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PRELIMINARY
APOLLO GUIDANCE AND NAVIGATION SYSTEM
PERFORMANCE AND INTERFACE REQUIREMENTS
SPECIFICATION

Reissued
20 March 1963

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PRELIMINARY APOLLO GUIDANCE AND NAVIGATION
SYSTEM PERFORMANCE AND INTERFACE REQUIREMENTS SPECIFICATION

1. SCOPE

1.1 Scope. - This specification describes the functional performance, and interface requirements of the guidance and navigation (G&N) system for the Apollo spacecraft. It provides the general and specific criteria and constraints to which the design of the complete G&N system shall be made compatible in order to attain overall mission success and provide for the safe return of the astronauts under emergency abort and survival measures.

1.2. Specification Organization. - This specification is in three parts: (1) the overall environment under which the G&N system shall be designed to function and operate, (2) the G&N functional and performance requirements which the system shall meet in the different phases of the mission in order to attain the objectives, and (3) the G&N equipment interface requirements under which the design of the system equipment shall be matched physically, structurally, and operationally to other spacecraft subsystems equipment in order to function properly as an integral part of the spacecraft.

1.2.1 Overall Mission Environment. - The overall environment portion of this specification describes: (1) the basic objectives of the Apollo mission, (2) the constraints imposed on the mission, and (3) the specific trajectory and operating trajectory envelopes demanded by the mission constraints, the spacecraft, and the spacecraft internal subsystem characteristics. To provide a reference for G&N analysis, a base-point mission model is described in which a set of nominal reference trajectories for various mission phases have been defined. The model is established primarily to provide a common reference under which acceptance criteria of G&N system design and associated performance can be established.

1.2.1.1 Spacecraft Environments. - The spacecraft environments presented herein are those characteristics of the spacecraft and its subsystems which govern the design and operation of the G&N system. The dynamics and design of the G&N system shall have the bandwidth margin and system flexibility to meet all of the indicated subsystem constraints.

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1.2.2 Functional Requirements. - The functional requirements portion of this specification describes the G&N system capability to meet the mission objectives. The requirements indicated are based on the known design configuration of the Massachusetts Institute Of Technology (MIT) guidance system under development. This system consists of the primary functional units of the Inertial Measurement Unit (IMU), the micrologic computer, the sextant (SXT) and scanning telescope (SCT), and associated operating electronic units.

1.2.3 Equipment Interface Requirements. - The equipment interface requirements portion of this specification defines the physical and functional interfaces between the G&N system and other systems of the spacecraft.

1.3 Specification Coverage. - This document covers the requirements for the G&N system only. Performance and interface requirements for the back-up modes of operation, utilizing the stabilization control system (SCS), are not included.

2. APPLICABLE DOCUMENTS

2.1 Applicability. - The following documents of the issue in effect on the date of contract form a part of this specification to the extent specified herein.

SPECIFICATIONS

Military

MIL-E 9500

Technical Uninhibited Ethylene Glycol

PUBLICATIONS

Air Force

ARDC Model Atmosphere 1959

OTHER DOCUMENTS

North American Aviation, Inc., Space & Information Systems Division

Interface Control Document (ICD). - G&N Interface Control Documents define design details and specific contractor responsibilities of the G&N interface. After the principal contractor and the G&N associate contractor have jointly identified all G&N interface areas, the principal contractor will be responsible for preparing the necessary ICDs. The resultant ICDs, when jointly approved by the NASA, the principal contractor, and the G&N associate contractor will be listed in Appendix A of this specification.

2.2 Precedence. - The order of precedence in case of conflict shall be as follows:

- (a) The contract
- (b) This specification
- (c) Documents referenced herein.

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3. REQUIREMENTS

3.1 Overall Mission Environment. - The following general criteria and systems characteristics describe the overall Apollo mission environment within which the G&N system shall function. Mission and spacecraft characteristics are identified in a basepoint mission model which is designed to be the focal point around which G&N mission analyses will revolve. The basepoint mission model is described in terms of its associated trajectory environment. In addition, major spacecraft systems features are specified which potentially affect the trajectory environment and the precision of various G&N functions. This information is generally expanded to consider trajectory and spacecraft variances contributing to variabilities in mission environment within a given flight, and from flight to flight. It is not intended to describe all possible regions and motion in space attainable by the spacecraft, but to provide the general philosophies and criteria which illustrate the major mission variables. Thus, the mission environment data do not constitute design requirements, but are intended to provide a summary of the basic information required to analyze detail G&N problems of specific pertinence both to individual flight phases, and to complete missions.

3.1.1 Mission Objectives and Requirements. - The flight objectives for the mission are: (1) to launch the spacecraft from a specified site, (2) to establish a translunar trajectory having specified perilune characteristics, (3) to establish a transearth trajectory having specified perigee characteristics, and (4) to return the spacecraft safely to a selected landing site along a specified geographic trace. These objectives must be met within an overall mission success probability of 0.900. Further, in the event of emergencies the crew must be returned to earth with a crew safety probability of 0.999. The spacecraft shall perform the above flight objectives at all lunar declinations and solar orientations within the conceptual constraints of the mission technique. Within this general operational pattern, mission flight dates will generally depend upon preferred illumination characteristics. However, attention will be given to retrieval considerations associated with nominal earth return phases and potential aborts during earth launch and boost. During all mission phases the spacecraft shall have abort capability to return and land the crew within emergency environmental limits. The time from abort to landing shall **not** exceed the emergency mission time from lunar launch to earth landing, thus establishing a maximum limit for design requirements for non-nominal secondary modes of operation. Within this broad framework of mission features, trajectory profiles and related spacecraft characteristics will be used to describe the Apollo operational environment for the G&N system.

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3.1.2 Trajectory Environment. - The trajectory environment in which the G&N system shall function is illustrated with a detailed basepoint lunar landing mission, followed by general discussions of trajectory parameter variations for each of the mission phases. The basepoint mission model is presented for the purpose of providing a complete description of trajectory and systems characteristics upon which G&N mission analyses relating events and influences between flight phases may be based.

3.1.2.1 General Mission Trajectory Criteria. - The basepoint lunar landing mission described below is based upon current assumptions which are subject to revision and will be updated as detail information becomes available. The trajectory has been selected at a time of month when the moon is at a mean distance from the earth and the translunar plane inclination to the moon orbital plane is approximately mid-way between allowable limits. The intention being to represent a "median" environment for G&N operations. The prevailing lighting conditions are daylight at launch (shortly before dusk), daylight at translunar injection, dawn at lunar landing, darkness at transearth injection, and daylight at entry and recovery at Woomera, Australia. The mission has been constructed from generalized trajectory data and maintains continuity in time from launch to final recovery. An exact general mission trajectory (GMT) at launch, however, is not specified. A restricted 3-body model consisting of the earth, the moon, and a massless vehicle has been used to generate the typical mission. Impulsive maneuvers have been assumed except for the powered flight phases at launch and translunar injection. The typical mission will eventually be replaced by precision reference trajectories in a true ephemeris model.

3.1.2.2 Basepoint Mission Characteristics. - A typical lunar mission trajectory has been obtained by utilizing the results of generalized trajectory data which are based upon a simplified model. A launch date has been chosen, such that the true earth-moon distance is compatible with the distance assumed in the simplified model. A number of operational methods and criteria had to be assumed in order to generate the trajectory. The methods and ground rules are still under investigation and are subject to change. The basepoint mission from launch to recovery is presented in the paragraphs that follow.

3.1.2.2.1 Launch and Boost. - Launching of the basepoint model occurs on 19 January 1967 from Cape Canaveral with a launch azimuth of 70 degrees. The spacecraft is boosted into a 100 nautical mile parking orbit which is inclined 34.3 degrees to the earth's equatorial plane. The boost phase requires 706 seconds of powered flight during which the spacecraft travels

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through a central earth angle of 23.5 degrees. A parking orbit coast of 57.2 minutes (227.1 degrees) is required prior to initiating the translunar injection phase.

3.1.2.2.1.1 Launch and Boost Configuration. - The earth launch configuration, shown in Figure 1, is comprised of the following system elements: the Saturn C-5 launch vehicle which includes S-IC, S-II, and S-IVB stages, the lunar excursion module (LEM), the service module (SM), the command module (CM), and the launch escape system (LES). Vehicle gross weight, moments of inertia, and center of gravity (c. g.) at stage separation points during boost are listed in the following table. Time histories of weight, c. g., and moments of inertia during boost into the parking orbit are shown in Figures 2 through 7.

Table I. Boost Configuration Dimensional Data

Event	Weight/lb	cg/in. from STA		I_{pitch} , Slug ft ² x 10 ⁻⁶	I_{roll} Slug ft ²
Lift Off	6,000,000	1,077	100	627.0	2,350,706
S-IC Cutoff	1,769,625	1,703	100	305.0	2,350,706
S-II Ignition and LES Jettison	1,368,980	527	1,664	88.5	484,604
S-II Cutoff	448,730	1,129	1,664	28.5	484,604
S-IVB Ignition	360,930	390	2,643	10.2	91,135

3.1.2.2.1.2 Parking Orbit Configuration. - During the earth parking orbit the vehicle configuration is comprised of the partially expended S-IVB, the LEM, the SM, and the CM. Dimensional data for this configuration are listed in the following table.

Table II. Parking Orbit Configuration Dimensional Data

Event	Weight/lb	cg/in. from STA		I_{pitch} Slug ft ² x 10 ⁻⁶	I_{roll} Slug ft ²
Parking Orbit	282,427	442	2,643	9.5	91,135

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3.1.2.2.2 Translunar Injection. - Translunar injection begins over the Pacific Ocean and requires 312 seconds of powered flight during which the spacecraft travels through a central angle of 23.5 degrees. Translunar injection occurs at a velocity of 35,826.2 fps at a latitude of 15.4 degrees S and a longitude of 175.6 degrees W. At this time the trajectory plane inclination relative to the plane of the lunar orbit is 35 degrees.

3.1.2.2.2.1 Translunar Injection Configuration. - The translunar injection configuration begins with the earth parking orbit configuration, Table II, and extends to the nearly expended S-IVB upon which the LEM, SM, and CM are staged. Dimensional data at S-IVB cut-off terminating the translunar injection maneuver are listed in the following table. Time histories of these dimensional data during the translunar injection maneuver are shown in Figures 6 and 7.

Table III. Translunar Injection Configuration Dimensional Data

Event	Weight (lb)	cg (in.) from STA		^I pitch, Slug ft ² x10 ⁻⁶	^I roll Slug ft ²
S-IVB Relight	277,427	390	2,643	9.5	91,135
S-IVB Cutoff	130,300	754	2,643	3.35	91,135

3.1.2.2.3 Translunar Midcourse and Lunar Orbit Injection. - The spacecraft reaches perilune after 70.15 hours of coasting flight and injects into a 100 nautical mile circular orbit without making a plane change. Orbit injection occurs near the extended earth-lunar line of centers and the injection maneuver will not be visible from the earth. The resultant orbit, at the time of orbit injection, has an inclination of 174.8 degrees relative to the lunar orbital plane and 171 degrees relative to the lunar equator. The entire spacecraft makes one pass over the intended landing area 78.5 minutes after orbit injection. The lunar landing area is assumed to be located at 44.1 degrees W selenographic longitude (leading edge) and 6.3 degrees N selenographic latitude.

3.1.2.2.3.1 Translunar Midcourse Configuration. - The translunar configuration begins with the injection configuration, Table III, but subsequently involves two different arrangements of the basic spacecraft modules. During the LEM transposition docking maneuver the LEM remains attached to the S-IVB and the spacecraft configuration is reduced to the fully tanked SM and the CM. After LEM transposition the spacecraft configuration is comprised

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of the SM, CM, and LEM, with the LEM attached to the forward end (apex) of the CM.

3.1.2.2.3.2 Lunar Orbit Injection Configuration. - The lunar orbit injection configuration is initially the same as the **translunar midcourse** configuration less midcourse expendables.

3.1.2.2.4 LEM Operations. - The LEM injects into an elliptic orbit with a period equal to the circular orbit period (126.25 min) 171.4 minutes after lunar orbit injection. The LEM makes one pass over the vicinity of the intended landing area at an altitude of approximately 50,000 feet. The landing maneuver is initiated in the vicinity of perilune during the second revolution. Landing occurs approximately 2.5 hours after leaving the spacecraft. The LEM stays on the lunar surface for approximately 29 hours. The LEM effects rendezvous with the spacecraft 33.1 hours after initial separation.

3.1.2.2.5 Transearch Injection. - The transearch injection phase begins 37.9 hours after lunar orbit injection during which the spacecraft has made approximately 18 revolutions. The injection is such that the CM returns to Woomera along a 34-degree inclined plane to the earth's equatorial plane. Entry occurs 72.6 hours after transearch injection with an entry angle of -6.4 degrees. At this time the vehicle is at an altitude of 400,000 feet at 92 degrees E longitude and 11.5 degrees S latitude. An entry range of 2770 nautical miles is required to reach the landing site. The mission duration from launch to entry has been 180.6 hours. During this time the spacecraft has spent 70.15 hours in translunar coast, 37.9 hours in the lunar vicinity, and 72.6 hours in transearch coast.

3.1.2.2.5.1 Transearch Injection Configuration. - The transearch injection configuration shown in Figure 8 is comprised of the partially expended SM and the CM, with variations in dimensional characteristics due to expendable materials consumed during the transearch injection maneuver.

3.1.2.2.5.2 Transearch Midcourse Configuration. - Two basic configurations exist during the transearch midcourse phase which include the nearly expended SM with the CM, and the CM alone just prior to entry.

3.1.2.2.6 Entry and Recovery. - Following separation of the SM prior to entry, the CM is assumed to be properly oriented for that portion of the flight which will take place within the atmosphere. The entry flight phase is assumed to occur upon reaching an altitude of 400,000 feet with an entry

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flight-path angle of -6.4 degrees at zero degrees roll attitude ($+C_{Lmax}$). A maximum load factor of approximately 5.3 g's will be encountered just prior to pullout. Following the initial pullout which occurs at approximately $200,000$ feet, the vehicle path will be controlled to achieve the desired range of 2770 nautical miles at touchdown. For the mission described, touchdown occurs at Woomera which is located at 31.0 degrees S latitude and 137.4 degrees E longitude. The elapsed time from entry at $400,000$ feet to the recovery altitude of $25,000$ feet is 855 seconds during which the vehicle traverses a total range of 2770 nautical miles. A drogue chute is deployed at $25,000$ feet and, 35 seconds later, at an altitude of $15,000$ feet, three main chutes deploy. Land impact (310 feet altitude) occurs approximately 385 seconds after main chute deployment at an impact velocity of 24.9 feet per second (fps). In the event of impact at sea level it would occur at approximately 575 seconds after main chute deployment at an impact velocity of 23.1 fps. The total elapsed time for the sequences described is approximately 1275 seconds (21.25 min) from entry to touchdown.

3.1.2.2.6.1 Entry Configuration. - The entry configuration is comprised of the CM alone, but the dimensional characteristics change during entry due to consumable materials expended and heat shield ablation.

3.1.2.2.6.2 Recovery and Landing Configuration. - To be described at a later date.

3.1.2.2.7 Basepoint Mission Summary. - The basepoint mission discussed above is summarized by mission phases in Table IV. The summary is in the form of a time line which presents the time for each phase and the cumulative time since launch. The spacecraft configuration during each phase is also indicated. It should be noted that during the LEM maneuver phases the CM and SM remain in the lunar parking orbit.

3.1.2.3 Detailed Flight Phases. - The detailed flight phases discussed below are related to the basepoint mission model, apply to actual flights, and form a frame of reference for the G&N functional requirements portion of this specification.

3.1.2.3.1 Boost. - Launch of actual spacecraft will occur at Cape Canaveral, Florida, latitude 28.5 degrees N, longitude 279.5 degrees E. The launch azimuth will be within range safety limits, about 70 to 115 degrees, and will vary with launch delay. Two launch windows are available each day. The minimum launch delay period will be two hours. The maximum launch delay will vary with the time of month, but cannot exceed 5.5 hours for the



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Table IV. Base Point Mission Summary

Configuration	Phase	Phase Time	Total Mission Time to End of Phase (hours)
CM + SM + LEM	Boost to Orbit	706.0 sec	0.20
	Earth Parking Orbit Coast	57.2 min	1.15
	Translunar Injection	312.0 sec	1.24
	Translunar Coast	70.06 hrs	71.30
	Lunar Orbit Injection	320.0 sec	71.39
	Lunar Parking Orbit Coast	171.4 min	74.25
LEM	EPO Injection	35.3 sec	74.26
	EPO Coast	148.07 min	77.03
	Perilune retro (50K to 1K)	336.0 sec	77.12
	Hover, translation, and touchdown (1K to surface)	127.0 sec	77.16
	Surface stay time	29.23 hrs	106.40
	Ascent to transfer ellipse	356.25 sec	106.50
	Ascent ellipse coast	52.96 min	107.38
	Parking orbit injection and rendezvous	5.26 sec	107.38
	Lunar Parking Orbit Coast	108.78 min	109.19
	Transearth Injection	127.5 sec	109.24
CM + SM	Transearth Coast	72.61 hrs	180.62
	Entry	21.25 min	180.97

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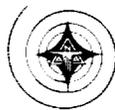
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specified azimuth limits. The launch window need not be centered about the 90-degree azimuth (due east) launch. The time of day at which launch occurs depends on the time of the month, lunar declination, launch azimuth, and the launch window utilized. Mission constraints, such as specified lighting conditions for certain mission phases, may make one of the daily launch windows strongly preferable, and perhaps eliminate one launch window from consideration.

3.1.2.3.1.1 Boost Vehicle. - The Saturn C-5 vehicle will be used to boost the Apollo spacecraft into the parking orbit and perform translunar injection on lunar missions. The C-5 vehicle consists of an S-IC stage, an S-II stage, an S-IVB stage, and a launch vehicle guidance unit. The Apollo spacecraft for the landing mission will consist of a LES, CM, SM and LEM.

3.1.2.3.1.2 Boost Trajectory. - The C-5 boost trajectory utilizes a gravity turn type ascent through the earth's atmosphere followed by an optimum steering program during the exo-atmospheric portion of boost to orbit. In general, the change-over from a gravity-turn trajectory to optimum steering occurs after S-II (second stage) ignition and LES tower jettison. The boost trajectory to earth parking orbit requires 2-1/2 stages to maximize the translunar injected payload. Therefore, each of the three stages will be ignited in the nominal boost trajectory to obtain the nominal 100 nautical mile parking orbit altitude. The following boost trajectory criteria were used for the basepoint mission profile:

- (a) Spherical earth with the center at the geometric center
- (b) An eastward launch from Atlantic Missile Range (AMR)
- (c) Patrick Air Force Base atmosphere with no wind or guidance disturbances
- (d) An earth-fixed coordinate system (i. e., the velocity presented must be increased by the earth's rotation to obtain inertial velocities)
- (e) A liftoff thrust to weight ratio equal to 1.25
- (f) Nominal specific impulse in all stages
- (g) S-IC five-engine cutoff sequence of 1-4 with the outboard engines burning four seconds after the inboard cutoff

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- (h) Coast of 1-1/2 seconds between S-IC burnout and S-II ignition
- (i) Jettison of LES weighing 6600 pounds 20 seconds after S-II ignition
- (j) Eleven-second vertical boost followed by a gravity turn initiated with a pitch-over angle of 89.78 degrees from the horizontal
- (k) A steering program to maximize weight in the 100 nautical mile orbit after LES jettison
- (l) Zero-nozzle cant angle and instantaneous thrust buildup and decay
- (m) Coast of 2-1/2 seconds between S-II burnout and S-IVB ignition.

Figures 9, 10, and 11 present flight parameters vs time and include the points of configuration change for this representative trajectory. There are several factors that cause the boost trajectory to vary from the normal. These variations are: winds, thrust deviations, off-normal mixture ratio settings, propellant loading inaccuracies, drag deviations, and atmospheric deviations.

3.1.2.3.2 Earth Parking Orbit. - The earth parking orbit for future lunar missions will have an altitude of 100 nautical miles and a corresponding orbital period of about 90 minutes. The inclination of the parking orbit plane to the earth equator plane is a function of launch azimuth at Cape Canaveral and can vary from 28.5 to about 37.2 degrees (for launch azimuths from 70 to 115 degrees). Trajectory considerations which affect the extent of the parking orbit coast are:

- (a) Lunar declination
- (b) Launch azimuth
- (c) Launch window selection.

Earth parking orbits of one revolution or less are sufficient to satisfy the constraints imposed by these trajectory considerations. Other factors, however, may also affect the parking orbit coast duration. Systems check-out and orbit determination will require some minimum amount of time. Multiple parking orbits may be used to advantage to reduce the required

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Ground Operational Support System (GOSS) coverage area. Orbit decay properties may also become significant for extended parking orbit durations.

3.1.2.3.3 Translunar Injection. - At translunar injection a match will exist between the required injection conditions and the powered flight conditions at cut-off. At an injection radius of 3565 nautical miles (altitude = 122 nautical miles) and a flight path angle of three degrees, the injection velocity for free-return circumlunar trajectories can vary between 35,796 and 35,875 fps in inertial frame coordinates depending on the time of month and the desired lunar approach trajectory plane.

3.1.2.3.3.1 Lunar Approach Trajectory Plane. - Since the circumlunar injection velocity is the primary variable governing the orientation of the lunar approach trajectory plane, its magnitude may be established on a particular launch date by the landing site location if overfly of the landing site is chosen as the operational requirement. The injection velocity for non-free-return trajectories can vary from a minimum of about 35,747 fps to a maximum established by the booster and spacecraft capability. The injection point latitude and longitude depend upon:

- (a) Time of the month
- (b) Launch window selection
- (c) Launch azimuth
- (d) Number of parking orbit revolutions.

The injection point latitude is between ± 37.2 degrees for launch azimuth limits of 70 to 115 degrees. Long parking orbit coasts result in translunar injections occurring over the general area of the Pacific Ocean, and short parking orbit coasts result in injections occurring over the Atlantic Ocean and Africa. The use of multiple parking orbit revolutions shifts the injection points westward about 22 degrees for each additional revolution. The loci of translunar injection points are established by compromise between the requirements for launch flexibility, system checkout time (parking orbit revolutions), desirable lighting conditions, and GOSS station coverage capabilities.

3.1.2.3.3.2 Translunar Injection Maneuver. - The third stage of the C-5 launch vehicle (S-IVB) performs the translunar injection maneuver from the earth parking orbit. The basepoint trajectory shown for translunar injection

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utilizes a gravity turn maneuver. This trajectory is extended to include conditions for the maximum energy requirements for free-return circumlunar trajectories. Lower energy requirements can be satisfied by earlier termination of the propulsion maneuver. Regardless of the type of maneuver selected, constant attitude, gravity turn, etc., the burning time, range angle, altitude, and flight-path angle for translunar injection are approximately the same. Figures 12, 13, and 14 present the basepoint mission flight parameters for this maneuver. The trajectory shown is co-planar. The out-of-plane translunar injection requirements to accommodate a reasonable launch window time capability are extremely small (approximately 1 degree) and can be easily accommodated within the C-5 vehicle launch window propellant allowance of 200 fps.

3.1.2.3.4 Translunar Coast. - The transit time from translunar injection to arrival at a perilune altitude of 100 nautical miles can vary from about 59 to 77 hours for circumlunar trajectories having "direct" returns (includes inclinations up to 90 degrees with respect to the lunar orbit plane). Non-circumlunar trajectories will have a minimum transit time which will depend on the booster capability. The maximum transit time is about 120 hours. The inclination of the translunar plane, defined at injection, to the earth equator plane will be between 28.5 and 37.2 degrees for launch-azimuth limits of 70 to 115 degrees. The corresponding inclinations of the translunar plane to the moon orbit plane range from zero to about 65.8 degrees.

3.1.2.3.5 Lunar Orbit Injection. - The orientation of the lunar approach trajectory plane with respect to the lunar orbit plane on a given day of the month is a function of the circumlunar trajectory injection velocity. The inclination to the moon orbit plane will not exceed 13 degrees and the resulting inclination to the lunar equator plane will not exceed 20 degrees. For some landing sites near the lunar equator, the translunar injection velocity can be selected so that the lunar approach trajectory plane contains the landing site and no plane change is required at lunar orbit injection. The ΔV required for lunar orbit injection at an altitude of 100 nautical miles with no plane change varies between 2970 to 3250 fps. Lunar orbit injection occurs over the non-visible portion of the moon within approximately 13 degrees of lunar central angle from the extended earth-lunar line. The actual powered-flight phase is initiated just prior to perilune of the incoming trajectory. The SM injects the CM and LEM modules into lunar orbit from the hyperbolic approach path. A representative co-planar injection maneuver is presented for a typical 80 nautical mile orbit altitude using a constant inertial attitude. This maneuver mode has been selected because of its

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simplicity and its negligible effects on injected payload. Figures 15 and 16 show the flight parameters and steering functions for this maneuver. It is noted that part of the powered trajectory descends below the desired circular orbit altitude. Orbital velocity conditions are attained approximately 120 nautical miles beyond the normal perilune point of the free-return trajectory.

3.1.2.3.6 Lunar Orbit Operations

3.1.2.3.6.1 Parking Orbit. - The ephemeris of the lunar parking orbit must be established prior to the initiation of the landing maneuver so that the maneuver can be properly executed. The number of revolutions required to establish the ephemeris and perform any other prelanding functions affect the required initial orientation of the orbit. The orbit must be favorably oriented with respect to the landing site at the time the landing maneuver is to be initiated. It is not anticipated that there will be any special plane change maneuvers for the spacecraft once the orbit has been established at orbit injection.

3.1.2.3.6.1.1 Parking Orbit Orientation. - The orientation of the parking orbit must be such that the plane change maneuver requirements during descent and ascent of the LEM are not excessive. Therefore, the required parking orbit inclination and nodal position at the time the landing maneuver is initiated are determined by the location of the landing site and the anticipated stay-time on the lunar surface. Any variation in the orientation requires either a variation in the number of orbit revolutions or in the plane change requirement for the LEM. The LEM will be capable of returning to the parking orbit at any time.

3.1.2.3.6.2 LEM Descent. - The landing maneuver is initiated by injecting the LEM into an elliptical orbit having a nominal perilune altitude of approximately 50,000 feet above the intended landing area and a period equal to that of the initial circular parking orbit. The injection into the elliptic orbit could include a plane change maneuver so that the LEM would pass over the intended landing area. The characteristic velocity requirement (ΔV) is a function of the characteristics of the initial circular orbit and the plane change requirement. The variation in the ΔV requirement is a function of the ability to accurately establish the initial circular parking orbit and the parking orbit altitude. The number of revolutions will be dictated by the ability of the spacecraft crew to properly identify the landing area, but the number cannot vary appreciably since the nodal shift due to lunar.

