[redacted]

This Document Contains 374 Numbered Pages
Copy No. 26 of 400 Copies, First Series.

NASA Contract NAS 5-302

[redacted]

PROJECT APOLLO

A Feasibility Study of an Advanced Manned Spacecraft and System

FINAL REPORT

VOLUME III. TRAJECTORIES, NAVIGATION AND GUIDANCE
Book 1

authors ~~Program Manager:~~ Dr. G. R. Arthur
~~Project Engineer:~~ H. L. Bloom

Prepared for:

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

(NASA Contract NAS 5-302)

May 15, 1961 368 p 26 refs

3536000

(1) GENERAL ELECTRIC Co.
(3) MISSILE AND SPACE VEHICLE DEPARTMENT
A Department Of The Defense Electronics Division
3198 Chestnut Street, Philadelphia 4, Penna. PA
(2)

[redacted]

[redacted]

TABLE OF CONTENTS

	Page
I. TRAJECTORY ANALYSIS	I-1
1.0 Lunar Orbit and Return Mission	I-3
1.1 Selection of Launch Date	I-3
1.2 Powered Ascent Trajectory	I-10
1.3 Free-Flight Transfer Trajectory	I-14
1.4 Emergency Return Trajectories	I-17
1.5 Lunar Satellite Orbit	I-34
1.6 Return from Lunar Orbit	I-48
1.7 Re-entry Trajectory Study	I-64
2.0 Earth-Orbit Development Flights	I-83
II NAVIGATION AND GUIDANCE	II-1
1.0 Introduction and Summary	II-1
2.0 Navigation and Guidance Concepts	II-7
2.1 Introduction	II-7
2.2 Mission Description	II-7
2.3 Insertion Guidance	II-18
2.4 Midcourse and Lunar Orbiting	II-18
2.5 Re-entry Management and Guidance	II-36
2.6 Abort and Emergency Return	II-48
2.7 Digital Computer Simulation of the APOLLO Mission	II-48
3.0 Navigation and Control System Design	II-57
3.1 Introduction	II-57
3.2 System Description	II-59
3.3 On-Board Digital Computer	II-73
3.4 Inertial-Stellar Stable Platform	II-113
3.5 Celestial	II-123
3.6 System Summary	II-139

[redacted]

TABLE OF CONTENTS (Continued)

	Page
4.0 Crew Functions	II-143
4.1 Introduction	II-143
4.2 Considerations Affecting Navigation, Guidance and Control Subsystem Design	II-144
4.3 Human Factors Aspects of Navigation, Guidance and Control Subsystem Design	II-146
4.4 Manual Back-up Subsystem Operation	II-191
4.5 Major Crew Decisions	II-194
4.6 Future Work	II-198
5.0 Error Analysis	II-199
5.1 General	II-199
5.2 Summary of Results	II-202
5.3 Insertion Errors	II-204
5.4 Outgoing Midcourse	II-209
5.5 Transition from Outgoing Leg to Terminal Phase	II-226
5.6 Injection Into Orbit Around the Moon	II-226
5.7 Transition from Terminal Phase to Return Leg	II-229
5.8 Return Midcourse Error Analysis	II-229
5.9 Transition from Midcourse to Re-Entry	II-239
5.10 Re-Entry Error Analysis	II-239
6.0 Required Modifications for Alternate Missions	II-247
6.1 Earth Orbit	II-247
6.2 Rendezvous with Earth Satellite	II-247
6.3 Lunar Landing	II-273
7.0 References	II-277

TABLE OF CONTENTS (Continued)

	Page
APPENDIX	
T-B - System Using OAO Astrotrackers	A-1
T-C - Saturn Boost Guidance	A-35
T-D - On-Board Trajectory Calculation	A-51
T-E - Computation of Sensitivity Coefficients	A-61
T-F - Navigational Position and Velocity Estimate	A-65
T-G - Linearity of Coefficients	A-99
T-H - Sensitivity Coefficients	A-103
T-I - Lunar Orbit Characteristics	A-113
T-J - Re-Entry Energy Management	A-121
T-K - Simulation Results	A-207
T-L - Analysis of Velocity Correction Loop	A-215
T-M - Analysis of APOLLO Attitude Control System	A-225
T-N - Re-Entry Vehicle Attitude Control	A-237
T-O - General Information on Star-Tracker	A-251
T-P - General Information on IR Horizon Sensors	A-257
T-Q - Investigation of Dome Feasibility	A-265
T-R - Lunar Orbit Escape	A-269
T-S - Analysis Theory	A-283
T-T - Patched Conic Errors	A-291
T-U - Sphericity Effect on Altitude	A-293
T-V - Selection of Alternate Stars	A-297
T-W - Re-entry Corridor Considerations	A-301

[redacted]

CHAPTER 1

TRAJECTORY ANALYSIS

[redacted]

I. TRAJECTORY ANALYSIS

The flight missions to be undertaken under the APOLLO manned vehicle program are, in chronological order of a logical development program:

- Low altitude circular earth orbits
- Eccentric earth orbits
- Continuous circumlunar flights
- Lunar orbit and return flights
- Lunar landing and return flights

All but the last of the missions tabulated can be accomplished using a Saturn C-2 booster system. This mission, which requires an APOLLO vehicle weighing significantly more than 15,000 lb, would require either a booster with increased payload capacity or the utilization of assembly-in-orbit techniques.

The trajectory analysis conducted during this study by General Electric's Missile and Space Vehicle Dept. has concentrated on the lunar orbit and return mission with lesser attention being given to the continuous circumlunar missions and to the developmental earth-orbit missions.

This Chapter is organized along the lines of a mission profile for the lunar orbit and return mission in the sense that the various phases of the flight from launch to landing are discussed in sequence.

The first section discusses the variation of lunar declination at possible firing times and relates this to the Saturn insertion conditions required to send the APOLLO vehicle to the Moon. Insertion loci, making use of a coasting phase in the Saturn ascent trajectory, are presented for all possible lunar declinations. Recommendations for the choice of two standard launch azimuths is made.

The Saturn-powered ascent trajectory is considered in the next section. Since the Saturn vehicle design, as well as its performance capabilities, has not been firmly defined, few specific conclusions are made.

The third section considers the free-flight transfer trajectory to the Moon. A brief summary of the general study of outbound trajectories is followed by a discussion of a sample trajectory for illustrative, as well as subsequent study purposes.

Emergency return trajectories initiated during the outbound transfer trajectory are discussed next. The results of the general parametric study are presented. This is followed by a discussion of the location, relative to the rotating earth, of the re-entry points of emergency return trajectories initiated from a typical trajectory.

Section 1.5 contains a discussion of some aspects of lunar satellite orbits, followed by a summary of a study of the velocity increments required to establish the orbit. Following this is a description of a lunar satellite orbit established immediately from the sample outbound transfer trajectory. The gravitational perturbations by the earth and sun on this lunar satellite orbit are shown to be small.

The transfer trajectory returning from the satellite orbit is described next. A general discussion of the lunar dis-orbit maneuver from various revolutions is presented. This is followed by a description of the initial and terminal parameters of a family of return trajectories initiated during the 18th orbital revolution. One of these trajectories, which returns to the earth and results in a landing along the Atlantic Missile Range, is selected for further description and discussion.

The final portion of the mission description is a discussion and summary of the re-entry trajectories at escape velocity. The trajectory time histories, including aerodynamic heating and deceleration information, is given for the APOLLO D-2 and R-3 configurations. The maneuver area, including cross-range, attainable by both configurations is also shown.

Aspects of the various phases which are also applicable to the continuous circumlunar mission are included in the various sections.

Chapter I is concluded by a short discussion of circular and elliptical earth satellite orbits for early developmental flights.

[redacted]

1.0 Lunar Orbit and Return Mission

As discussed above, the lunar orbit and return mission would be preceeded by earth orbital and continuous circumlunar missions to test the APOLLO vehicle system. The lunar orbit mission, however, has received a major portion of the effort under the APOLLO study. This mission profile can be described as follows: The APOLLO vehicle is launched by the Saturn booster system on a powered-ascent trajectory, employing a parking or coasting orbit phase, which terminates when the APOLLO vehicle has reached the appropriate combination of velocity and position vectors for the free-flight transfer trajectory. The vehicle then follows an unpowered, except for mid-course correctional maneuvers, trajectory to the point of closest approach to the Moon. At this point, the APOLLO vehicle's velocity relative to the Moon is reduced to a value which corresponds to approximately circular orbital velocity at that lunar altitude. After measurements are taken and the circular orbit is well defined, a second thrusting phase is employed to reduce the perilune altitude of the orbit to the final, required value over the near face of the Moon. The position of this velocity change, relative to the Earth-Moon line, can be selected to optimize the orientation of the lunar satellite orbit for the nominal return trajectory. After the required orbiting duration, the vehicle's velocity relative to the Moon is again increased by the on-board propulsion system to establish the return trajectory to earth. Midcourse maneuvers may again be performed to insure that the APOLLO vehicle will re-enter the earth's atmosphere, at an altitude of 400,000 ft, at a path angle such that neither the allowable skip nor "g" limits are exceeded. The vehicle then flies through the atmosphere and maneuvers to a landing at the recovery site.

1.1 SELECTION OF LAUNCH DATE

The proposed schedule of manned APOLLO vehicle launchings will cover a considerable span of time. The early, unmanned developmental circular and elliptical earth orbits would begin in 1964 and extend to 1966. Manned flights would begin in 1965.

[redacted]

The continuous circumlunar flights, which are the first missions to require consideration of the relative position of the Moon, are scheduled to begin in 1966. The lunar orbit and return missions could begin in 1967 and possibly extend for several years. The ultimate mission of lunar landing and return, which requires greater booster capability, could begin in 1970 and extend for a number of years, as discussed in Volume IX.

From the above tentative schedule, it can be seen that flights to the vicinity of the Moon will probably cover a time period of at least four years.

Since the continuous circumlunar missions, which are in the nature of developmental flights for later lunar orbit and landing missions, will approach the Moon most closely on the far side, it would be preferable to schedule these launchings to result in approaches to the Moon during a time of new Moon so that additional information about the far side of the Moon could be obtained.

Missions involving entry into a lunar orbit and return to earth, however, would be scheduled to have the lunar orbiting phase occur at a time in the vicinity of full Moon so that the near side would be illuminated for purposes of visual and sensor observation of surface features to aid in the selection of potential landing sites. Because of the varying contrast of surface features with the angle of illumination, flights could be scheduled to arrive at the Moon at various times of the month ranging from first to last quarter. Alternately, the lunar orbiting phase could be extended to a maximum of 6 or 7 days, consistent with the 14 day total mission constraint to have the contrast change somewhat.

In view of the above discussion, which indicates that firings would occur to coincide with various lunar phases over a time span of several years, the launch and insertion capability of the Saturn-based APOLLO vehicle should be designed for maximum flexibility.

During an 18.6 year cycle, the inclination of the Moon's orbital plane relative to the equator varies between limits of approximately 18.5 and 28.5 degrees. When the Moon's declination limits are ± 28.5 degrees, nearly-coplanar launches are possible from Cape Canaveral.

In order to study the most general case of 3-dimensional trajectories, the month of July, 1961 was chosen for trajectory studies by General Electric's Missile and Space Vehicle Dept. During this particular month, the lunar declination limits are approximately ± 19.2 degrees, nearly the smallest possible values. Figure I-1-1 presents the lunar declination time history for July, 1961 showing that the maximum positive declination is reached on 12 July and maximum negative declination on 26 July. Also indicated along the curve are the lunar declinations corresponding to full Moon and new Moon during the entire year (by tics labelled above and below the curve, respectively).

