS shall always be in one of the following modes:

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3.3.1.2.2.1 Alignment Mode. - During this mode of operation, the AGS shall be capable of aligning to any inertial reference frame as determined by the alignment submode scheme selected via the DEDA. Entry into the ALIGNMENT mode shall always be made by inserting one of the following submode instructions.

- (a) IMU ALIGN Submode
- (b) LUNAR ALIGN Submode
- (c) BODY AXIS ALIGN Submode

The AGS shall have the capability to initialize the navigation computation during the ALIGNMENT mode. The AGS shall cause the attitude error signals to the CES to be reduced to zero during the ALIGNMENT mode. On exit from one of the ALIGNMENT Submodes and entry into the INERTIAL REFERENCE mode, the AGS shall be capable of performing all the abort guidance functions required by this specification within 2 seconds.

3.3.1.2.2.1.1 IMU ALIGN Submode. - When AGS alignment is to be accomplished using PGNC'S Euler angle signals, the IMU ALIGN submode shall be utilized. This submode shall be initiated by utilizing the DEDA to apply an instruction (IMU align instruction). In the IMU ALIGN submode the AGS shall align to the inertial reference frame as determined by the PGNC'S Euler angle signals.

3.3.1.2.2.1.2 LUNAR ALIGN Submode. - When AGS alignment is to be accomplished on the lunar surface and the PGNC'S is inoperative, the LUNAR ALIGN submode shall be utilized. In this submode, the AGS shall use the local vertical and the AEA stored azimuth data. After the lunar align submode is complete, the AGS shall provide an "Alignment Complete" indication via the DEDA upon operator request.

The ASA accelerometer outputs shall be used to determine the local vertical with respect to the instantaneous vehicle coordinates.

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3.3.1.2.2.1.2 (Continued)

The vehicle azimuth angle with respect to the CSM orbit plane at lunar touchdown shall be determined by either of two methods:

- (a) If no IMU/CDU failure has occurred, IMU ALIGN shall be entered and the value of azimuth stored in the AEA.
- (b) If an IMU/CDU failure does occur prior to touchdown, the azimuth information stored in the AEA shall be used. The instruction to store the information shall be entered through the DEDA at the time it is expected to be stored.

A correction factor shall be added to compensate for the effects of lunar rotation during the lunar stay time.

The correction factor ($\Delta\sigma$) will be supplied via the voice link and inserted using the DEDA where $\Delta\sigma$ is the change in azimuths due to lunar rotation and CSM plane change.

3.3.1.2.2.1.3 BODY AXIS ALIGN Submode. - When AGS alignment is to be accomplished in orbit and no external information is available, the BODY AXIS ALIGN submode shall be utilized. This submode shall be initiated by utilizing the DEDA to apply an external instruction (BODY AXIS ALIGN instruction). The AGS shall align its inertial reference frame to the body reference axes as defined by the ASA input axes. During this submode, the vehicle motion will be restricted to:

- (a) Vehicle limit cycling as given in Table IV.
- (b) Vehicle slewing rates up to 2 deg/sec.

3.3.1.2.2.2 Inertial Reference Mode. - When the AGS is in the INERTIAL REFERENCE mode, it shall provide attitude error information and engine on/off commands which will be used for vehicle stabilization and/or for directing vehicle abort. When in the INERTIAL REFERENCE mode, the AGS shall always be in one of the following submodes:

- (a) Followup Submode

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3.3.1.2.2.2 (Continued)

- (b) Attitude Hold Submode
- (c) Automatic Submode
- (d) Semi-Automatic Submode
- (e) CSM Acquisition Submode

The INERTIAL REFERENCE mode shall be initiated upon application of an external instruction utilizing the DEDA. The submode shall be determined from discrete signals from the CES and the DEDA as defined herein.

3.3.1.2.2.2.1 FOLLOW-UP Submode. - With the AGS in the INERTIAL REFERENCE mode and the "Follow-Up" signal present, the AGS shall be in the FOLLOW-UP submode. While in the FOLLOW-UP submode the AGS shall be capable of performing the following functions:

- (a) Maintain an inertial reference frame and output total attitude signals for display.
- (b) Maintain the Attitude error signals to the CES at zero unless the Display Steering Commands in Follow-up and Auto signals are present. In this case, error signals will be available on the FDAI for monitoring by the astronaut, but will not be used by CES.
- (c) Utilize accelerometer inputs from the ASA to calculate LM vehicle present position and velocity in the inertial reference frame.
- (d) Solve the LM abort guidance problem using an explicit guidance scheme, in accordance with the guidance option selection.
- (e) Accept initializing information either from the PGNS downlink or the DEDA.
- (f) Command Engine-ON if either the Ascent Engine-ON signal or the Descent Engine-ON signal are present and command Engine-OFF otherwise.

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3.3.1.2.2.2 ATTITUDE HOLD Submode. - The AGS shall be in the ATTITUDE HOLD submode when the attitude hold submode conditions of Figure 4 are satisfied. In this submode the AGS shall output attitude error signals which will maintain the vehicle attitude which existed upon entering this submode.

The ATTITUDE HOLD submode shall be entered in the absence of the other INERTIAL REFERENCE submodes and at the following times during an abort operation:

- (a) During staging as signaled by the Abort Stage Signal and thereafter for a prescribed time between 1 and 10 seconds after receipt of the Ascent Engine-ON signal.
- (b) At lift-off from the lunar surface and thereafter for a prescribed time between 1 and 10 seconds after receipt of the Ascent Engine-ON signal.
- (c) Upon computer recognition of velocity to be gained threshold prior to termination of AGS controlled powered maneuvers.

During the ATTITUDE HOLD submode, the AGS shall perform the functions detailed in paragraphs 3.3.1.2.2.2.1 (a), (c), (d), (e), (f) of this specification except during staging and at lift-off when engine-ON is commanded.

3.3.1.2.2.2.3 AUTOMATIC Submode. - The AGS shall be in the AUTOMATIC submode when the automatic submode conditions of Figure 3 are satisfied. In this submode, the AGS shall output attitude error signals and engine commands which are necessary to guide the vehicle during the burns specified in the Coelliptic Flight Plan (see paragraph 3.1.2.2). During this submode, the AGS shall also perform the functions detailed in paragraph 3.3.1.2.2.2.1 (a), (c), (d), (e).

3.3.1.2.2.2.4 SEMI-AUTOMATIC Submode. - The AGS shall be in this submode when the semi-automatic submode conditions of Figure 4 are satisfied. When in this submode, the AGS shall generate attitude error signals to reorient LM to the desired thrust direction. An Engine-ON command will be generated on completion of this maneuver provided ullage has been detected. During the SEMI AUTOMATIC submode, the AGS shall also perform the function detailed in paragraph 3.3.1.2.2.2.1 (a), (c), (d).

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3.3.1.2.2.2.5 CSM Acquisition Submode. - The AGS shall be in this submode when the CSM acquisition submode conditions of Figure 4 are satisfied. During this submode, the AGS shall orient the LM vehicle z-axis toward the AGS computed direction of the CSM. During this submode, the AGS shall also perform the functions detailed in paragraph 3.3.1.2.2.2.1 (a), (c), (d). The x-axis is directed parallel to the CSM orbit plane.

3.3.1.2.2.2.6 Calibration Mode. - The requirements of this mode are that it provides the capability for in-flight gyro drift and accelerometer bias calibration and lunar pre-launch gyro drift calibration. These requirements are given in 3.1.3.9. The calibration mode shall provide the following functions:

- (a) In-flight calibration/IMU align
- (b) Lunar surface calibration
- (c) Earth prelaunch calibration.

3.3.1.2.2.3 AGS Operation in Each Submode. - The operations performed by the AGS in each of the submodes are given in Table V.

The operations performed in a submode can be classified as those which are always performed in the submode and those which are performed on command via the DEDA. The former are identified by a "1" and the latter by a "2" in Table V. The exceptions to this are that when LM and CSM initialization via the downlink or LM initialization via the DEDA are commanded, CSM Navigation, and the Guidance computation are not performed. In the Calibration mode, commanded via the DEDA, the AGS performs Inflight Gyro Drift Calibration, IMU Align and Inflight Accelerometer Bias Calibration, and the additional operations indicated in Table V when LM is in coasting flight. When on the lunar surface, as indicated by the lunar surface signal (an internal flag set when Store Lunar Azimuth is commanded at lunar touchdown), the AGS performs Lunar Surface Gyro Drift Calibration and the additional operations indicated in Table V.

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No. LSP-300-3B3.3.2 Attitude Reference, Guidance and Other Requirements. -

3.3.2.1 Inertial Reference Alignment Accuracy. - The AGS shall be aligned to the accuracy specified within a 3 minute period. The single axis alignment error is defined as the average of the angular rotation about an axis of the desired inertial reference required to rotate each of the nominally orthogonal axes of the AGS inertial reference frame to be coincident with the corresponding axes of the AGS inertial coordinate frame.

- (a) IMU Alignment Accuracy - The AGS inertial reference frame shall be aligned to the PGNCS command angles with a maximum error of 4 arc minutes at the completion of the IMU alignment submode.
- (b) Lunar Alignment Accuracy - The AGS inertial reference frame shall be aligned to within 10 arc minutes, maximum, per axis of the gravity vector at the completion of the lunar align submode and shall be aligned to within 10 arc minutes of the stored value of azimuth. (See note.)
- (c) Body Axis Alignment Accuracy - The AGS inertial reference frame shall be aligned to within 3 arc minutes per axis of the ASA frame at the completion of the Body Axis Align Submode.

NOTE: If, as a result of the mission environment specified in 3.2.3 the linear acceleration and angular rates exceed $\pm 25^\circ/\text{sec}$ and $90 \text{ ft}/\text{sec}^2$, and the inertial sensors encounter their stops, loss of attitude reference and velocity may result, at which time the performance requirements of this specification will not be met. Immediately after the shock, the AGS shall meet all the requirements of this specification after subsequent alignment and initialization, or after a lunar align provided the loss of azimuth reference due to the shock does not exceed 1.0 degrees.



3.1.3.1.2 Attitude Error Signal Resolution. - The attitude error signals shall have a resolution of one arc minute or less per axis.

3.1.3.1.3 Attitude Error Signal Quantization. - The attitude error signals shall be quantized to 1/4 degree (15 arc minutes).

3.1.3.1.4 Attitude Error Signal Drift. - The attitude error signals shall be drift-free. The drift shall be less than 0.1 arc minutes per hour. The drift shall be caused by the gyro error and shall be less than 0.1 arc minutes per hour.