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motion about the earth may displace the landing area outside the maneuver capability of the LEM.

3.1.2.3.6.2.1 LEM Elliptic Orbit. - To establish the elliptic orbit, thrust is applied in the orbit plane perpendicular to the flight path. The required burn time is 28.44 seconds; precise motor cutoff is critical since each additional second of burn time results in a decrease of 16,000 feet in perilune altitude. Figures 17 and 18 illustrate the trajectory required for this maneuver.

3.1.2.3.6.2.2 LEM Landing. - The terminal landing maneuver is initiated in the vicinity of perilune. The landing maneuver includes a hover or translation of up to 2.2 minutes or both with the final touchdown occurring between 5.9 and 8.5 minutes after initiation of the terminal landing maneuver. For the trajectory presented, the maneuver is assumed to be co-planar with maximum angular acceleration and pitch rates of $10^{\circ}/\text{sec}^2$ and $10^{\circ}/\text{sec}$, respectively. The retro maneuver terminates at 1000 feet with a vertical descent velocity of 10 ft/sec. This phase of the landing trajectory is shown in figures 19 and 20.

3.1.2.3.6.2.2.1 LEM Landing Trajectory. - To provide a complete trajectory to the lunar surface, a constant descent rate and translational capability is assumed. The LEM is capable of translating 1000 feet horizontally during descent and the total ΔV required from an altitude of 1000 feet will not exceed 700 fps. Figure 21 illustrates the maneuver described.

3.1.2.3.6.3 LEM Ascent. - The LEM ascent is performed by first injecting into a 50,000-foot circular orbit. A Hohmann transfer is then performed to the circular orbit of the CM and SM and rendezvous is effected. The time of launch, number of orbit revolutions prior to the Hohmann transfer, and initiation of the Hohmann transfer must be preplanned so that the required relative positions of the vehicles for rendezvous are established. Any variation in the parameters above will require a non-Hohmann orbit transfer maneuver. The injection into the orbit of the CM and SM could include a plane change maneuver.

3.1.2.3.6.3.1 LEM Staging. - To maximize the LEM performance capability and provide the maximum spacecraft reliability, it is assumed that the LEM will be staged at the lunar surface. Therefore, the LEM landing propellant tanks, landing legs, and used equipment or payload will be left on the lunar surface prior to ascent and rendezvous with the Apollo spacecraft. To determine a typical co-planar launch trajectory, a T/W ratio of 0.4 is used in addition to the limiting pitch acceleration and rate previously stated. The

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trajectory assumes continuous thrust to a 50,000-foot lunar parking orbit. Figures 22 and 23 show this phase of the mission.

3.1.2.3.6.3.2 LEM Orbit Transfer and Rendezvous. - At the correct spatial position in its 50,000-foot circular orbit, the LEM initiates orbit transfer to rendezvous with the Apollo spacecraft. As the LEM approaches the apolune position in its transfer, the closing rendezvous is initiated. The powered phases of this orbital transfer are shown in Figure 24.

3.1.2.3.7 Transearth Injection. - Initiation of the transearth injection maneuver is executed after the LEM rendezvous has been accomplished and the crew recovered. The number of revolutions after rendezvous is a function of the recovery time, the time required to perform pre-injection tasks, and the desired transearth transit time. The maneuver required is a function of the characteristic of the parking orbit at the time of orbit injection, the time spent in the orbit, i. e., the total number of revolutions, and the terminal constraints at perigee which must be satisfied.

3.1.2.3.7.1 Terminal Constraints. - The terminal constraints which must be satisfied are the earth atmospheric entry angle, geocentric conic inclination, and the entry epoch. The required entry angle is specified by the characteristics of the entry vehicle and must be such that capture is insured without exceeding the aerodynamic heating or loads limitations. The inclination of the terminal conic relative to the earth equatorial plane is nominally 34 degrees. The inclination relative to the moon orbital plane, however, depends upon the time of month and can vary between approximately 5.4 and 62.6 degrees. The position of the node of the geocentric conic relative to the earth-lunar line of centers is essentially invariant with transit time. Therefore, the combination of transearth injection epoch and transit time must result in the proper geographic node at the time of entry.

3.1.2.3.7.2 Variable Parameters. - The transearth injection parameters which are varied to satisfy the terminal constraints are the injection epoch and the end-boost velocity, azimuth, and flight-path angle. On any given epoch, the required end-boost conditions to satisfy the terminal conditions can be determined. The ΔV requirements could be excessive, however, so the injection which results in the minimum ΔV requirement will be selected. There is one such epoch during each revolution. The position of the vehicle at the time of minimum ΔV injection will, in general, be over the non-visible side of the moon. Thus, the transearth injection maneuver cannot be observed from the earth.

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3.1.2.3.7.3 Trajectory. - The transearth injection trajectory presented in Figure 25 is based upon a co-planar, gravity-turn type maneuver. The trajectory shown does not include any part of the LEM equipment or possible lunar payload returned to the earth.

3.1.2.3.8 Transearth Coast. - The transearth coast time is determined by the transearth injection conditions. The transit time could vary between 60 and 90 hours. Thus, if the vehicle is required to initiate the transearth injection maneuver during any given parking orbit revolution, an entry epoch can be obtained which will satisfy the terminal geometric constraints.

3.1.2.3.9 Entry. - The entry trajectory environment, including the entry from an abort condition, will establish much of the Apollo spacecraft systems and structural design requirements. The major items influencing the trajectory environment are: (1) mission requirements and constraints, (2) corridor depth and the vehicle position in the corridor, (3) CM aerodynamic design, (4) trajectory shape or maneuver history, (5) atmospheric deviations, and (6) piloting considerations. Considerations of these items will result in variations in the following factors: (1) aerodynamic heating and temperature histories, (2) maneuver loads and load durations, (3) aerodynamic lift and control effectiveness, (4) dynamic response characteristics, (5) target touchdown dispersions, and (6) landing area prediction. The influence of these factors will establish the trajectory bounds and the overall system requirements. Typical entry flight characteristics associated with the lunar mission previously described are shown in Figures 26, 27, and 28, and show those items that must be considered in describing trajectory environment and design criteria.

3.1.2.3.9.1 Assumptions. - The intended touchdown site for the mission illustrated herein was contained in the initial entry plane. At the entry interface the CM was 2770 nautical miles from the intended destination. Other assumptions and conditions in the example are:

- (a) Aerodynamic Trim Design; $L/D = 0.5$, $W/C_{DA} = 64.3$ psf
(Hypersonic Mach No.)
- (b) Path control by roll about the velocity vector.

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- (c) Entry guidance based upon approximate analytical expressions for range prediction to obtain the proper bank angle commands
- (d) ARDC 1959 Standard Atmosphere, no winds
- (e) Spherical non-rotating earth.

Since this particular mission does not correspond to the range requirement limits, it is not representative of critical flight environment conditions. Continuous path control can be maintained throughout the mission. Vehicle trim attitude maintained by aerodynamics is nearly continuous. The aerodynamic heating histories are well below design limits; maneuver loads are not excessive. Reduction of the second peak g can be accomplished by proper adjustment of the constants used in the steering mechanization. The steering logic and constants used herein result in acceptable trajectories for the required ranges, expected entry conditions, and reasonable variation in the CM aerodynamic design. They have not been optimized around the particular set of design data used in this example.

3.1.2.3.9.2 Entry Range. - Transearth injection and coast characteristics satisfying the 100 percent of month lunar launch capability necessitate entry range capability between 1375 nautical miles and 5000 nautical miles for a recovery at either of two preselected sites with a specified earth trace. The small variations in entry velocity (± 100 fps) arising from normal variations in transearth transit time (60 to 90 hours) for the nominal initial path angle of -6.4 degrees are insignificant. Thus, the entry environment primarily reflects: (1) variations in entry range requirements, (2) expected atmospheric deviations from a preselected reference, (3) expected position and velocity deviations at the entry interface, and (4) entry guidance and control tolerances.

3.1.2.3.9.2.1 5000 Nautical Mile Entry Range. - Missions that require a 5000-nautical mile entry range to reach the specified destination will be most critical to systems requirements, particularly if large variations in flight conditions at the entry interface are included in the design requirements. For values of lift to drag (L/D) such as 0.5 a 5000 nautical mile range is not attainable with entirely atmospheric deceleration paths, with the possible exception of entry along the overshoot boundary.

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Thus, to achieve the maximum range for most initial entry conditions, some amount of controlled atmospheric exit must occur. Since aerodynamic path control is ineffective during an exit condition, exit errors and errors propagated during exit must be corrected during the subsequent reentry. It is not possible to correct sizeable guidance and control errors made prior to reentry because of the low L/D and the nature of the reentry trajectory. Thus, the ability to arrive at the desired destination hinges delicately on exit velocity and path angle. Typical values of range sensitivity to exit conditions and reentry correction capability are listed in the following table. The actual exit conditions are determined by the initial corridor conditions (positions and velocities) and the entry guidance steering parameters. Trajectory dispersions at exit are usually dominated by probable dispersions in the initial entry conditions (corridor). The achievement of acceptable touchdown dispersions at the intended site under maximum range conditions necessitates careful selection of the nominal corridor condition. Thus, the allowable corridor dispersions for long range are well below those required for capture.

Table V. Atmospheric Exit Sensitivity
 X = 5000 NM (longitudinal range)
 L/D = 0.5

Exit Conditions		Exit Sensitivity		Reentry Range Correction Capability
V_x fps	γ_x Deg.	$\partial X / \partial V_x$ NM/fps	$\partial X / \partial \gamma_x$ NM/Deg.	δX_{RE} NM
25050	0.45	2.15	1400	±700
24570	2.65	2.35	740	±500

3.1.2.3.9.2.2 Maximum Entry Range. - In addition to severe spacecraft guidance and control requirements, maximum range is critical to heatshield design, CM stabilization, and control fuel requirements and produces challenging piloting problems if manual override capability is required at any point in the mission. Heating loads are highest for entries along the overshoot boundary. Trajectory shaping based upon minimum atmospheric exit time results in maximum heating loads for a specific entry condition and CM design. During high altitude and atmospheric exit flight phases,

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CM attitude control must be exercised by the jet reaction system since aerodynamic trim is limited to minimum dynamic pressure levels of approximately 10 to 20 psf.

3. 1. 2. 3. 9. 2. 3 Minimum Entry Range. - The ability to touchdown at the intended site for the minimum range condition (1375 nautical miles) is primarily a function of the initial position in the corridor and the ability to exercise manual control during relatively high load-time histories. Although not expected to be critical to systems design, the minimum range limits are those corresponding to the expected overshoot boundary and maximum L/D values. If the expected dispersions in corridor flight characteristics are small, then entry guidance logic and steering constants can be simplified; i. e., adjustment in constants, etc. for corridor differences would be unnecessary.

3. 1. 2. 3. 9. 2. 4 Lateral Range. - Lateral range requirements associated with terminating the flight at the proper destination are determined by the initial entry dispersions. Lateral range capability, however, is dependent strongly upon the trajectory shape following the initial atmospheric penetration, the initial entry maneuver or bank angle, and the total range required. For a given CM design and entry guidance logic, the lateral range capability is dictated by the steepest entry path angle.

3. 1. 2. 4 Other Requirements. - Other mission requirements that affect trajectory environment and G&N system design are: (1) the capability of touchdown at an alternate, but preselected site, and (2) the ability to abort the mission at any point regardless of specific touchdown location. Operational contingencies, such as weather, may necessitate the alternate site requirement. Storms may cover areas as large as 2000 nautical miles by 200 nautical miles and require rather high flexibility in the G&N systems design. The vehicle final maneuver capability of "guaranteed footprint" will be highly dependent upon initial position and velocity dispersions at entry. To afford large guaranteed footprints, tight space guidance and control are required.

3. 1. 2. 5 Basepoint Vehicle Systems Characteristics

3. 1. 2. 5. 1 Launch Vehicle. - The 3-stage C-5 launch vehicle is shown in Figure 1. The over-all length of the vehicle is approximately 360 feet. The maximum diameter of the first two stages of the vehicle is 396 inches.

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3. 1. 2. 5. 1. 1 Saturn S-IC Stage. - The S-IC stage utilizes five F-1 engines producing a total sea level thrust of 7, 500, 000 pounds. The sea level specific impulse of the engines is 263.58 seconds. Main stage propellant capacity is limited to 4, 400, 000 pounds including the engine mixture ratio tolerance reserve. The propellants for the first stage are liquid oxygen (LOX) and RP-1. The S-IC thrust varies during boost as a function of altitude. At engine cutoff, which is very close to vacuum levels, the thrust produced is 8, 625, 000 pounds. Thrust termination for the S-IC stage is accomplished by shut-down of the central F-1 engine. Four seconds later, the four outboard gimballed F-1 engines are cutoff at which time the retro rockets on the S-IC stage and the ullage rockets on the S-II stage are ignited.

3. 1. 2. 5. 1. 2 Saturn S-II Stage. - The 396-inch diameter second stage of the Saturn is powered by five 200, 000-pound thrust J-2 engines producing a total thrust of 1, 000, 000 pounds in a vacuum. The vacuum specific impulse for this stage of 424 seconds. The stage propellant capacity is 930, 000 pounds. The propellants for the S-II stage are LOX and liquid hydrogen (LH₂). The S-II stage ignition occurs 1.5 seconds following the S-IC stage cutoff. At the end of S-II stage boost, all J-2 engines are shut down simultaneously. Ignition impulse and cut-off impulse characteristics, along with detailed stage separation and retro sequences, will be provided when available.

3. 1. 2. 5. 1. 3 Saturn S-IV B Stage. - The 260-inch diameter third stage of the Saturn is powered by a single 200, 000-pound thrust J-2 engine having a specific impulse of 422 seconds. The propellants for the S-IV B are LOX/LH₂ and the stage propellant capacity is 230, 000 pounds including flight performance and launch window reserves. S-IV B ignition occurs 2-1/2 seconds following thrust termination of the S-II stage. Ullage and ignition sequences, restart sequences, ignition impulse, and cut-off impulse characteristics will be provided when available.

3. 1. 2. 5. 1. 3. 1 Thrust Vectors Control Characteristics. - System response characteristics, maximum angular accelerations, and attitude rates will be provided when available.

3. 1. 2. 5. 1. 3. 2 Reaction Control Characteristics. - The reaction control system (RCS) on the Saturn S-IV B launch vehicle stage will be utilized for orientation and attitude stabilization during earth parking orbit operations and during the LEM transposition and docking maneuver after translunar injection.

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3.1.2.5.2 Service Module. - The SM propulsion system utilizes a single, ablative, gimballed, thrust chamber having a nozzle expansion ratio of 60:1 and a nominal initial thrust level of 21,900 pounds with an average specific impulse of 319.5 seconds. The maximum number of engine starts expected is 12. The expected variations in thrust and specific impulse vs time are shown in figure 29.

3.1.2.5.2.1 Propellant. - The propellants used in the propulsion system are storable and hypergolic; the oxidizer to fuel mixture ratio being 2:1. The oxidizer propellant is nitrogen tetroxide (N_2O_4). The fuel propellant is a 50-50 by weight mixture of hydrazine (N_2H_4) and unsymmetrical-dimethylhydrazine ($(CH_3)_2N_2H_2$). The maximum propellant capacity for the service module is 45,000 pounds; however, most lunar missions will be flown with the propellant tanks off-loaded.

3.1.2.5.2.2 Ignition Impulse. - The ignition impulse does not exceed $1,535 \pm 285$ pounds per second. The ignition impulse is defined as the integration of thrust with time from the onset of the electrical command to the time at which 90 percent of the rated thrust is achieved. This time interval will be within 0.225 - 0.300 seconds.

3.1.2.5.2.3 Cutoff Impulse. - The cutoff impulse is $4,570 \pm 670$ pounds per second. The cutoff impulse is defined as the integral of thrust with time from the command signal to terminate thrust to the time at which the thrust has decayed to 10 percent of the rated thrust value. This time interval will be within 0.225 - 0.300 seconds.

3.1.2.5.2.4 Velocity Changes. - The minimum total impulse for minimal velocity changes by the SM propulsion system is $6,740 \pm 955$ pounds per second. This figure is based upon initiation of engine cutoff at the time at which the thrust has risen to 90 percent of the rated thrust value.

3.1.2.5.2.5 Thrust Vector Control Characteristics

3.1.2.5.2.5.1 Finite Thrusting Mode. - Finite thrusting is considered to be those events such as lunar orbit injection, transearth injection, and space abort, where the effects of thrust initiation and thrust decay sequences and transients on spacecraft steering dynamics are small compared to long period, equilibrium factors. The thrust chamber is gimballed with deflection capabilities of ± 8.5 degrees in both pitch and yaw, and maximum

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gimbal rates of ± 0.35 radians per second. The engine gimbaling transfer function is of the form:

$$\frac{\dot{\delta}}{\delta_c} = \frac{K}{S + 40} \quad \frac{\delta}{\delta_c} = \frac{K}{S(S + 40)}.$$

3.1.2.5.2.5.2 Midcourse Correction Mode. - Transition from reaction control to thrust vector control and steering transients introduced by thrust misalignment and center of gravity uncertainties during thrust build-up and decay affect the precision with which small changes in velocity can be made. The normal transition sequence from reaction control to thrust vector control is to coarse-align the spacecraft with the reaction control system to within ± 5 degrees of the required attitude with attitude rates at thrust vector control initiation no greater than 0.50 degrees per second. Within these limits, the resulting maximum deviations in equivalent thrust direction as a function of delivered total impulse for translunar and transearth inertia configurations will be provided when availability permits.

3.1.2.5.2.6 Reaction Control System Characteristics. - The SM RCS is used for spacecraft attitude stabilization and control throughout the mission from launch vehicle separation at translunar injection to CM separation prior to entry. It also is used for translational control; during docking maneuvers; as ullage rockets, during service propulsion ignition; and, with certain limitations, for midcourse guidance maneuvers which are below the minimum impulse capabilities of the service propulsion system.

3.1.2.5.2.6.1 Attitude Stabilization and Control Mode. - The SM RCS is utilized to orient and stabilize the spacecraft during ballistic flight for navigation sightings. The attitude control propellants consumed for this purpose are charged to G&N functions. Maximum attitude control propellant allowances are dependent upon optimization of overall mission performance parameters between increased navigational precision with large numbers of sightings and increased midcourse maneuver requirements to compensate for lower navigation accuracies. Regardless of optimum trends, however, navigation accuracy requirements must be met prior to any planned maneuver in order to allow a reasonable period for nominal execution and establishment of any special emergency procedures. Attitude control propellant consumption as functions of duty cycle characteristics and configuration moments of inertia will be provided when available.

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During navigation sightings the spacecraft can be controlled to the following rates:

Minimum roll rate, $\dot{\phi}$ (min) = 1.2 arc minutes per second

Minimum pitch rate, $\dot{\theta}$ (min) = 0.4 arc minutes per second

Minimum yaw rate, $\dot{\psi}$ (min) = 0.4 arc minutes per second

Maximum roll rate, $\dot{\phi}$ (max) = 20 degrees per second

Maximum pitch rate, $\dot{\theta}$ (max) = 20 degrees per second

Maximum yaw rate, $\dot{\psi}$ (max) = 20 degrees per second

Maximum angular accelerations in roll, pitch, and yaw as functions of configuration moments of inertia will be provided when available.

3. 1. 2. 5. 2. 6. 2 Orbital Docking Mode. - The SM RCS will be utilized for translational control along all three spacecraft axes during the LEM transposition docking maneuver and during lunar orbital docking in the event of a LEM failure. Spacecraft acceleration characteristic during both of these operations will be provided when available. The effects of thrust profile characteristics and spacecraft steering transients on translational velocity control will be provided when available along with overall propellant allowances.

3. 1. 2. 5. 2. 6. 3 Midcourse Correction Mode. - During translunar, transearth, and lunar orbital mission phases the SM RCS is utilized for guidance maneuvers which are below the minimum impulse capabilities of the service propulsion system. It may also be used as an alternate means of applying guidance corrections during translunar and transearth flight phases in the event of a service propulsion failure. Velocity control characteristics in this mode of operation are similar to those for translational control in the X axis and will be provided along with additional data for potential spacecraft weights and inertias and the maximum continuous thrusting period as affected by time between firing when available.

3. 1. 2. 5. 3 Command Module. - The CM is rolled about the velocity vector to control the direction of the aerodynamic lift, thereby providing the capability for guiding along a desired trajectory. Major CM aerodynamic performance coefficients and attitude control characteristics of the basepoint configuration are presented in the following paragraphs.

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3.1.2.5.3.1 Aerodynamic Characteristics. - The CM lift-to-drag design ratio, L/D is 0.5 and the ballistic coefficient, $\frac{W}{C_D A}$ is 64.3 pounds per feet. The effects of heat shield ablation and normal entry consumables on these parameters will be provided when available.

3.1.2.5.3.2 Reaction Control System. - The CM reaction control system is utilized to orient and stabilize the spacecraft for entry after separation from the SM and to control bank angle (roll) and provide attitude stabilization about all three body axes during entry. Maximum angular accelerations and attitude rates as affected by entry aerodynamics will be provided when available.

3.1.2.5.4 Ground Operational Support System Navigational Role. - GOSS capabilities shall include radio, optical, and information-system elements of AMR Tracking Installations, of Apollo Tracking Net, and of Deep Space Tracking Network exclusive of its Central Computing Facility. GOSS shall also include information processing capability in the Apollo Integrated Mission Control Center. The navigational requirements of GOSS are:

- (a) To furnish present and predicted trajectory and maneuver data to Apollo Intergated Mission Control Center.
- (b) To afford the Apollo crew a continuously available source of alternate navigation and guidance data for any intervals in which more accurate information is not available from on-board sources. This will include, but not be limited to such contingencies as equipment malfunction, weather impairment of visual sensing, crew preoccupation with non-catastrophic emergencies, and computer diversion to higher priority tasks.
- (c) To make available to the on-board G&N system such ground-based radar sensings as may be desired for input data into the on-board orbit determination routines.
- (d) To monitor, by telemetry, the performance of on-board systems and components.

3.1.2.5.4.1 Operational Use. - The most accurate navigational information obtainable at any given time, whether from GOSS or other sources, shall be available to the crew as required.

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3.2 G&N Performance Requirements

3.2.1 Description of G&N Tasks. - The overall Apollo G&N task is described and expanded into subtasks or phases in a chronological sequence in which the objective of G&N activity is different between phases (see figure 30). The G&N activities shall include all operations associated with the rational capability of the Apollo spacecraft to determine where it is, where it is going, where it should be going, and how to get there. This activity includes operations performed by automatic G&N equipment, the GOSS, and the astronauts. The requirements imposed by each G&N phase on the design of the G&N system are presented in detail and the basic ground rules which are to be followed in all phases are stated in the paragraphs that follow. The breakdown of the G&N task is described below.

3.2.1.1 Prelaunch Checkout. - The G&N system shall be started, checked out, calibrated, aligned, and initialized through procedures which shall be compatible with the launch vehicle countdown.

3.2.1.2 Boost Monitoring. - Spacecraft motion shall be monitored from launch up to insertion into an earth parking orbit.

3.2.1.3 Earth Orbit Injection Evaluation. - Following injection into earth orbit and prior to completion of a quarter revolution about the earth, a navigational procedure shall be applied to ensure that a safe earth orbit has been achieved.

3.2.1.4 Earth Orbit Determination. - Parameters defining the earth orbit shall be determined prior to injection onto a translunar trajectory.

3.2.1.5 Translunar Injection Monitoring. - Spacecraft motion shall be monitored (or controlled) during transfer from the earth parking orbit to a translunar trajectory.

3.2.1.6 Translunar Injection Evaluation. - Following injection on the translunar trajectory, a navigational procedure shall be applied to determine if a satisfactory translunar trajectory has been achieved relative to the probabilities of mission success and crew safety.

3.2.1.7 Translunar Midcourse Guidance and Navigation. - The effects of translunar injection errors and errors in the equations used to describe motion along the translunar trajectory shall be determined and controlled during translunar flight.

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3. 2. 1. 8 Lunar Orbit Injection. - Spacecraft motion shall be controlled during a transfer from the translunar trajectory to a lunar parking orbit.

3. 2. 1. 9 Lunar Orbit Injection Evaluation. - Following injection into lunar orbit and prior to completion of a quarter lunar revolution, a navigational procedure shall be applied to ensure that a satisfactory lunar orbit has been achieved relative to the probabilities of mission success and crew safety.

3. 2. 1. 10 Lunar Orbit Determination. - To be defined at a later date.

3. 2. 1. 11 Transearch Injection Control. - Spacecraft motion shall be controlled during transfer from the lunar parking orbit to a transearch trajectory.

3. 2. 1. 12 Transearch Injection Evaluation. - Following transearch injection, a navigational procedure shall be applied to determine if a satisfactory transearch trajectory has been achieved relative to the probabilities of achieving the selected landing site via a normal entry, and crew survival.

3. 2. 1. 13 Transearch Midcourse Navigation and Guidance. - The effects of transearch injection errors and errors in the equations used to describe motion along the transearch trajectory shall be determined and controlled during transearch flight.

3. 2. 1. 14 Entry Control. - CM motion shall be controlled during entry into the earth's atmosphere and the CM shall be directed to the vicinity of a preselected landing site.

3. 2. 1. 15 Abort Control. - Spacecraft motion shall be prescribed and controlled to achieve a satisfactory probability of safe retrieval of the crew in the event the mission cannot be continued.

3. 2. 2 Basic Requirements. - The following ground rules shall be applied to the G&N system for all mission phases.

3. 2. 2. 1 Accuracy. - The guidance and navigational precision shall be sufficient to control spacecraft dispersion to a degree consistent with overall fuel budgeting. Mechanization of guidance for major maneuvers shall provide for power phase trajectories which deviate from optimum steering (fuel-wise) within limits consistent with overall fuel budgeting. The

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combination of spacecraft weight and booster capability point to the desirability of saving fuel by using high precision to reduce dispersion. This combination also indicates that sophisticated mechanization should be considered to ensure the near-minimum fuel power tracts are followed during major burning phases. In no case, however, should complexity, development time and cost, or ease of operation be degraded to achieve a level of precision or sophistication which shall reduce the fuel chargeable to G&N component inaccuracies or mechanization approximations to a value significantly smaller than that chargeable to established spacecraft characteristics.

3.2.2.2 Reliability. - Attainment of the maximum mission reliability and crew safety shall be the most important single consideration in the design, construction, and operation of the G&N system. Full use of astronaut capabilities shall be applied; however, automatic systems shall be employed whenever reliability and safety are enhanced to the extent that mission reliability is optimized. Techniques involving in-flight testing, calibration, and maintenance and application of redundant components and systems shall be implemented as required to achieve the desired probabilities of mission success and crew safety. Specifically, in the event of an IMU malfunction during injection, an SCS attitude reference shall be employed in establishing a desired constant attitude, and cutoff shall be provided for by an integrating accelerometer on the longitudinal axis of the spacecraft. The probability of failure of the primary G&N system to function successfully through all mission phases shall be less than 0.006. The probability of the G&N system failing to have the capability of safely returning the crew to earth shall be less than 0.0001.