The need for flexibility in launching the APOLLO vehicle to the Moon when it is at various declinations can be seen in this graph. For example, full Moon in June occurs when the Moon is at its maximum negative declination, whereas the Moon is at its maximum positive declination during the full Moon in December. Therefore, if artificial delays of months, in order to allow the Moon to reach a particular combination of phase and declination, are to be avoided the Saturn must be capable of inserting the APOLLO vehicle on a wide spread of free-flight trajectories.

A table of the dates of apogee and perigee of the Moon's orbit for a portion of the year is also included. The position of the Moon in its orbit will influence the insertion velocity required and the vehicle's time of flight to the Moon. For example, the difference in flight time between an apogee and perigee shot will be approximately 10 hours if the vehicle's arrival velocity is held constant, or conversely, the arrival velocity can differ by 200 to 400 ft/sec for trajectories with constant flight time.

1.1.1 Insertion Loci

The geometry of the transit trajectory from launch to the point of close approach to the Moon is sketched in Figure I-1-2. The trace of the APOLLO vehicle's trajectory from the launch point to the vicinity of the Moon is indicated by the arc of the shaded plane. The vehicle is launched at an inertial azimuth, ψ_L, from Cape Canaveral which is located at a latitude β_L and longitude, λ_L. Some distance down-range, the powered-flight phase is terminated and the vehicle is inserted into its free-flight transfer trajectory to the Moon. The minimum distance from launch to the insertion point

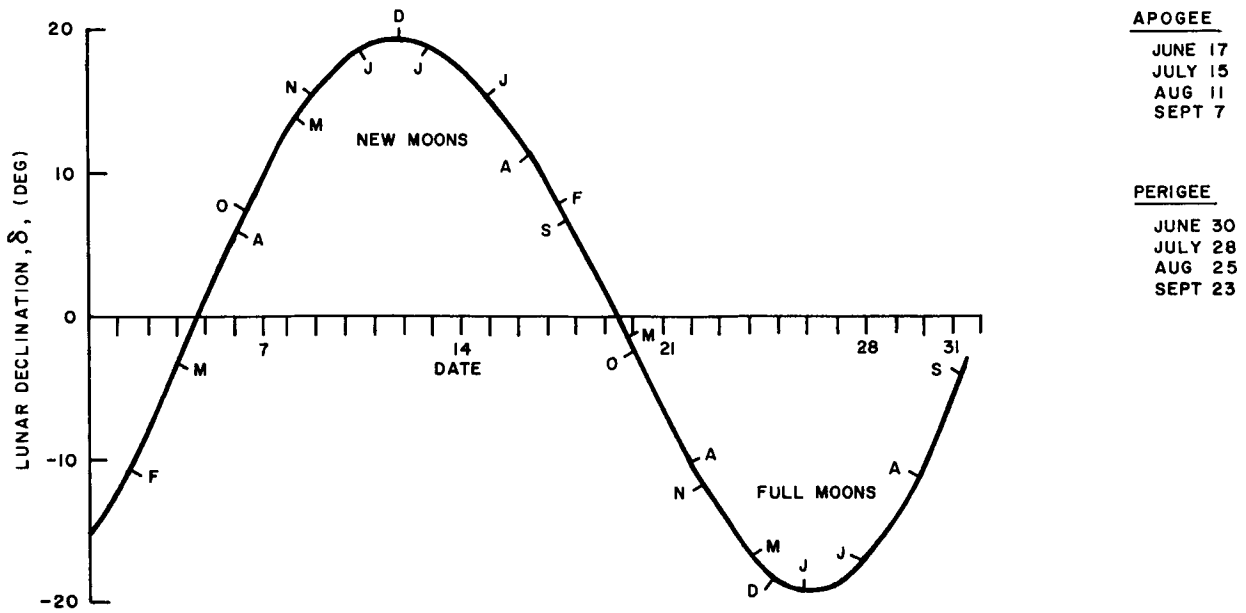


Figure I-I-1. Lunar declination -- July, 1961

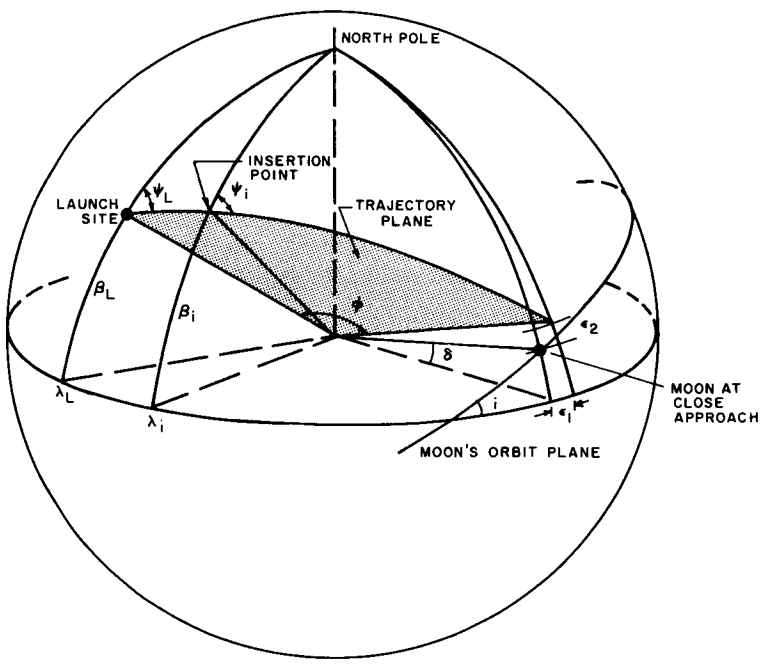


Figure I-1-2. Transit trajectory geometry

is limited by the continuously-powered ascent range of the Saturn booster system and can be extended by use of a coasting arc in a circular earth orbit during the ascent trajectory. For freedom in the selection of the type of approach at the Moon, the terminal point of the trajectory can be assumed to be offset from the projected position of the Moon at that time by two angles, ϵ_1, and ϵ_2. The offset angle ϵ_1 can be used to vary the lunar altitude of the point of closest approach and the angle ϵ_2 can be used to adjust the lunar latitude of that point.

The total in-plane trajectory angle, ϕ, required for various lunar declinations, which can be computed by solving the appropriate spherical triangles, is shown in Figure I-1-3 for representative azimuths. Several limiting conditions are shown on the curves. The first of these is the maximum lunar declination of ± 28.5 degrees. Second, a launch azimuth of 120 degrees east of north has been assumed as representative of the south-east AMR range safety limit. The combination of shallow insertion path angle, relatively low insertion altitudes, and a long Saturn-powered ascent range combine to limit the total in-plane angle to values greater than approximately 205 degrees as shown. This value corresponds to the continuously powered ascent trajectory. The difference between a given value of the angle ϕ and this limiting value is equal to the coasting arc required for that combination of launch azimuth and lunar declination.

Restricting the launch azimuth to angles less than 120 degrees is seen to limit the lunar declinations accessible to a vehicle employing a non-coasting ascent trajectory to values between -28.5 and approximately -14 degrees. More positive declinations require the use of a coasting arc, the minimum values of which can be found from the angle ϕ along the limiting azimuth curve.

Figure I-1-4 presents the insertion loci for a typical insertion velocity-path angle combination plotted as latitude and longitude. Traces of trajectories leaving Cape Canaveral at several azimuths with varying lengths of coasting arcs are shown with a lunar declination overlay. The limits of Figure I-1-3 can be seen on this graph. The declinations accessible with zero coasting arc are seen again. The length of the coasting arc required to reach the Moon at other declinations can be determined. All lunar declinations could be reached by the Saturn-boosted APOLLO vehicle if the launch azimuth were swung from a value of about 85 degrees to 120 degrees with a 36 degree

[redacted]

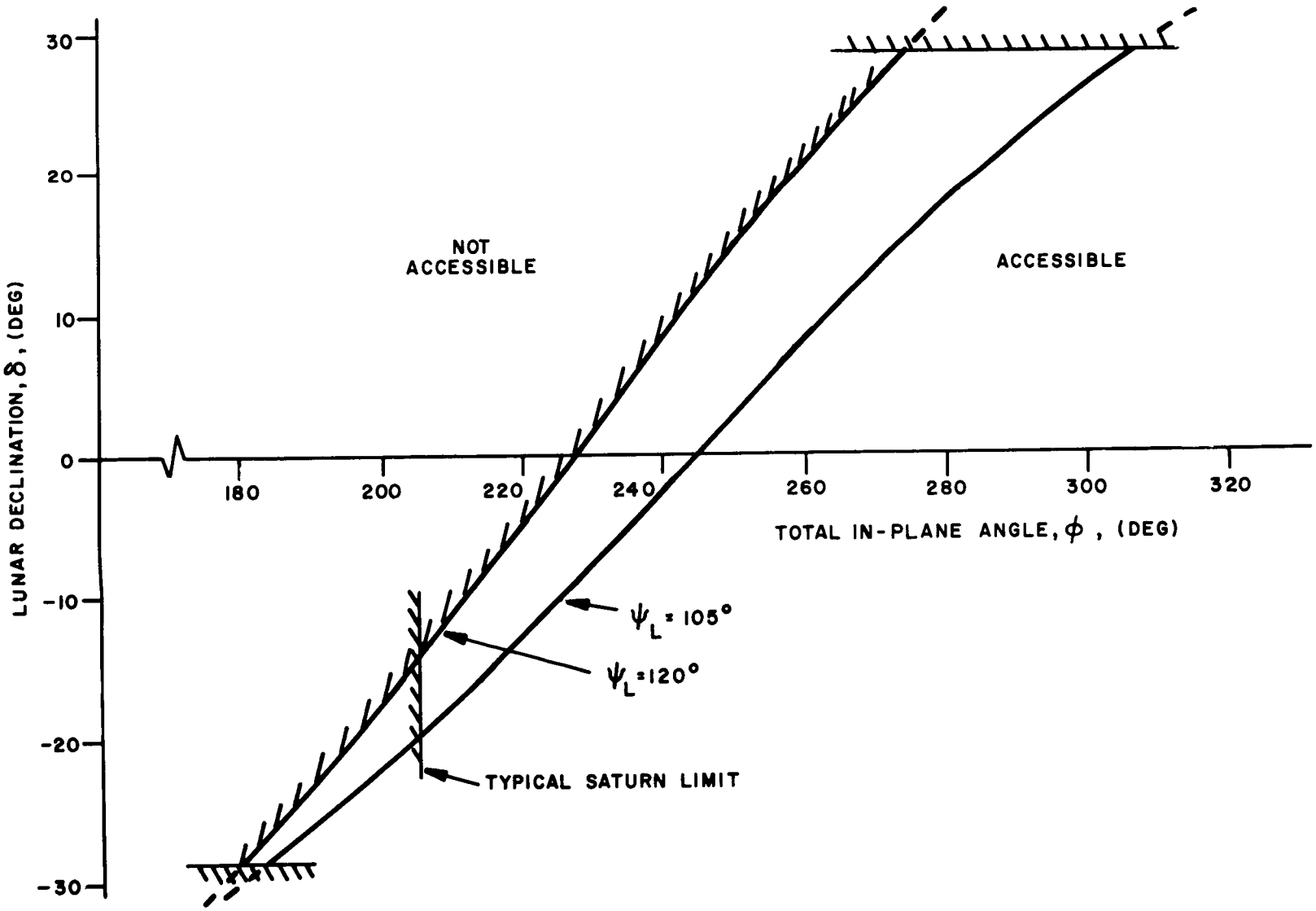


Figure I-1-3. Range safety limits on accessible lunar declinations

[redacted]