3.1.3.1.5 Attitude Error Signal Accuracy. - The AOS shall provide attitude error signals to the navigation system with an accuracy of 0.1 arc minutes per hour. The accuracy shall be based on the accuracy of the gyro error and shall be less than 0.1 arc minutes per hour.

3.1.3.1.6 Attitude Error Signal Reliability. - The attitude error signals shall be reliable. The reliability shall be based on the reliability of the gyro error and shall be less than 0.1 arc minutes per hour. (See para. 3.1.3.4).

3.1.3.1.7 Attitude Error Signal Integration. - The attitude error signals shall be integrated. The integration shall be based on the integration of the gyro error and shall be less than 0.1 arc minutes per hour.

3.1.3.1.8 Attitude Error Signal Output. - The attitude error signals shall be output to the navigation system. The output shall be based on the output of the gyro error and shall be less than 0.1 arc minutes per hour.

3.1.3.1.9 Attitude Error Signal Input. - The attitude error signals shall be input to the navigation system. The input shall be based on the input of the gyro error and shall be less than 0.1 arc minutes per hour.

3.1.3.1.10 Attitude Error Signal Processing. - The attitude error signals shall be processed. The processing shall be based on the processing of the gyro error and shall be less than 0.1 arc minutes per hour.

3.1.3.1.11 Attitude Error Signal Storage. - The attitude error signals shall be stored. The storage shall be based on the storage of the gyro error and shall be less than 0.1 arc minutes per hour.

3.1.3.1.12 Attitude Error Signal Retrieval. - The attitude error signals shall be retrieved. The retrieval shall be based on the retrieval of the gyro error and shall be less than 0.1 arc minutes per hour.

3.1.3.1.13 Attitude Error Signal Display. - The attitude error signals shall be displayed. The display shall be based on the display of the gyro error and shall be less than 0.1 arc minutes per hour.

3.1.3.1.14 Attitude Error Signal Control. - The attitude error signals shall be controlled. The control shall be based on the control of the gyro error and shall be less than 0.1 arc minutes per hour.

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3.3.2.9 Orbit Insertion. - For aborts subsequent to the start of powered descent and prior to the end of powered ascent, the AGS shall have the capability of inserting the LM initially into a standard elliptical orbit having a minimum clear pericyynthion of 30,000 feet above the mean lunar radius. Capability shall be provided to perform a fully automatic orbit insertion maneuver as described above using initialization data from any one of the following sources: PGNCs, CSM Voice Link, MSFN, whenever they are available. Accuracy and availability of these initialization sources are given in paragraph 3.2.11.

3.3.2.10 Coelliptic Orbit Sequence. - For aborts prior to the start of powered descent or after insertion into the elliptical orbit, the AGS shall have the capability to perform a series of coelliptic rendezvous burns as described in paragraph 3.1.2.2. For aborts after the end of powered ascent, the AGS shall have the capability to complete the unfinished burns of the coelliptic sequence. For the Design Mission, an orbit with a minimum clear pericynthion of 30,000 ft. above the mean lunar radius shall result after each of these burns.

The burns for for the above cases shall result in an intercept trajectory passing within a sphere having a 5 n. mi. (3 sigma) radius with its center at the CSM. At the point of intersection of the LM and the sphere, the magnitude of the vector difference between the nominal AGS computed relative velocity (i.e., velocity of LM with respect to the CSM) and the actual relative velocity shall be less than 30 fps (3 sigma). This requirement shall be contingent upon the target constraints and the AGS performance of miscourse corrections based upon RR navigation updates as specified in paragraph 3.3.2.8 and to cases where the TPI burn is completed in less than one revolution of the CSM after the CSI maneuver. Capability shall also be provided to perform a coelliptic orbit sequence as described in 3.1.4.2, using initialization data from the following sources: PGNCs, CSM Voice Link and MSFN, whenever they are available. Accuracy and availability of the initialization sources are given in paragraph 3.2.11.

3.3.2.11 Direct Intercept. - The direct intercept maneuver described in 3.1.2.3. shall only be used for aborts prior to powered descent, and after the end of powered ascent. The direct transfer burn shall result

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in an intercept trajectory passing within a sphere having a 5 n. mi. radius (3 sigma) with its center at the CSM. At the point intersection of the LM and the sphere, the magnitude of the vector difference between the nominal AGS computed relative velocity (i.e., velocity of LM with respect to the CSM) and the actual relative velocity shall be less than 30 fps (3). This requirement shall be contingent upon the AGS performance of midcourse corrections based upon RR navigation updates as specified in paragraph 3.3.2.8. Capability shall also be provided to perform the Direct Intercept Maneuver using initialization data from any one of the following sources: PGNCs, CSM Voice Link, and MSFN, whenever they are available. Accuracy and availability of these sources are given in paragraph 3.2.11.

3.3.2.12 External Delta V. - Capability shall be provided to perform a velocity change based on velocity components obtained from an external source and inserted via the DEDA, (See paragraph 3.1.2.4).

3.3.2.13 Engine On/Off Commands. - In the semi-automatic mode, the AGS shall be required to initially turn on either the ascent or descent engine if a state of ullage has been detected by the AGS.

The AGS shall detect ullage when the accumulated velocity increments along the X-axis exceeds 0.2 ft/sec in each 2 second computer cycle, for 3 consecutive cycles. The AGS shall detect loss of ullage when, in any 2 second computer cycle, the accumulated velocity increments along the vehicle X-axis is less than 0.2 ft/sec.

The AGS shall continue to maintain an engine ON signal until the velocity to be gained (V_{ax}) decreases below a preselected threshold.

3.3.2.14 Earth Prelaunch Calibration/Compensation. - Shall be based on a scheme which uses the known latitude of the launch site, a self alignment to local vertical and a known azimuth angle with respect to true north known to 0.1 degree. The calibration scheme shall also use a measurement of the X-gyro channel spin axis mass unbalance made a maximum of 120 days prior to completion of the lunar landing mission.

The implementation of this scheme shall have a calibration capability within a maximum of 20 minutes such that the compensated AGS attitude drift rate during the mission environment, when the AGS is operating, shall be less than the following:

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3.3.2.14 (Continued)

<u>Days After Calibration</u>	<u>Environment</u>	<u>X Channel</u>	<u>Y & Z Channels</u>
0 to 8 days	Coasting	2.49°/hr (3 σ)	0.90°/hr (3 σ)
0 to 8 days	Boosting	*	1.03°/hr (3 σ)

* { uncorrelated bias 1.03°/hr (3 σ)
 * { uncorrelated g Sensitive 0.96°/hr/g (3 σ)
 * { correlated bias and g Sensitive 2.32 (1-accel. along x-axis) °/HR (3 σ)

3.3.2.15 AGS In-Flight Calibration/Compensation. - Shall be initiated via a DEDA command and shall be performed based on a scheme which compares the PGNS reference with the AGS attitude drift rate. For a calibration time of not more than 5 minutes, during coasting phase, compensation shall be performed such that the resultant AGS attitude drift rate during the mission environment, when the AGS is operating, shall be less than the following:

<u>Days After Calibration</u>	<u>Environment</u>	<u>X Channel</u>	<u>Y & Z Channels</u>
0 to 3 days	Coasting	0.73°/hr (3 σ)	0.73°/hr (3 σ)
0 to 3 days	Boosting	*	0.89°/hr (3 σ)

* { uncorrelated bias 0.89°/hr (3 σ)
 * { correlated bias and g Sensitive 2.35 (1-accel along x-axis) °/HR (3 σ)

The PGNS total attitude drift rate shall be assumed to be 0.15°/hr (3 σ) for the purposes of this requirement. A filtering technique shall be implemented in order to minimize noise present in the PGNS and AGS angle data.

3.3.2.16 Accelerometer Channel In-Flight Calibration/Compensation. - The AGS upon DEDA command shall have the capability to calibrate/compensate AGS accelerometer channel bias. For a calibration time of not more than 4 minutes, during the coasting phase, compensation shall be performed such that the resultant AGS accelerometer channel bias during the mission environment, when the AGS is operating, shall be less than the following:

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3.3.2.16 (Continued)

<u>Days After Calibration</u>	<u>X, Y & Z Channels</u>
0 to 3 days (Boost Environment)	230 x 10 ⁻⁶ g's (3σ)

3.3.2.17 Lunar Surface Prelaunch Calibration/Compensation. - The AGS shall have the capability to automatically calibrate/compensate AGS Attitude drift rates while LM is resting on the lunar surface. This operation shall be initiated by a DEDA command after the appropriate constants are inserted via the DEDA for use in the calibration process.

The implementation of this operation shall have a calibration capability within a maximum of 5 minutes, such that the compensated AGS attitude drift rate during the mission environment, when the AGS is operating, shall be less than the following:

<u>Days After Calibration</u>	<u>X Channel</u>	<u>Y and Z Channels</u>
0 to 1 day (Coasting)	0.77°/hr (3σ)	0.84°/hr (3σ)
0 to 1 day (Boosting)	*	1.00°/hr (3σ)
* { uncorrelated	0.78°/hr (3σ)	
{ uncorrelated g sensitiv	0.46°/hr/g (3σ)	
{ correlated	2.35" (1/6-accel along x-axis)°/HR(3σ)	

3.3.2.18 Gyro Channel Scale Factor Pre-Flight Calibration/Compensation. - The AGS shall have the capability to compensate AGS gyro channel scale factor. The compensation coefficients shall be entered into the AEA via the DEDA. The AGS gyro channel scale factor error after compensation and in the mission environment when the AGS is operating, shall be less than the following:

<u>Days After Calibration</u>	<u>X, Y & Z Channels</u>
0 to 120 days	935 x 10 ⁻⁶ deg/deg (3σ)

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3.3.2.19 Accelerometer Scale Factor Channel Pre-Flight Calibration/Compensation. - The AGS shall have the capability to compensate AGS accelerometer channel scale factor. The compensation coefficients shall be entered into the AEA via the DEDA. The AGS accelerometer channel scale factor error after compensation, and in the mission environment when the AGS is operating, shall be less than the following:

<u>Days After Calibration</u>	<u>X, Y & Z Channels</u>
0 to 120 days	$940 \times 10^{-6} \text{ g/g } (3\sigma)$

3.3.2.20 Accelerometer Bias Pre-Flight Calibration/Compensation. - The AGS shall have the capability to compensate AGS accelerometer channel bias. The compensation coefficients shall be entered into the AEA via the DEDA. The AGS accelerometer channel bias after compensation during the mission environment when the AGS is operating, shall be less than the following:

<u>Days After Calibration</u>	<u>X, Y & Z Channels</u>
0 to 120 days	$615 \times 10^{-6} \text{ g } (3\sigma)$

3.3.2.21 Combined Rates. - The AEA shall be capable of following gyro input rates up to and including $25^\circ/\text{sec.}$ consecutively or concurrently about all vehicle axis with a maximum computational lag error of $\pm 0.70^\circ$ per axis.