3.2.2.3 On-Board Capability. - The CM G&N system shall be designed to function as a self-contained system. G&N requirements shall be met independent of support from earth-based systems.

3.2.2.4 Acceptance and Transmission of Data. - The G&N system shall be designed to be capable of accepting GOSS data to upgrade the G&N performance and reliability. The system shall permit pertinent information relative to G&N performance to be transmitted to GOSS for evaluation and computation to provide back-up guidance.

3.2.2.5 Astronaut Override Capability. - Pertinent information shall be available and displayed to the astronaut to allow him to evaluate the performance of the G&N system and to take over spacecraft control or select alternate modes of operation.

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3. 2. 2. 6 Abort Capability. - The G&N system shall be designed to accept astronaut initiated abort commands as required to achieve the desired probability of crew survival. The integrated abort structure shall impose minimal restrictions on the design and operation of the G&N system.

3. 2. 2. 7 Operation by a Single Astronaut. - The G&N system shall allow a single astronaut to perform all tasks required for crew survival.

3. 2. 2. 8 Back-up Alignment Capability. - The G&N system shall provide the astronauts with external reference data from the optical and display elements which shall enable them, with the aid of the SC system, to manually align the vehicle thrust vector in any inertial direction for making emergency injection maneuvers, midcourse ΔV corrections, and earth de-orbit velocity changes. The errors in thrust vector alignment introduced by the G&N system shall not exceed $\pm 1.5V$ degrees.

3. 2. 3 Functional Requirements

3. 2. 3. 1 Prelaunch Checkout. - Prelaunch checkout is divided into two areas: (1) guidance and navigation functional checkout; (2) erection and alignment of the IMU. Checkout of the G&N system shall verify post installation checkout results, and establish the "operational condition" (see 6. 1. 1) of the system. The G&N system shall demonstrate an "operational condition" commensurate with Apollo mission success. Erection and alignment of the IMU is an operational mode that is associated with prelaunch activities and shall be considered a part of prelaunch checkout. Erection and alignment shall be accomplished commensurate with boost monitoring requirements. Re-erection and realignment shall be accomplished as necessary.

3. 2. 3. 1. 1 Subsystem Checkout

3. 2. 3. 1. 1. 1 Self Checks. - The Ground Support Equipment (GSE) and Apollo Guidance Control (AGC) computers shall be capable of self check both independently and in combination. Such checks shall establish the operational conditions of the GSE and the AGC.

3. 2. 3. 1. 1. 2 Sextant (SXT) and Scanning Telescope (SCT). - The scanning telescope and sextant subsystem shall be capable of both precision requirements and dynamic response commensurate with mission success. The SCT shall be capable of providing the astronaut with a field

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of view consistent with proper target identification. The SCT and associated servo loop shall be capable of slaving the SXT assembly to the target with prescribed accuracy. The SXT shall be capable of determining the proper celestial angles (as determined by the astronauts) to prescribed accuracies.

3.2.3.1.1.3 Inertial Measurement Unit (IMU). - The IMU shall be capable of erection and alignment to an "operational condition" consistent with Apollo mission success.

- (a) The IMU drift rate (time dependent) errors shall establish an "operational condition" consistent with Apollo mission success.
- (b) The IMU g-sensitive (acceleration dependent) errors shall be of such a magnitude as to define an "operational condition" consistent with mission success.
- (c) The IMU dynamic response characteristics shall describe an "operational condition" commensurate with mission success.
- (d) IMU operation must be satisfactory when the G&N system is subjected to a "worst case" vibration and temperature environment.

3.2.3.1.1.4 Coupling Display Unit. - All coupling display units shall be in an "operational condition" consistent with astronaut G&N system monitoring and astronaut G&N requirements.

3.2.3.1.1.5 Ground Communication Links (Radar, Telemetry). - All ground communication links shall be in an "operational condition" consistent with mission success.

3.2.3.1.2 G&N System Functional Checkout. - The G&N system shall respond to predetermined input signals (simulated if necessary) in order to define a satisfactory "operational condition." These predetermined input signals shall be used to check out critical areas of G&N system operation. Normal G&N system functional operation shall be monitored by the AGC and GSE to determine if a state of satisfactory "operational condition" exists.

3.2.3.1.2.1 Erection and Alignment. - G&N scale factors, biases, and pulse rates shall be determined for each relevant component of the G&N system.

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These constants shall be determined with accuracies consistent with a satisfactory "operational condition," and shall be updated from time to time as necessary. The IMU shall be erected and aligned to accuracy commensurate with a satisfactory "operational condition." Realignment several times to verify G&N system operation shall be necessary.

3.2.3.2 Ascent-to-Parking Orbit. - The G&N system shall perform as follows during ascent to parking orbit:

- (a) Monitor position, velocity, acceleration, attitude rates and vehicle attitude along the trajectory
- (b) Predict the burnout conditions and orbital parameters based on the best estimate of the current state vector and the desired thrust attitude computed in accordance with the adaptive guidance technique mechanized in the Saturn guidance computer
- (c) Take over ascent guidance through use of on-board equipment in the event of Saturn G&N malfunction. (The G&N system shall be capable of accepting data from ground operational systems to assist in evaluating flight performance for abort.)

3.2.3.2.1 Monitoring. - The G&N system shall be capable of determining the trend of the trajectory and that the vehicle is within statistical limits for the totality of possible non-nominal conditions. The monitoring procedure will provide the criteria necessary to determine whether to: (1) continue as is, (2) take over and continue ascent guidance from within the CM, (3) abort before boost cutoff, or (4) abort from orbit. The criteria will be based on the detection of abnormal guidance conditions through evaluation of the quantities mentioned above, inputs from abort sensors concerned with the current state of the propulsion system, vehicle structural integrity, and inputs from ground instrumentation including range safety.

3.2.3.2.2 Orbital Prediction. - The G&N system shall provide real-time orbital prediction during SII and SIV-B powered flight to determine if the predicted burnout conditions will result in an orbit within the permissible eccentricity and tolerance on orbital inclination. If these conditions are not met, the entry angle and speed shall be computed to determine whether it is safe to continue into an orbit encompassing the defined entry interface or to abort as soon as possible to one of the launch abort landing sites.

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3.2.3.3 Earth Orbit Injection Evaluation. - Following injection into earth orbit and prior to completion of a quarter-revolution about the earth, navigational procedures shall be applied to establish that a satisfactory earth orbit has been achieved commensurate with the desired probabilities of overall mission success and crew survival.

3.2.3.4 Earth Orbital Navigation. - The earth orbital phase begins with successful injection into earth orbit and ends with a de-orbit maneuver or the translunar injection phase. The minimum time in earth parking orbit shall not be less than that required to checkout the spacecraft and determine earth orbit. The maximum time may be as long as required to satisfy total mission timing for return to a primary landing site. The upper limit shall be approximately four orbits to ensure that primary landing sites will be available in case of abort during the translunar injection maneuver. The platform shall be capable of alignment (using SCT or SXT and two star vectors) within 15 minutes and at least once per orbit while in earth orbit (mainly dependent upon drift parameters). The G&N shall determine parameters defining the earth orbit from SCT observations for the following purposes:

- (a) Initialization of CM G&N prior to translunar injection
- (b) Determination of thrust initiate
- (c) Velocity to be gained
- (d) Guidance matrices and parameters.

The G&N system also shall account for air density uncertainty and oblateness effects while in earth orbit. The optimum navigational procedure (landmark identification and data filtering techniques) shall be dependent upon the time in orbit. The SCT angular error must be less than ___ arc seconds. The SCT shall be capable of manual control to determine local vertical within _____ minutes to an uncertainty of _____ seconds.

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3.2.3.4.1 Earth Orbit Attitude Control. - The attitude control in pitch, roll, and yaw maneuver shall not exceed a duty cycle in terms of number and time of re-orientation and dwell time in each sighting mode. Duty cycle will be provided when availability permits.

3.2.3.5 Translunar Injection. - If Saturn G&N is to perform the guidance computations for translunar injection, the Apollo G&N system shall be capable of monitoring the injection maneuver. The position, velocity, acceleration, and attitude of the vehicle shall be monitored to within an accuracy of _____ feet, _____ feet per second, _____ feet per second squared and _____ degrees during the maneuver.

3.2.3.5.1 Translunar Injection Evaluation. - The G&N system shall be capable of evaluating the injection conditions within _____ hours after injection through data obtained from on-board midcourse navigation and GOSS. Injected position, velocity, flight path angle, and the inclination of the flight plane relative to the moon's orbital plane shall be determinable within prescribed uncertainties. A determination shall be made of the propellant required to make the outbound velocity corrections necessary to put the spacecraft on a circumlunar trajectory within specified limits. This trajectory requirement must be met to continue the mission as planned. If the propellant required to accomplish this maneuver is in excess of statistical limits, abort criteria shall be applied to determine whether: (1) sufficient propellant is available to orbit the moon, (2) propellant is available only for the attainment of a circumlunar trajectory in keeping with the total propellant needs estimated for safe return to the earth entry interface, or (3) to abort from a point on the outbound mid-course trajectory.

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3.2.3.5.2 Translunar Injection Control. - Attitude control in the pitch, roll, and yaw maneuvers shall not exceed a specified duty cycle in terms of number and time of reorientation and dwell-time in each sighting mode. Duty cycle will be provided when available.

3.2.3.6 Translunar Midcourse Phase. - The translunar midcourse phase is that segment of the mission starting at the end of translunar injection and ending at the required conditions prior to injection into lunar orbit.

3.2.3.6.1 Translunar Midcourse Functional Requirements. - The G&N system functional requirements during the translunar midcourse are: (1) to monitor the position and velocity of the spacecraft and provide the displays necessary for mission evaluation, (2) to compute and control the velocity corrections needed to assure arrival at the point of lunar orbit injection with the required terminal conditions, and (3) to provide the capability for mission abort at any point along the trajectory. The analyses required to present the detailed translunar midcourse requirements, such as the specific displays required for mission evaluation, the position and velocity deviations and uncertainties allowable at the point of lunar orbit injection, etc., are not yet available. These specific requirements will be supplied in the subsequent updating of this document.

3.2.3.6.2 Translunar Midcourse Operational Requirements. - The design of the G&N system shall not place constraints on the translunar midcourse operation that are not compatible with other mission functions the spacecraft is required to perform. The G&N system shall be compatible with the requirement for maintaining the telecommunication and the spacecraft orientation required for the control of thermal environment.

3.2.3.6.3 Translunar Midcourse Control. - The attitude control in the pitch, roll and yaw maneuver shall not exceed a specified duty cycle in terms of number and time of reorientation, and dwell time in each sighting mode. Duty cycle will be provided when availability permits. The total ΔV expended for course correction shall not exceed _____ feet per second.

3.2.3.7 Lunar Orbit Injection. - To obtain a lunar orbit, thrust initiation, steering, and cutoff commands shall be generated to perform a transfer from the translunar trajectory to a lunar parking orbit. A decision rule is required to specify in detail the characteristics of the desired lunar parking orbit (such as mean radius and inclination) based upon the deviation of the spacecraft from the nominal trajectory and G&N performance history

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during translunar midcourse operation. The criteria for initiation of the maneuver and the steering, thrust initiation, and cut off accuracy requirements will be determined through G&N analysis based upon allowable dispersion in lunar orbit parameters and the portion of SM fuel chargeable to mechanization of the injection maneuver. Preliminary results of the G&N analysis indicate that an inertially fixed thrust attitude will be sufficiently near an optimum (with respect to fuel) steering maneuver so that specifying an inertial direction, a velocity to be gained, magnitude, and a thrust initiation time may be sufficient information to define the maneuver. The velocity cutoff error would be within ___ ft/sec in magnitude and ___ deg in attitude. A circular orbit shall be achieved within specified deviations in perilune of ___ feet, inclination of degrees and period of ___ seconds. The system shall be capable of a degree out-of-plane maneuver, and of steering the vehicle in consideration of the geometrical and vehicle constraints as well as the energy requirements.

3.2.3.7.1 Lunar Orbit Injection Control. - The standard deviation of the ΔV correction required to remove arrival deviations shall not exceed ___ fps.

3.2.3.7.2 Lunar Orbit Evaluation. - Following injection into lunar orbit and prior to completion of a quarter lunar reorbit, a navigation procedure will be applied to determine if a satisfactory lunar orbit has been achieved commensurate with the probabilities of mission success and crew survival. The G&N system shall be capable of providing guidance for altering the lunar orbit or initiating and controlling an abort using the SM propulsion system as required by the desired probabilities of mission success or crew survival if the orbit is unsatisfactory.

3.2.3.7.2.1 Lunar Orbit Control Requirement. - Attitude control in pitch, roll and yaw shall not exceed a specified duty cycle in terms of number and time of re-orientation, and dwell time in each sighting mode. Duty cycle will be provided when availability permits.

3.2.3.7.3 Lunar Orbital Navigation. - The lunar orbital phase begins with injection into lunar orbit, and is completed when transearth injection is begun. The vehicle shall inject into a near circular lunar orbit. The platform will be aligned by using SXT and SCT and two star vectors within 15 minutes and at least once per orbit while in lunar orbit (dependent upon drift parameters). The G&N shall be capable of determining a complete set of orbital parameters (position and velocity) from SCT observation for: (1) initialization of guidance system, (2) determination of CM ephemeris, and (3) initialization of CM G&N prior to transearth injection

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and determination of thrust initiated, velocity to be gained, guidance matrices and parameters. The G&N system shall also account for lunar tri-axiality while in lunar orbit. The SCT shall be capable of making a sighting at least every 30 minutes. The optimum navigational procedure (landmark identification and data filtering techniques) depend upon the time in orbit. The SCT error shall be less than _____ sec.

3.2.3.7.3.1 Attitude Control. - Attitude control in pitch, roll, and yaw shall not exceed a specified duty cycle in terms of number and time of re-orientation and dwell time in each sighting mode. Duty cycle will be provided when availability permits.

3.2.3.8 Transearch Injection. - The Apollo G&N shall be capable of performing the computations required to initiate transearch injection, determine the desired thrust attitude profile, and signal thrust cutoff to put the vehicle on a hyperbolic escape trajectory relative to the moon. The guidance scheme shall have the flexibility to provide a launch opportunity once every lunar orbit. It shall be capable of orienting the plane of the departure conic to permit return to alternate landing sites.

3.2.3.8.1 Transearch Injection Evaluation. - Following injection onto the transearch trajectory, a navigational procedure shall be applied to determine if a satisfactory trajectory has been achieved commensurate with the desired probabilities of mission success and crew survival. The G&N system must be capable of altering the transearch trajectory using the SM propulsion system to insure that entry conditions allow a completely successful entry to be accomplished or to initiate and control an abort as required by the desired probabilities of mission success and crew survival if the transearch trajectory is unsatisfactory.

3.2.3.8.2 Control for Transearch Injection Evaluation. - The attitude control in pitch, roll, and yaw maneuver shall not exceed a specified duty cycle in terms of number and time of re-orientation and dwell time in each sighting mode. Duty cycle will be provided when availability permits.

3.2.3.9 Transearch Midcourse Functional Requirements. - The transearch midcourse phase is that part of the lunar mission starting from the end of transearch injection and ending with the conditions required for a safe entry.

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The functional requirements for the transearth midcourse phase are:

- (a) Monitor position and velocity of the spacecraft and provide the displays necessary for mission evaluation
- (b) Provide the velocity correction capability required for a safe entry.

3. 2. 3. 9. 1 Compatibility. - The transearth midcourse interface requirements are the same as for the translunar case with the following exception:

The SM engine shall correct any ΔV above 11. 3 ft/sec while any correction made by the reaction control system (RCS) shall be less than 11. 3 ft/sec.

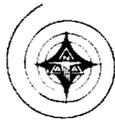
3. 2. 3. 9. 2 Operational. - The transearth midcourse operational requirements are the same as for the translunar case.

3. 2. 3. 9. 3 Transearth Midcourse Control. - The attitude control in the pitch, roll, and yaw maneuver shall not exceed a specified duty cycle in terms of number and times of re-orientation and dwell time of each sighting mode. Duty cycle will be provided when availability permits. The G&N system shall be designed such that the ΔV requirements for attaining the required earth return entry-corridor shall not exceed ft/sec.

3. 2. 3. 10 General Entry Requirement. - Two entry guidance systems shall be employed during entry: (1) primary, and (2) survival. The primary system must be capable of computing position and velocity vectors from measured accelerations and a gravity field mathematical model. In addition, the system must steer the CM in a manner such that survival and touchdown requirements are satisfied. Survival includes considerations relative to: (1) life support systems, (2) human tolerance load levels, (3) system tolerances, (4) piloting, and (5) retrieval areas.

3. 2. 3. 10. 1 Range Requirements. - The range requirements include: (1) landing at the selected site within touchdown accuracy of nautical miles, and (2) controlling the lateral motion such that cross-range limits are not exceeded in order to satisfy ground tracking requirements.

3. 2. 3. 10. 2 Override Capability. - The astronaut must have the capability to override the primary system in case it malfunctions. The survival system shall display to the astronaut the necessary information for determining whether the survival boundaries will be exceeded. The survival system shall also monitor the primary system to determine whether it has malfunctioned. The survival system shall be capable of entering the survival mode at any time during entry.

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3.2.3.10.3 Primary Navigation. - Upon completion of initialization of the guidance state vector in the primary G&N system for entry, the G&N system shall continuously establish the current state vector. Position, velocity, and measured acceleration vectors shall be available in the computer.

3.2.3.10.4 Primary Guidance. - A change in roll angle about the stability axis shall be determined in the computer and presented to the attitude control system. The commanded roll angle change shall be based upon steering functions which will cause the vehicle to follow a trajectory which satisfies the survival and touchdown requirements. The flexibility of the steering equations must be such that for any variation of the velocity and position vectors within the entry corridor, the vehicle shall move along a trajectory which satisfies the survival and touchdown requirements including lunar and earth orbit missions as well as the abort situation.

3.2.3.10.4.1 Primary Functional Interface. - The primary G&N system shall be capable of receiving position and velocity information from the earth communications link during the initialization phase. The capability of reading out position and velocity information stored in the computer for communication to earth must exist.

3.2.3.10.4.2 Initialization of the Primary G&N System. - Requirements to be determined.

3.2.3.10.4.3 State Vector Determination. - The uncertainties of the computed state vector by the AGC shall be determined by considering the following error sources:

- (a) Initial position and velocity
- (b) Initial platform alignment
- (c) Acceleration sensing errors (accelerometer and gyro)
 - (1) bias
 - (2) g
 - (3) g^2
 - (4) random vibration
 - (5) temperature gradient
- (d) Platform stiffness
- (e) Sampling rate (guidance computation cycle)
- (f) Computer round-off
- (g) Steering equations

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- (h) Oblate earth if earth oblateness is not considered in the navigation equations
- (i) Time of flight if rotating earth model is not used in the navigation equations.

3.2.3.10.4.3.1 Landing Site Position Error. - The major axis of the 1σ probability ellipse must be less than _____ nautical miles.

3.2.3.10.4.3.2 Internal Systems Check. - Information shall be presented to the astronaut during the initialization phase to enable him to determine whether the primary G&N power supply, accelerometers, gyros, and the computer are operating. Calibration shall not be required for the initialization phase.

3.2.3.10.4.4 Primary Display. - The following information shall be displayed to the astronauts from the G&N primary system for monitoring purposes:

- (a) Commanded roll angle
- (b) Roll angle
- (c) Velocity relative to air mass.

3.2.3.10.4.4.1 Monitoring of the Primary System. - Expected acceleration time history of the primary system shall be monitored. Expected commanded roll angle of the primary system shall be monitored.

3.2.3.10.4.5 Primary Control. - The capability must exist for the astronaut to override the roll command from the primary system and go to the survival mode. The G&N system must have the capability to return to the primary guidance mode after an override maneuver. The commanded roll maneuver shall not require the consumption of more than _____ pounds of attitude control propellant during all normal entries.

3.2.3.10.4.5.1 Platform Dynamics Design Requirements. - Platform dynamics design requirements are:

- (a) Unlimited motion about the roll axis (wind axis) and pitch axis
- (b) Rotation about the yaw axis of at least ± 60 degrees from the nominal

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- (c) Angular rates of at least 30 degrees/second about each axis
- (d) Angular accelerations of at least 11 degrees/second².

3.2.3.10.4.6 G&N Primary System Reliability. - The probability of failure of the G&N primary system to complete the required functions during entry shall be less than _____. In-flight-maintenance, prior to the 400,000 foot-altitude interface, is allowed in computing the probability.

3.2.3.10.5 Survival G&N System. - Survival boundaries shall be displayed to the astronaut and represented by a G-time history. Acceleration time history shall be displayed and compared to the survival boundaries. The acceleration shall be measured in a direction such that errors in measured accelerations are minimum due to uncertainties in C_D and C_L . Roll angle about the stability axis shall be displayed.

3.2.3.10.5.1 Survival System Initialization. - At a time not greater than minutes prior to the expected touchdown time, the following operations shall be accomplished:

- (a) Gyro for measuring roll angle shall be spun-up and aligned with the spin vector normal to plane of trajectory
- (b) Accelerometer output shall be nulled for biasing purposes
- (c) The proper G-time overlay corresponding to required range shall be selected
- (d) Tests shall be performed during the initialization phase to determine whether the power supply, accelerometer, gyro, and clock are operating; calibration shall not be required
- (e) The t_0 time for starting the system shall be a function of the earth's air density profile.

3.2.3.10.5.2 Measurement. - Measurements provided by the survival G&N system shall meet the following accuracy requirements:

- (a) The 1σ gyro roll angle measurement error shall be not greater than _____ degrees during normal entry.

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- (b) The 1σ accelerometer error shall be not greater than g's during normal entry.
- (c) The 1σ error in starting the clock shall be not greater than _____ seconds.
- (d) The scale of the G-time display shall be at least _____ g/in. and _____ sec/in.

3.2.3.10.5.3 Information Provided. - The astronauts shall be provided information concerning roll maneuvers in order to avoid the consumption of more than _____ pounds of attitude control propellant.

3.2.3.10.5.4 Survival G&N System Reliability. - The probability of failure of the survival entry guidance system to complete the required functions during entry shall be less than _____. Inflight maintenance prior to 400,000 feet altitude interface is allowed in computing the probability.

3.2.3.11 Guidance and Navigation Abort Requirements. - The G&N system shall be designed to provide abort guidance capability for all applicable mission phases with minimum change in the G&N system.

3.2.3.11.1 Near Earth Abort. - The region referred to as "near earth" extends from second stage ignition through translunar injection and includes the following phases: (a) launch abort, (b) orbital abort, and (c) abort from translunar injection. Since survival is of utmost importance, the following primary considerations must be observed in selecting an abort trajectory:

- (a) G-load and heating criteria must not be exceeded
- (b) Radiation tolerance must not be exceeded
- (c) Abort guidance capable of reaching a pre-selected recovery site will be included
- (d) Minimum return-time trajectories will be determined for situations such as support system failures and solar flares.

3.2.3.11.1.1 Launch Abort. - In case of a catastrophic abort during launch, such as imminent booster explosion, a first-burn period shall be initiated to escape the effect of overpressure. The abort guidance mode then shall be entered, and landing site control or g-load limiting can be initiated.

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Guidance shall be furnished in case of abort for landing site control and for entry g-load limiting ($g_e < 10 g$). The scheme must perform with wide variations in vehicle performance encountered during aborts up to orbital injection.

3.2.3.11.1.2 Earth Orbital Abort. - The G&N shall be capable of initiating an orbital abort from any one of three consecutive orbits to one of seven landing sites. Orbital abort will generally be accomplished in the same manner as a normal de-orbit maneuver.

3.2.3.11.1.3 Abort from Translunar Injection. - The G&N must maintain abort capabilities throughout the injection maneuver, and assure return to the primary landing site under non-critical abort conditions, or return to one of six alternate sites in case of catastrophic failures such as life support systems or solar flare. Abort translunar injection is a critical phase since aborts near cutoff result in very long flight times and possible passage through both Van Allen belts. Landing sites for abort from translunar injection must be selected carefully so as to satisfy entry corridor and propellant requirements.

3.2.3.11.2 Midcourse Abort. - The midcourse phase extends from translunar injection to the moon's sphere of influence. The guidance technique shall provide a smooth transition from normal midcourse operations to abort. The G&N system shall be capable of returning the CM to one of seven preselected recovery areas with the appropriate time of arrival for entry.

3.3 Equipment Interface Requirements

3.3.1 Design. - The G&N system shall be designed to assure physical and functional interface compatibility with other Apollo spacecraft systems, ground-based equipment, and the crew in all ground and in-flight modes of operation.

3.3.2 Electrical. - The G&N system shall operate within design limits specified in 3.3.2.1 for the spacecraft dc supply, or its simulated equivalent.

3.3.2.1 DC Power Distribution System. - The following characteristics apply to all primary system dc loads whether supplied from fuel cells, entry battery, or post-landing battery, in the normal or emergency operating mode. Values given are sensed at the load side of the isolating diodes.

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- (a) DC Steady State Voltage Limits — The dc steady state voltage limits shall be 25 to 30 volts.
- (b) DC Transient Voltage Limits — The dc transient voltage limits shall be 21 to 32 volts, with recovery to steady state conditions within 1.0 second.
- (c) DC Ripple Voltage Limits — The dc ripple voltage shall be 250 millivolts maximum peak to peak.

3.3.2.2 Ripple Voltage. - The G&N system shall not contribute in excess of 50 millivolts, peak to peak, to the total dc source ripple voltage content.

3.3.2.3 Negative Connection. - Negative power return terminals shall be isolated from the associate contractor's chassis and shall be grounded at one point only: the negative bus.

3.3.2.4 Umbilicals. - Use of electrical umbilicals on the CM for service to the G&N system shall be held to a minimum.

3.3.3 Thermal Control. - The G&N system shall operate within design performance limits for all mission phases and system modes with the following characteristics of system cooling provided at the interface between it and the environmental control system,

Navigation Base Assembly (NBA) Coolant Requirements

Inlet Temperature = $45^{\circ}\text{F} \pm 3^{\circ}\text{F}$

Flow Rate = 33 lbs/hr \pm 3 lbs/hr

Pressure Drop = 1.5 PSI max. between points 1 and 2
(see coolant routing schematic) at
33 lbs/hr and inlet temperature of 45°F .

3.3.3.1 Fluid Circulation. - Fluid circulation shall terminate during entry into the earth's atmosphere at approximately 100,000 feet.