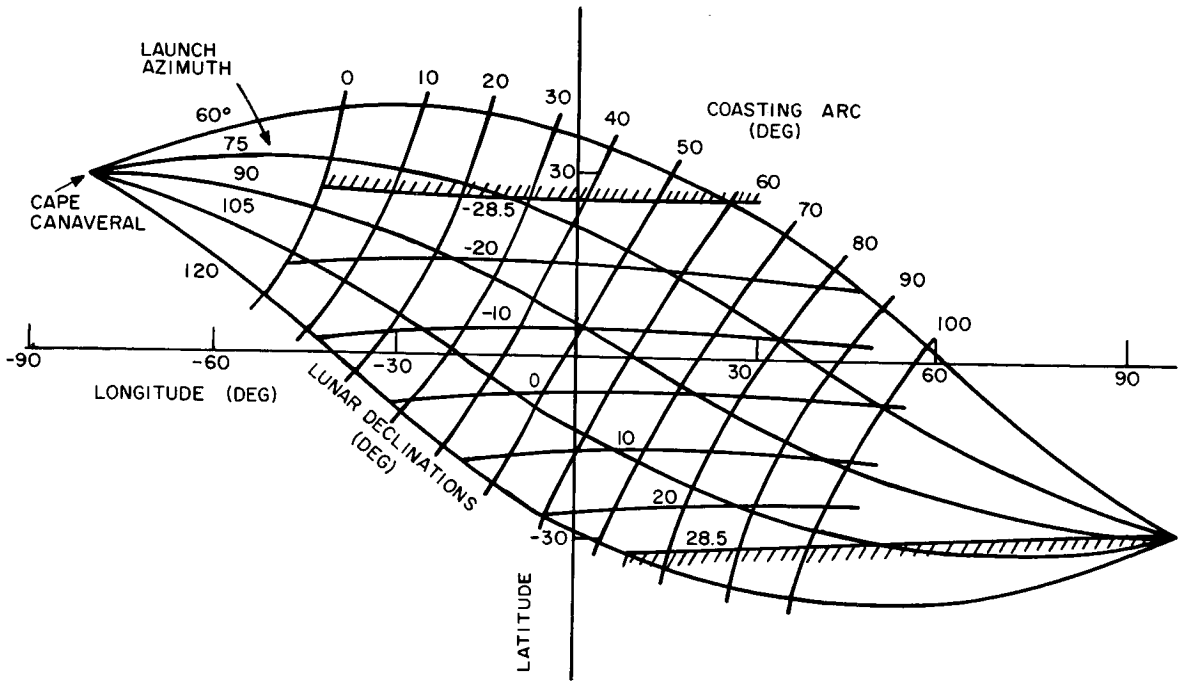


Figure I-1-4. Insertion loci required for lunar missions

(zero coasting arc) powered ascent range and then by fixing the azimuth at 120 degrees by using coasting arcs of as much as 73 degrees.

An interesting aspect of this graph is that nearly all launch azimuths are in a southeasterly direction because of the lunar declination limits and the characteristics of the booster (shallow injection path angles and a long ascent range).

From the points of view of the location of abort recovery areas and ground-based tracking and communication facilities, it would be advantageous to select a minimum number of launch azimuths to be used for all firings. Since typical Saturn-powered ascent trajectories are relatively shallow, the tracking net must be located nearly along the trajectory ground trace.

On the basis of the above considerations it is recommended that a launch azimuth of 108 degrees, which passes down the Atlantic Missile Range, be used for firings to the Moon when it is at declinations ranging from -20 degrees to +28.5 degrees.

This launch azimuth produces an earth trace passing nearly over Ascension Island which could be used for pre- or post-insertion tracking, depending on the target lunar declination. Coasting arcs from zero to about 100 degrees, corresponding to coasting times from zero to about 25 minutes, will be required. To cover the remaining lunar declinations from -20 to -28.5 degrees, a launch azimuth of 85 degrees would be used. This azimuth, which would be used only for the small percentage of time during which the Moon is at declinations below -20 degrees, would require coasting arcs up to a value of 30 degrees, equivalent to coasting times as long as about 7.5 minutes.

These non-due east azimuths will result in some small Saturn payload penalty in addition to the payload loss resulting from shaping the powered ascent trajectory to accommodate a coasting period.

If this azimuth-induced payload loss is considered to be unacceptable, a single launch azimuth of 90 degrees could be used. The accessible lunar declinations would range from approximately -26 to +28.5 degrees, or nearly the complete variation. The coasting arc and time required to reach the Moon at a declination of +28.5 degrees, however, would be about 154 degrees and 38 minutes, respectively. This azimuth would require the use of ship-board tracking and communication equipment. In addition, the abort trace would pass over relatively inaccessible areas of Africa.

1.2 POWERED ASCENT TRAJECTORY

Since the Saturn C-2 booster system, which will be used to launch the APOLLO vehicle to the Moon, is not completely defined as yet, only a small study of Saturn payload capabilities was conducted.

Shortly after the APOLLO study began, a report* was made available to General Electric's Missile and Space Vehicle Dept. as a source of Saturn information. This paper presented performance calculations and time histories of a family of escape trajectories for an 8-engine vehicle configuration. This data was used to determine a tentative powered ascent trajectory for payload, free-flight insertion, and abort studies.

---

* "Saturn C-1 and C-2 Booster Systems", NASA Project APOLLO Working Paper No. 1002, Nov. 9, 1960 (Conf.)

After the mid-term briefing an additional set of Saturn performance curves, based on 7-engines operating during first stage, were made available by Dr. R. F. Hoelker of NASA-MSFC.

1.2.1 Payload Considerations

From an over-all APOLLO aspect it is desirable to have the capability of carrying the largest possible APOLLO weight as Saturn payload consistent, of course, with constraints imposed by insertion requirements.

The study of insertion loci, discussed in Section 1.1.1 of this volume, made use of a powered-ascent range of about 2160 n mi. With this long powered-ascent range and the generally shallow (less than 5 degrees insertion path angles), it was found necessary to make use of a parking orbit to achieve flexibility in the selection of lunar declinations. In order to use a parking orbit, the powered-ascent trajectory must be at a path angle of approximately zero degree when the vehicle's velocity is equal to the local circular orbit value. Case III of Ref.* met this requirement at a flight time of 580 seconds. The path angle at escape velocity for this case equals 3.0 degrees without path shaping. The altitude of the orbit is 650,000 ft at a range of 1230 n mi from launch. The additional powered-range increment to achieve escape conditions is 930 n mi during an additional 194 sec of powered flight.

The Saturn payload weight for this family of trajectories, relative to the value of escape velocity for Case I (γ = 1.6 degrees), is given in Figure I-1-5 for various values of insertion velocity. The payload weight penalty of Case III at escape velocity is about -560 lb. The slope of the curve with respect to insertion velocity is such that a payload gain of 450 to 650 lb, relative the value for escape velocity for Case III, results from the range of insertion velocities for the outbound transfer trajectories discussed in Section 1.3. These burnout velocities are from 200 to 300 ft/sec less than local escape velocity.

---

* "Saturn C-1 and C-2 Booster Systems", NASA Project APOLLO Working Paper No. 1002, Nov. 9, 1960 (Conf.)

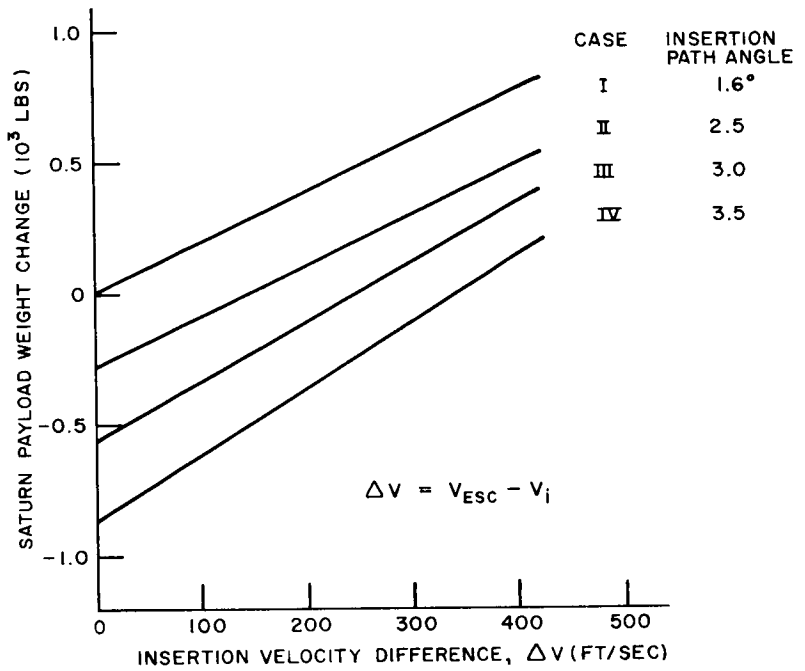


Figure I-1-5. Relative saturn payload weight

An Ames study†, indicates that a velocity increment of about 7000 ft/sec is required for abort insertion velocity for this path angle-altitude combination. The basic APOLLO vehicle is designed with an on-board propulsion capability of 7500 ft/sec so that this criterion could be satisfied if this powered-ascent trajectory were used.

The later, NASA-MSFC curves present parametric performance calculations based on a 7-engine Saturn configuration. Considerable path-shaping is employed for both continuously powered as well as parking orbit type of ascent trajectories. Figure I-1-6 (reproduced from Figure 22 of the NASA-MSFC set of curves) shows a Saturn payload reduction of only about 200 lb resulting from the use of a parking orbit relative to the optimum continuous trajectory with an injection altitude of 140 km for each case. The injection path angle for both of these trajectories is approximately 3.5 degrees.

The set of MSFC curves also present the results of an abort study which makes use of ballistic re-entry trajectories which are initiated by varying velocity increments