3.3.2.22 Attitude Error Signal Range. - The deviation of the actual attitude from the commanded shall have a maximum magnitude of 180° . The range of attitude error signals issued by the AGS shall be ± 15 degrees.

NOTE: The numerical values specified in the previous paragraphs shall be considered as the absolute value of the 3σ dispersion plus the absolute value of the mean of the performance values obtained during any 300 sec interval during the total time interval associated with each performance number (i.e., 3 day, 8 day, 120 days, as applicable).

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3.3.2.23 Drift Rate. - All errors resulting from the AEA processing limitations shall add up to an attitude reference drift rate of not more than 0.2 degrees per hour. The processing limitations shall include fictitious coning errors, coning errors, and ASA modeling errors.

3.3.2.24 Total Attitude Signal Accuracy. - The total attitude signal shall be accurate to ± 0.5 degrees, maximum, per axis except when within ± 0.5 degrees of either of the two attitude points of singularity. In this case, the maximum error in the angle " α " shall be less than 0.5 degree and the maximum error in the sum of the angles $\alpha + \delta$ shall be less than 0.5 degree.

3.3.2.25 Acceleration Range. - The AEA shall be capable of processing accelerometer range of ± 90 feet/second² with a velocity increment of (.003125) feet/second per pulse.

3.3.2.26 Flight Director Attitude Indicator. - The sines and cosines of the Euler angles α , β and γ that are output by the AGS to the FDAI for total attitude display shall be defined by the following transformation between vehicle axes and AGS inertial coordinates axes:

$$\begin{pmatrix} X \\ Y \\ Z \end{pmatrix}_{\text{Inertial}} = \begin{bmatrix} c\alpha & c\gamma & -s\alpha & s\beta & s\gamma & s\alpha & c\gamma & +c\alpha & s\beta & s\gamma & c\beta & s\gamma \\ & & -s\alpha & c\beta & & & c\alpha & c\beta & & & & -s\beta \\ -c\alpha & s\gamma & -s\alpha & s\beta & c\gamma & s\gamma & +c\alpha & s\beta & c\gamma & & c\beta & c\gamma \end{bmatrix} \begin{pmatrix} X \\ Y \\ Z \end{pmatrix}_{\text{Body}}$$

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4 QUALITY ASSURANCE PROVISIONS

4.1 Test Requirements. - This section of the specification establishes the general test requirements to be followed during the AGS test program. The program shall consist of the following test categories:

- (a) Design Verification Tests (4.2)
- (b) Qualification Tests (4.3)
- (c) Acceptance Tests (4.4)
- (d) System Simulation Tests

4.2 Design Verification Tests. - Design verification tests shall include all tests conducted to substantiate the correctness of the design for its intended mission under simulated ground and flight environments and off-design conditions.

4.2.1 Critical Environmental Tests. - Those tests which are conducted on early production test units, (assemblies or sections) under selected critical environments to provide confidence that the test unit shall pass qualification design limit load tests. Selection of environments shall be made by the vendor subject to Grumman approval.

4.2.2 Specific Tests. - As a culmination to design verification the test unit (section or assembly) shall be subjected to the tests listed in the following subparagraphs. Tests applicable to this requirement of design verification shall fulfill the following essentials:

- (a) The tests shall be performed on production equipment.
- (b) The test sequence shall be in the order presented herein. Deviations from the sequence can be made subject to Grumman approval.
- (c) Successful completion of portions of these Design Verification Tests will permit performance of corresponding portions of the Qualification tests as approved by Grumman.
- (d) No replacement of parts, adjustments or maintenance shall be permitted during the selected environmental tests except when approved by Grumman.

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4.2.2 (Continued)

(d) (Continued)

- (1) If a failure occurs during the tests, preliminary analysis shall be performed and Grumman immediately notified of the failure. Action may be initiated such as the replacement of parts only with the approval of Grumman.
- (2) If a redesign or a retest or both of any part, component or assembly of the section under test is indicated from (a) above, the selected environmental tests shall be considered incomplete until retest has been completed to the satisfaction of Grumman. At the discretion of Grumman, redesign and retesting may be required of any part, assembly or equipment which fails to survive the selected environments or exhibits damage which is in excess of the minimum acceptable operating mode. Failure of any part or component of a redundant circuit or mode shall be considered failure of the equipment. At the completion of the selected environmental tests, in addition to the functional check specified, the equipment shall be visually inspected and its circuitry checked to the extent allowable, short of disassembly, in order to ascertain to the highest degree possible the condition of the parts and components, within the equipment.

4.2.2.1 Selected Environments. - The test AGS section shall be successfully subjected to the selected environments of the operational cycle in accordance with the Selected Environments table of the respective assembly (AEA, DEDA and ASA) Design Control Specifications.

4.2.2.2 Overstress Tests. - At the completion of the selected environmental tests, the test unit (section or assembly), AGS shall be tested to failure under systematically increasing dynamic and environmental stresses. Deviation of performance from the minimum acceptable operating mode shall constitute a failure. The equipment shall dwell long enough at each increment of overstress to stabilize conditions and complete the abbreviated operational test, when applicable.

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4.2.2.2.1 Selection of Stresses. - The Failure Mode Prediction Analysis shall provide the basis for the selection of critical stresses to be employed in the overstress tests. Only conditions from the launch and post-launch phase of the mission shall be used for the overstress tests. If the critical mission stresses are due to a combination of dynamic and environmental conditions, the tests shall be performed under that combination of environments. If the critical stresses are due to several dynamic and environmental conditions which are not in combination in the mission, the test increments shall be performed with each condition imposed separately. Each increment of the test conditions shall be increased in proportion to its values at mission levels.

4.2.2.2.2 Input Parameters. - The input parameters, such as mass flow, voltage, current, frequency, etc., shall be maintained at the values chosen for the selected environmental tests.

4.2.2.2.3 Limited Life Items. - The test items for which operating time or cycles may produce lower failure modes shall be tested with each increment proportioned to the exposure time or number of cycles used in the Flight Simulation tests.

4.2.2.3 Analysis of Results. - An engineering analysis of the data generated by the overstress tests including a correlation with the Failure Mode Prediction Analysis shall be performed for Grumman evaluation.

4.3 Qualification Tests. - Qualification tests shall be performed using two production AGS's to demonstrate attainment of design objectives including margins of safety. The qualification tests listed below shall be performed on the AGS:

(a) Design Limit Tests (4.3.4)

(b) Endurance Tests (4.3.5)

Design limit tests shall be performed on a section or an assembly level. Endurance tests shall be performed on a section level.

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4.3.1 General. - The test sequence shall be in the order presented herein. Deviation from the sequence can be made subject to Grumman approval. The test unit (assembly or section) and test apparatus shall be subjected to inspection by Grumman Quality Control representatives. At convenient times prior to and after the tests, the test unit (assembly or section) shall be examined to determine if it conforms to all requirements. At the option of Grumman, measurements shall be made of critical dimensions prior to start of the qualification tests. During the progress of tests, examinations may be made at the option of Grumman. The results of these examinations shall be a part of the qualification test data. Any test articles that have undergone qualification tests shall not be used for flight acceptance tests, nor shall they be used as flight equipment or flight spares. All parts or elements which have been subject to qualification tests, either individually or as part of a larger assembly shall be distinctively marked as qualification tested parts. Unless prior approval has been obtained from Grumman, items of equipment which have been qualification-tested shall not be used for qualification testing as part of a higher order assembly.

4.3.2 Parts Failure and Replacement. - No replacement of parts, adjustments or maintenance will be permitted during qualification tests except when approved by Grumman.

- (a) If a failure occurs during a qualification test, Grumman shall be notified immediately. Corrective action may be initiated only with the approval of Grumman.
- (b) In general, redesign of parts failed due to an inherent design deficiency during qualification tests shall be mandatory; however, in the event of circumstances which Grumman shall have the option of designating as mitigating, parts may be replaced and the test continued.
- (c) If a redesign or a retest or both of any part or component of the test unit (assembly or section) under test is indicated from (a) or (b) above, the qualification test of the test unit (assembly or section) in which the part or component is contained shall be considered incomplete until the retest of the part or component in question has been completed to the satisfaction of Grumman.

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4.3.2 (Continued)

- (d) At the discretion of Grumman, redesign and retesting may be required of any part which has completed the qualification test program, but fails or indicates weakness when retained to complete testing of other parts.

4.3.3 Inspection Before Tests. - The test unit (assembly or section) shall be completely inspected for compliance with the drawings and specifications prior to qualification testing. Deviations from the drawings and specifications shall be cause for rejection approved by Grumman.

4.3.4 Design Limit Tests. - The test unit (assembly or section) shall be successfully subjected to the Flight Simulation tests of the operation cycle at design limit levels as given in the design limit tables of the respective assembly design control specification.

4.3.5 Endurance Tests. - The test section shall be successfully subjected to a complete Operational Cycle including super flight simulation, at mission levels in accordance with the Endurance Test tables of the respective assembly design control specifications.

4.3.6 Inspection After Tests. - After completion of the qualification test program, each test unit shall be disassembled to the component or module level and subjected to a tear-down inspection. Measurements shall be taken, as necessary, to disclose excessively worn, distorted, or weakened parts. Photographs shall be taken of such discrepant parts and these photographs shall be included in the test report.

4.3.7 Post-Qualification Tests. - At the completion of the qualification tests, the test unit (assembly or section) shall be subjected to the following additional tests in order to increase confidence in equipment design life and strength. Post-qualification shall not constrain production nor be a cause for additional facility or special test equipment requirements. Equipment failures experienced during post-qualification testing shall be brought to the attention of Grumman for an evaluation of the impact on the flight worthiness of the equipment.

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4.3.7.1 Overstress Tests. - The post-qualification testing of the Design Limit test unit (section or assembly) shall consist of overstress tests in the same mode or condition as selected or the design verification overstress tests.