3.3.3.2 Coolant. - Water/Glycol mixture of the following composition shall be used as the coolant:

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- (a) Glycol (conforming to Specification MIL-E-9500) - 130 parts by weight
- (b) Distilled Water - 70 parts by weight
- (c) Inhibitor No. 1 (tiethanolamine phosphate) - 5 parts by weight
- (d) Inhibitor No. 2 (sodium mercapto benzo thiozole) - 1 part by weight

3.3.3.3 Couplings. - Self-sealing, quick-disconnect type couplings shall be used on the navigation base only. The maximum number of self-sealing, quick-disconnect type couplings shall be three (3).

3.3.3.4 Cold-Plate. - The responsibility for providing cold-plates shall be as follows:

- (a) Navigation base integral cold-plate shall be supplied by the associate contractor.
- (b) All other cold-plates will be supplied by the principal contractor.

3.3.4 Telemetry. - The G&N system shall be required to supply measurements for the purpose of ground monitor and data acquisition by providing the appropriate information signals to the operational telemetry and its associated recording equipment. The information signals shall be transmitted in digital-serial, analog, and digital parallel format.

3.3.4.1 Digital-Serial. - The guidance computer shall transmit digital information to the telemetry system in serial format, and shall transmit this information under command of the telemetry system.

3.3.4.1.1 Signal Composition. - The computer output information shall be in the form of 40-bit words, transmitted serially, at a maximum rate of 50 words per second.

3.3.4.2 Analog. - Analog signals within the G&N system that are to be telemetered shall be signal-conditioned at the source and voltage-normalized to a range of 0.0 to 5.0 volts dc.

3.3.4.3 Digital-Parallel. - The G&N system shall generate digital-parallel, or event signals to indicate changes in system mode or status.

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3.3.4.3.1 Signal Composition. - The digital-parallel signals shall have two voltage ranges:

Signal range "1" + 3.5 to + 10v dc

Signal range "0" 0 ± .5 v dc

3.3.4.4 Telemetry Transmission. - The telemetry system shall process guidance and navigation into forms suitable for transmission to earth by radio link. Conditioned analog data shall be commutated and digitized.

3.3.5 Ground Support Equipment (GSE). - The G&N system shall be capable of being satisfactorily checked and monitored by automatic checkout equipment.

3.3.5.1 Carry On Equipment. - Carry on equipment shall be used for GSE in order to minimize umbilical requirements and to optimize G&N system pre-launch checkout.

3.3.5.2 Auxiliary Equipment. - The G&N system shall be capable of being maintained with a minimum of non-checkout auxiliary equipment.

3.3.6 Crew. - The interface between the crew and the G&N system shall not exceed the capabilities and limitations of one human operator for successful operation.

3.3.7 Accessibility. - All attachment fittings, wiring and tubing, supports, and accessory fastenings shall be easily accessible during the proper sequence of assembly of the G&N system into the spacecraft, and such attachments, fittings, and fasteners as shall require manipulation for the purpose of service, maintenance, or repair of the equipment shall be easily accessible to the astronauts, in any mode of dress, during flight and shall not require the use of special tools.

3.3.8 Stabilization and Control. - The G&N system shall provide total attitude signals, attitude error signals, minimum impulse signals, engine on commands, and mode switching signals to the stabilization and control system.

3.3.8.1 Total Attitude Signals. - The G&N system shall provide analog signals to the total attitude display on the flight director attitude indicator.

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The signals shall be proportional to the sine and cosine of the three stable platforms gimballed angles. The total attitude angles shall be in the spacecraft stability axes.

3.3.8.2 Attitude Error Signals. - The G&N system shall provide pitch, roll, and yaw attitude error signals to the stabilization and control system. The signals shall be available in both body and stability axes. The stabilization and control system shall select the appropriate axes to be used for spacecraft control. Provisions shall be made to switch these signals out in case of guidance equipment failure.

3.3.8.3 Minimum Impulse Signals. - The G&N system shall provide roll, pitch, and yaw minimum impulse signals to the stabilization and control system to accomplish fine control of the spacecraft. The signal shall be a dc voltage transmitted by a switch closure.

3.3.8.4 Engine On Commands. - The G&N system shall provide the stabilization and control system with an engine-on signal.

3.3.8.5 Mode Switching Signals. - As a function of mission requirements, the G&N system shall provide mode switching commands in the form of dc signals.

3.3.9 Displays and Controls. - The G&N system shall provide all controls and displays necessary for acceptable and accurate operation of the guidance and navigation equipment by the crew. All display and control elements shall be compatible with the crew interface requirements and limitations as specified in paragraph 3.3.6. The controls and displays necessary for the operation of the guidance and navigation equipment **will** be allotted an area within the lower equipment bay. In addition, an area **will** be allotted on the main display panel for such redundant controls and displays which are required for crew operation from the couch position.

3.3.10 Weight. - The weight of the guidance and navigation system shall not exceed 350 pounds. This weight shall be based upon the following items of the subsystem and shall include all intra package wiring, connectors, component hardware, encapsulating compounds where used, entrapped fluids, and environmental protection equipment and devices where required.

- (a) Coupling Display Unit (CDU)
- (b) Map and Data Viewer (MDV)

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- (c) Displays and Controls - AGC
- (d) Displays and Controls - NAV
- (e) Eyepieces
- (f) Tracker
- (g) Spares
- (h) Displays and Controls (Main Panel)
- (i) Inertial Measurement Unit (IMU)
- (j) Navigation Base (NVB)
- (k) Sextant (SXT)
- (l) Scanning Telescope (SCT)
- (m) Computer (AGC)
- (n) Power and Servo Assembly (PSA)
- (o) Junction Boxes (JBX)

3.3.11 Environment. - The G&N system shall operate within design limits under normal and emergency environmental conditions, and shall comply with test requirements to be supplied when available.

3.3.12 Master Clock Synchronizing Pulse. - The master clock for the Apollo spacecraft shall be located within the G&N system computer package. The master clock shall provide the spacecraft central timing equipment with a synchronizing signal in the form of a pulse train with a frequency of 512 KC. This signal shall be available at all times and is not dependent upon the operation of the G&N computer or system.

3.3.13 Display Lamps. - A section of the G&N display panel on the lower equipment bay and a section of the main display panel will be allocated for lamps to be energized by signals from the G&N system. These status, caution, and failure lamps shall indicate conditions within the G&N system.

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In addition two display lamps shall be provided at the G&N display panel on the lower equipment bay which will be activated by spacecraft systems other than the G&N system.

3.3.14 In-Flight Test System. - The principal contractor will provide an on-board in-flight test system for diagnosis and isolation of malfunctions within the G&N system and to support in-flight maintenance. The G&N system shall supply hard-wired conditioned signals and soft-wired unconditioned signals for use by the inflight test system.

3.3.14.1 Hard-Wired Signals. - The hard-wired signals shall be used to indicate the status of the system and shall permit gross diagnosis of system performance and malfunctions.

3.3.14.2 Soft-Wired Signals. - The soft-wired signals shall be accessible with a test probe and shall enable the crew to isolate malfunctions within the G&N system to the smallest replaceable component.

3.3.15 Electromagnetic Interference. - The design of the G&N system shall incorporate principles consistent with minimizing the generation of, and susceptibility to, electromagnetic interference. The associate contractor shall coordinate with the principal contractor in establishing guidance and navigation module and system grounding and shielding concepts, and shall participate in a NASA-approved program to test, report, and control electromagnetic interference. These principles shall also extend to the ground support equipment.

3.3.16 Leakage Rates

3.3.16.1 Internal Leakage Rates. - The G&N associate contractor shall supply to the principal contractor a list of all hermetically sealed components, the composition of the pressurization gas, the pressurization level, and the leakage path. The principal contractor will assign to each component a maximum allowable leakage rate internal to the CM.

3.3.16.2 External Leakage Rates. - The G&N system associate contractor shall supply to the principal contractor those components required to penetrate the CM pressure bulkhead, including information regarding the size of the penetration and the method and material used to maintain the pressure integrity of the CM. The principal contractor will assign to each component a maximum allowable leakage rate external to the CM.

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4. QUALITY ASSURANCE

4.1 Reliability. - The detailed reliability of the interfaces of the G&N system shall be determined at a later date.

5. PREPARATION FOR SHIPMENT

Not applicable to this interface specification.

6. NOTES

6.1 Definitions

6.1.1 Operational Condition. - Criteria for a satisfactory "operational condition" shall be established when the G&N system is operating in a realistic "worst case" environment as specified for the following conditions:

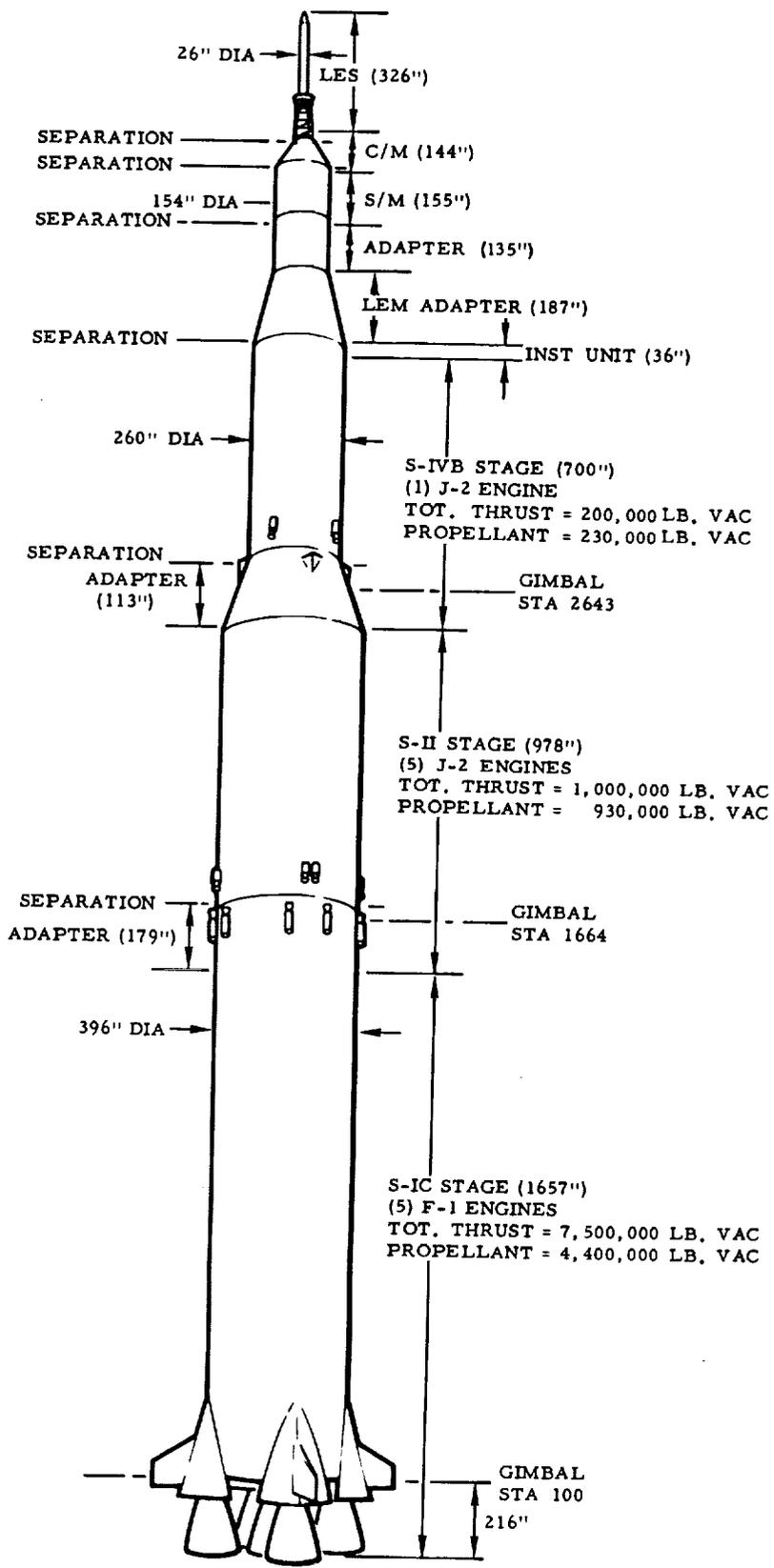
- (a) Temperature
- (b) Humidity
- (c) Pressure
- (d) Fungus Growth
- (e) Atmospheric Chemical Composition*
- (f) Acceleration
- (g) Shock
- (h) Acoustic Noise
- (i) Radiation
- (j) Radio Frequency Interference

* Atmosphere depends upon the particular mission phase that is being simulated.

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Operation in a realistic "worst case" environment should be proven by the experimental results of tests completed on a sufficiently large number of G&N systems relevant to each mission phase. The constraints are imposed by each particular mission phase to establish a specified probability of mission phase success due to proper operation of the G&N system. These tests are performed before prelaunch checkout.

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SATURN C-5 4318" (359-83')

NOT TO SCALE

Figure 1. Boost Configuration





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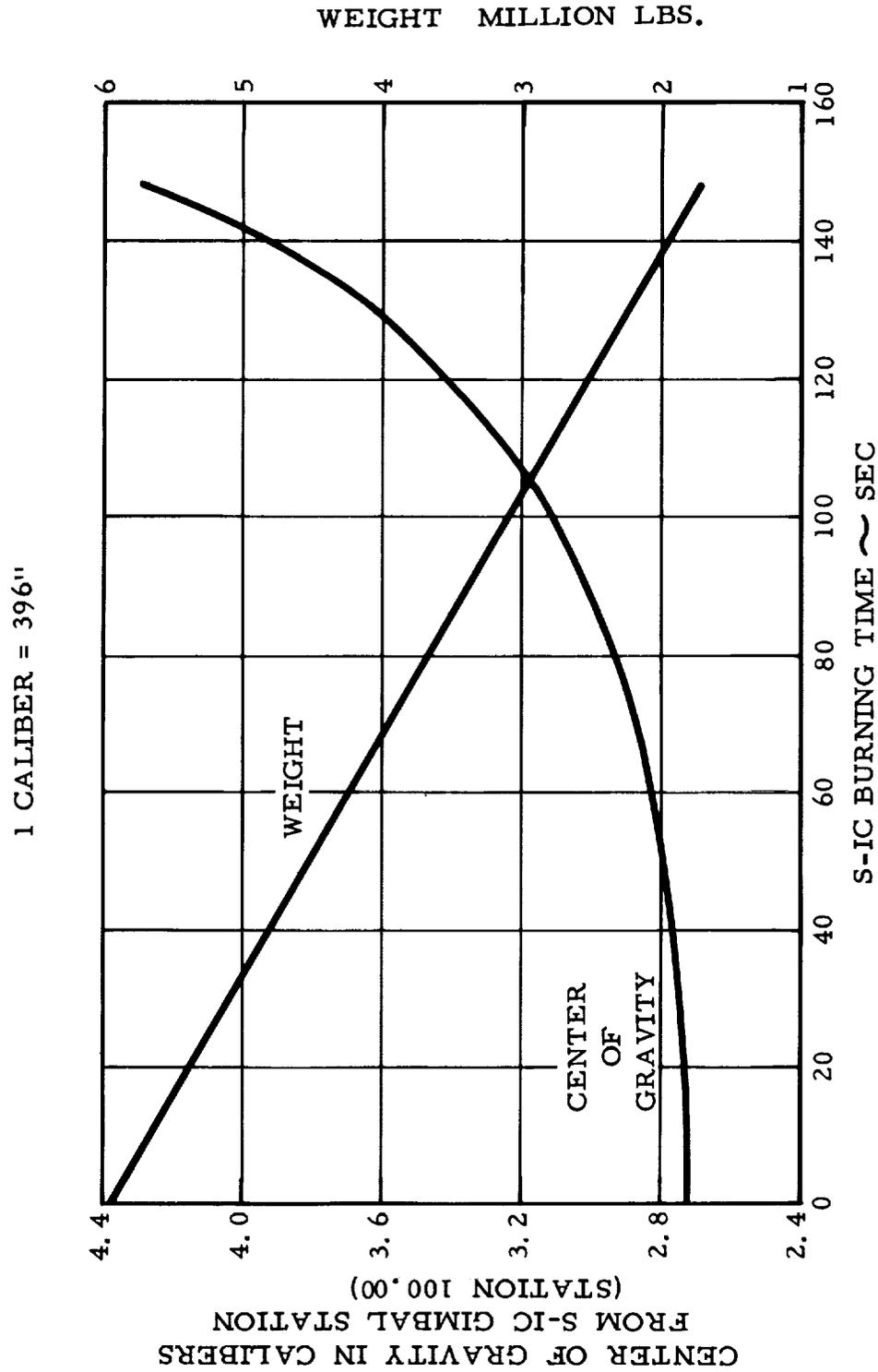


Figure 2. Weight and Center of Gravity During First Stage Boost

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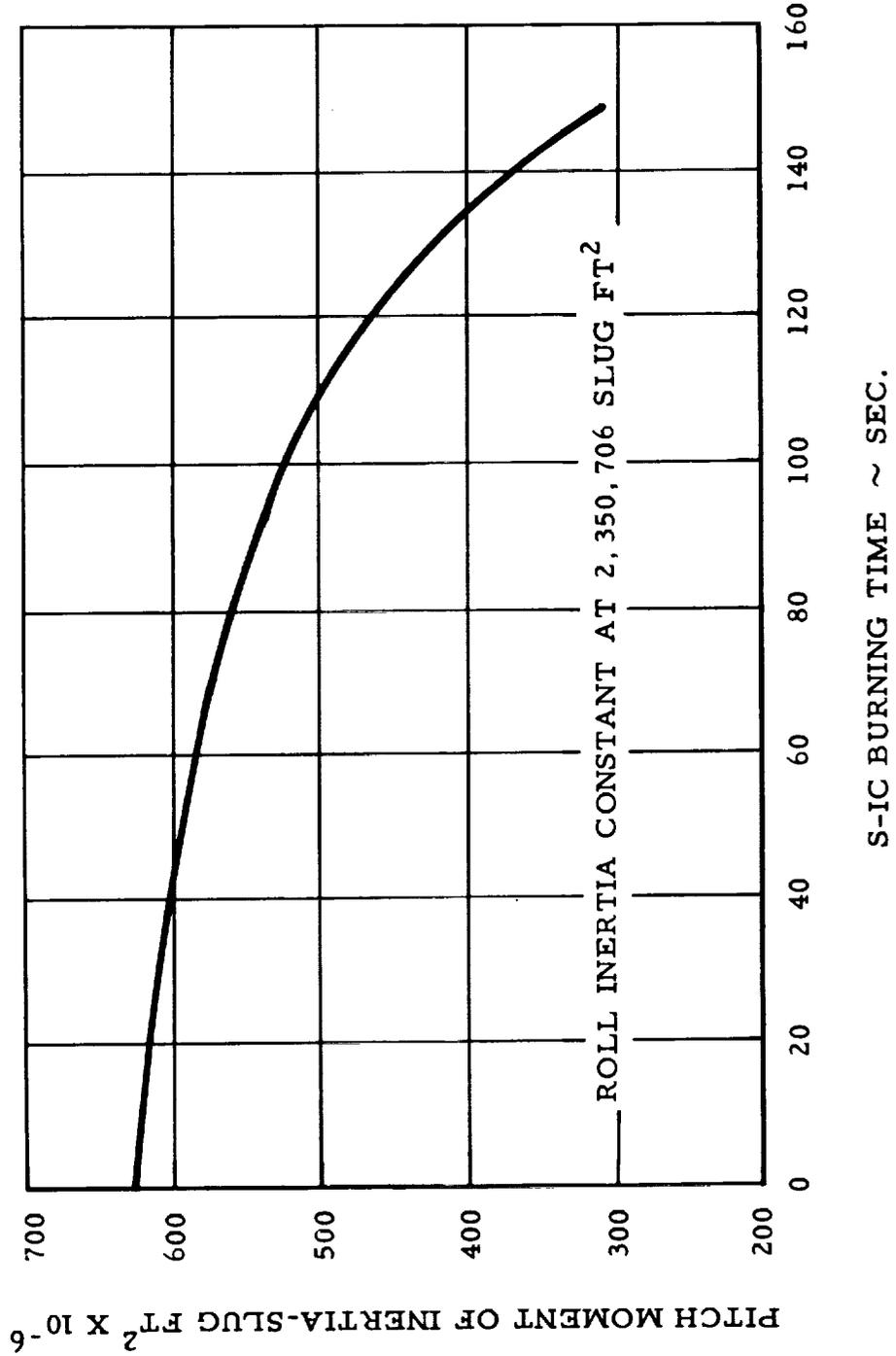


Figure 3. Moments of Inertia During First Stage Boost

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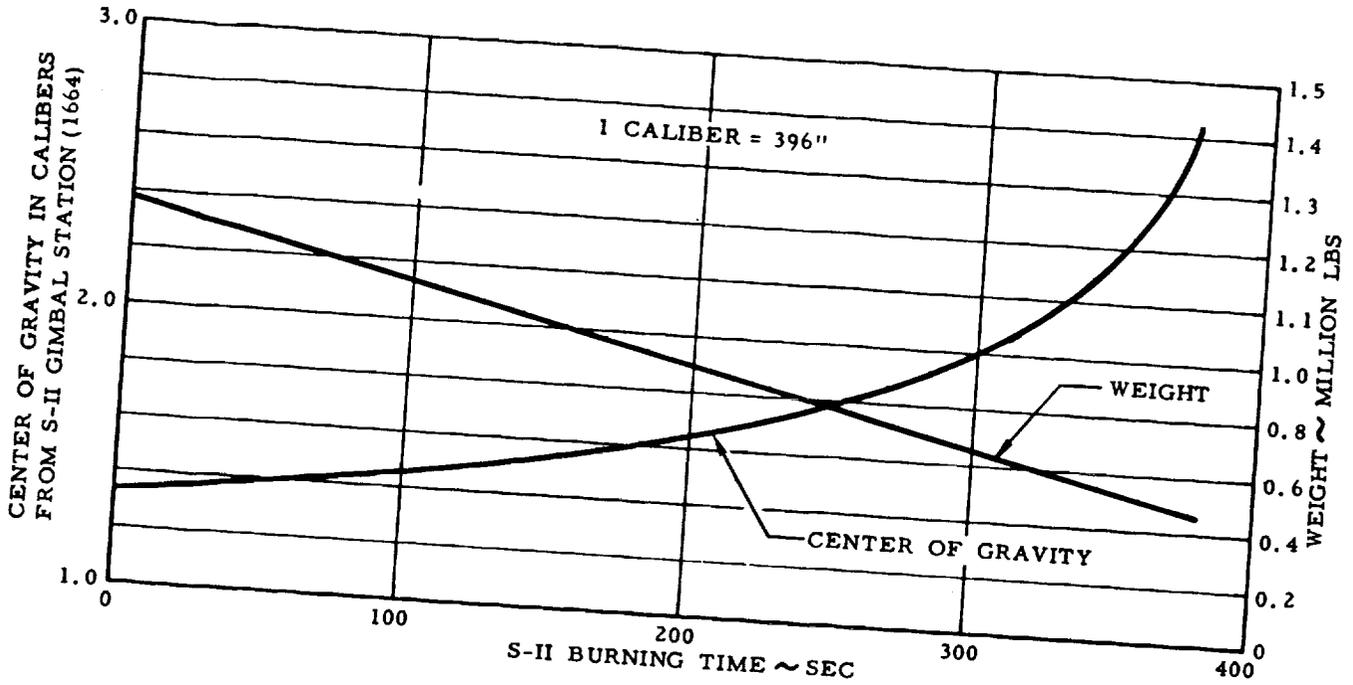


Figure 4. Weight and Center of Gravity During Second Stage Boost



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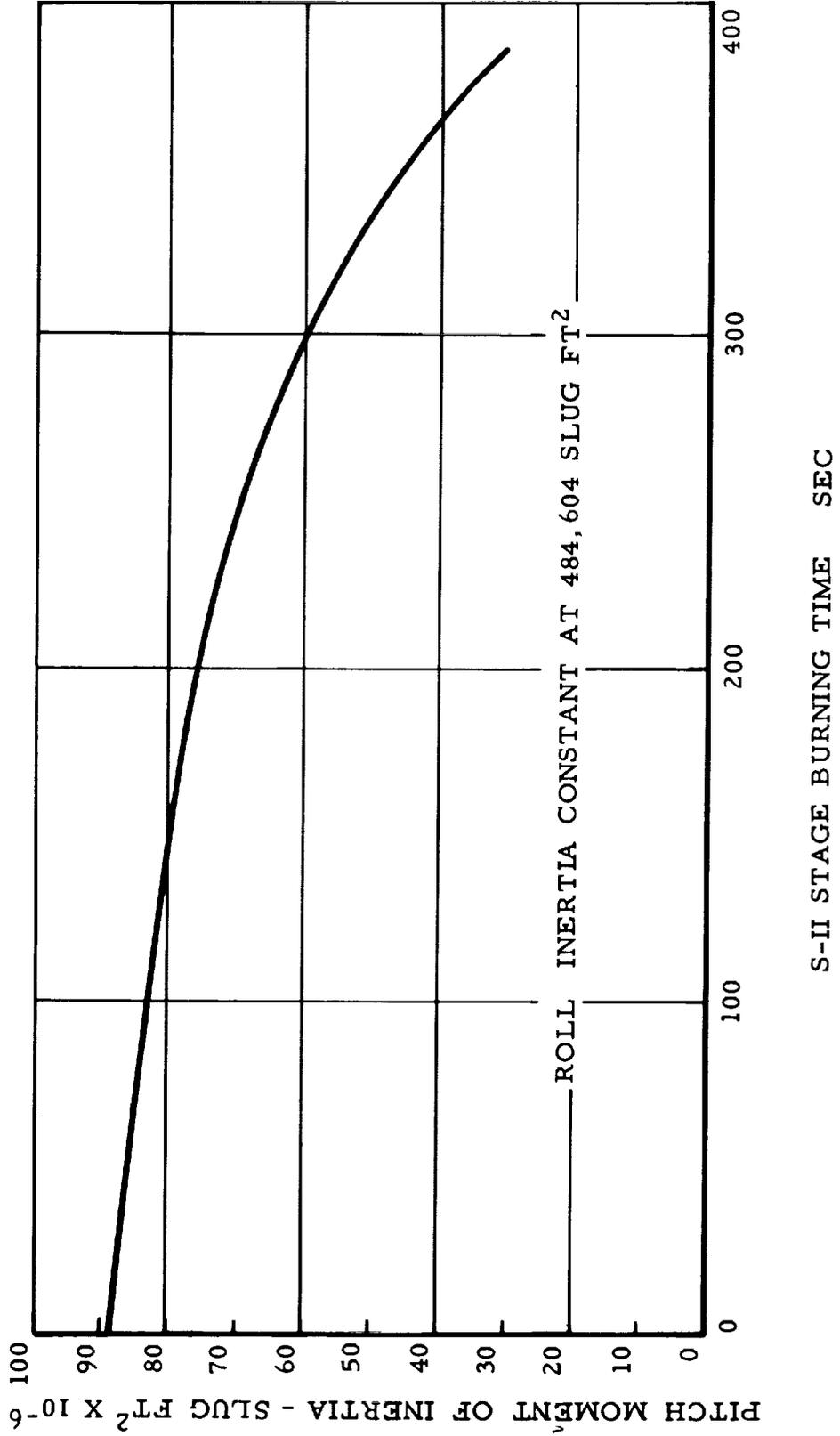