† "Abort Rocket Requirements at Super-orbital Speeds", NASA-ARC, Jan. 10, 1961

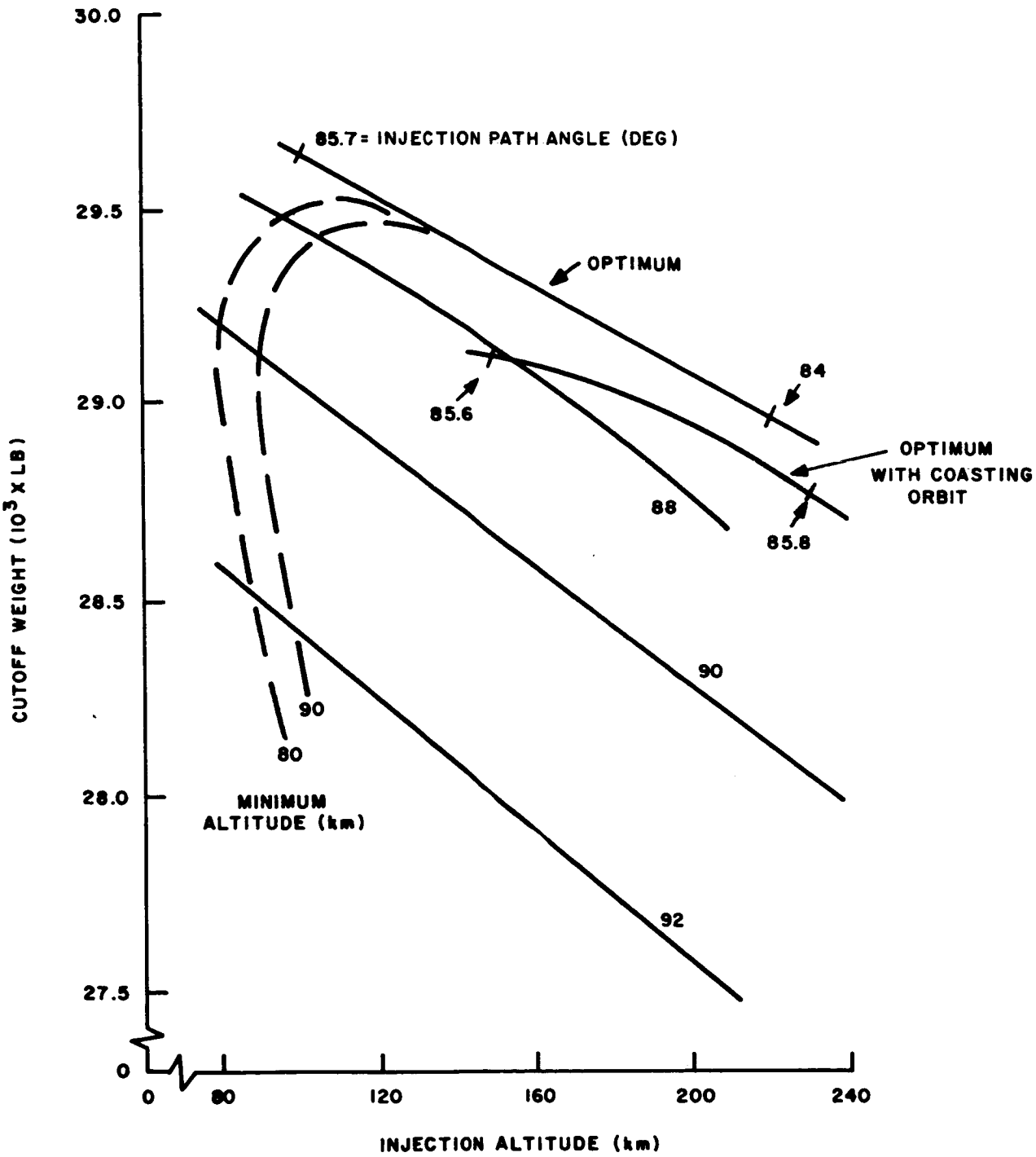


Figure I-1-6. Cutoff weight vs injection altitude

applied at either one of two inertially-fixed thrust directions depending on the vehicle's velocity. Insufficient data is available to compare the results of this study with General Electric's Missile and Space Vehicle Department powered flight abort study described in detail in Chapter III of Volume II.

1.3 FREE-FLIGHT TRANSFER TRAJECTORY

The free-flight trajectories for Project APOLLO were computed using the General Electric "n-body" trajectory program on an IBM-7090. The motion of the vehicle, represented by a body of zero mass, was influenced by the gravitational attractions of the earth, sun, and Moon.

The basic "n-body" trajectory program was augmented by an "initial conditions" subroutine. This program determines the insertion parameters, both in a form suitable for input to the "n-body" trajectory program, and also as the corresponding quantities such as latitude, longitude, and azimuth relative to the rotating earth. This is done by assuming the vehicle to move in a plane and solving the appropriate spherical triangles. Values of insertion velocity, path angle, and range from Cape Canaveral, including both the powered-ascent plus coasting range are required as inputs. Estimates of the free-flight time and in-plane transit angle are required, along with the right ascension of the Moon at the desired approach time, to determine the launch time.

At this point, a computation is made in the sub-routine, and the time of arrival, given by the sum of launch time and time of flight, is compared with that used to determine the position of the Moon. If these do not agree, an iteration process is performed until they do.

Two offset "aim angles", ϵ_1, and ϵ_2, discussed in Section 1.1.1, are available to shift the position of the point of closest approach relative to the Moon.

The target date, for the largest percentage of trajectories computed, is July 26, 1961. On this date, the Moon is near its maximum negative declination of about -19 degrees. Full Moon occurs on July 27 and the Moon reaches perigee of its orbit on July 28. The

trajectories are launched at 14-1500 hours Universal Time (U.T.) on July 23, 1961. Most of the times cluster around 1425 hours U.T., giving a local time of $08^h 53^m$ U.T. on July 23, 1961.

Outbound transfer trajectories have been run for a spread of about 150 ft/sec in insertion velocity. The flight time (time interval from insertion to perilune) is shown in Figure I-1-7. The flight time, which is seen to vary between 50 and 70 hours for this range of insertion velocities, is also a relatively weak function of the altitude of the lunar close approach. This altitude dependence is only of the order of a few tenths of an hour over the lunar altitudes of interest so that "representative" flight time can be plotted as a function of velocity, as shown.

The latitude and longitude coordinates of the insertion point corresponding to these trajectories are shown in Figure I-1-8 as functions of the insertion velocity. Comparison with the insertion loci of Figure I-1-4 show that they correspond to initial positions for powered-flight arc of 36 degrees with zero coasting arc. The launch azimuths vary from approximately 105 to 110 degrees over the range of velocities considered.

1.3.1 Sample Outbound Transfer Trajectory

After completion of the general study of outbound transfer trajectories summarized briefly above, and an analysis of the velocity increments required to enter the lunar orbit, discussed in a subsequent section, a sample outbound trajectory was selected for further study.

This trajectory is defined by an insertion velocity of 36,100 ft/sec with zero path angle at an initial radial distance of 3492 n mi. The launch date is July 23, 1961. The powered-flight arc is 36 degrees and the coasting arc is zero. The launch azimuth is 105 degrees east of north, resulting in an insertion point located at 16 degrees north latitude and 46 degrees west longitude. The free-flight time to a perilune altitude of 1122 n mi is 71.2 hours. The unbraked approach velocity at perilune is 6311 ft/sec.

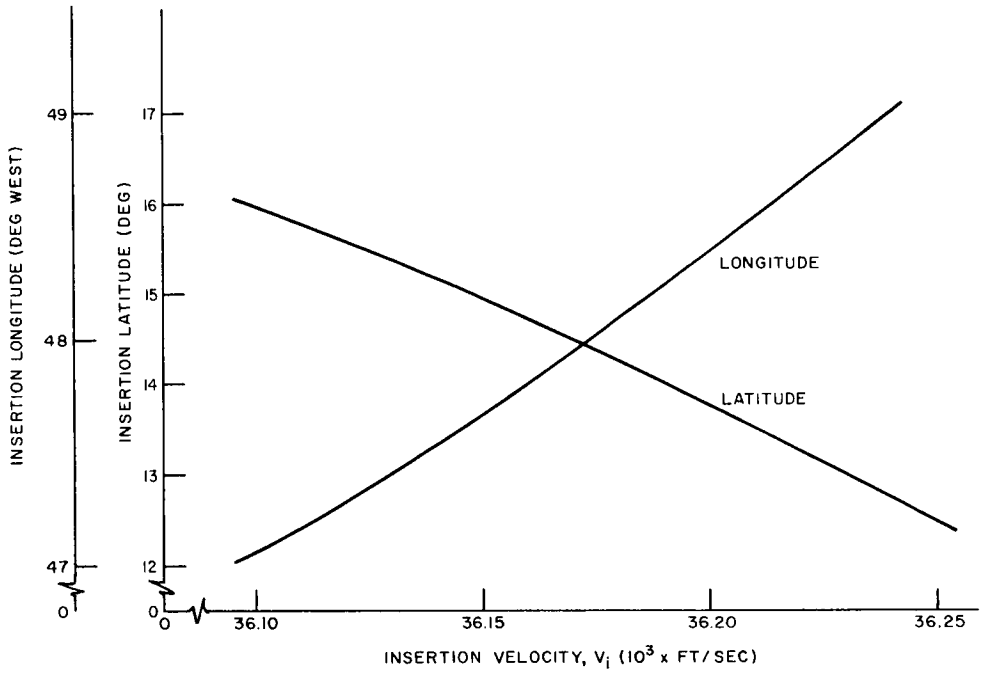


Figure I-1-7. Outbound transit time vs insertion velocity

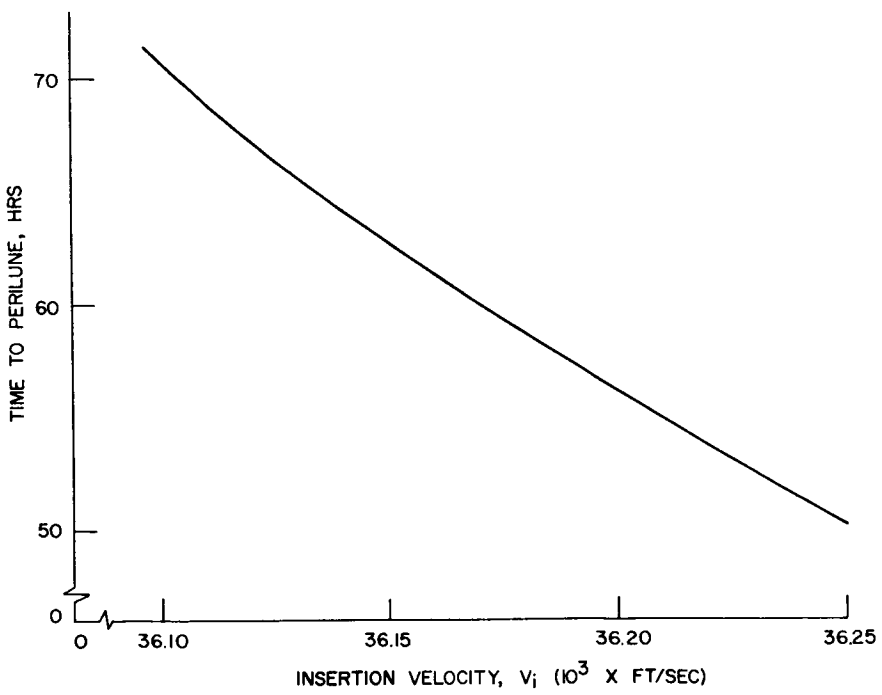


Figure I-1-8. Insertion point coordinates vs insertion velocity

Figure I-1-9 presents the x-y, or earth-equatorial projection of the trajectory to the vicinity of the Moon. The time of flight from insertion is indicated at several points. The Moon's position is also shown at those times. The relatively low-energy nature of the ellipse relative to the earth can be seen, as well as the fact that the Moon does not appear to perturb the ellipse significantly for the first 50 to 60 hours of flight. Figure I-1-10 presents the y-z projection of the trajectory. The plane of this projection passes approximately through the Moon's position midway through the flight. Figure I-1-11 presents the x-z projection, looking generally along the trajectory toward the Moon. The Moon's position just after reaching maximum negative declination can be seen in this figure.

1.3.2 Earth Surface Trace

The earth-surface trace of the sample outbound transit trajectory, described above, is presented in Figure I-1-12. The powered ascent trajectory covers an arc of 36 degrees from Cape Canaveral at an azimuth of 105 degrees east of north. The free-flight insertion point is at 15 degrees north latitude and 46 degrees west longitude. The trace crosses the equator at 20 degrees west longitude approximately 6 minutes after insertion. The maximum negative latitude reached is 32 degrees, which is characteristic of this azimuth from Cape Canaveral.

The eastward extreme of the early portion of the trace is at a longitude of about 56 degrees and occurs at a flight time of 1.6 hours. The trace then turns westward and makes three sweeps over the western hemisphere. The point at which the vehicle reaches the Moon, after 71.2 hours of flight, occurs at 19 degrees south latitude and 137 degrees west longitude. This is about 20 degrees west of the longitude of Goldstone, so that tracking and communication with the APOLLO vehicle during its approach to the Moon can be accomplished from that station. This portion of the flight is about 10 degrees east of the tracking site in Hawaii, which could act in a back-up capacity.