4.4 Acceptance Tests. - Each deliverable assembly of the AGS shall be subjected to an acceptance test as per LSP-300-37, LSP-300-33 and LSP-300-39. After each assembly has successfully completed the assembly acceptance test, the AEA, DEDA and the ASA shall be subjected to an AGS acceptance (compatibility) test. The acceptance (compatibility) test shall demonstrate that:

- (a) The AGS satisfies all applicable requirements of this specification under the restricted set of initial conditions, operating modes and operating sequences of paragraph 3.1.4 (Design Mission).
- (b) Conforms to the approved design corresponding to the particular AGS production model under test.

AGS's to be used on flight spacecraft shall not contain an assembly which has been subjected to more than two acceptance tests, nor an assembly which has been subject to environments of an intensity high than acceptance test levels, nor an assembly which has accumulated an excess of 200 hours total running time, during acceptance testing. No more than 150 hours shall be accumulated during the assembly level acceptance testing. No more than 50 hours shall be accumulated during the compatibility testing.

4.5 Systems Simulation Tests. -

4.5.1 Scope. - Sufficient System Simulation Tests shall be run to verify that the computations required of the AEA meet all the requirements of this specification and the applicable subassembly specifications. These tests shall not use actual flight hardware but math. models that simulate the hardware and associated environment where required. The simulations may be divided into Open Loop Checkout and Closed Loop Simulations.

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4.5.1.1 Open Loop Checkout. - The Interpretive Computer simulation (ICS) shall be used in open loop operations with drivers consistent with this specification to checkout selected functions as required prior to their testing in closed-loop operation. The ICS shall provide the computation characteristics of the AEA and shall accept the hard wired and soft wired programs as written for the AEA.

4.5.1.2 A flight simulation (ICS/FS) wherein the ICS and the N-stage dynamics are integrated, will be used to simulate aborts.

5 PREPARATION FOR DELIVERY

5.1 Preservation, Packaging and Packing. - Preservation, packaging and packing shall be in accordance with Section 5 of referenced specifications listed in 2.1.

6 NOTES

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TABLE I

ABORT GUIDANCE SECTION FUNCTIONS

A. EARTH PRELAUNCH

- Preflight checkout and calibration

B. EARTH LAUNCH THROUGH TRANSILUNAR FLIGHT

- Nonfunctioning (off condition)

C. LUNAR ORBIT (PREPARATION FOR LANDING MISSION)

- OFF-STANDBY OPERATE Conditions
- SELF Test
- IN FLIGHT CALIBRATION/COMPENSATION
- ALIGN INERTIAL REFERENCE to PGNS
- Initialize Absolute Time via the DEDA
- Initialize LM and CSM State Vectors using PGNS Downlink
- AGS in FOLLOWUP Submode before Descent Orbit Injection

D. LUNAR DESCENT ORBIT INJECTION (NON-ABORT CONDITION)

- FOLLOWUP Submode
- Maintain inertial reference
- Maintain Navigation
- Continually solve for Abort Trajectory which will either Intercept the CSM in its Orbit Directly or execute concentric Flight plan (depending on option selected)
- Provide Engine-ON commands
- Provide Data for Inflight Monitoring
- Provide Analog and Digital Telemetry
- Accept orbit insertion targeting data

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TABLE I (Continued)

E. LUNAR DESCENT ORBIT INJECTION (ABORT CONDITION)

- AUTOMATIC Submode
- Maintain Inertial Reference
- Continually Solve for Abort Trajectory
- Provide Engine-ON Commands
- Provide Steering Signals and Engine-OFF Command as required to Steer LM to Rendezvous
- Provide data for Inflight Monitoring
- After Engine Shutdown AGS Functions same as in para. "O."

F. LUNAR DESCENT COAST (NON-ABORT CONDITION)

- FOLLOWUP Submode
- Maintain Inertial Reference
- Maintain Navigation
- Accept PGNS Alignment Information - Realign
- Accept PGNS Downlink Information - Reinitialize LM and CSM
- Continually Solve for Abort Trajectory
- Provide Data for Inflight Monitoring
- Provide Digital Telemetry
- Accept Orbit insertion targeting data via the DEDA

G. LUNAR DESCENT COAST (ABORT CONDITION)

- Automatic Submode
- Maintain Inertial Reference
- Maintain Navigation
- Continually solve for Abort Trajectory
- Provide Engine-ON Commands

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TABLE I (Continued)

G. (Continued)

- Provide Steering Signals and Engine-OFF Command as Required to Steer LM to Rendezvous with the CSM or to a Parking Orbit
- After Engine Shutdown AGS Functions same as in para. "M" - Ascent Boost (Abort Condition)
- Provide Data for Inflight Monitoring

H. POWERED DESCENT (NON-ABORT CONDITION)

- Same as "D"-Lunar Descent Injection (Non-Abort Condition)

I. POWERED DESCENT (ABORT CONDITION)

- Same as "E" - Lunar Descent Injection (Abort Condition)

J. LUNAR SURFACE (PRELAUNCH CHECKOUT, NON-ABORT CONDITION)

- OFF-STANDBY-OPERATE Condition
- Lunar Preflight Checkout and Calibration/Compensation
- In SELF TEST
- Align Inertial Reference to PGNS
- Initialize CS4 and Absolute Time through DEDA
- Initialize LM through PGNS Downlink
- AGS in FOLLOWUP Mode before Liftoff

K. LUNAR SURFACE (PRELAUNCH CHECKOUT, ABORT CONDITION)

- OFF-STANDBY-OPERATE Condition
- Lunar Preflight Checkout and Calibration/Compensation
- Self Test Mode
- Enter LUNAR ALIGN Submode
- Accept Azimuth Alignment Data through DEDA
- Accept LM, CSM and Absolute Time Data through DEDA
- Initialize LM, CSM, Absolute Time via DEDA

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TABLE I (Continued)

K. (Continued)

- Enter Inertial Reference Mode
- Continually Solve for Abort Trajectory from Lunar Surface which will either Intercept the CSM directly or Place LM in a Parking Orbit
- Provide Data for Inflight Monitoring
- Provide Digital and Analog Telemetry
- On Entry into AUTOMATIC Submode Function as described under para "E".

L. LUNAR ASCENT BOOST (NON-ABORT CONDITION)

- Same as "H" - Powered Descent (Non-Abort Condition)

M. LUNAR ASCENT BOOST (ABORT CONDITION)

- Same as "I" - Powered Descent (Abort Condition)

N. LUNAR ASCENT COAST (NON-ABORT CONDITION)

- Same as "F" - Lunar Descent Coast (Non-Abort Condition)

O. LUNAR ASCENT COAST (ABORT CONDITION)

- ALTITUDE HOLD Submode
- Maintain Inertial Reference
- Acquisition Submode
- Accept Radar Data to Update Navigation
- Continually solve for Parking or Intercept Trajectory
- Provide Inflight Monitoring Data
- Provide Analog and Digital Telemetry

P. LUNAR ASCENT MIDCOURSE CORRECTION (ABORT CONDITION)

- Maintain Inertial Reference

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TABLE I (Continued)

P. (Continued)

- Provide Attitude Reorientation Commands on Entry into SEMI-AUTOMATIC Submode
- On Entry into AUTOMATIC Submode Function as described under para "E"

Q. CSM/ RENDEZVOUS (ABORT AND NON-ABORT CONDITIONS)

- Maintain Inertial Reference
- Maintain Navigation
- FOLLOWUP or ATTITUDE HOLD Submode

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TABLE II

MISSION PROFILE

Phase	Time (Minutes)		Environment
	Boost Total	Non- Boost Total	
Prelaunch*		120.0 days	Prelaunch Unpackaged
Earth launch & ascent	11.8		Launch & Boost
Earth orbit through transposition	5.2	210.9	Translunar
Continue translunar trip through insertion into lunar orbit	6.4	3627.4	Translunar
Coast in lunar orbit (LEM checkout)		223.3	Translunar
Total preseparation	23.4	4661.6	
LEM separation to insertion		20.0	Descent
Insertion and Hohmann Transfer orbit	0.6	57.9	Descent
Powered descent from pericyynthion to hover	8.4		Descent
Hover to touchdown	1.14		Descent

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No. LSP-300-3BTABLE II (Continued)

Phase	Time (Minutes)		Environment
	Boost Total	Non- Boost Total	
Post landing checkout		75.0	Lunar Stay
Exploration		65.0	Lunar Stay
Prelaunch preparation		100.0	Lunar Stay
Total lunar stay for mission success estimate		2153.5	Lunar Stay
Additional lunar stay for crew safety estimate		1200.0	Lunar Stay
Powered ascent	7.2		Ascent
Coast (approximately) 30 mins		28.	Ascent (no vibration)
Coelliptic sequence initiation, RCS horizontal burn	0.5		Ascent
Coast to apocynthion		49.	Ascent (no vibration)
Circularization burn at 65 mi., RCS burn	0.5		Ascent
Coast in 65 mi. parking orbit		45.	Ascent (no vibration)

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TABLE II. (Continued)

Phase	Time (Minutes)		Environment
	Boost Total	Non- Boost Total	
Terminal Phase Initiation, RCS burn	0.5		Ascent
Coast 140° central angle, includes 2 midcourse corrections with RCS burns of 24 and 4 sec. (V=30 and 5 fps)		48.	Ascent (no vibration)
Rendezvous, includes 3 RCS burns of approximately 4, 8, and 4 sec (V=5, 10 and 5 fps)		10.	Ascent (no vibration)
Docking		15.	Ascent

*This phase consists of pad time after calibration on site and does not include 4.5 years of storage or 850 hours of operating test time.

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TABLE III

LM ENGINE PERFORMANCE

A. DESCENT ENGINE

1. Throttle range - 1050 - 6,300 lb. with a step to nominal max thrust.
2. Nominal maximum thrust - 9,880 lb. \pm 1.5% with no erosion
3. Nominal Isp - 303.5 seconds, one sigma deviation 15 sec.
4. Thrust Profile - 30% for 3 sec, 10% for next 23 sec., Maximum Thrust thereafter "Defined in TBD "
5. Thrust Cut-off impulse* - \pm 100 lb-sec (3 sigma)
6. Thrust misalignment (to be revised)
 - a. Thrust uncertainty = 0.5° cone
 - b. Mount misalignment = 2° cone
 - c. Gimbal angle - 6° cone
7. The nominal descent engine impulse occurring between AGS Engine-Off command and complete shut-down shall be variable between 800 and 1600 lb-sec. The AGS shall be designed to allow for the variation from engine to engine.