Figure 5. Moments of Inertia During Second Stage Boost

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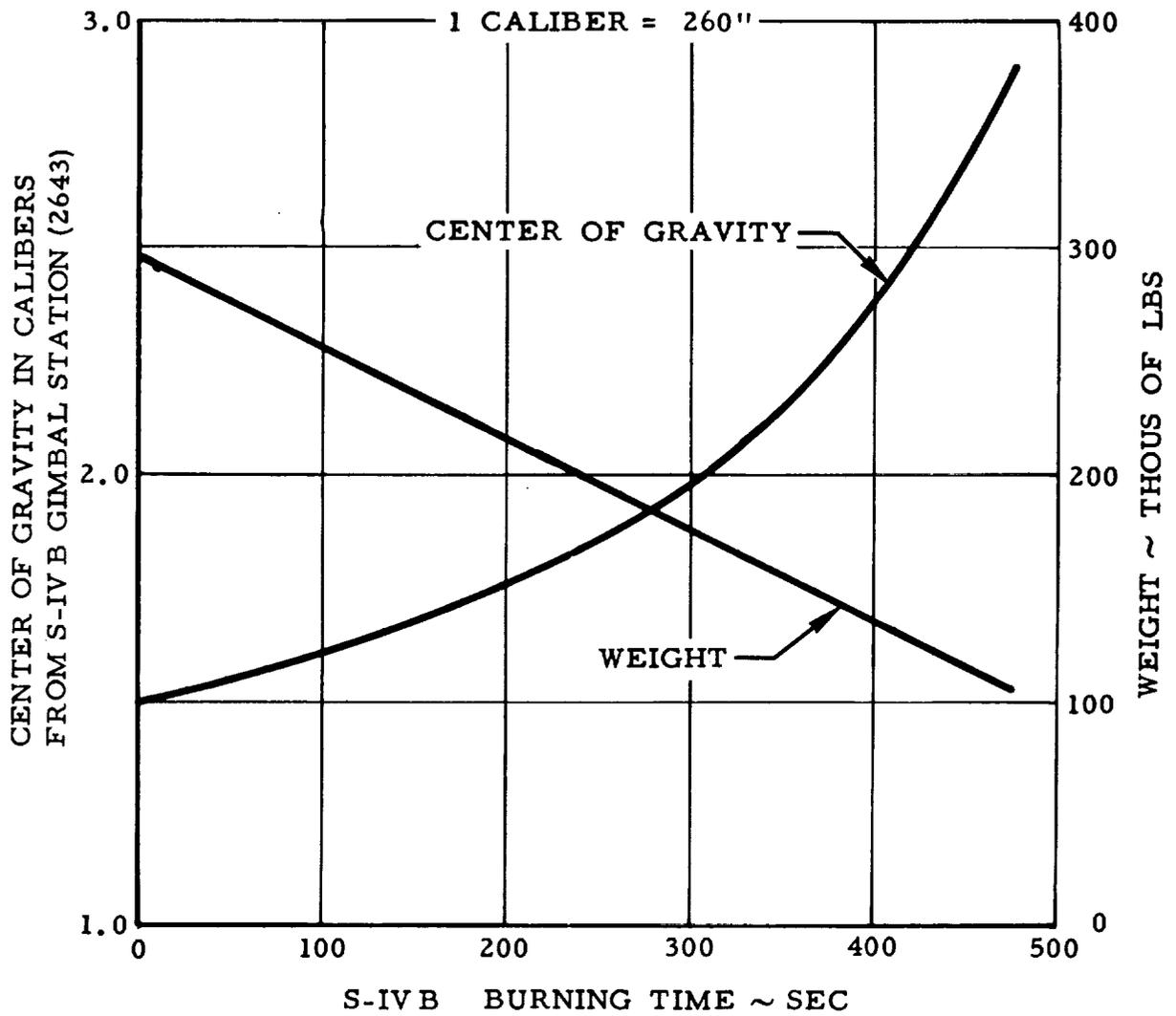


Figure 6. Weight and Center of Gravity During S-IVB Boost

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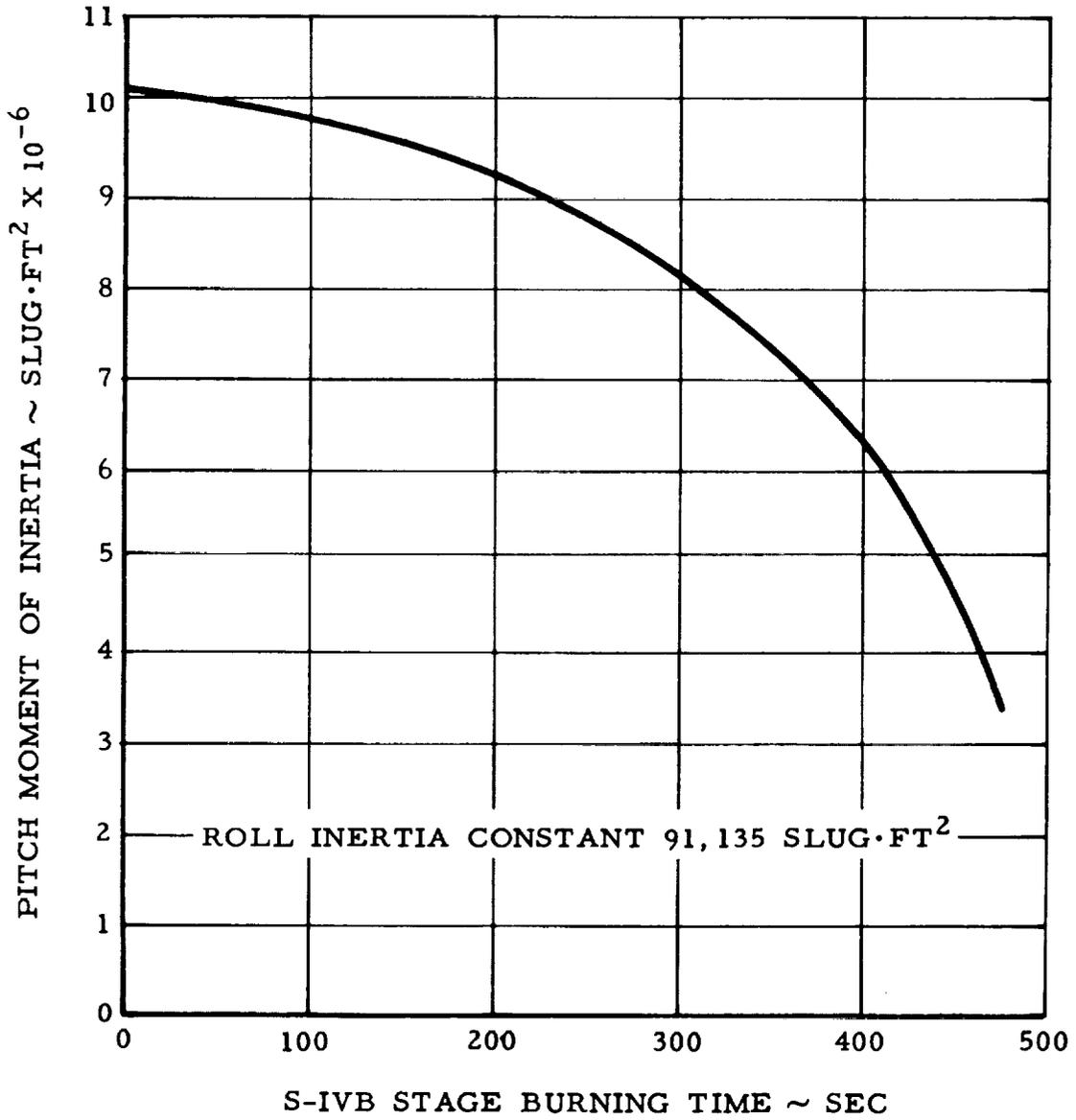


Figure 7. Moments of Inertia During S-IVB Boost

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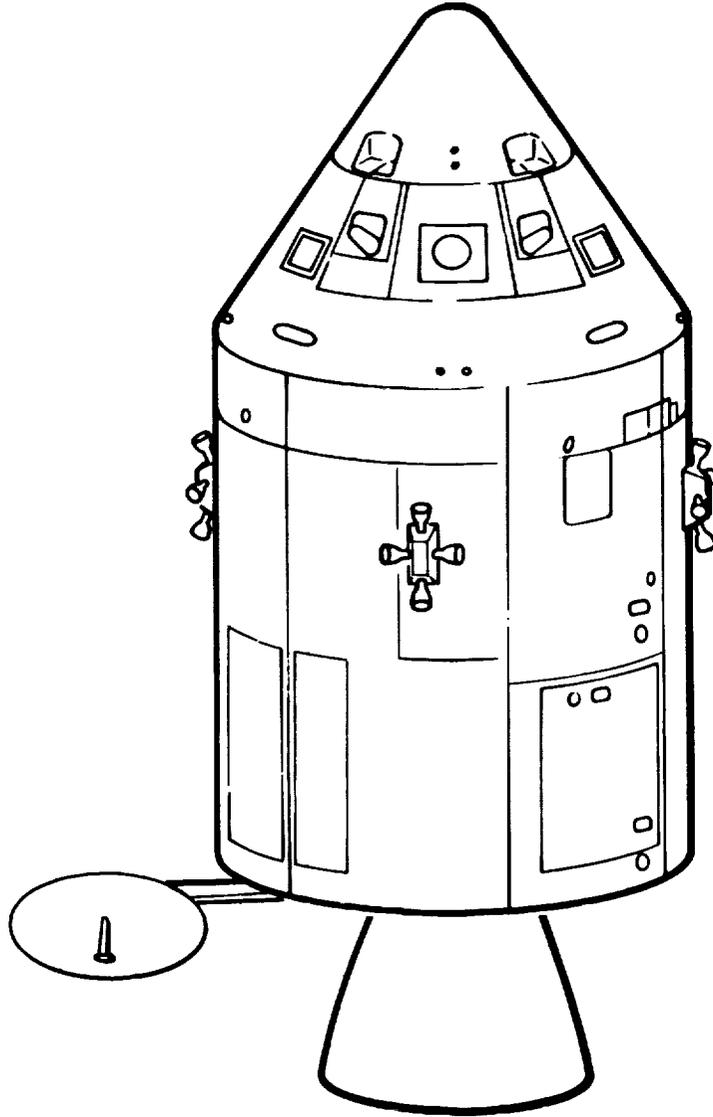
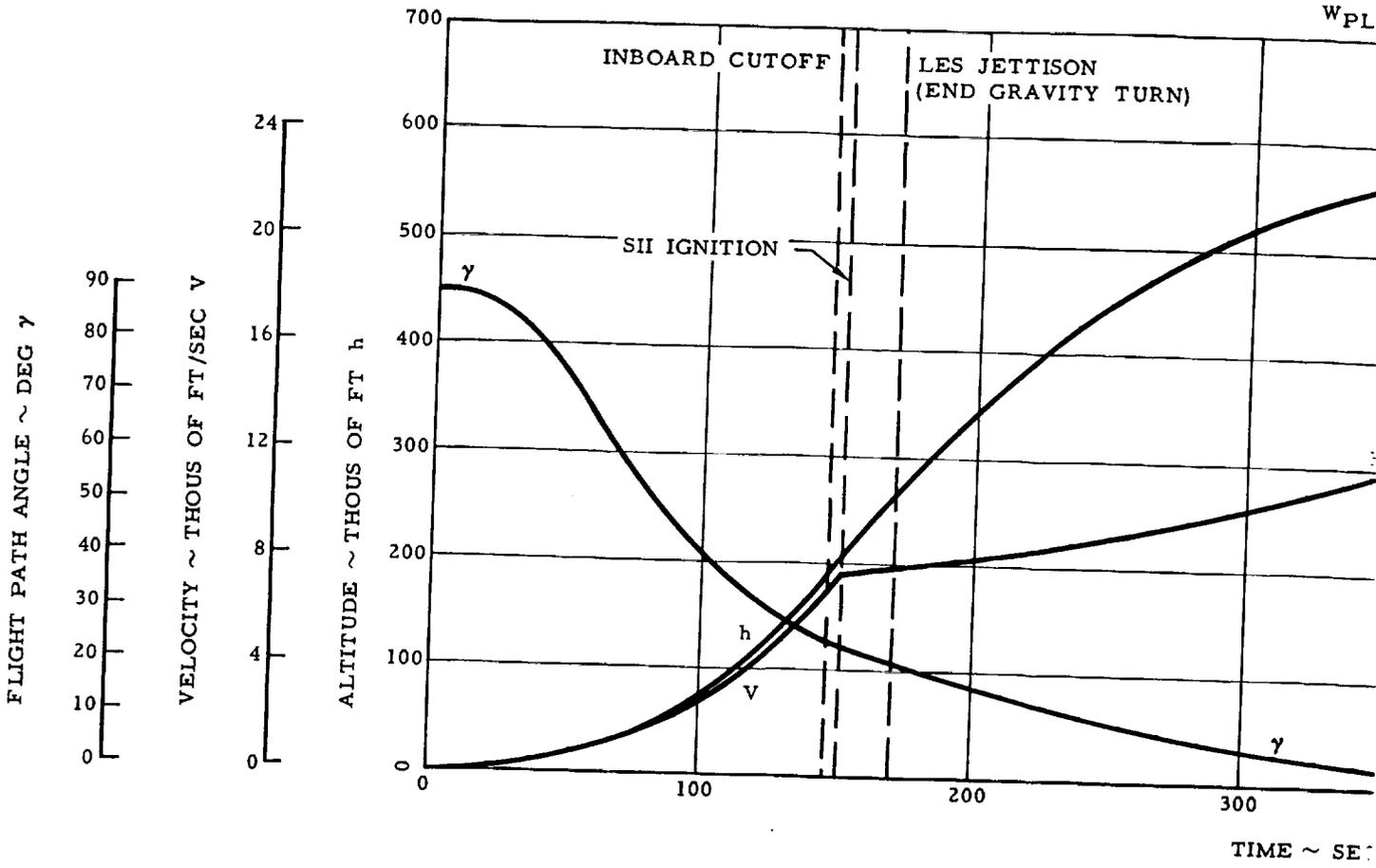


Figure 8. Transearth Configuration

C-5 THREE STAGE
VIA 100 N.
(LIFTOFF T
WPL





C-5 THREE STAGE TO TRANSLUNAR INJECTION
 VIA 100 N. MI. PARKING ORBIT
 (LIFTOFF TO ORBIT INJECTION)
 $W_{PL} = 91,525$ LBS

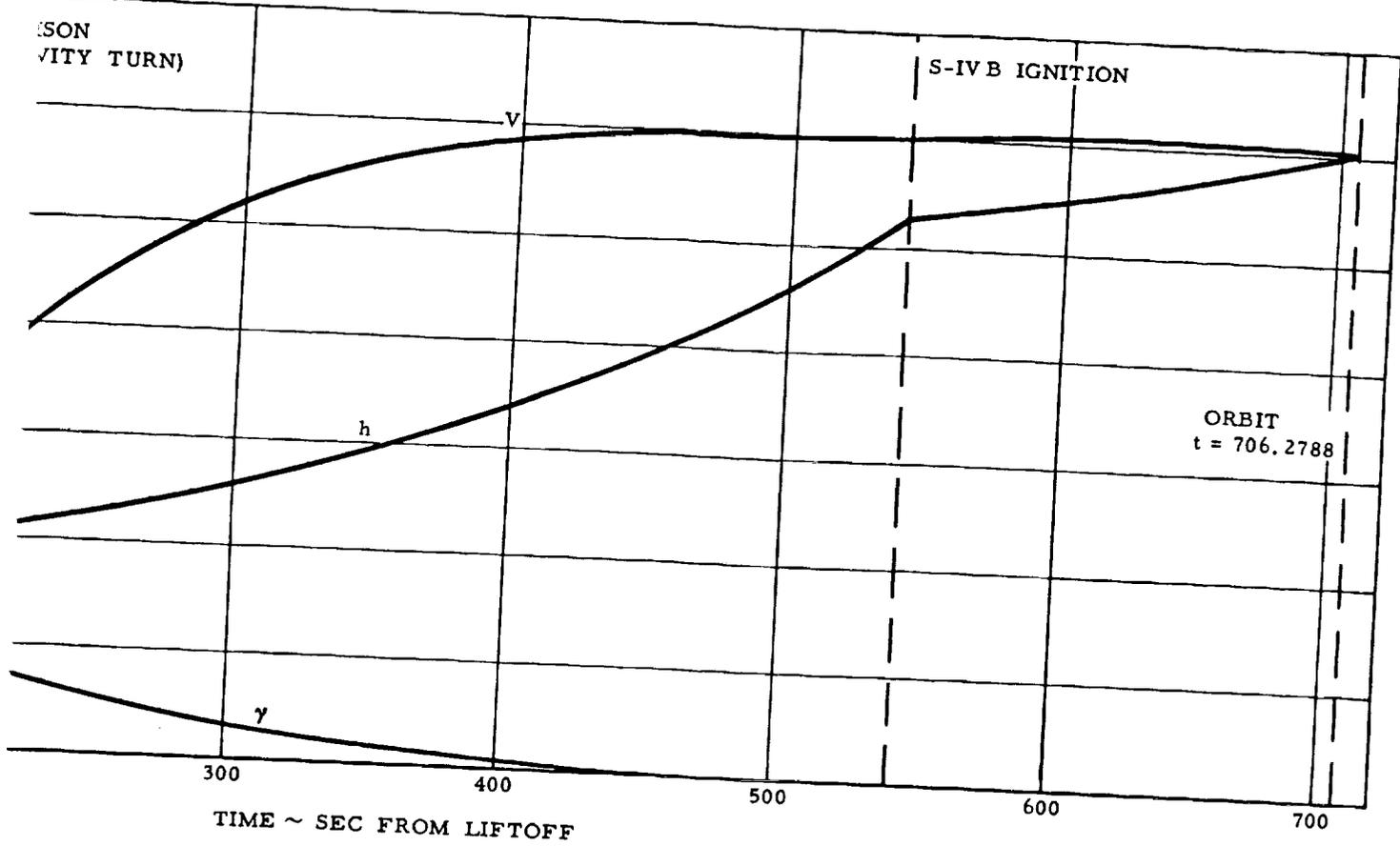


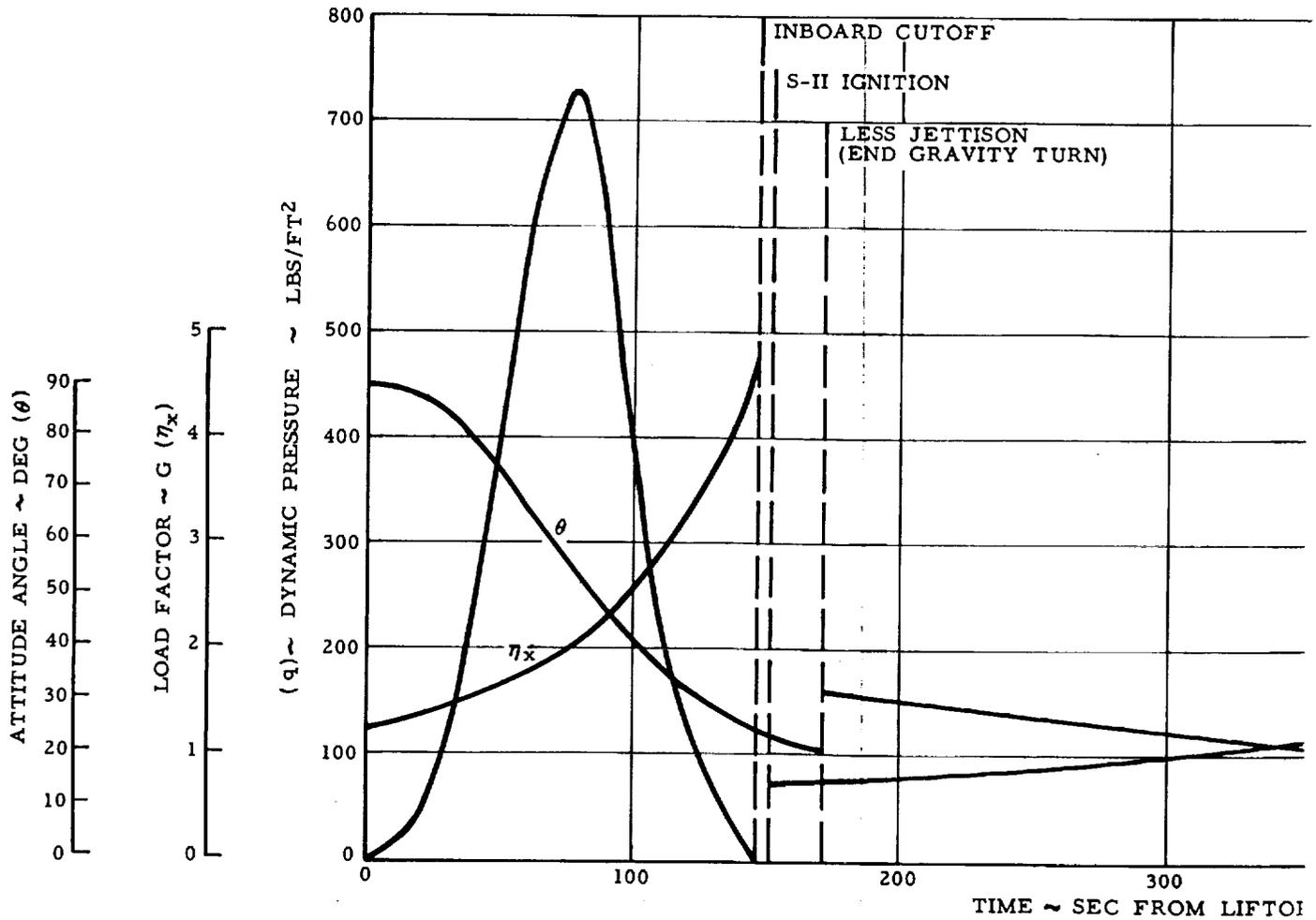
Figure 9. Flight Parameters for C-5 Boost to Parking Orbit - Velocity, Flight Path Angle, and Altitude vs Time

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C-5 THREE STAGE TO TRAN.
VIA 100 N. MI. PARF
(LIFTOFF TO ORBIT
 $W_{PL} = 91,525$ L





C-5 THREE STAGE TO TRANSLUNAR INJECTION
 VIA 100 N. MI. PARKING ORBIT
 (LIFTOFF TO ORBIT INJECTION)
 WPL = 91,525 LBS

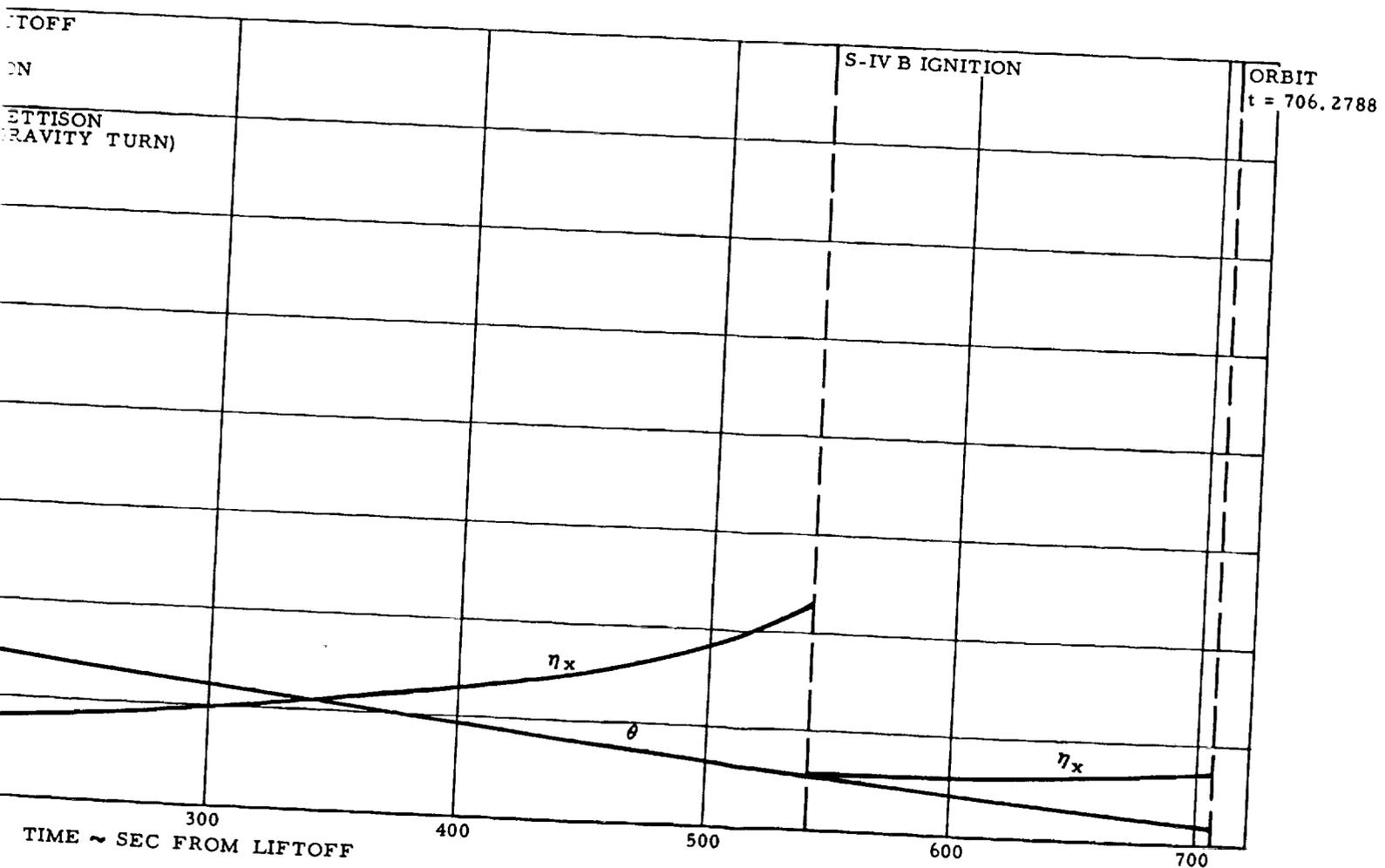


Figure 10. Flight Parameter for C-5 Boost to Parking Orbit - Attitude Angle, Load Factor, and Dynamic Pressure vs Time

2



C-5 THREE STAGE TO TRANSLUNAR ORBIT
VIA 100 N MI. PARKING ORBIT
(LIFTOFF TO ORBIT)
 $W_{PL} = 91,525$ LBS.