1.4 EMERGENCY RETURN TRAJECTORIES

Since the APOLLO vehicle is to be manned, provision must be made to return the vehicle and crew to a landing on the earth in the event of an emergency occurring

[redacted]

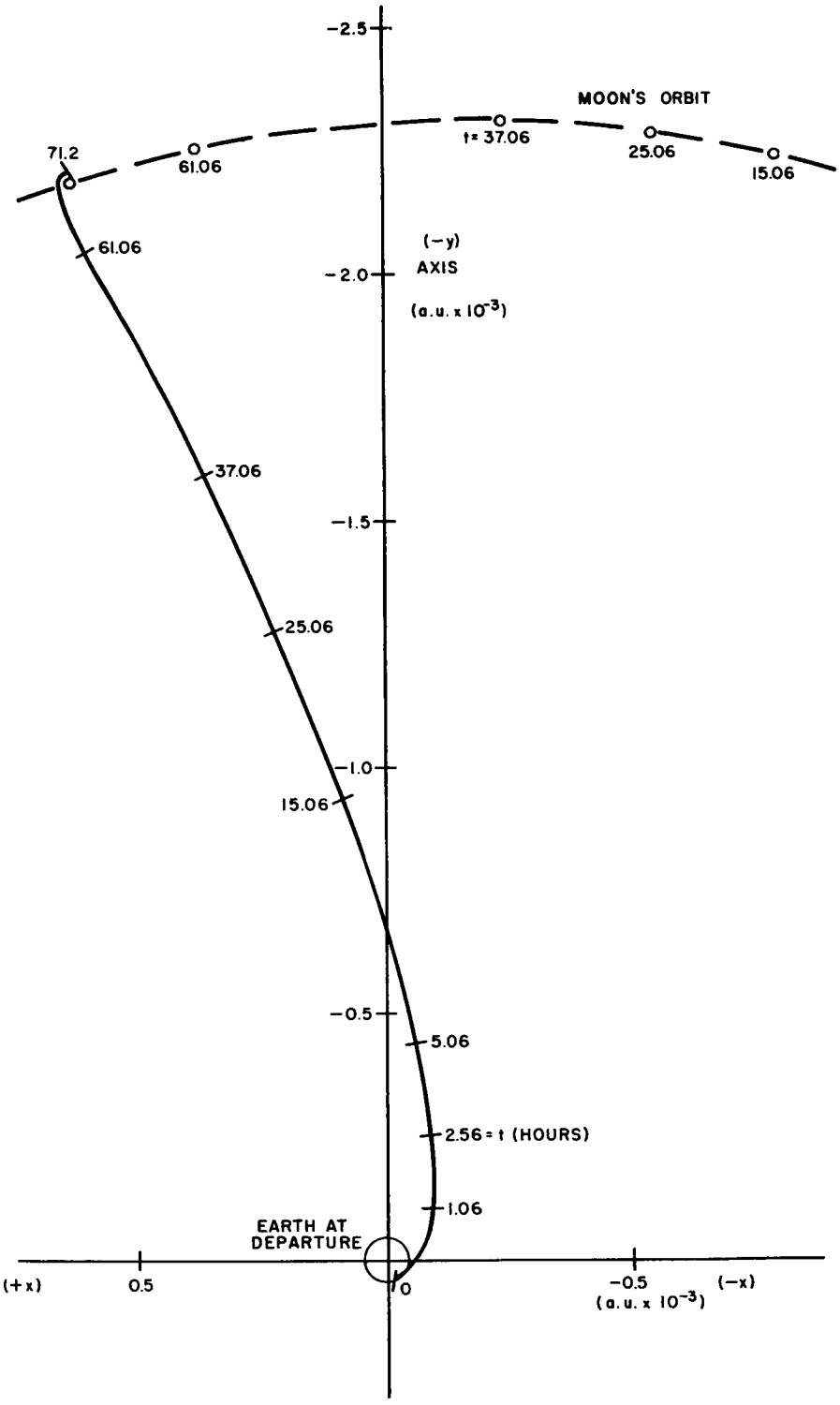


Figure I-1-9. Sample outbound transfer trajectory (x-y projection)

[redacted]

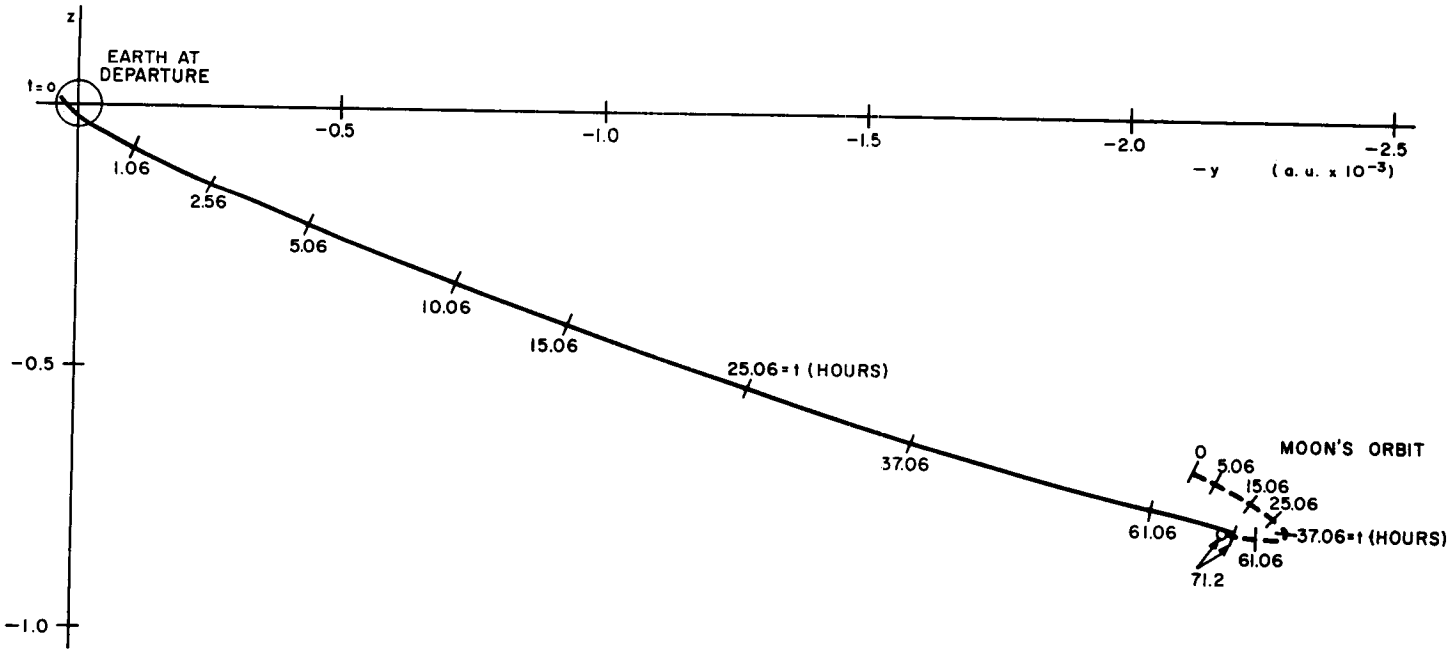


Figure I-1-10. Sample outbound transfer trajectory (y-z projection)

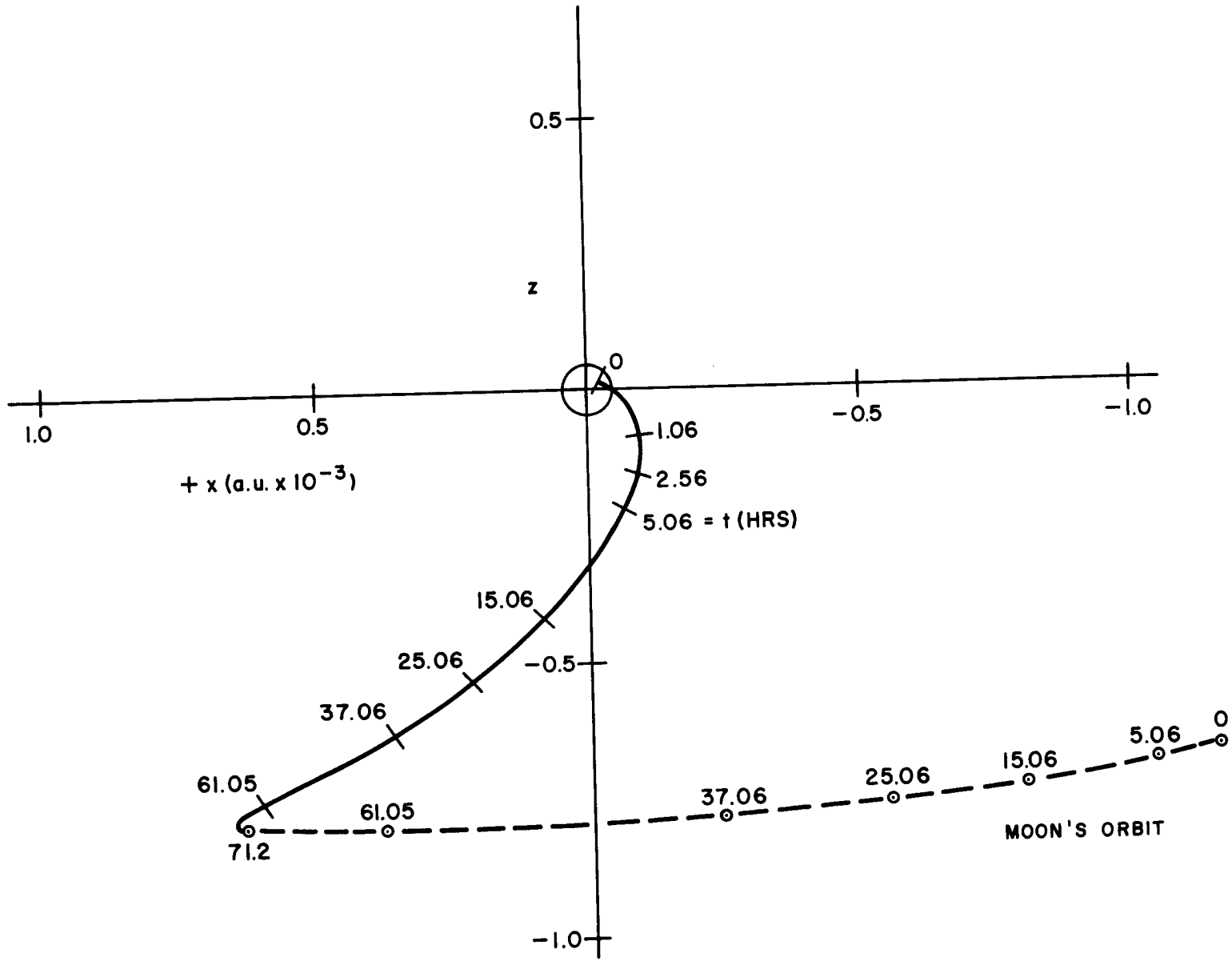


Figure I-1-11. Sample outbound transfer trajectory (x-y projection)

[redacted]