*Nominal value unknown

B. ASCENT ENGINE

1. Nominal thrust - 3500 \pm 52.5 lb.
2. Nominal Isp - 311., one sigma deviation: 2 sec.

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TABLE III (Continued)

B. (Continued)

3. Thrust Cut-off Profile:

- a. Thrust decay to 10% rated thrust in 0.20 seconds
- b. Engine cut-off command delay

4. The nominal cut-off impulse shall be variable between 300 and 450 lb-sec. The AGS shall be designed to allow this variation from engine to engine. The 3 sigma deviation about the nominal shall be 75 lb-sec.

5. Thrust misalignment (to be revised.)

- a. Thrust uncertainty = 0.3° cone
- b. Mount misalignment = 2° cone

6. The delay between engine-on command and ascent engine ignition shall not exceed 1 sec. during launch from the lunar surface.

7. The Ascent Engine will be canted as follows:

$$\delta\psi = 2.0^\circ \text{ (about Z)}$$

$$\delta\theta = 3.5^\circ \pm 2^\circ \text{ (about Y)}$$

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TABLE IV
 BODY AXES ATTITUDE DYNAMICS
 (INITIAL LFM WEIGHT (32579 POUNDS))

	Axis	Separation	Start of Powered Descent	Hover	Lift-off	Docking
Limit Cycle Period + 0.3 deg deadzone	t_p	62.7	62 sec	35.4 sec	17.4 sec*	8.1 sec
	t_q	127	1.5-125	1.5-79	16.9	13.5
	t_r	127	1.5-125	1.5-89	28	8.2
Limit Cycle Period + 5 deg deadzone	t_p	1050 sec				135 sec
	t_q	2110				223
	t_r	2110				137
Angular Acceleration Capabilities	$\dot{\phi}$	5.1 $\frac{\text{deg}}{\text{sec}^2}$				38.4 deg/sec ²
	\dot{q}	2.53				24
	\dot{r}	2.53				39
Attitude Rate Response Time for Small (less than 0.5°/sec) commands (95% steady state)	T_p	0.2 sec				0.09 sec
	T_q	0.4				0.16
	T_r	0.4				0.10

*Applies for no moment unbalance. Moment unbalances during powered ascent can cause limit cycle operation with a maximum frequency of 3.26 cps.

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TABLE IV (Continued)

	Axis	Separation	Start of Powered Descent	Hover	Lift-off	Docking
Attitude Rate Response Time for Large (e.g., 10 deg/sec) commands (95% steady state)		1.86 sec				0.25
		3.76				0.4
		3.76				0.24
Attitude Hold Time Constant	T_p	1.43 sec				0.36 sec
	T_q	1.34				0.33
	T_r	1.34				0.36
Attitude Rate Limits, (Upper)	Manual Control		All axes: 21 deg/sec \pm 1°/sec			
	Automatic Control		Roll and Yaw: 5 deg/sec \pm 5%, Pitch: 10 deg/sec \pm 5%			

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TABLE V

AGS Operations Performed in Submodes

Operation	IMU Align	Lunar Align	Body Axes Align	Followup	Attitude Hold	Semi Automatic	Automatic	Acquisition	Calibration	Notes
PGNS Downlink Processing	2	2	2	2	2	0	0	0	2	(1) Zero attitude errors are output
Initialize	2	2	2	2	2	0	0	0	2	(2) Performed only on lunar surface if lunar surface signal is present
Gyro Data Processing and Compensation		1		1	1	1	1	1	1	(3) Not performed when lunar surface signal is present (input via DEDA)
Accelerometer Data Processing and Compensation	1	1	1	1	1	1	1	1	1	(4) Engine OFF command is output
Update Direction Cosines	1	1		1	1	1	1	1	1	(5) Not performed for 2 seconds when initialize via downlink or LM initialize via DEDA is commanded
IMU Align (6)	1								1 (3)	
Lunar Align		1								
Body Axes Align			1							(6) PGNS Euler Angles Processed only when IMU align operation is performed
Compute Attitude errors	(1) 1	(1) 1	(1) 1	(1) 1		1	1	1	1 (1) 1	
Telemetry	1	1	1	1	1	1	1	1	1	
RR Data Processing	0	0	0	2	2	0	0	2	0	
DEDA Processing*	1	1	1	1	1	1	1	1	1	
FDAI	1	1	1	1	1	1	1	1	1	

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TABLE V (Continued)

Operation	IMU Align	Lunar Align	Body Axes Align	Followup	Attitude Hold	Semi Automatic	Automatic	Acquisition	Calibration	Notes
LEM Navigation	1(5)	1(5)	1(5)	1(5)	1(5)	1	1	1	1(5)	
CSM Navigation	1(5)	1(5)	1(5)	1(5)	1(5)	1	1	1	1	Legend:
Inflight Self-Test	1	1	1	1	1	1	1	1	1	1 - These operations are always performed in the submodes indicated
Inflight Gyro Drift Calibration									1(3)	
Lunar Surface Gyro Drift Calibration *									1(2)	2 - These operations are performed on command via DEDA in the submodes indicated
Inflight Accelerometer Bias Calibration									1(3)	0 - These operations should not be commanded in the submodes indicated
Navigation Display	1	1	1	1	1	1	1	1	1	
Engine ON/OFF Commands	1(4)	1(4)	1(4)	1	1	1	1	1	1(4)	* - These operations are not described in the Flight Equations document (Ref 5).
Store Lunar Azimuth	2	0	0	2	2	0	0	0	0	

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TABLE VI

POWERED ASCENT COELLIPTIC RENDEZVOUS

	Mission Time (sec)	0	200	391.9	ALTITUDE (n. mi)	CSI BURN		CDM BURN		TPI BURN		TERMINAL END.
						2046.7	2096.4	5114.3	5167.3	6290.6	6309.3	
ALTITUDE (FT)	0	33165.	60000.		19.41	19.87	65.5	65.13	65.04	65.05	79.43	
V TOTAL (FPS)	15.2	2187	5510.		5454.9	5512.0	5264.0	5328.5	5328.3	5348.7	5286.9	
V VERTICAL (FPS)	0	212.7	0		55.01	57.79	-3.57	0	-8.54	11.06	1.89	
γ (DEG)	0	5.58	0		.00231	0.6	.00195	0	-.009	.118		
THRUST ACCEL (FPS ²)	11.476	14.245	0		1.21	1.22	1.22	1.23	1.23	1.23	0	
THRUST (LBS)	3636.	3636.	0		200.	200.	200.	200.	200.	200.	0	
ΔV (FPS)	0	261.6	5999.6		5999.6	6060.	6060.	6124.8	6147.8	6147.8		
WEIGHT (LBS)	10249	7062	5357		5357	5360	5314	5276	5271	5257		
σ_L (DEG)	0	1.858	8.992		98.88	101.568	256.78	259.412	315.62	316.565	478.88	
σ_{CM-Q} (DEG)	9.67	17.19	18.15		10.58	10.33	2.86	2.82	1.6	1.58	0	
λ_L (DEG)	.243	.1276	0		0	0	0	0	0	0	0	
λ_{Q-L} (DEG)	.43	.1276	0		0	0	0	0	0	0	0	
PITCH ATTITUDE (DEG)	90.	16.75	-5.5				5.81	10.82	29.4	31.04		
RANGE (N. MI)	176.92	302.3	329.4		19.6	187.5	52.57	51.68	31.95	31.60	0	
CSM VER VEL	0											
CSM TOT VEL	5287											
CSM ALT (N. M)	79.43											

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TABLE VII
TRAJECTORY PARAMETERS FROM IEM SEPARATION TO IO-GATE

	30M 32.8S	96M 32.8S	88M 31.441S	88M 46.441S	88M 48.441S	94M 47.691S	94M 49.691S	95M 77.511S	95M 81.511S	98M 37.878S
MISSION TIME	483859.	191154.	46391.	46494	46506	21108.	20909	9266	8642	115.
ALTITUDE (FT)	5193.3	5454	5588.5	5554.4	5536.3	1350.5	1334.4	639.3	609.9	12.23
TOTAL (FPS)	0	-188.2	-1.89	7.08	8.68	-99.25	-101.77	-156.09	-154.16	-1.69
VERTICAL (FPS)	0	-1.98	-1.93	7.305 ⁻⁰²	8.982 ⁻⁰²	-4.214	-4.374	-14.13	-14.64	-7.954
(DEG)	0	0	3.163	1.058	9.809	15.95	8.45	9.76	9.38	5.88
THRUST ACCEL (fps ²)	0	0	3150.	1050.	9715.	10128.	5356.	5689.	5447	3055
THRUST (LBS)	99.52	--	99.5	133.3	152.9	4477.	4494.	5310.	5348	64.8.
V (FPS)	32.579	--	32.57	32.460	32.398	20,958		19.286	9.153	17,261
WEIGHT (LBS)	-1.62	--	6.0	4.47	4.407	24.0	24.59	26.6	44.2	81.29
PITCH ATTITUDE (DEG)	-19.85	-88.16	-1.11	-13.85	-13.74	-1.27	-1.25	-363	.338	0
σ_L (DEG)	-0.639	-1.97	-9.01	-9.68	-9.69	-5.03	-4.56	-1.52	-1.35	5.03
σ_{CSM} (DEG)	0	0	0	0	0	0	0	0	0	0
CSM VERT (FSP)	5287	5287	5287	5287	5287	5287	5287	5287	5287	5287
CSM VEL TOTAL (FPS)	79.1	79.43	79.43	79.43	79.43	79.43	79.43	79.43	79.43	79.43
CSM ALT. N.MI (FPS)	0	0	0	0	0	0	0	0	0	0
λ_L (DEG)										

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GRUMMAN AIRCRAFT ENGINEERING CORPORATION

Bethpage, L. I., N. Y.
CODE IDENT 26512

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SPECIFICATION NO. LSP-300-3B