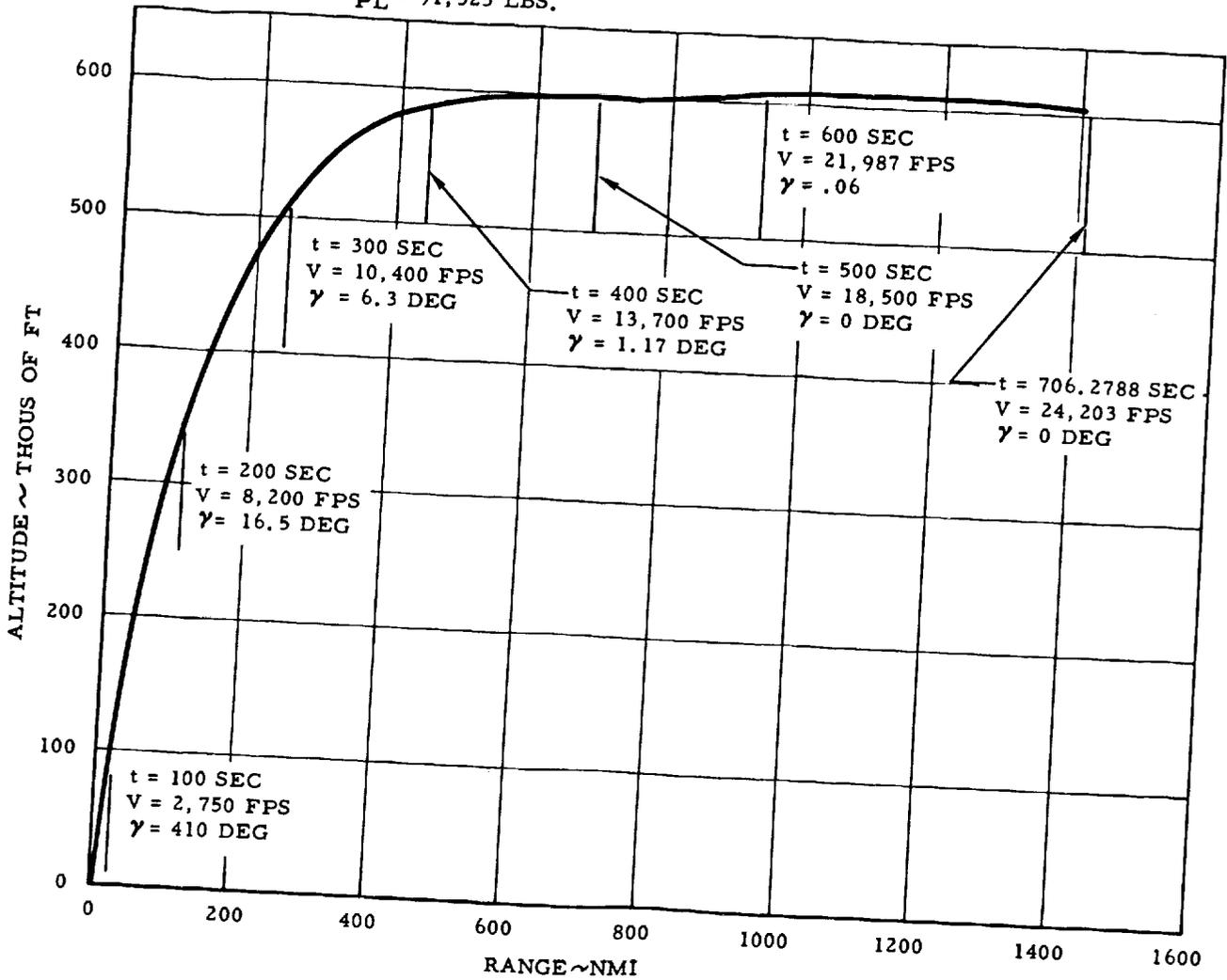
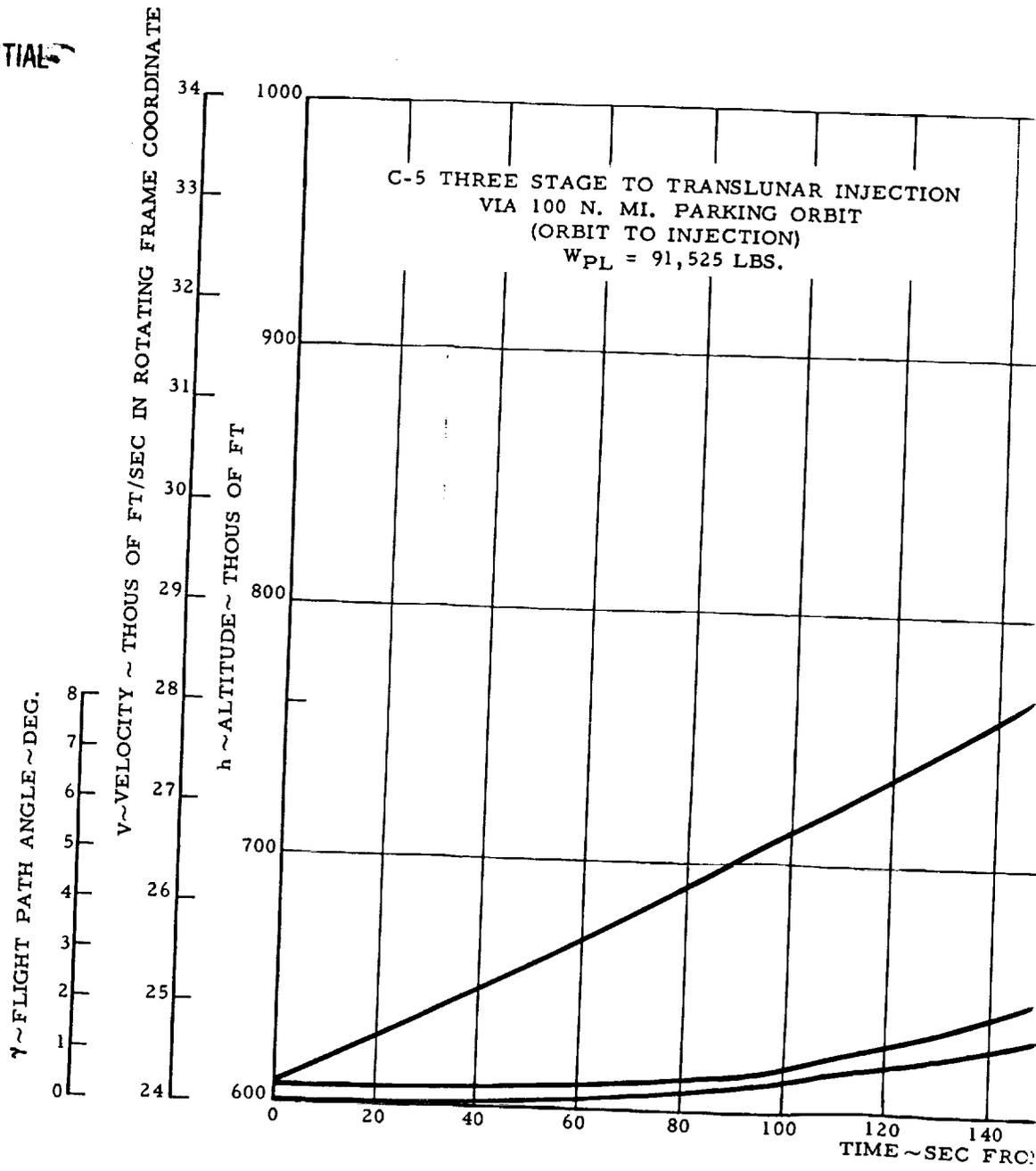


Figure 11. Boost Trajectory Profile for C-5 Boost to Parking Orbit

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34,277 FT/SEC

TRANS-LUNAR INJECTION
(PARKING ORBIT
INJECTION)
1,525 LBS.

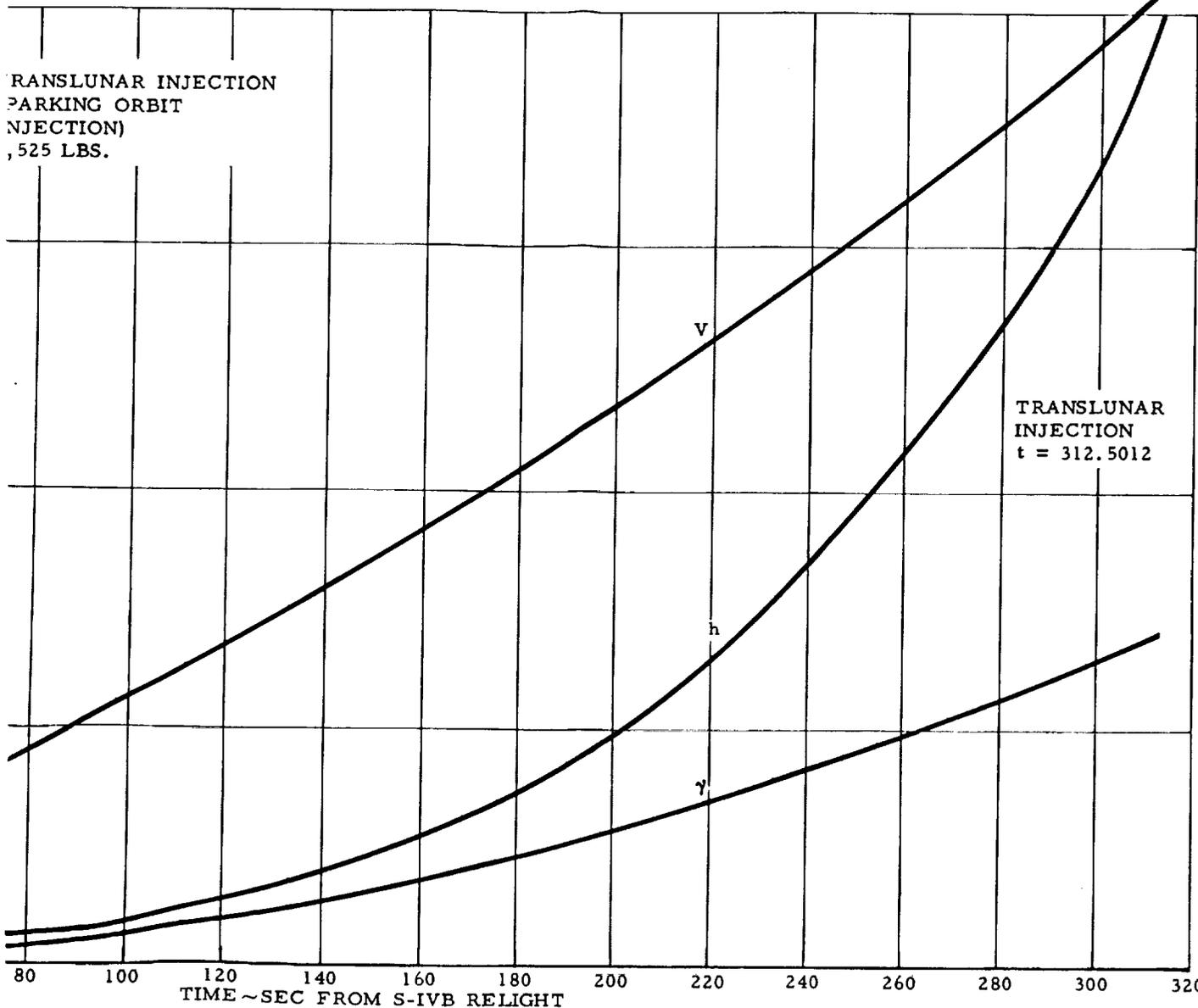


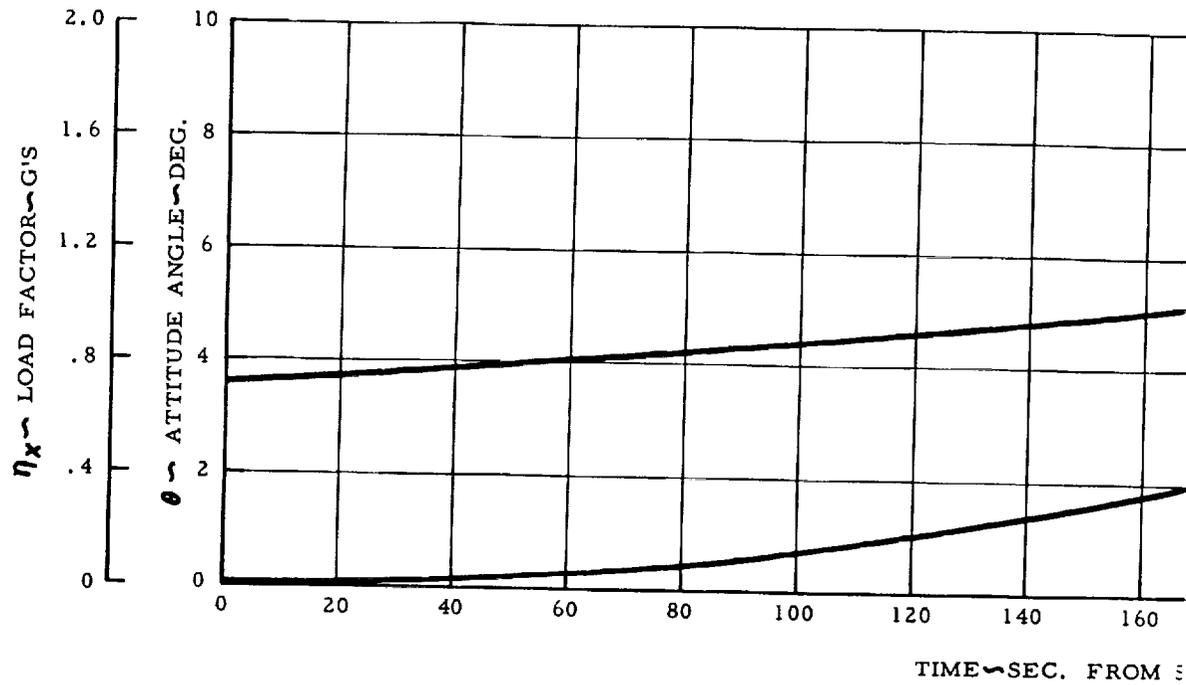
Figure 12. Flight Parameter for Translunar Injection - Flight Path Angle, Velocity, and Attitude Vs Time.

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C-5 THREE STAGE TO TRANSLUNAR INJECTION
VIA 100 N. MI. PARKING ORBIT
WPL = 91,525 LBS.
(ORBIT TO INJECTION)



63.1



O TRANSLUNAR INJECTION
II. PARKING ORBIT
91,525 LBS.
O INJECTION)

TRANSLUNAR
INJECTION
t = 312.5012

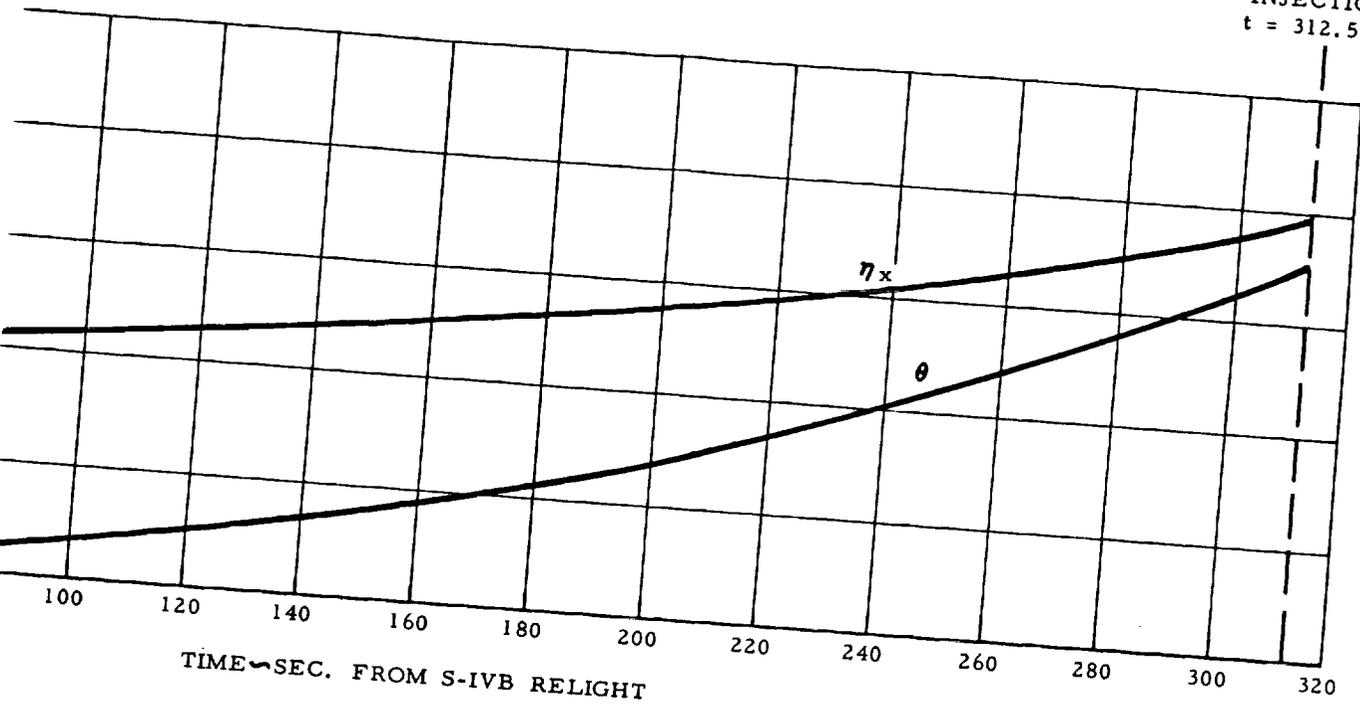


Figure 13. Flight Parameters for Translunar Injection - Load Factor And Attitude Angle Vs. Time



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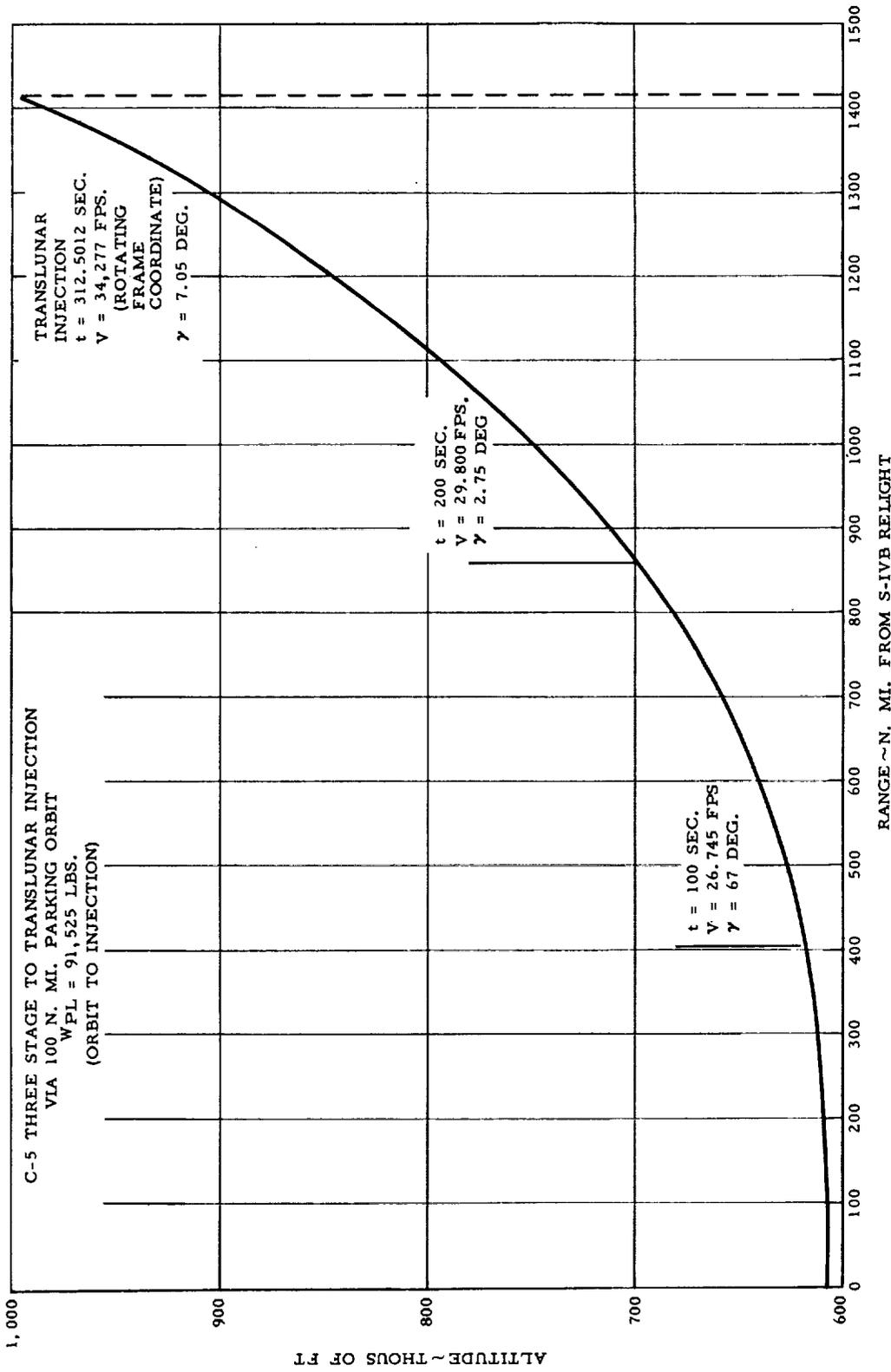


Figure 14. Translunar Injection Maneuver Profile

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S/M I_{SP} = 319.5 SEC
 T = 21,900 LB
 T/W_O = .258954
 T/W_F = .35063

TABULATION OF FLIGHT PARAMETERS

t	v	FLIGHT PATH ANGLE γ DEG.	BODY ATTITUDE α DEG.	STEERING ANGLE α DEG.
1.	8336.3	-7.9995	166.58	174.58
2.	8223.1	-7.2688	167.70	174.29
3.	8064.0	-6.3055	169.20	175.51
4.	7933.6	-5.5691	170.34	175.96
5.	7630.0	-4.0337	172.97	177.00
6.	7312.0	-2.6844	175.44	178.13
7.	6979.5	-1.5454	177.80	179.36
8.	6473.6	- .3304	181.01	181.34
9.	5896.4	+ .2917	184.19	183.90
10.	5284.16	0.0	187.00	187.00
a.	8325.8	-8.7555		
b.	8336.3	-7.9995		
c.	8346.0	-7.2437		
d.	8359.5	-6.0353		
e.	8370.5	-4.8275		
f.	8379.0	-3.6201		
g.	8387.2	-1.0899		
h.	8390.0	0.00		
i.	8387.2	+1.8099		
j.	8375.0	+4.2338		