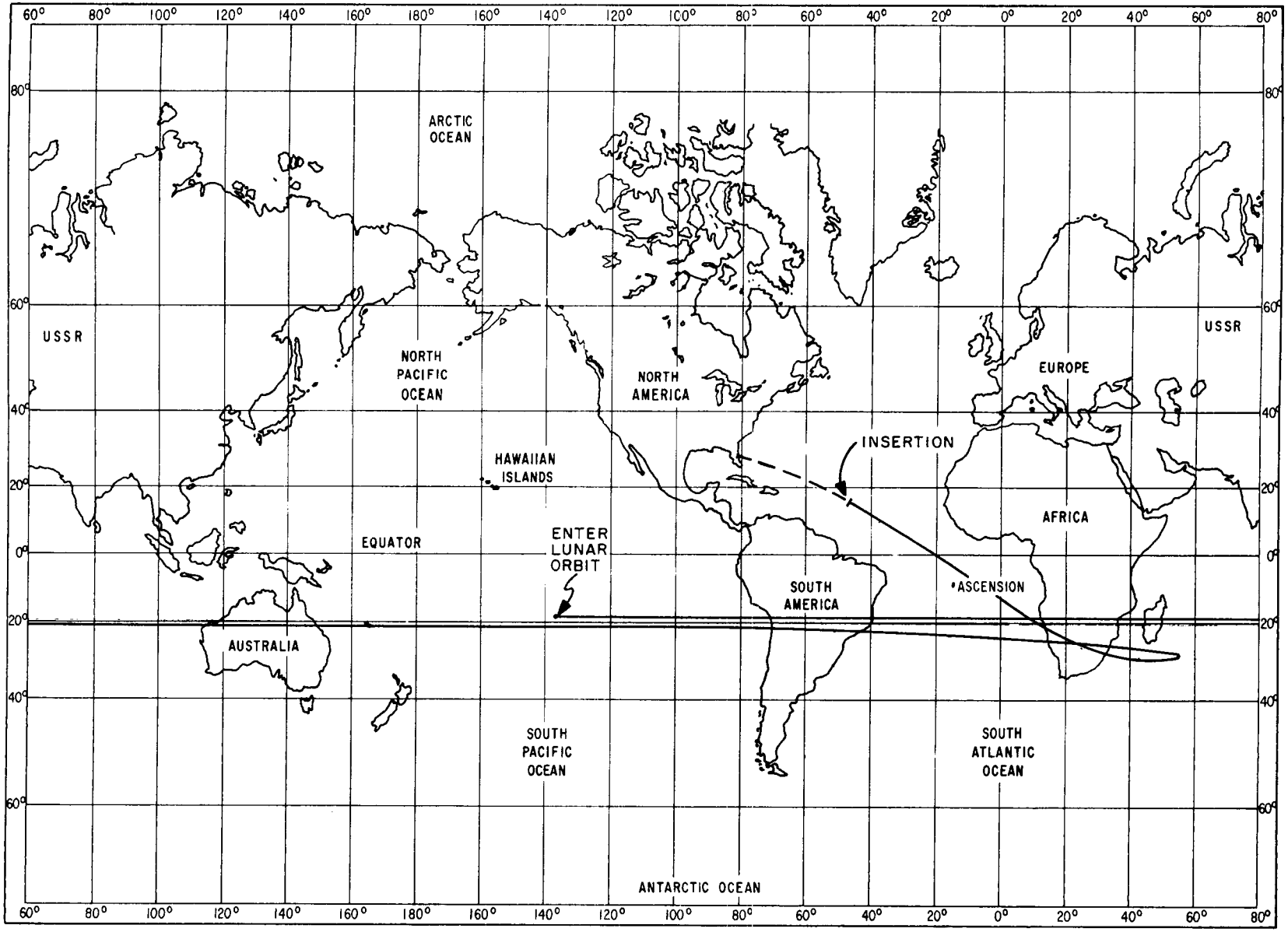


Figure I-1-12. Earth surface trace of sample outbound trajectory

[redacted]

during any phase of the lunar mission. These maneuvers, which would be used to shorten the return flight time or to modify the vehicle's mission, can be considered to be initiated during any of the three major phases of the free-flight portion of the lunar orbit and return mission. These major phases are: outbound transfer trajectory, lunar orbit, and return transfer trajectory. These phases will be discussed in the following sections.

The study of abort trajectories initiated on the launch pad and during the powered ascent trajectory are discussed in Chapter III of Volume II.

It can be assumed that the on-board propellants provided to enter and leave the lunar orbit will be available for use in effecting an emergency return to the earth. The velocity increment required to enter the range of lunar satellite orbits specified by NASA is of the order of 3300 to 3500 ft/sec, depending on the outbound transfer time. The velocity increment required to leave the lunar orbit and return to the earth has been shown to have a value of 2750 ft/sec or greater.

Some versions of the APOLLO vehicle are designed to have an on-board velocity potential of the order of 6000 to 7000 ft/sec plus an allowance for both outbound and return midcourse corrective maneuvers. This study of emergency return trajectories has considered velocity increments ranging from 1000 to 10,000 ft/sec, although a maximum value of 6000 to 7000 ft/sec is available.

For the case of the APOLLO vehicle on a continuous circumlunar mission, an on-board propulsion system with a capability of at least 4000 to 5000 ft/sec should be provided to accommodate abort maneuvers initiated during powered flight at velocities greater than circular orbital velocity, as well as for emergency returns initiated during free-flight.

1.4.1 Returns During the Outbound Trajectory

For purposes of this exploratory study the outbound transfer trajectory is approximated by an unperturbed elliptical orbit relative to an isolated earth. The instantaneous values of velocity and path angle for this trajectory are shown as a function of altitude in Figure I-1-13. Several points on this trajectory at altitudes ranging from about 100 to 100,000 miles were used to initiate the emergency return trajectories.

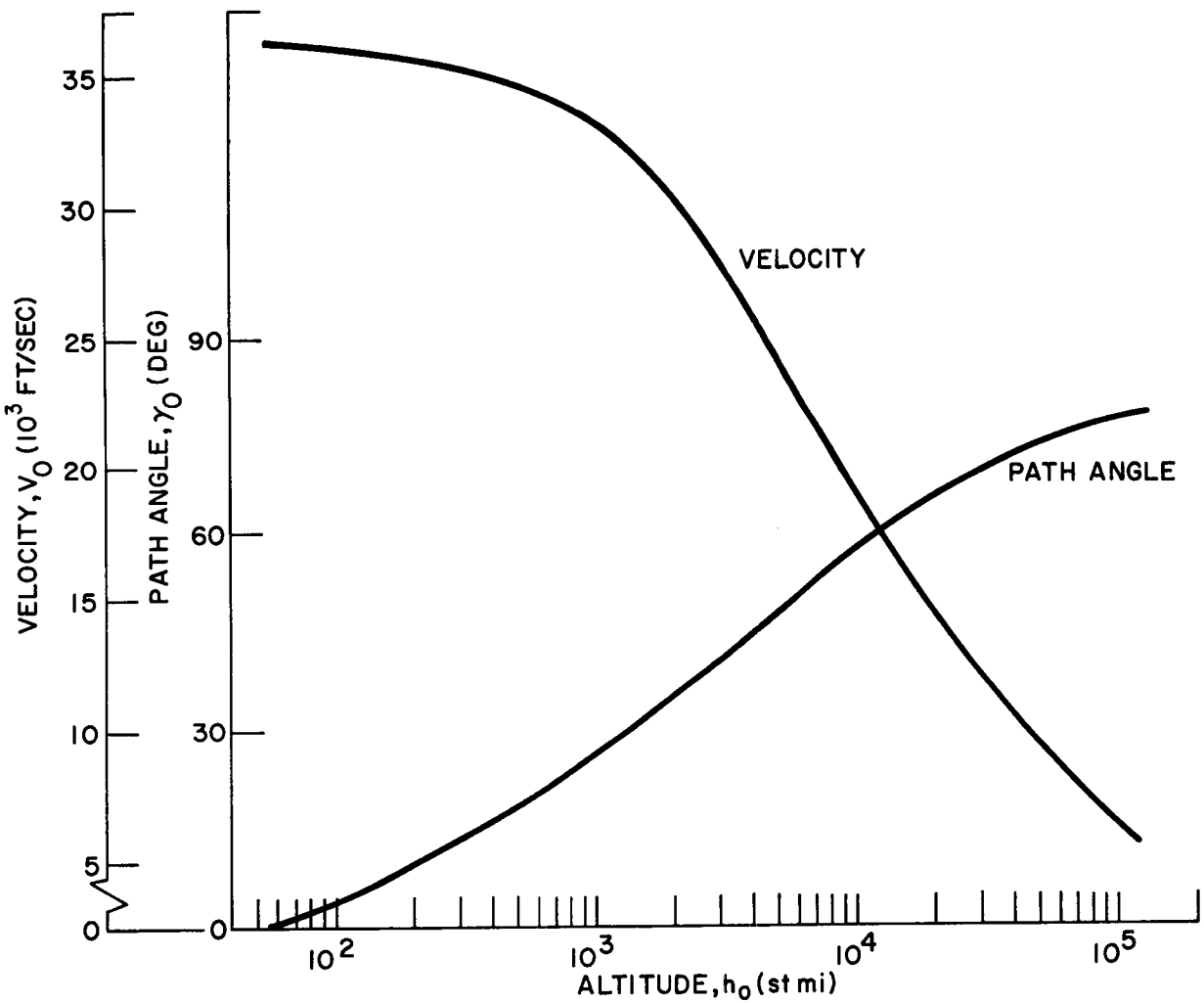


Figure I-1-13. Typical transit trajectory variables

Only one relatively simple class of emergency return maneuvers was investigated in this phase of the study. These are returns initiated by a single vector addition of the velocity increment. The geometry of the vector addition is shown in Figure I-1-14. At some radial distance, r_o, the vehicle's velocity vector on the outbound trajectory can be defined by its magnitude, V_o, at a path angle, γ_o, relative to the local horizontal.

The velocity increment, ΔV, is added in the plane of the trajectory at an angle, δ relative to the initial velocity. The positive sense of the angle is indicated. The emergency return trajectory is defined by the resulting velocity, V_1 at a path angle, γ_1.

The angle, δ, at which the velocity increment is added is dictated by the requirement that the re-entry path angle, γ_E, at the re-entry altitude of 400,000 ft be within acceptable limits. The limiting re-entry path angles for the APOLLO D-2 configuration, for example, are -4.8 and -7.2 degrees, with an average value of -6 degrees. The allowable limits on re-entry angle for the APOLLO R-3 vehicle are slightly greater. The above values are for a re-entry velocity which is equal to the local escape velocity. As the re-entry velocity is decreased, the allowable path angle tolerance increases.

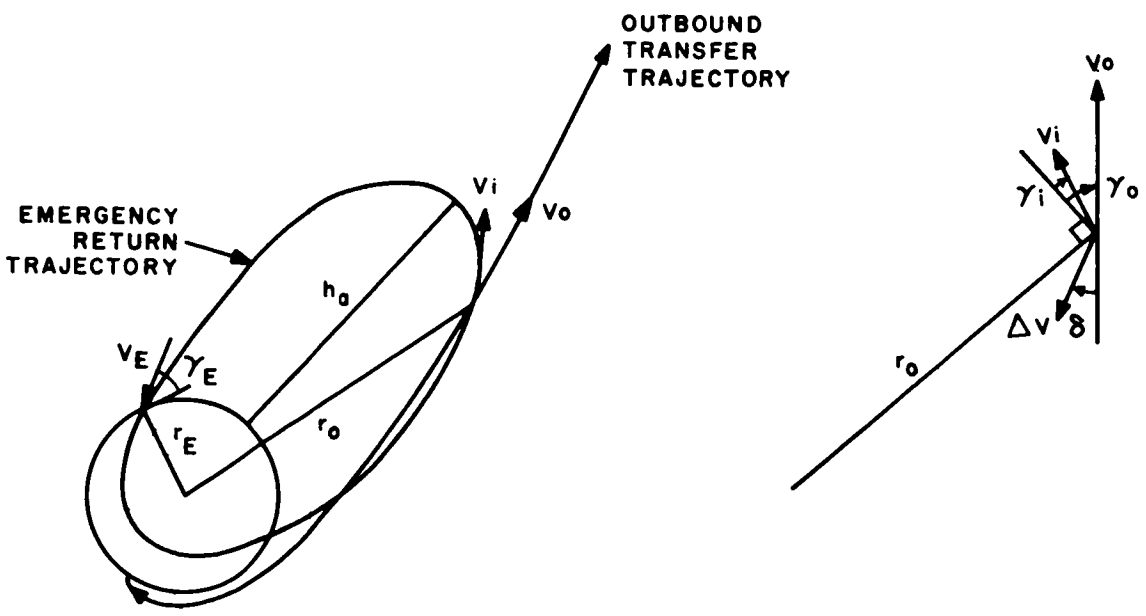


Figure I-1-14. Schematic of emergency return trajectory

[redacted]

For illustrative purposes, however, curves for emergency return trajectories defined by re-entry path angles of 0, -5, and -10 degrees will be presented for the parametric portion of the study.