TABLE VIII

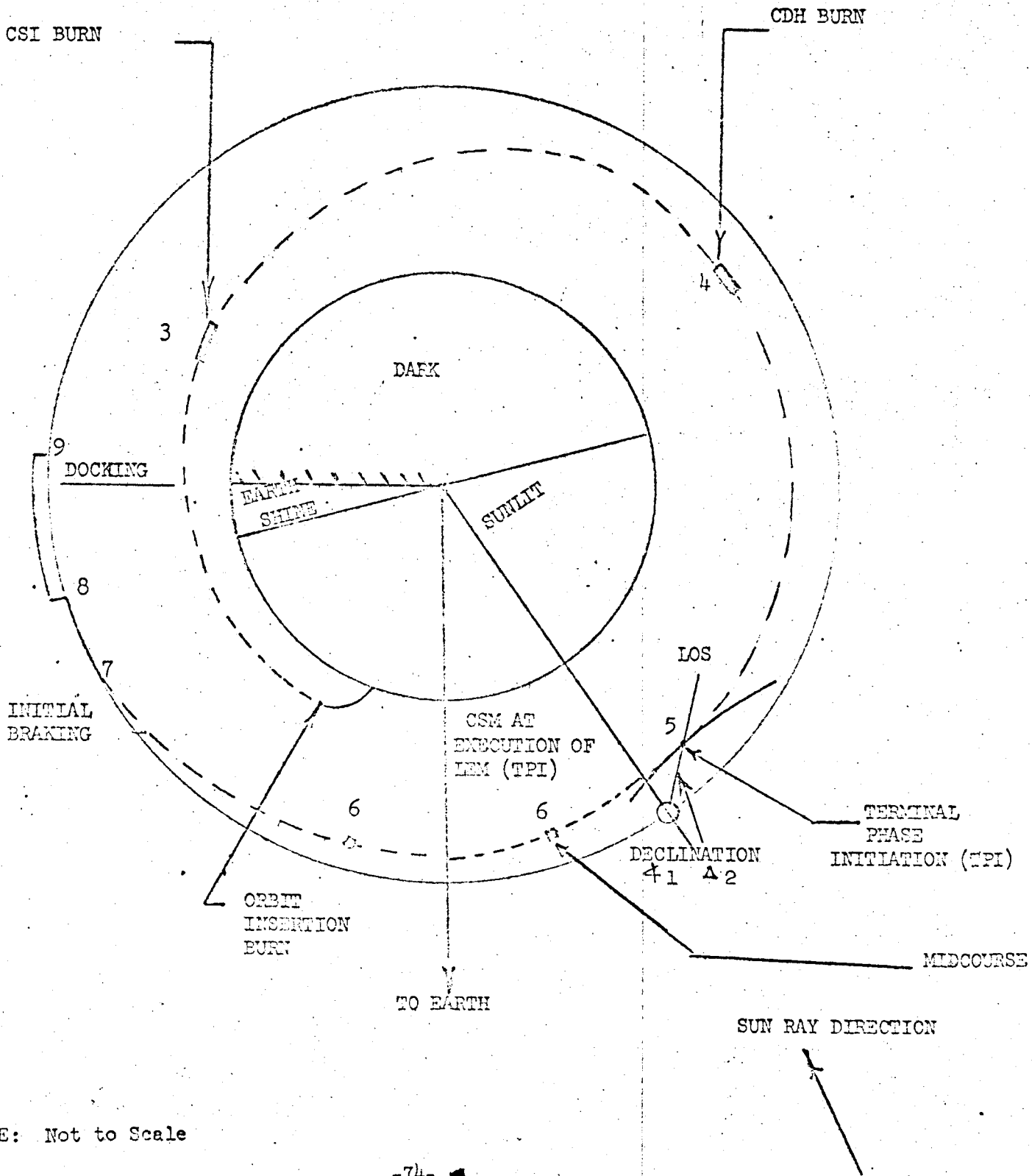
LM WEIGHT HISTORY

	(1)	(2)	(3)	(4)	(5)	(6)
1. Ascent stage inert weight (without ΔV fuel) lb. nominal	5476	5340	5340	5340	5246	5235
2. Ascent stage main engine fuel (usable) weight, lb. nominal	4771	4771	4771	0	4771	0
3. Ascent stage other expendables (RCS ΔV fuel) weight, lb. nominal	137	118	118	118	118	118
4. Inter stage weight, lb. nominal	zero	zero	NA	NA	NA	NA
5. Descent stage inert weight (without ΔV fuel) (lb. nominal)	4780	4771	NA	NA	NA	NA
6. Descent stage main engine fuel (usable) weight, lb. nominal	17415	zero	NA	NA	NA	NA
7. Descent stage other expendables, weight, lb. nominal	Neg.	Neg.	NA	NA	NA	NA
8. TOTAL WEIGHT	32579	15000	10229	5458	10135	5353

- NOTE: (a) Unit to unit weight variation of ascent stages shall be ± 300 lb. -600
- (b) Unit to unit weight variation of descent stage shall be ± 550 lb.
- (c) The uncertainty of the weight of fuel in the ascent and descent stage tanks at the beginning of the mission shall be 50 lb. (3 sigma).
- (1) Separation from CSM
- (2) Fuel depletion of descent stage
- (3) Immediately after staging (during descent phase abort)
- (4) Fuel depletion of ascent stage (during descent phase abort)
- (5) Ascent stage at nominal lift-off from lunar surface
- (6) Ascent stage at fuel depletion after lift-off from lunar surface

COELLIPTIC

FLIGHT PLAN



NOTE: Not to Scale

1/2" 10 X 10 TO THE CENTIMETER 46 1515
KUPPEL & ESSER CO. ALABAMA
KUPPEL & ESSER CO.

WATER BATH TEMPERATURE
WATER BATH TEMPERATURE
WATER BATH TEMPERATURE

(10)

-100

-80

-60

-40

-20

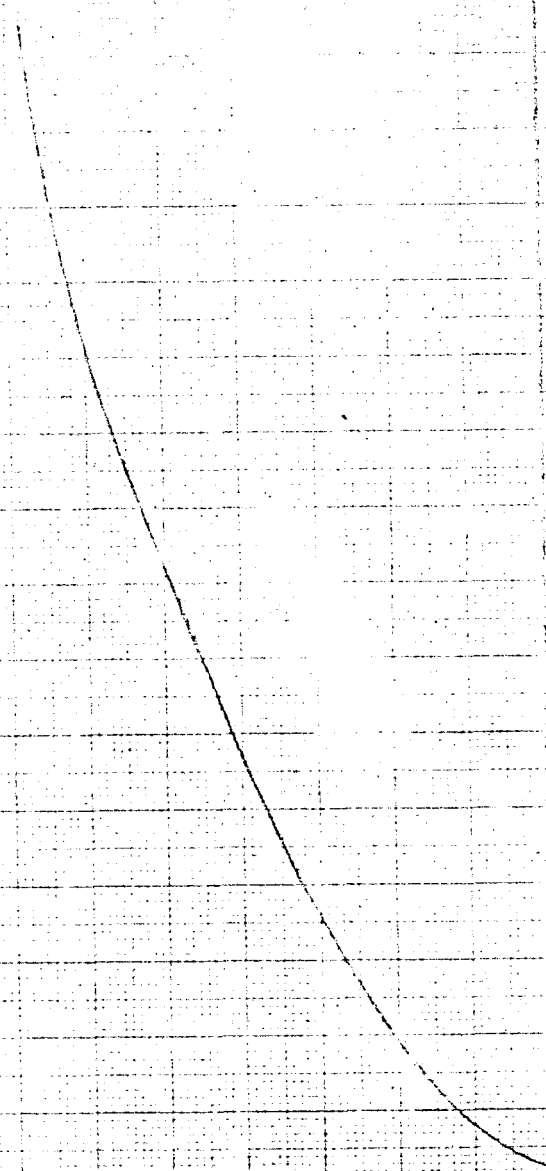
0

ALUMINUM BATH TEMPERATURE (10)

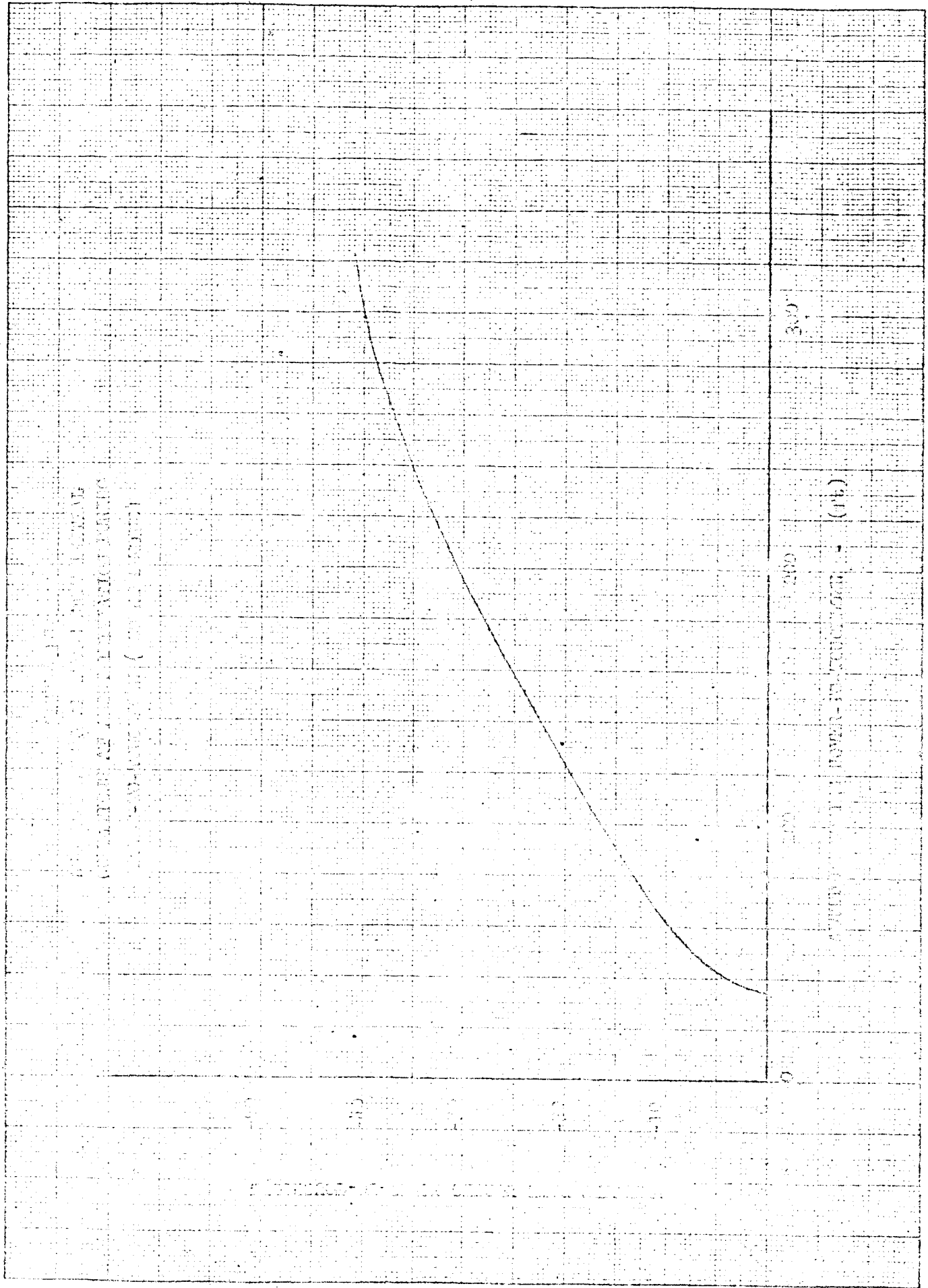
100

200

300



TYPE 10 X 10 TO THE CENTER LINE 46 1516
MAY 1964
MUTUAL WATER CO.