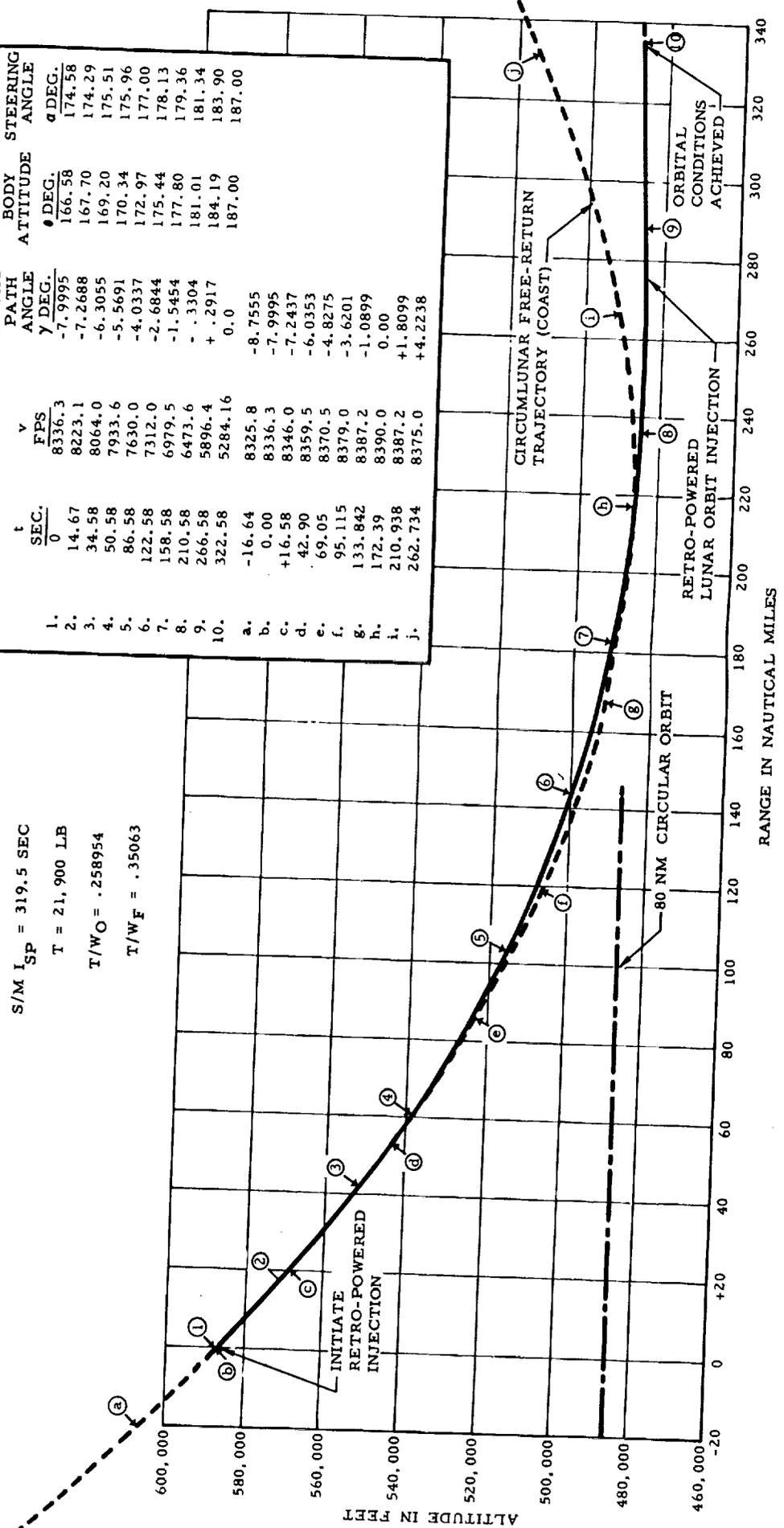


Figure 15. Lunar Orbit Injection Profile

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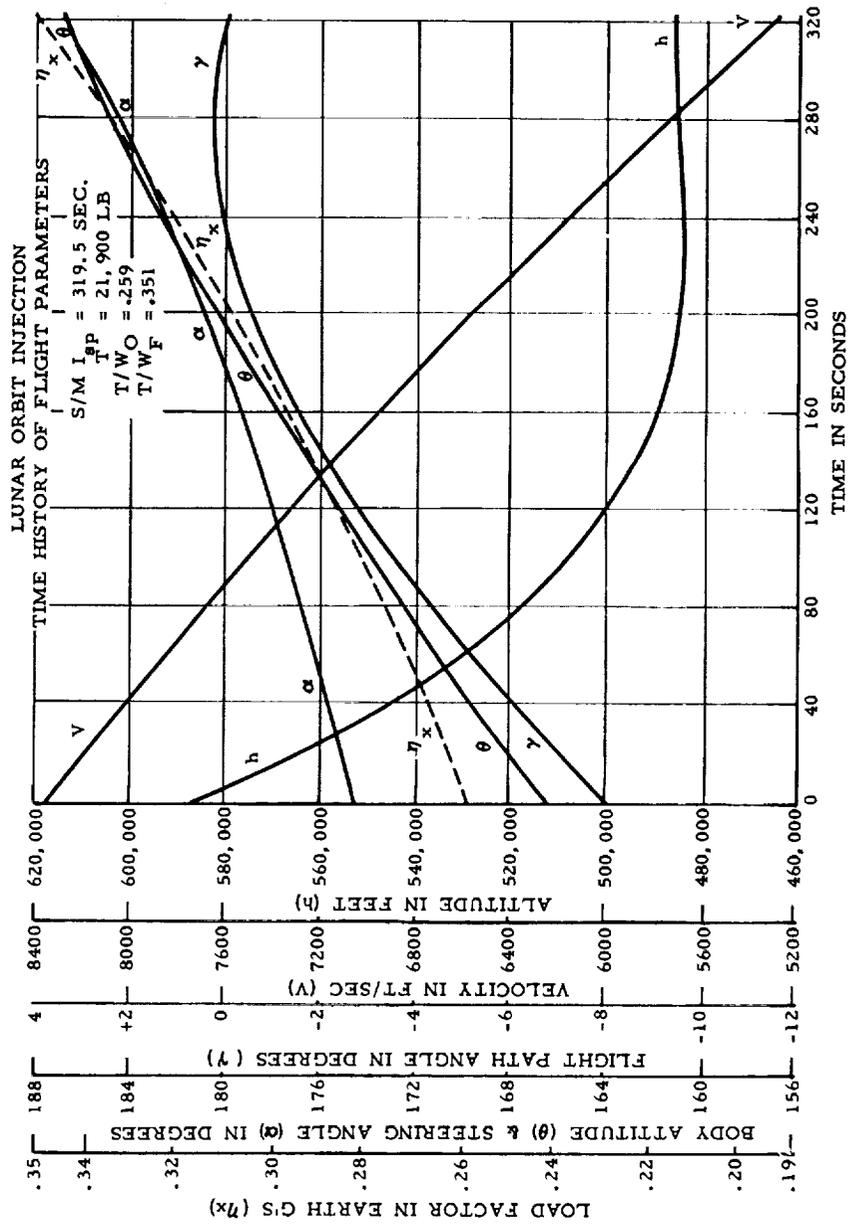


Figure 16. Flight Parameters for Lunar Orbit Injection

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LEM KICK INTO EQUAL-PERIOD ELLIPTICAL ORBIT
PERILUNE ALTITUDE = 50,000 FT
ALTITUDE - RANGE PROFILE

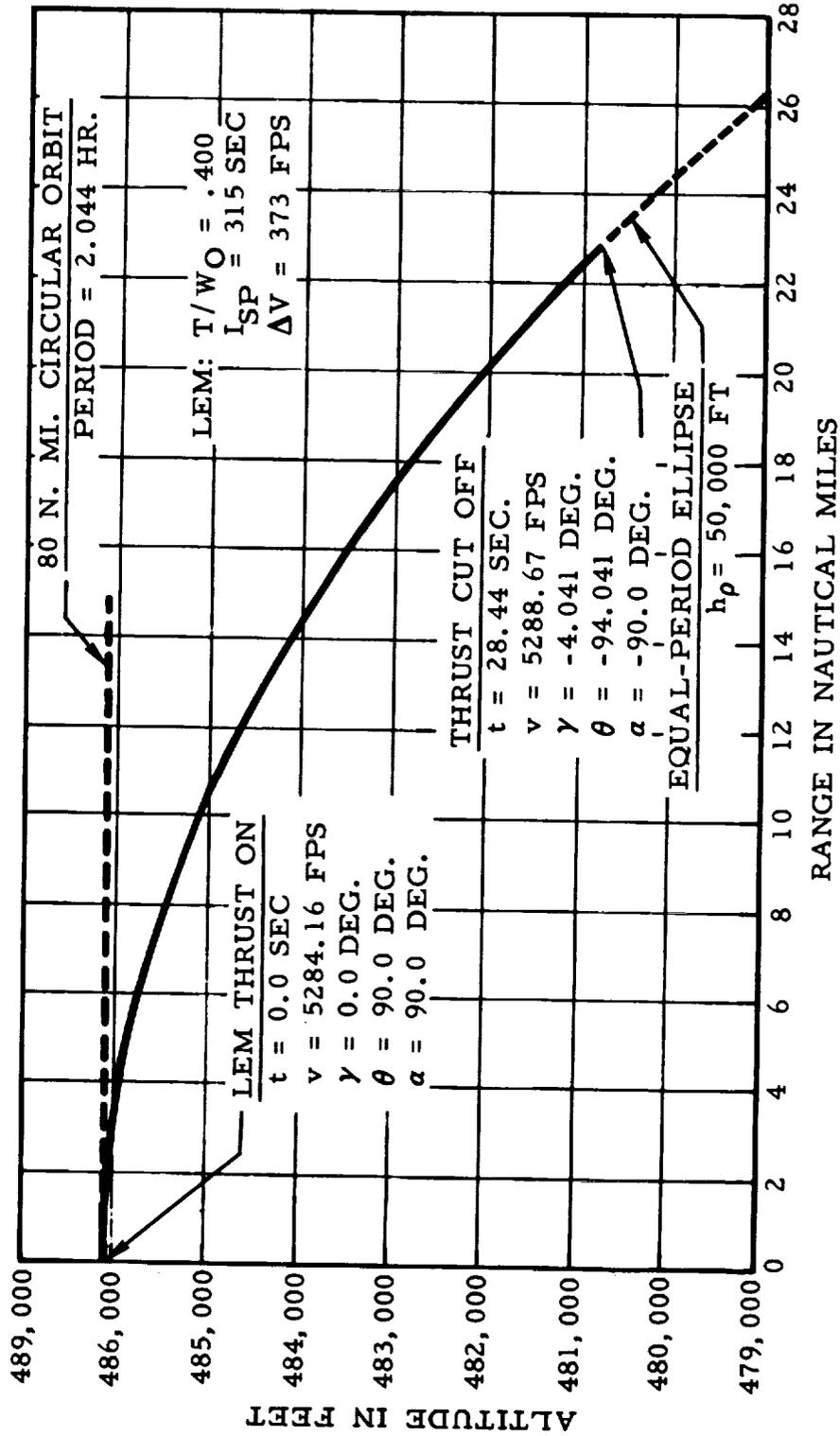


Figure 17. LEM Equal - Period Orbit Injection Profile

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LEM KICK INTO EQUAL - PERIOD ELLIPTICAL ORBIT
PERILUNE ALTITUDE = 50,000 FT

PARAMETRIC TIME HISTORY

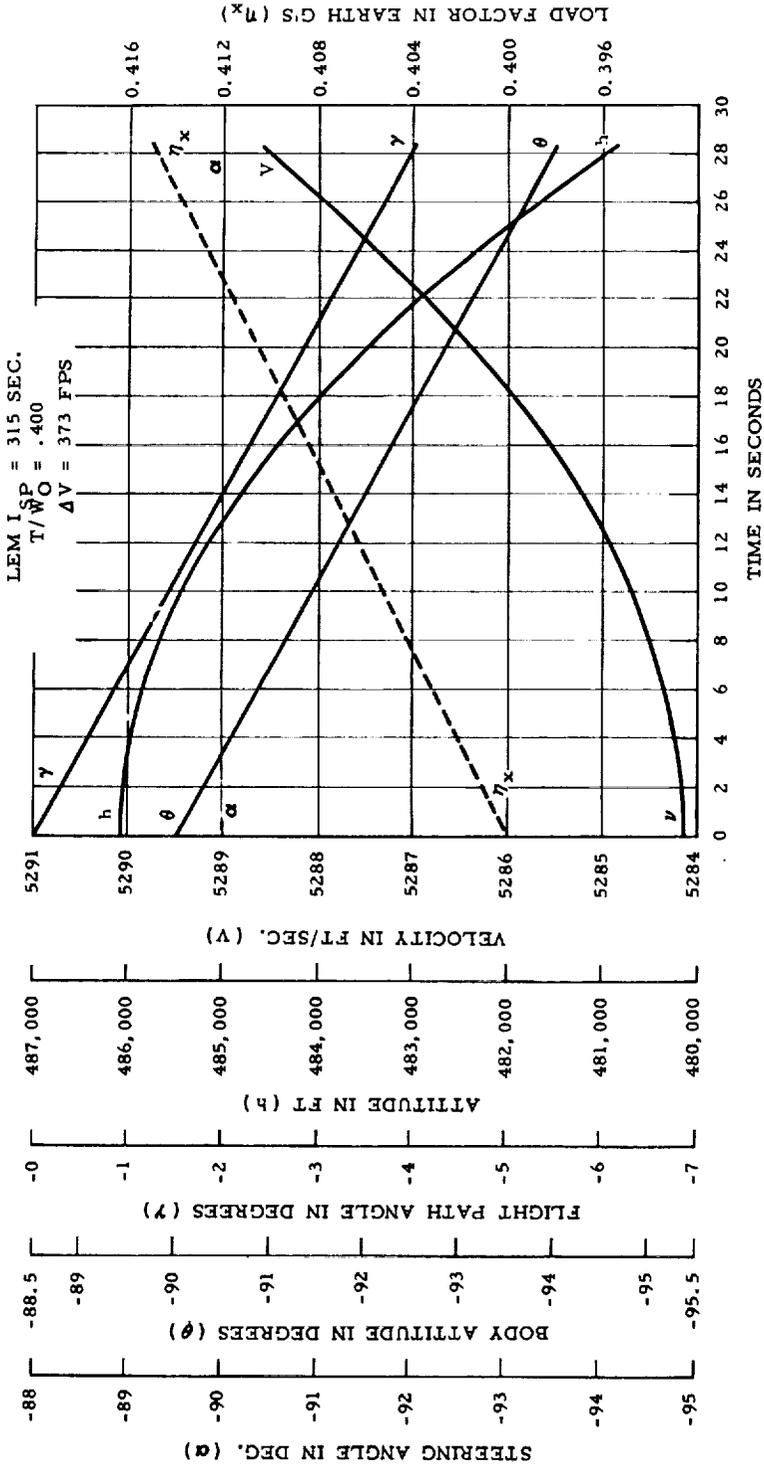


Figure 18. Flight Parameters for LEM Equal-Period Orbit Injection

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KEY TO FLIGHT PARAMETERS

	<u>t</u>	<u>v</u>	<u>γ</u>	<u>θ</u>	<u>α</u>
	<u>SEC.</u>	<u>FPS</u>	<u>DEG.</u>	<u>DEG.</u>	<u>DEG.</u>
1	0.0	5671	0.0	183.9	183.9
2	78.0	4565	-1.0	179.0	180.0
3	140.0	3610	-2.4	174.4	176.8
4	192.0	2740	-4.6	170.0	174.6
5	234.8	1990	-7.0	166.5	173.5
6	277.5	1220	-11.0	162.5	173.5
7	344.8	10	-90.0	90.0	180.0

LEM I_{SP} = 315 SEC

T/W_0 = .415

ΔV = 5895 FPS

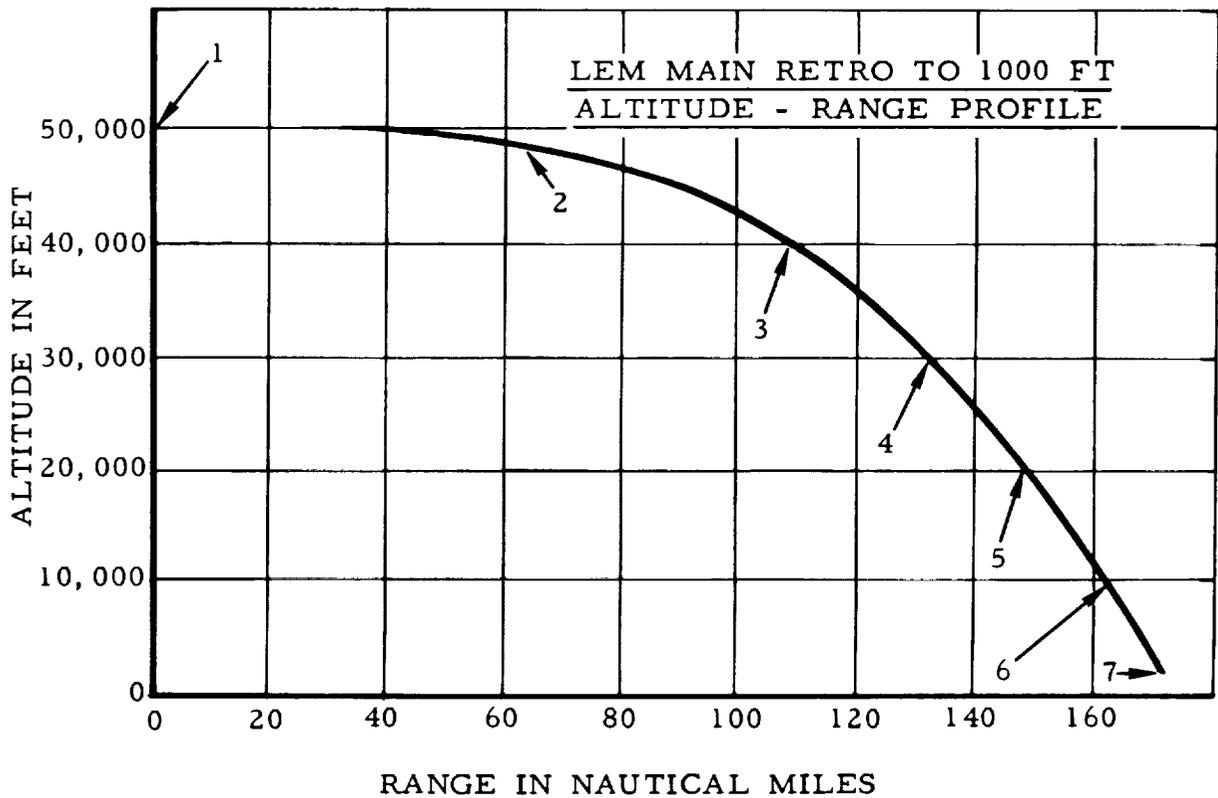


Figure 19. LEM Main Retro Maneuver Flight Profile

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LEM MAIN RETRO TO 1000 FT ALTITUDE
PARAMETRIC TIME HISTORY

LEM V_{SP} = 315 SEC.
 T/W_0 = .415
 ΔV = 5895 FPS

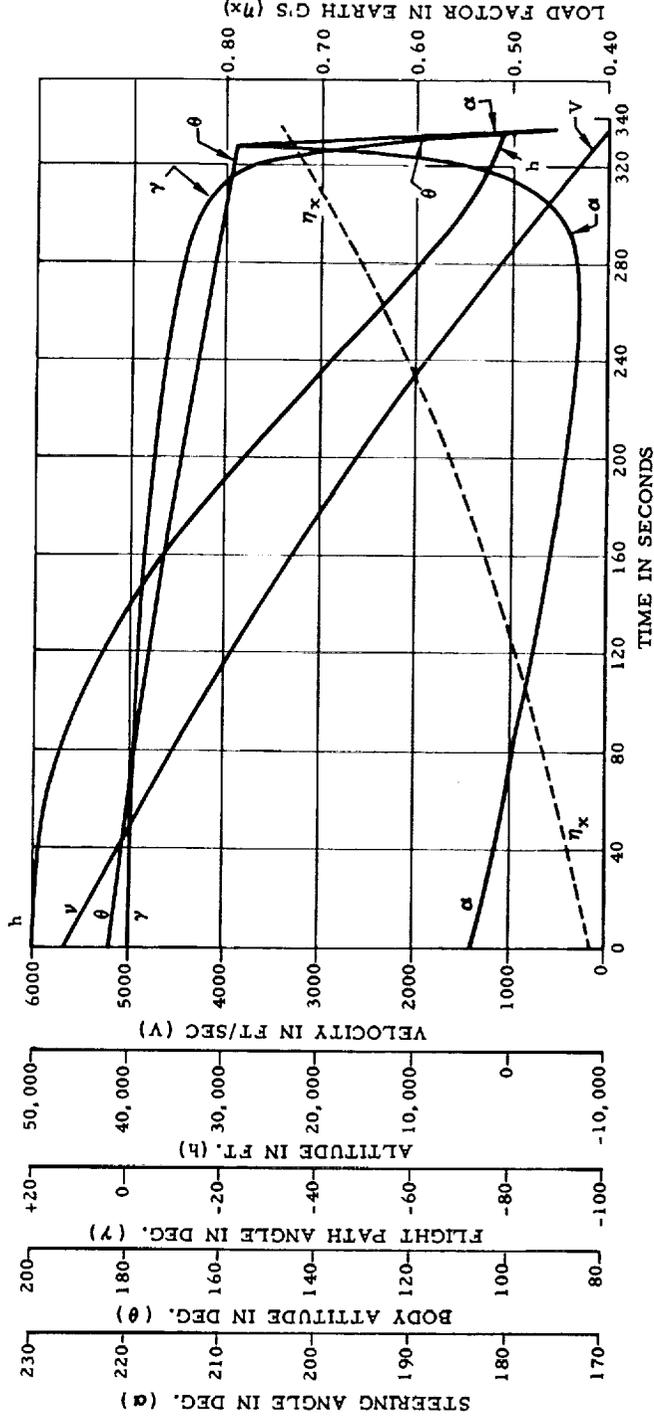


Figure 20. Flight Parameters for LEM Main Retro Maneuvers



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LEM FINAL DESCENT TO LUNAR SURFACE
ALTITUDE - RANGE PROFILE

LEM CHARACTERISTICS:

$I_{SP} = 315 \text{ SEC.}$
 $T/W = .165 \text{ (CONSTANT)}$
 $\Delta V = 700 \text{ FPS}$
 $h = 7.845 \text{ FPS (CONSTANT)}$
 $\theta = 2.653^\circ \text{ FROM VERT.}$
 (SEE SCHEMATIC)

	TIME SEC.	VELOCITY FPS	γ DEG.
①	0.0	7.85	-90.0
②	30.0	10.78	-46.73
③	45.0	13.58	-35.30
④	63.73	17.54	-26.37
⑤	90.0	12.11	-40.38
⑥	110.0	8.95	-61.28
⑦	127.46	7.85	-90.0

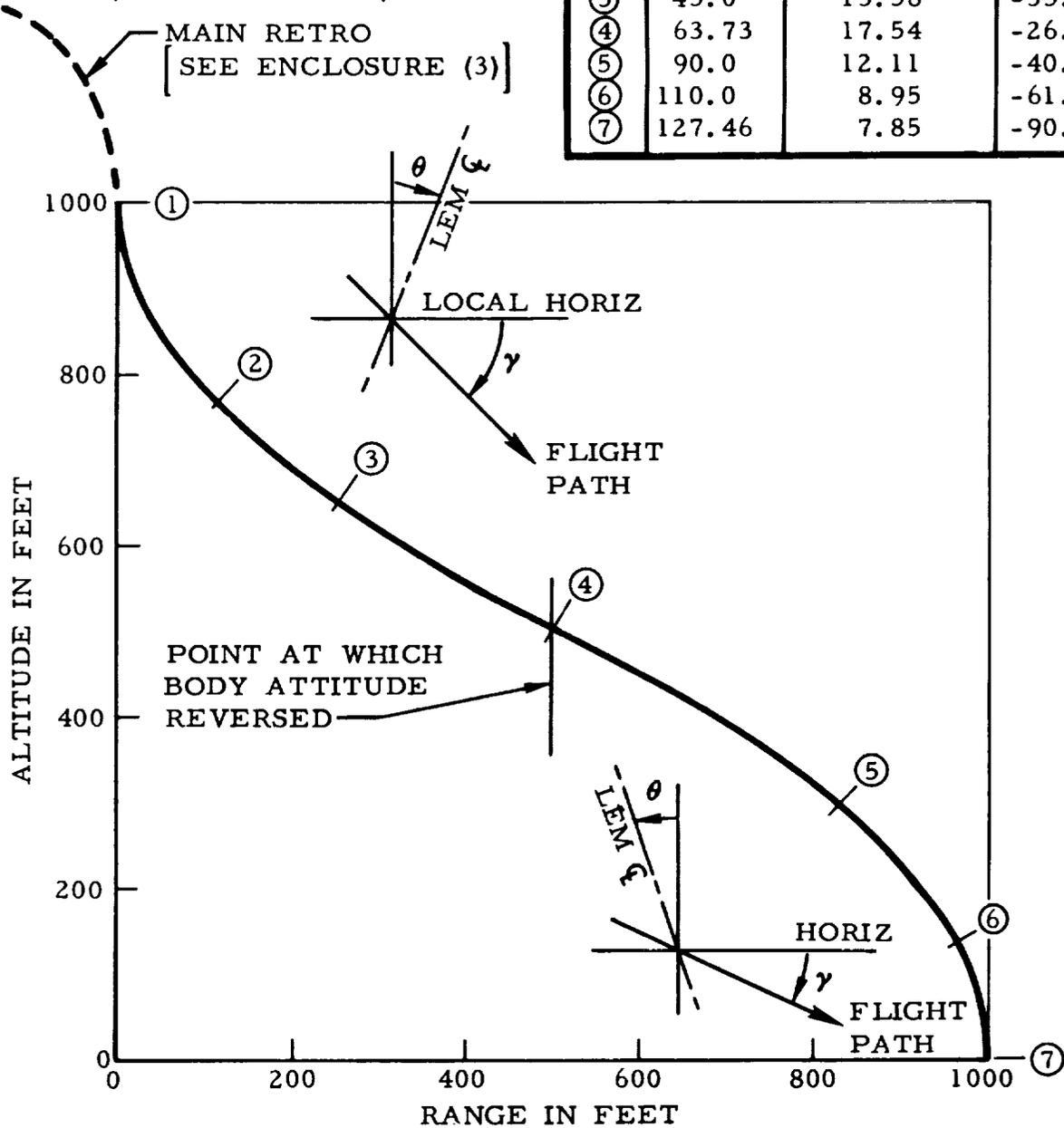


Figure 21. LEM Final Descent Flight Profile

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	<u>t, SEC.</u>	<u>V, FPS</u>	<u>h, FT</u>	<u>γ, DEG.</u>	<u>θ, DEG.</u>	<u>α, DEG.</u>
1	0.0	0.0	0.0	90.0	90.0	0.0
2	109.7	1315	12,500	8.4	23.3	15.05
3	169.5	2190	25,000	6.0	15.9	10.1
4	229.0	3170	37,500	3.7	8.2	+4.7
5	284.0	4225	46,250	1.8	1.3	-0.4
6	343.0	5481	50,000	0.0	-5.6	-5.6

$I_{SP} = 35 \text{ SEC.}$
 $T/W_0 = .400$

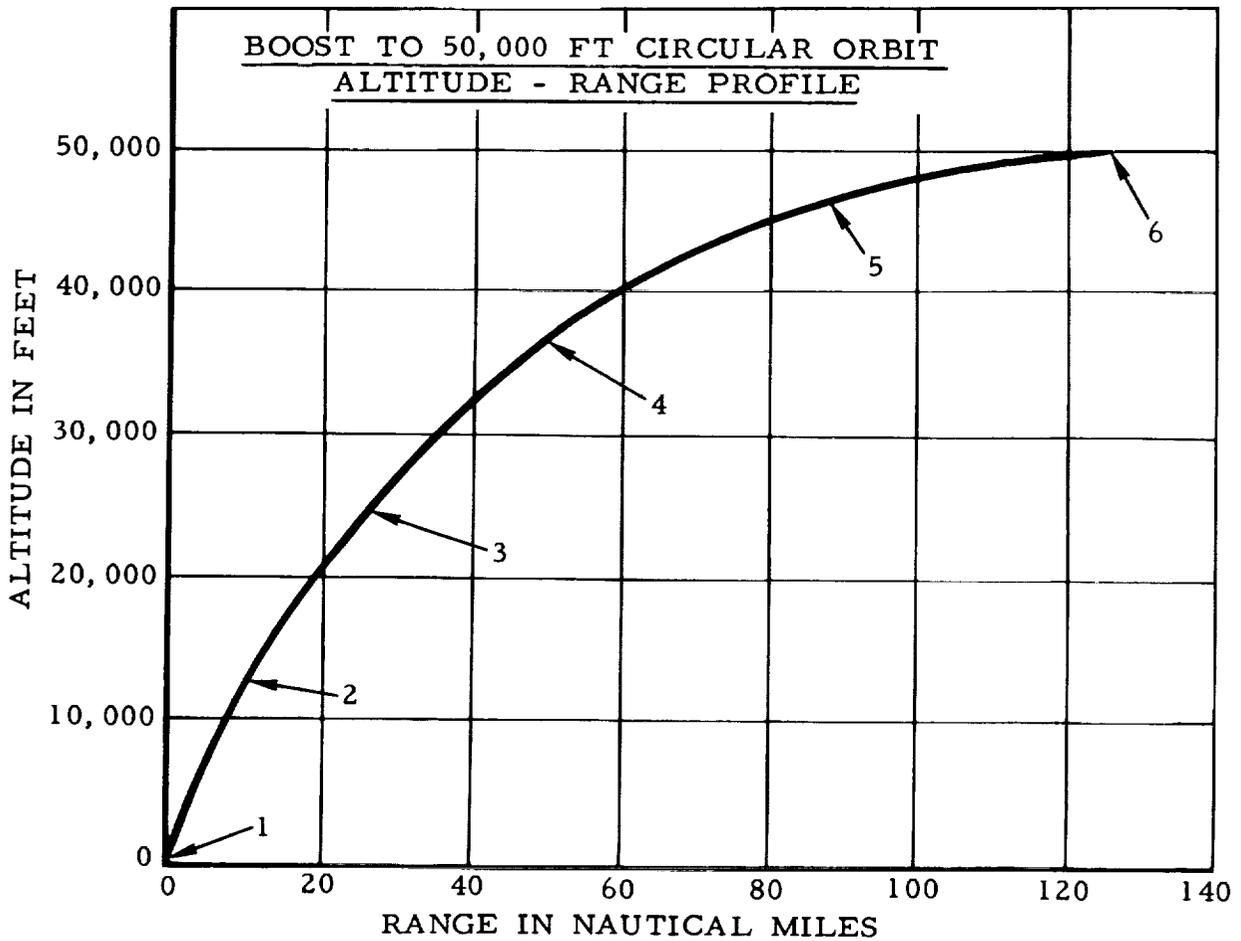


Figure 22. LEM Boost Profile to Circular Orbit

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LEM BOOST TO 50,000 FT CIRCULAR ORBIT
PARAMETRIC TIME HISTORY

LEM $I_{SP} = 315$ SEC.