The trajectory parameters of interest in a study of these emergency returns are apogee altitude, return flight time, re-entry path angle and velocity, as well as the angle δ required as a function of altitude for various velocity increments. In addition to these parameters, the re-entry latitude and longitude and the trajectory azimuth are required for the determination of the landing area.

Figures I-1-15, I-1-16 and I-1-17 present the angle δ required to establish the return trajectory. Comparison of the three graphs show relatively little shift of the angle δ with re-entry path angle for initial altitudes greater than 5000 miles. The divergence between graphs at lower altitudes arises from the shape of the trajectory required to achieve the required re-entry path angle.

The re-entry velocity, V_E, is shown as a function of initial altitude in Figure I-1-18 for the several values of velocity increment and the three re-entry path angles. The values of circular and escape velocity at an altitude of 400,000 feet are indicated for reference.

The apogee altitude, through which the return trajectory passes, is shown in Figure I-1-19 as a function of initial altitude for selected velocity increments. The line, labelled $h_o = h_a$ is included to indicate the altitudes at which the resultant velocity, V_1, is horizontal. This corresponds to apogee of the return trajectory. At higher initial altitudes the initial return path angle, γ_1, is negative so that the return is initiated after apogee.

These apogee points occur at an altitude of about 35,000 miles for a velocity increment of 8,000 ft/sec and at about 70,000 miles for a velocity increment of 6000 ft/sec. The approximate locations of the peaks of the Van Allen radiation belts are indicated along the apogee scale. The location of apogee altitudes relative to these peaks gives a rough indication of the relative magnitude of the radiation dosage encountered during the return.

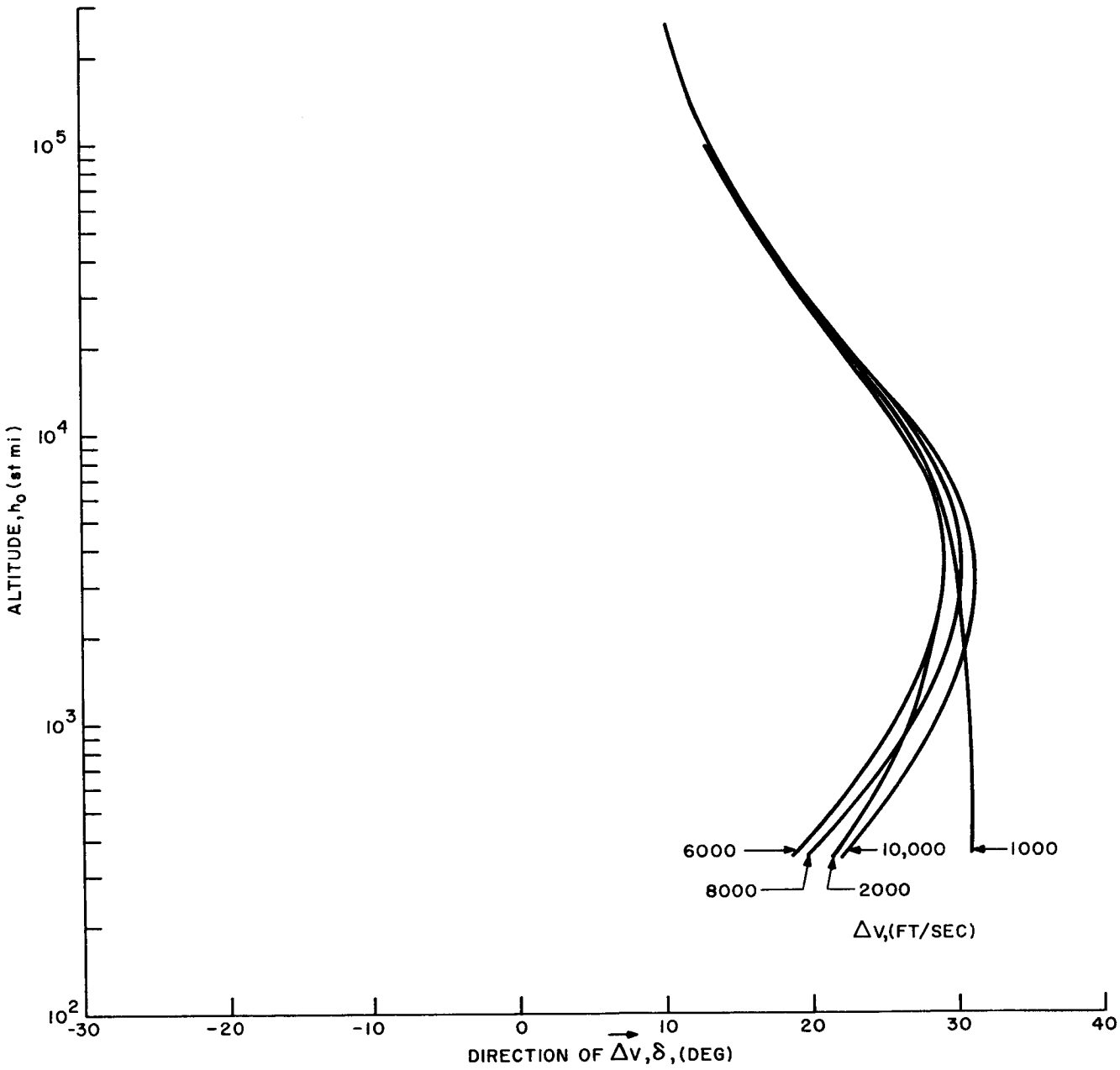


Figure I-1-15. Orientation of ΔV vector ($\gamma_E = -0°$)

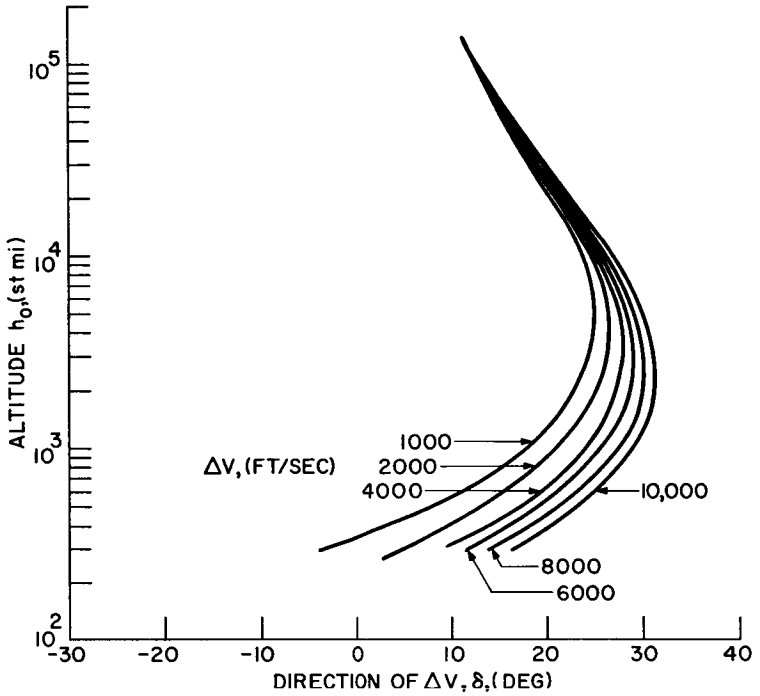


Figure I-1-16. Orientation of Δ V vector (γ_E = -5°)

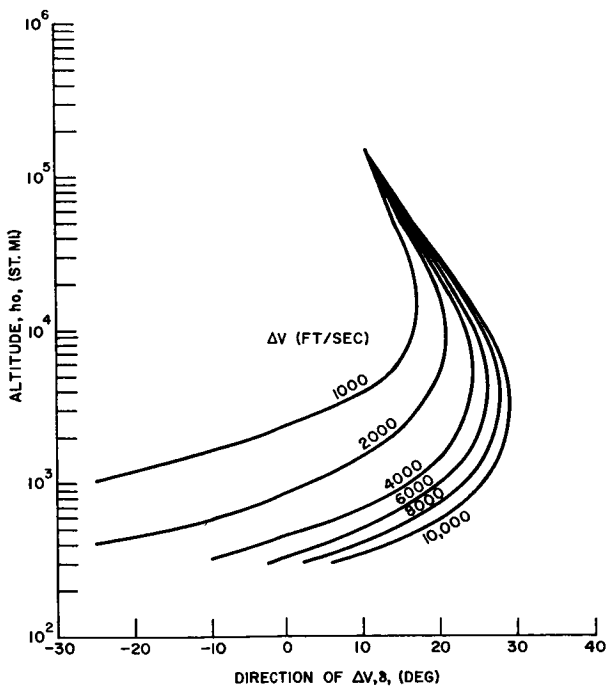


Figure I-1-17. Orientation of Δ V vector (γ_E = -10°)