IM VEHICLE ACCOUNT OF IOWA - WEIGHT DURING ACCOUNT

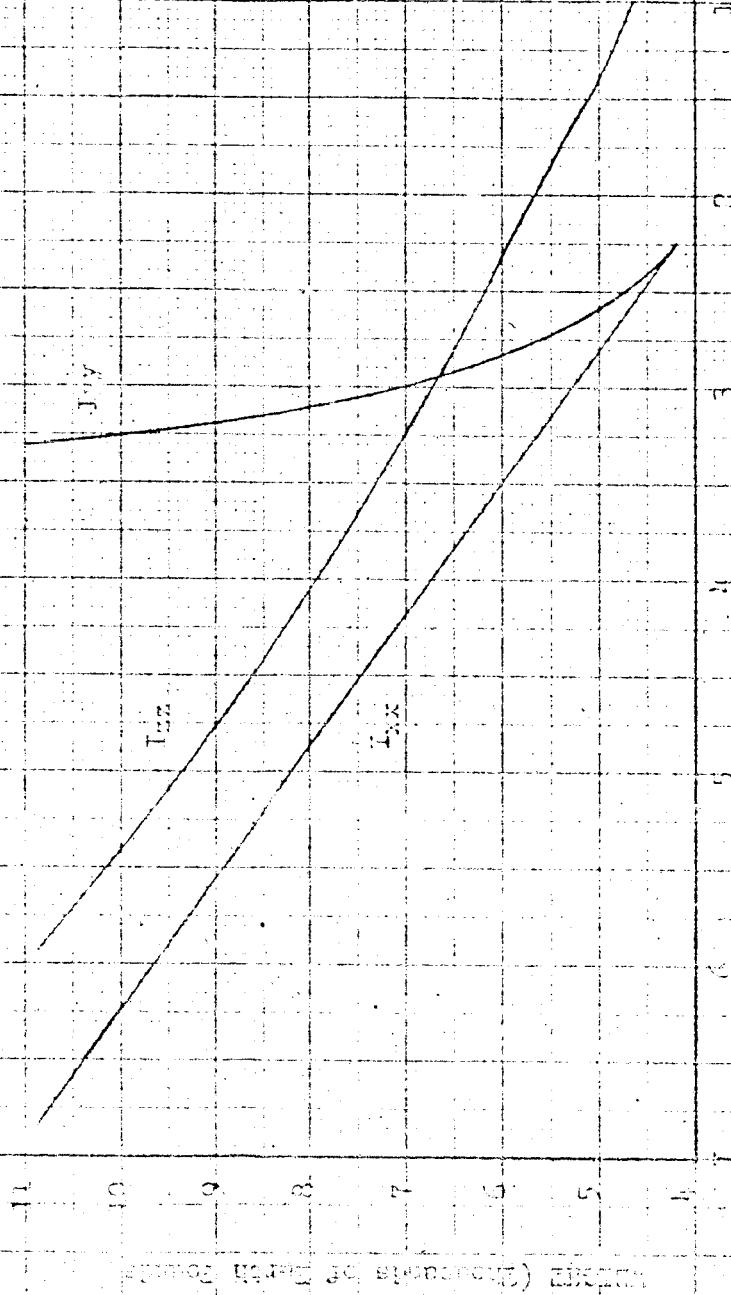


FIGURE 2A

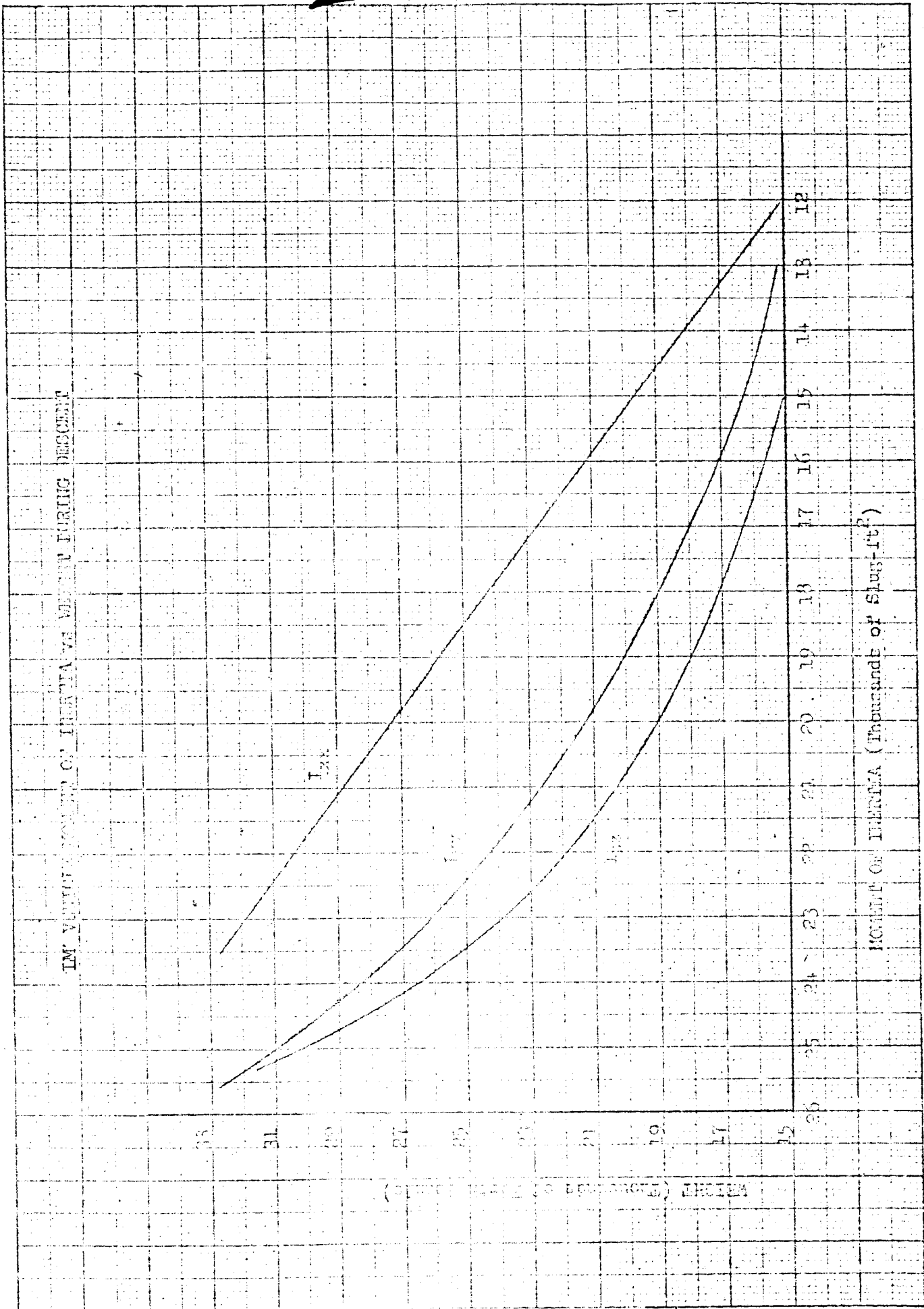


FIGURE 2B

46 1327

10 X 10-23-1038 H

KROGER & COMPANY

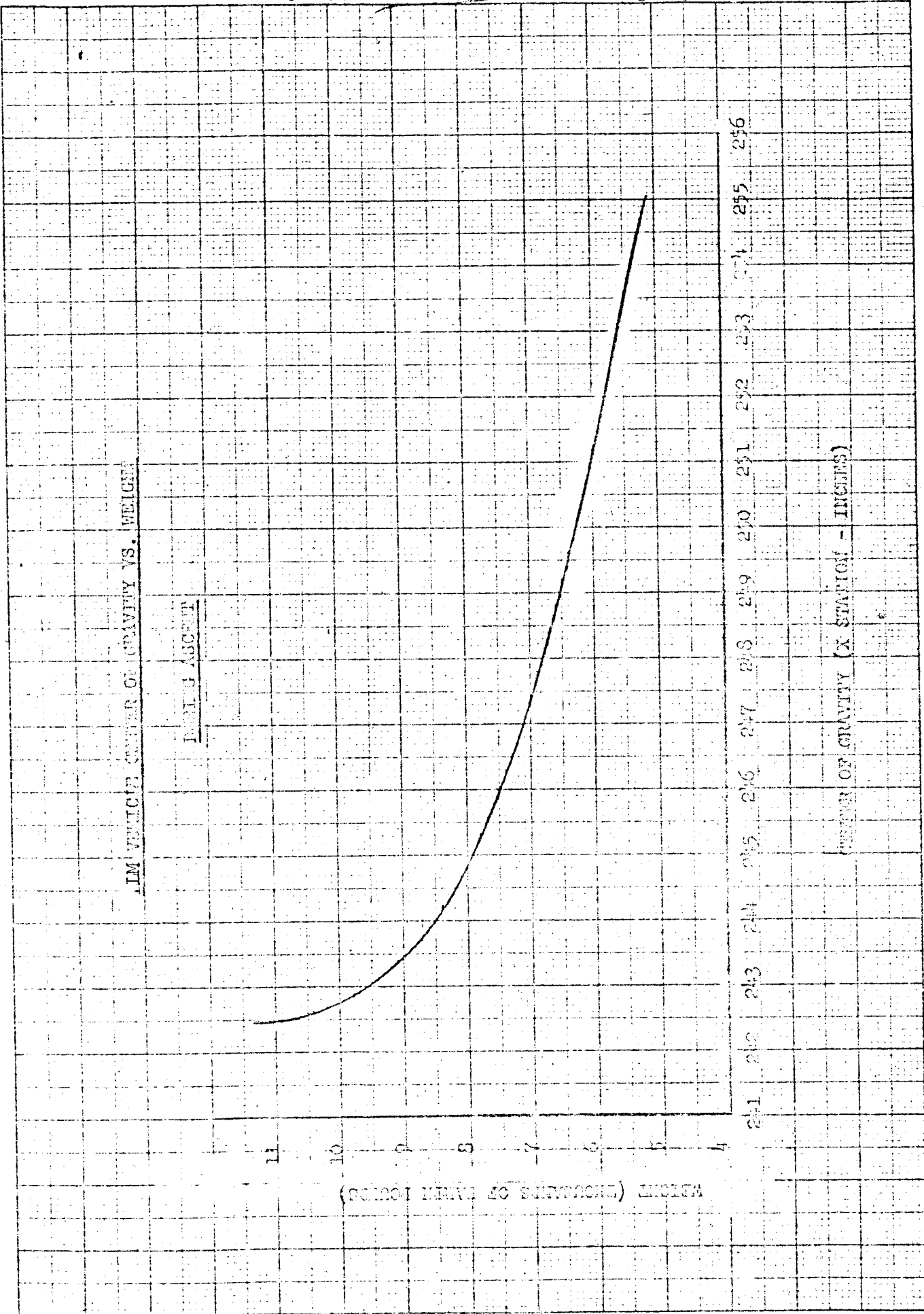


FIGURE 20

10 X 10 TO 1/8 INCH 46 1327
VEHICLE WEIGHT

LM VEHICLE CENTER OF GRAVITY vs WEIGHT DURING DESCENT

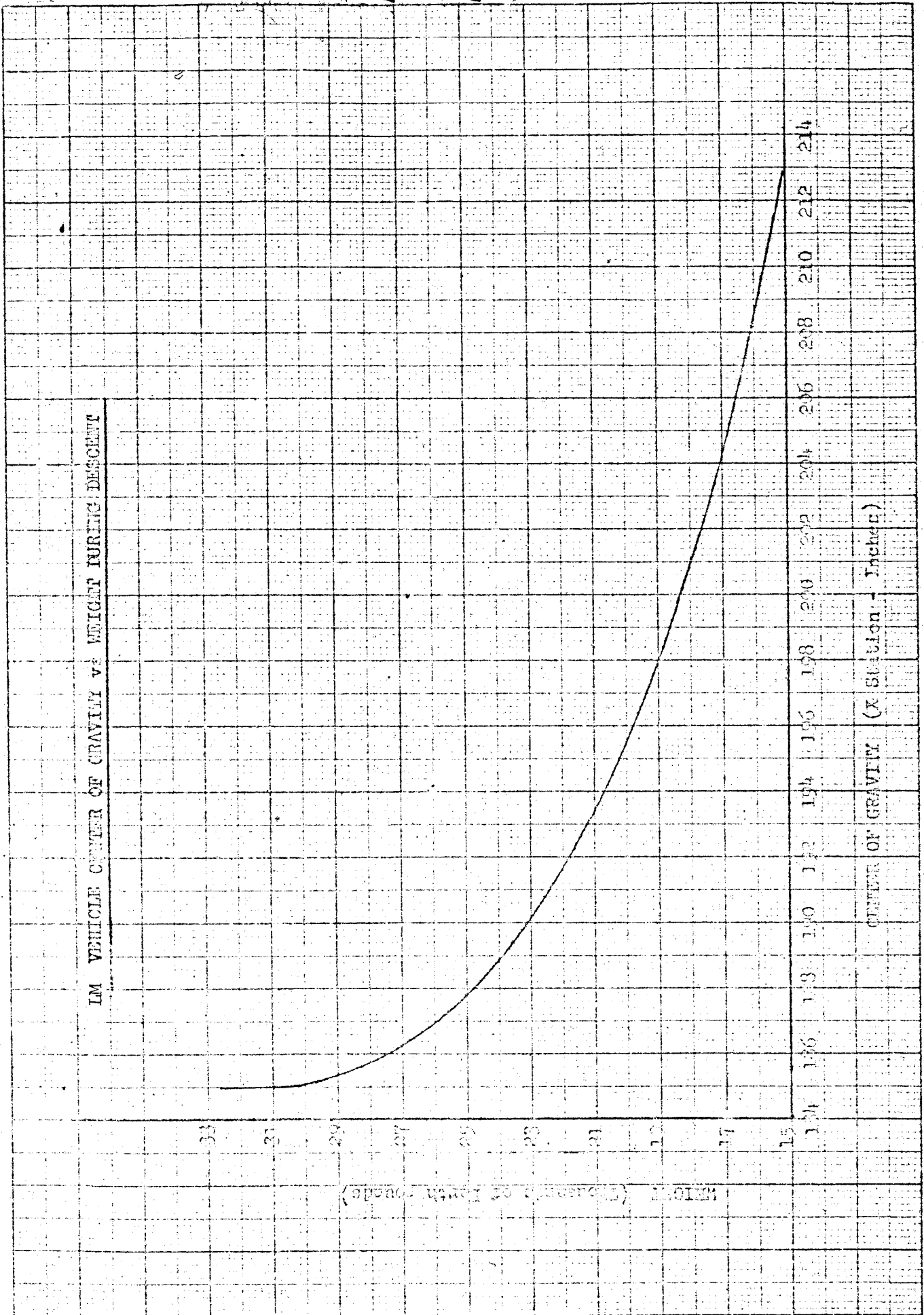


FIGURE 2D

SPECIFICATION NO. LSP-300-3B

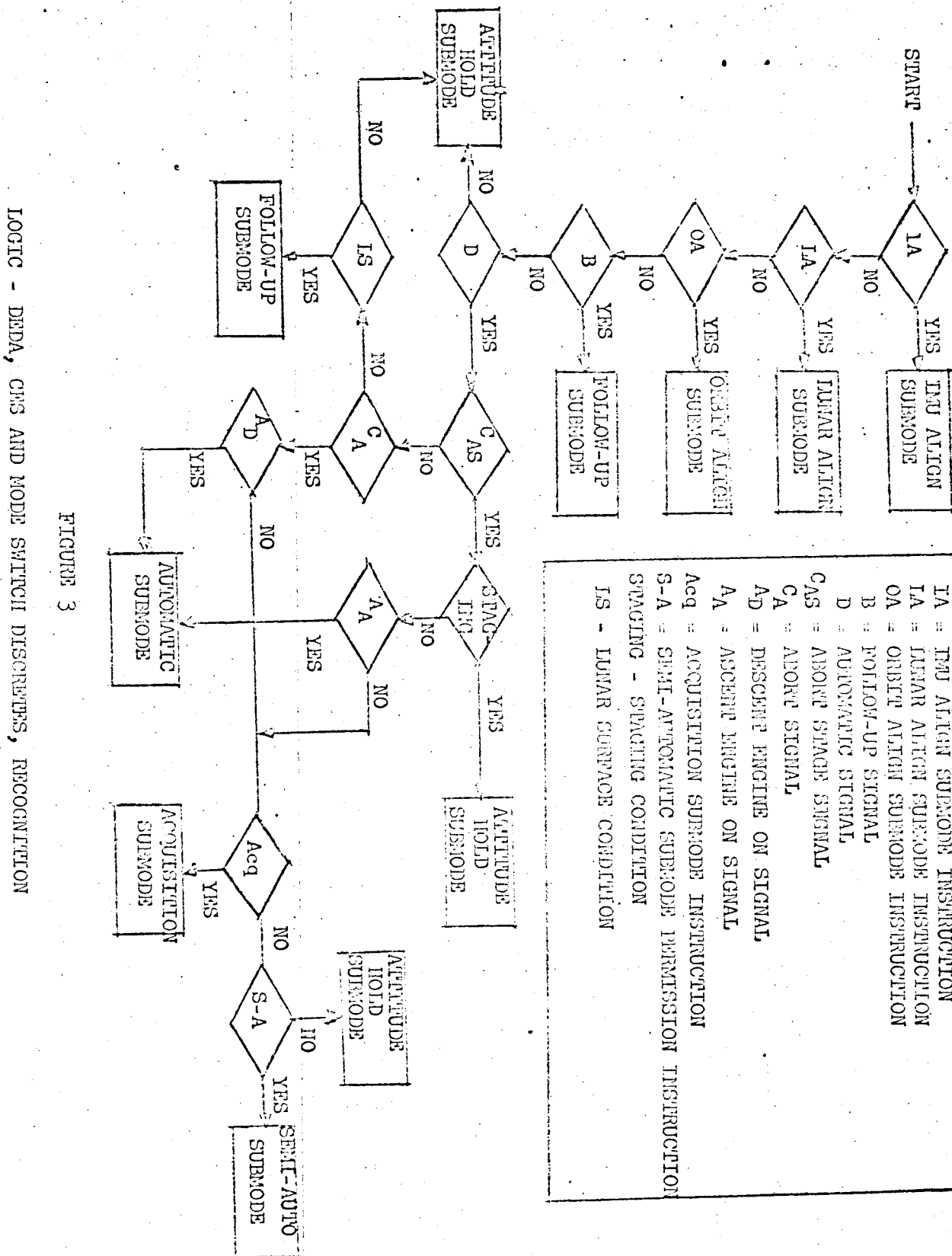


FIGURE 3

LOGIC - DEDA, CMS AND MODE SWITCH DISCREETERS, RECOGNITION

- IA = TRJ ALIGN SUBMODE INSTRUCTION
- IA-A = LUMAR ALIGN SUBMODE INSTRUCTION
- OA = ORBIT ALIGN SUBMODE INSTRUCTION
- B = FOLLOW-UP SIGNAL
- D = AUTOMATIC SIGNAL
- C-A = ABOUT STAGE SIGNAL
- CA = ABOVE SIGNAL
- AD = DISCREET ENGINE ON SIGNAL
- AA = ASCEND ENGINE ON SIGNAL
- Acq = ACQUISITION SUBMODE INSTRUCTION
- S-A = SEPT-AUTOMATIC SUBMODE PERMISSION INSTRUCTION
- SMAGING - SPAGING CONDITION
- IS - LUMAR COMPACT CONDITION