$T/W_0 = .400$

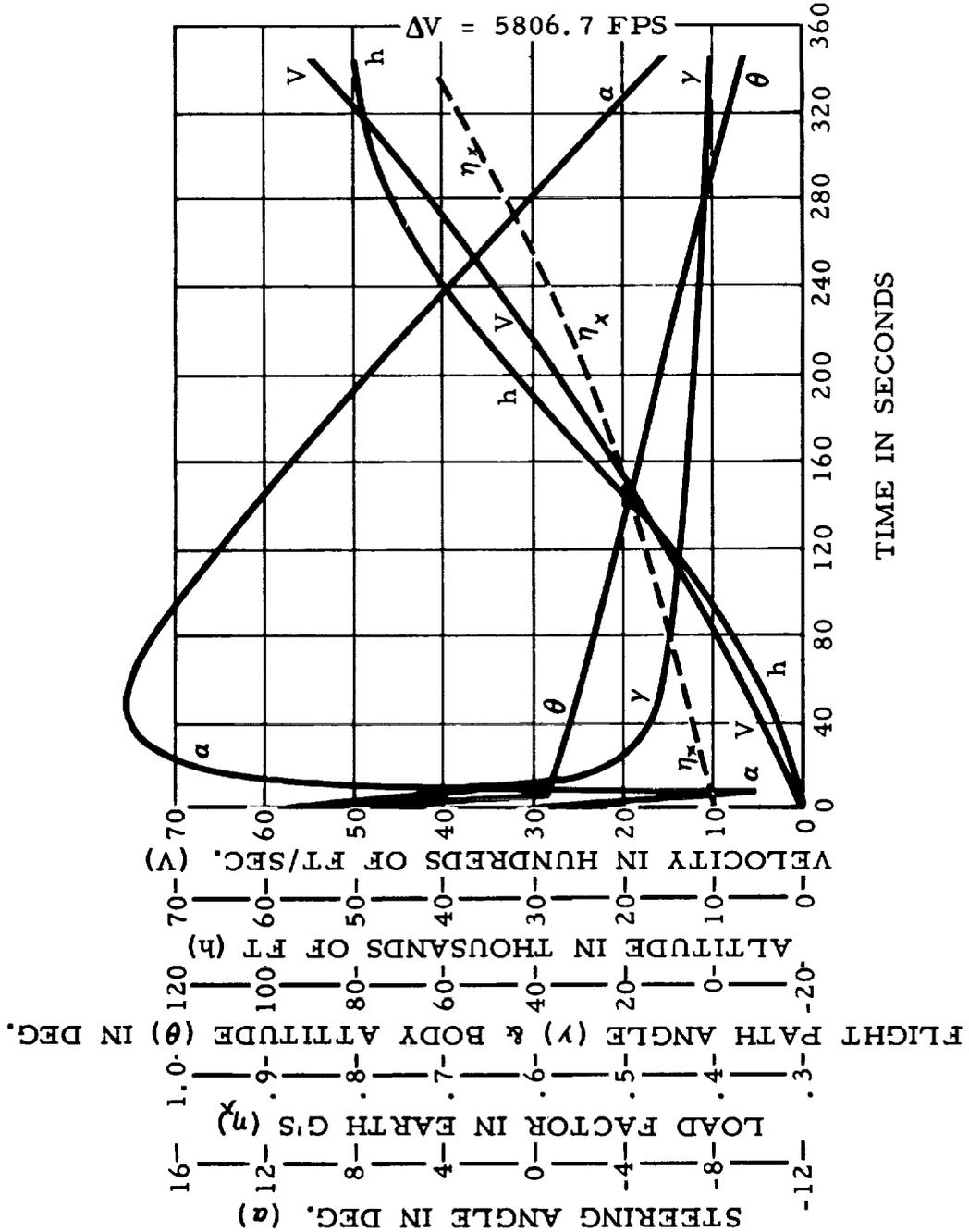
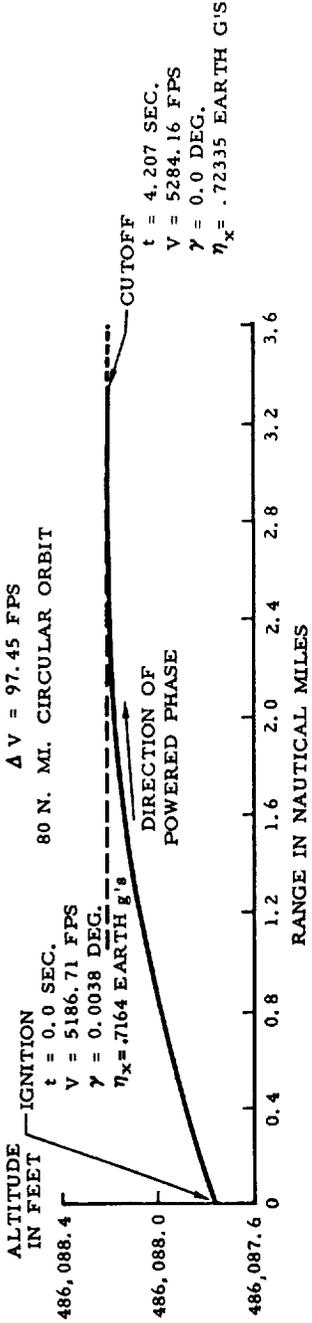


Figure 23. Flight Parameters for LEM Boost.



LEM KICK TO CIRCULARIZE
ELLIPTICAL COAST AT 80 N. MI.



LEM KICK INTO ELLIPTICAL TRANSFER ORBIT
FROM 50,000 FT CIRCULAR ORBIT

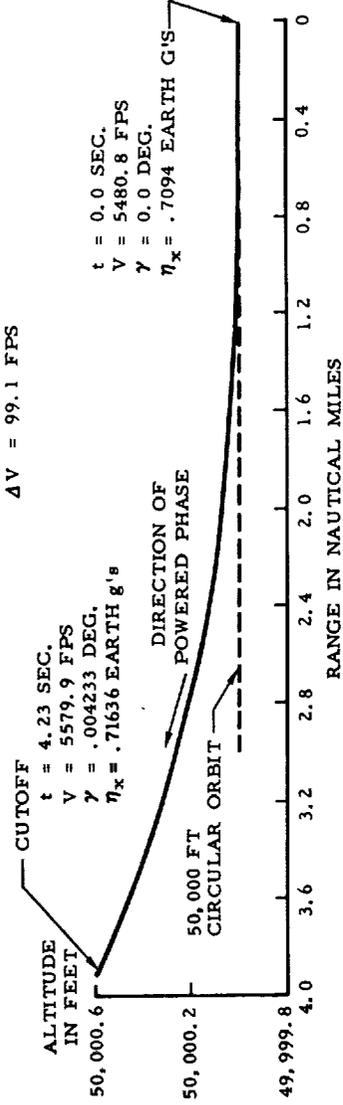
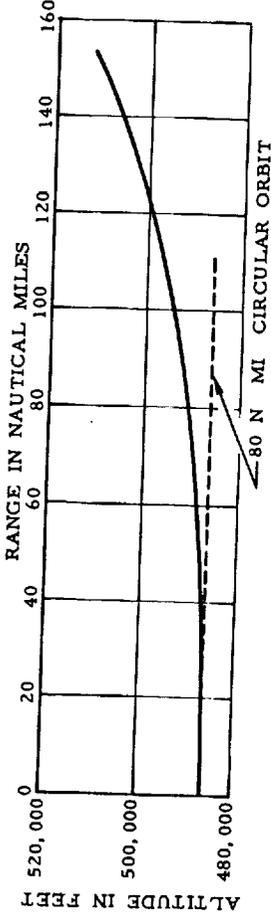


Figure 24. LEM Rendezvous Maneuver Profile.



TRANSEARTH INJECTION FROM 80 N MI ORBIT
ALTITUDE - RANGE PROFILE

S/M $I_{sp} = 319.5$ SEC
 $T = 21,900$ LB
 $T/WO = .650G$ (EARTH)
 $T/WEB = .927$

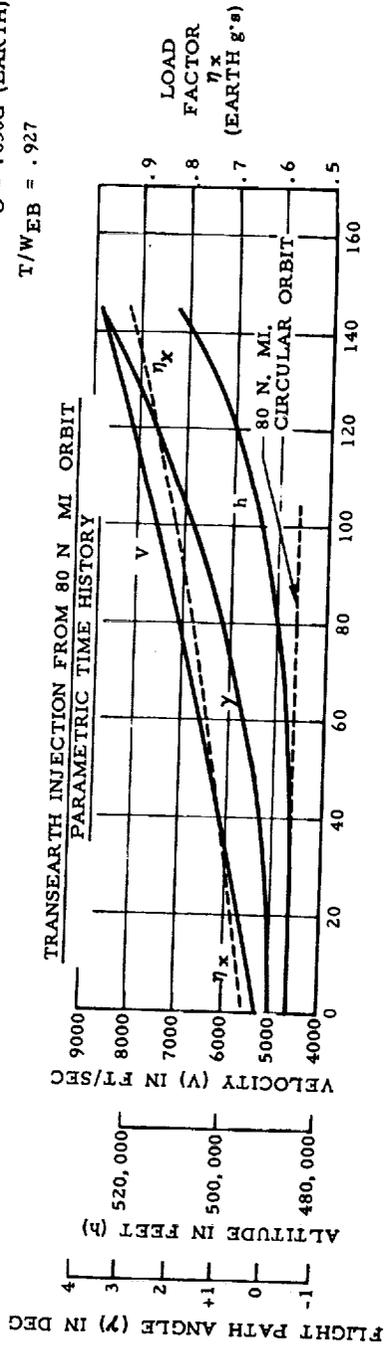


Figure 25. Transearth Injection Maneuver

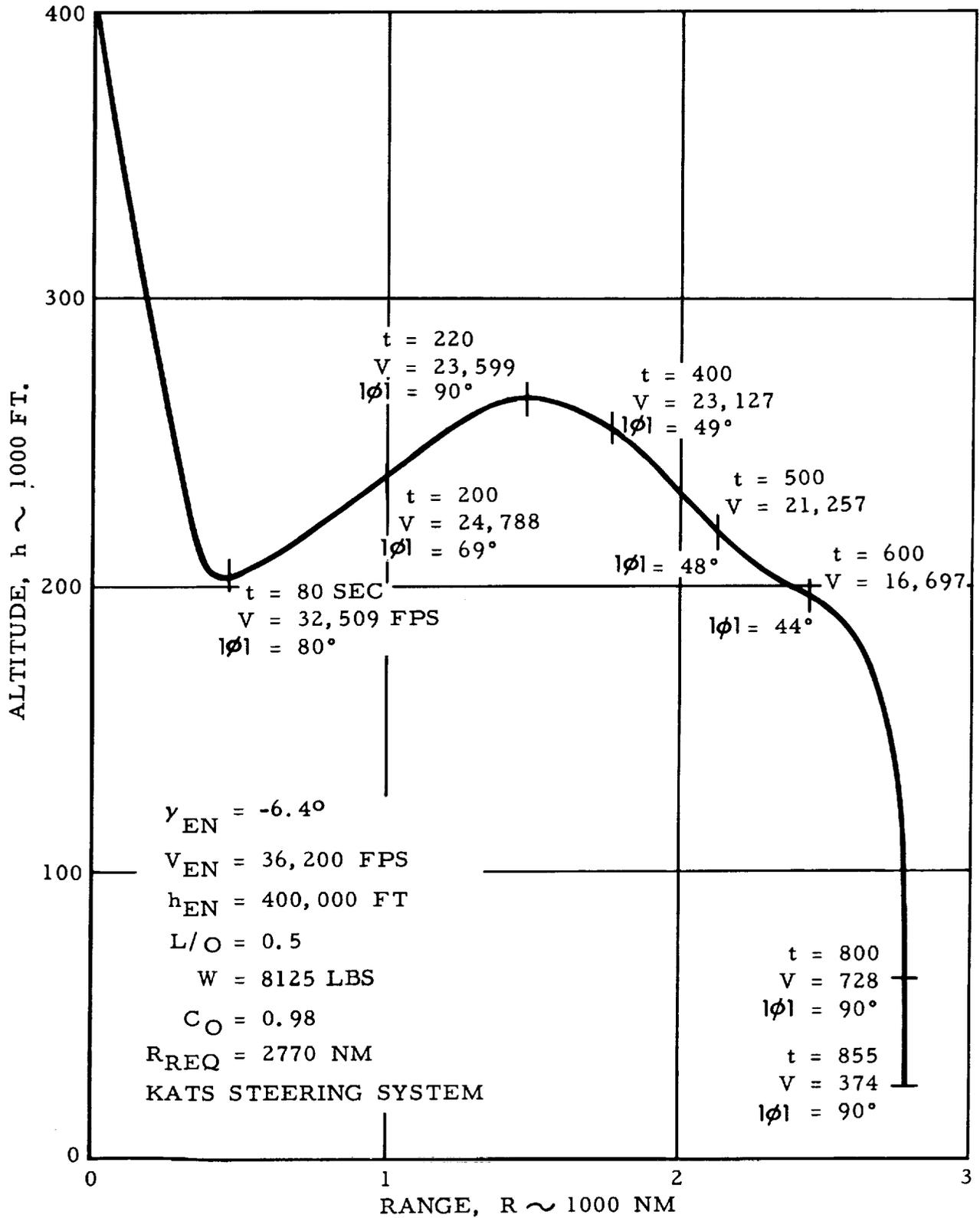


Figure 26. Entry Trajectory Profile(1959 ARDC Model Atmos.)



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RESULTANT LOAD FACTOR TIME HISTORY
1959 ARDC MODEL ATMOS

$\gamma_{EN} = -6.4^\circ$

$V_{EN} = 36,200 \text{ FPS}$

$h_{EN} = 400,000 \text{ FT}$

$L/D = 0.5$

$W = 8125 \text{ LBS}$

$C_D = 0.98$

$R_{REQ} = 2770 \text{ NM}$

KATS STEERING SYSTEM

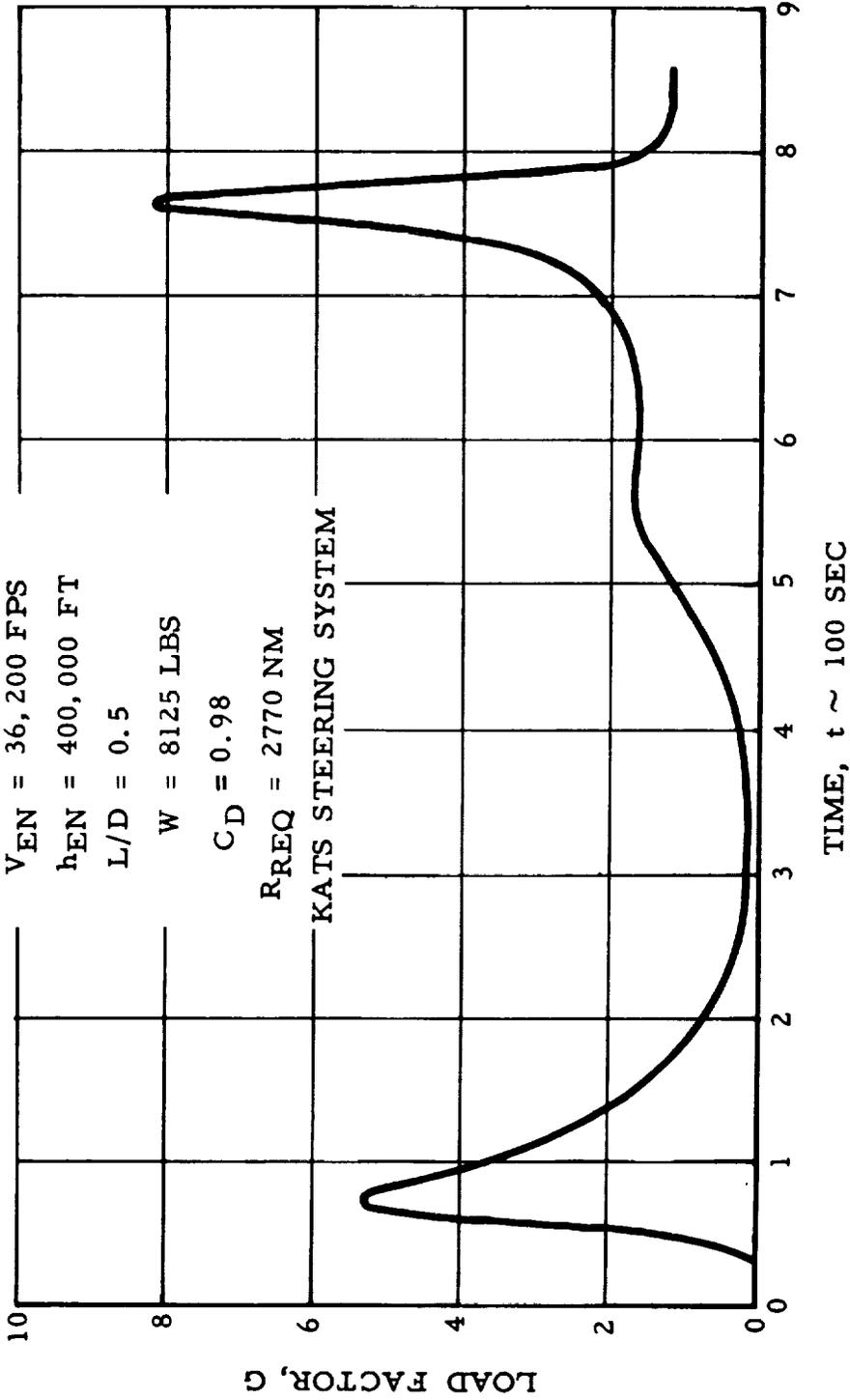
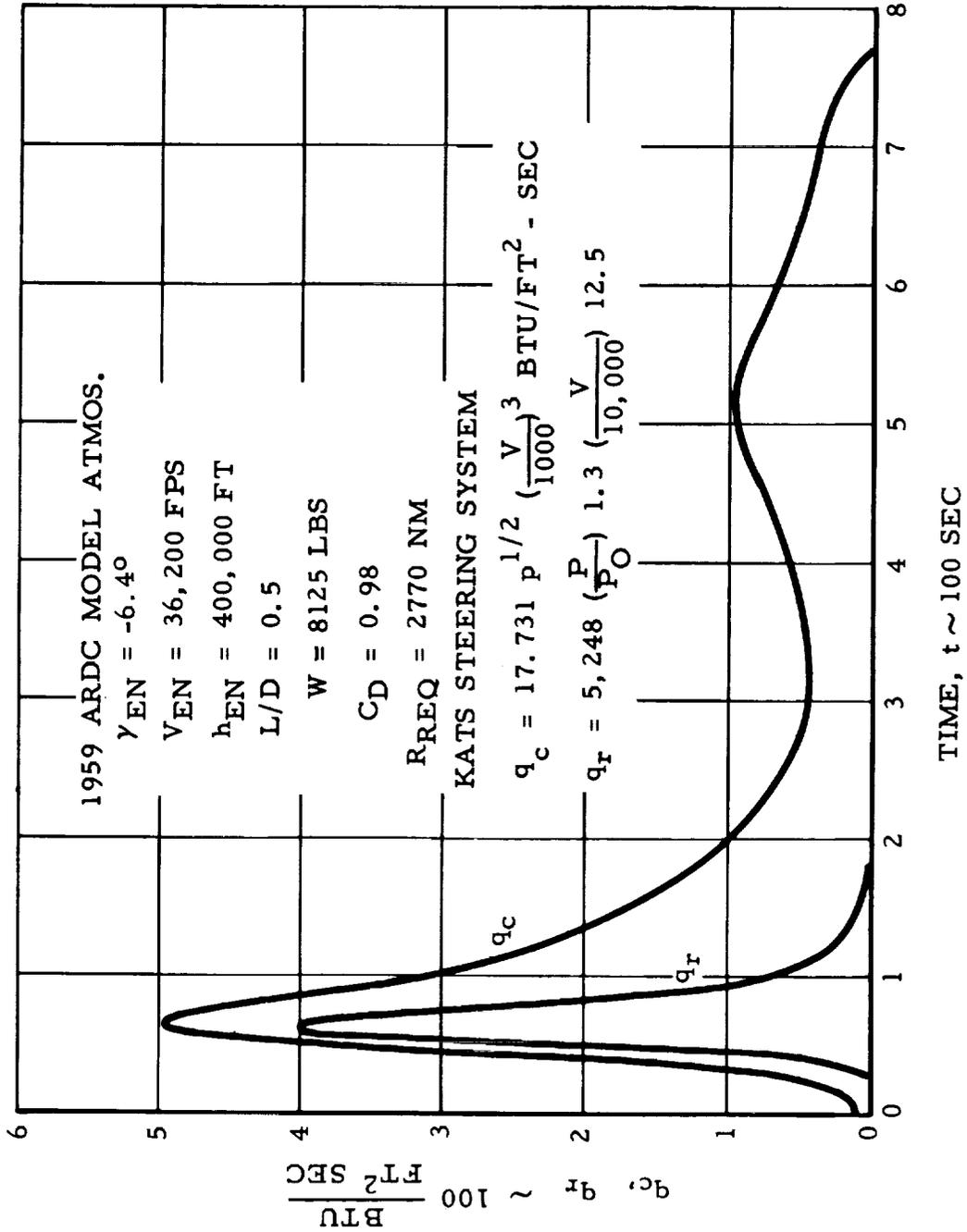


Figure 27. Load Factor During Entry

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CONVECTIVE AND RADIATIVE HEATING RATE,

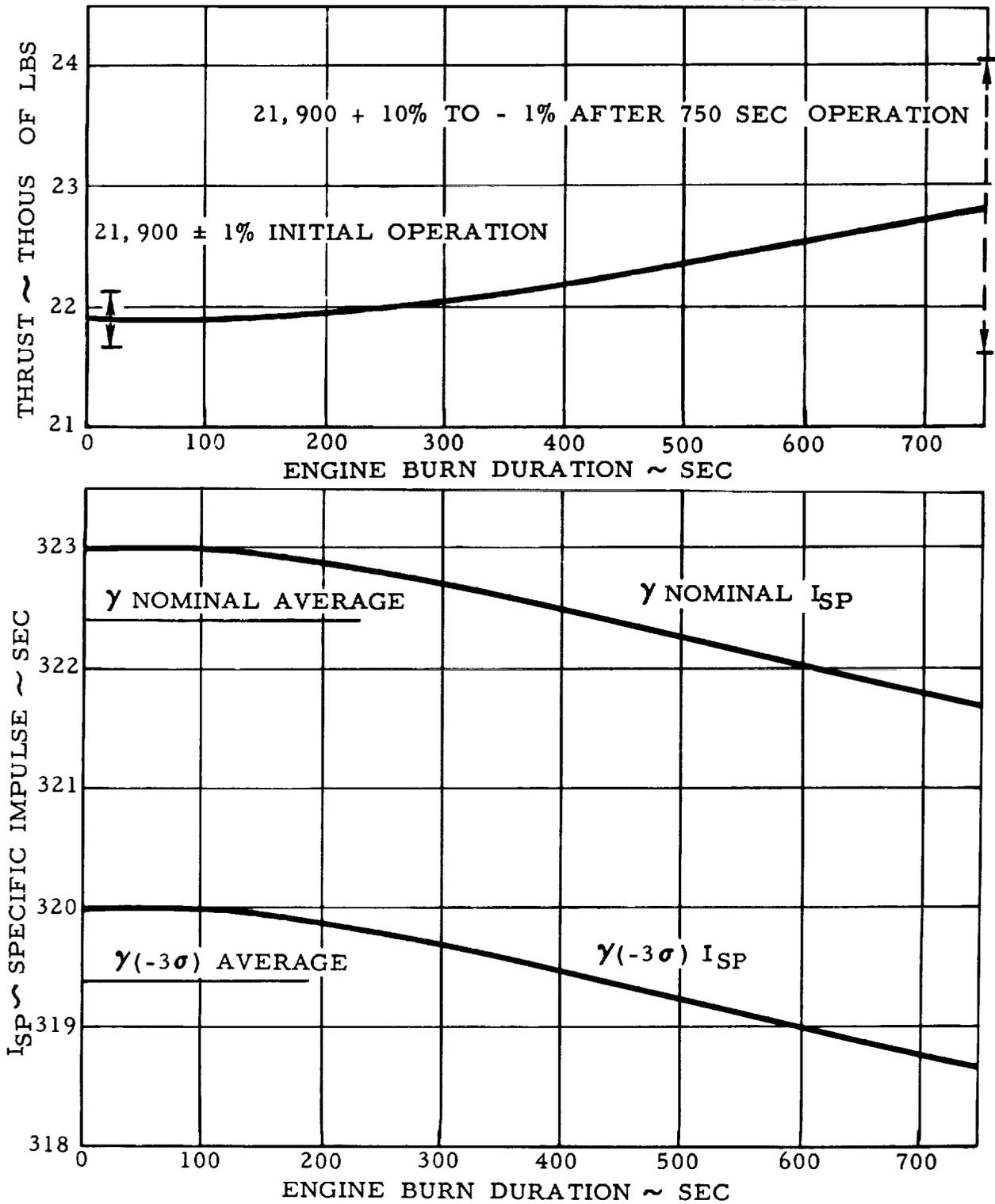
Figure 28. Representative Heating Rates During Entry

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APOLLO SERVICE MODULE ROCKET
ENGINE THRUST AND SPECIFIC IMPULSE



BASED ON PROCUREMENT SPEC MC 901-0009 & AEROJET MODEL SPEC AGC 10141

Figure 29. Service Propulsion Characteristics

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APOLLO CM-LEM G&N FUNCTION PROFILE