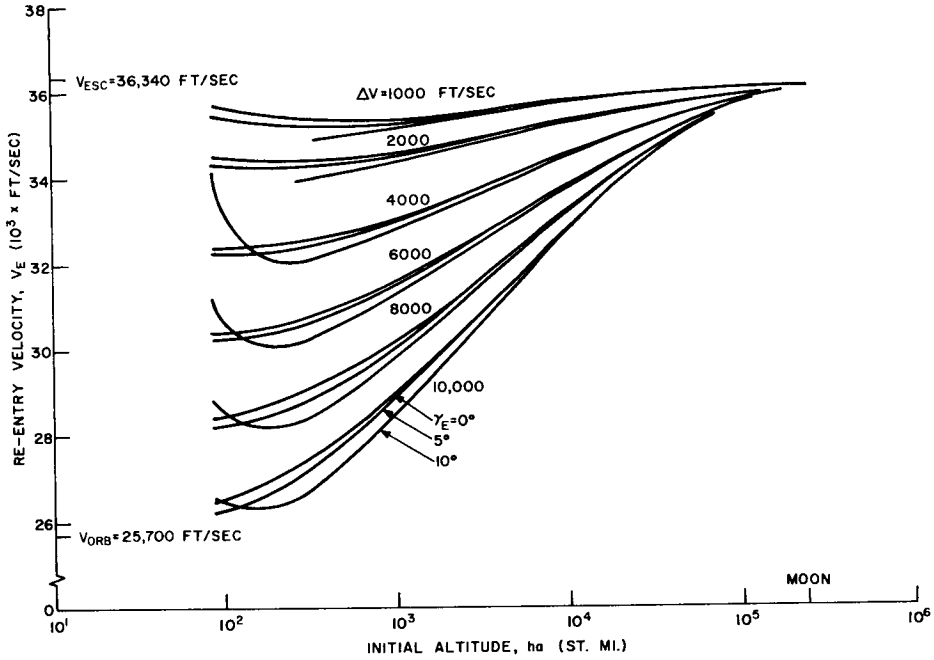


Figure I-1-18. Re-entry velocity vs initial altitude

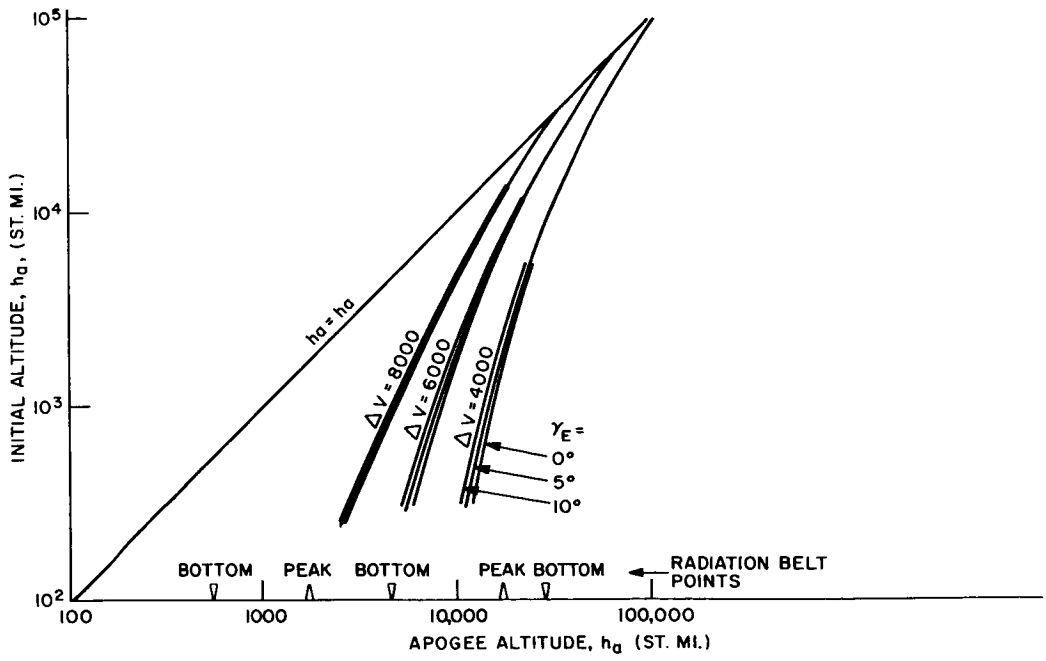


Figure I-1-19. Apogee altitude vs initial altitude for various ΔV's

The time of flight to return to perigee altitude is given in Figure I-1-19A. The change in flight time for the re-entry path angle variation considered in this study is of the order of one hour or less for all but the lowest initial altitudes. The return flight times from altitudes of 20,000 miles or more with velocity increment of 4000 ft/sec or less are longer than 1 day.

The times of flight times change slope at the altitudes for which the returns are initiated at apogee.

The latitude of the re-entry point is computed from the latitude and azimuth at the point of emergency return using the return transit angle. This transit angle varies from values of approximately 300 degrees for returns initiated at low altitudes to approximately 180 degrees at initial altitudes of the order of 100,000 miles for velocity increments of 4000 to 6000 ft/sec. The longitude of the re-entry point relative to the rotating earth is computed in a similar fashion, taking the return flight time into account.

Figure I-1-20 presents the earth surface trace of the early portion of the outbound transfer trajectory used in this portion of the study as the solid curve. This trace reaches 33 degrees south latitude over Africa and then approaches 20 degrees south latitude on its first pass around the earth. The locus of re-entry points is indicated by the dashed lines between 20 and 30 degrees north latitude for emergency returns defined by a re-entry path angle of -6 degrees. These trajectories are initiated by a representative velocity increment of 6000 ft/sec.

The numbered points on the dashed lines show the re-entry points of return trajectories initiated at the correspondingly numbered points on the outbound trajectory. The initial altitudes of these points vary from about 100 for point #1 to 100,000 miles for point #6.

The re-entry locus is seen to begin in the vicinity of Hawaii and then progresses westward with point #4 located in the vicinity of Formosa. The trace circles the earth a number of times for the entire outbound trajectory. Points #5 and #6, which also occur between the West Coast and Hawaii, are located on the first and second complete revolutions of the trace, respectively.

Since this re-entry locus passes over Africa and Asia, which are recovery regions to be avoided if possible, provision must be made to select the re-entry longitude as a function

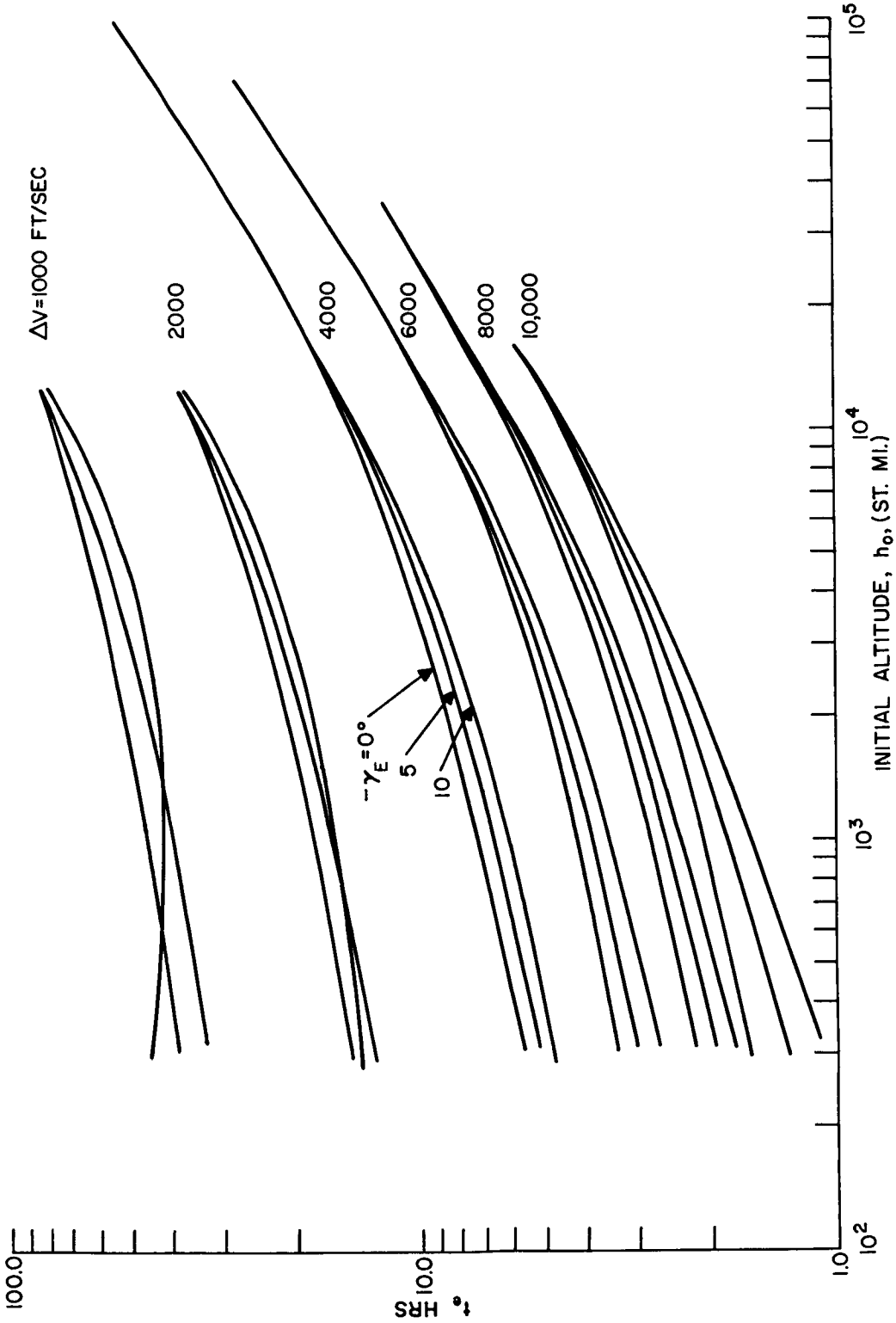


Figure I-1-19a. Time to re-entry vs initial altitude for various ΔV's

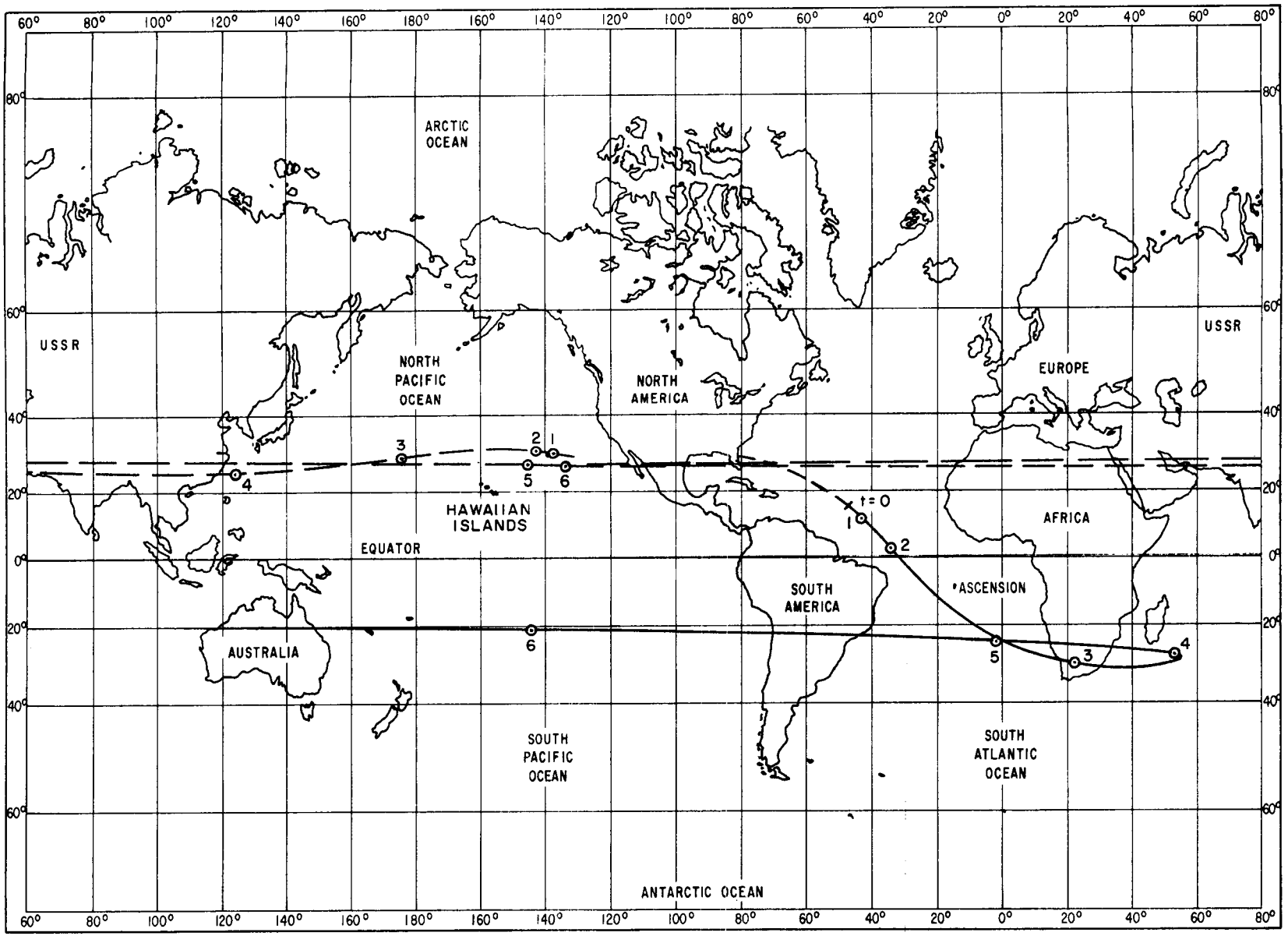


Figure I-1-20. Traces of outbound and emergency return points

of the initial altitude. Two techniques are available for doing this. The first is to change the velocity increment to some other value, added at the angle required to provide acceptable re-entry conditions. Since the velocity increment used for this re-entry locus is not the maximum available value, the re-entry longitude of these points could be shifted either toward the east or west. The other technique available is to delay the initiation of the return until the APOLLO vehicle is at a higher altitude. If this is acceptable from other considerations such as total time to return to earth, it would be a very efficient technique, as shown by the re-entry longitude difference of nearly 360 degrees between returns initiated at points #5 and #6.

In addition, re-entry latitude control could be obtained if desired by orienting the vector velocity increment so that it has a component normal to the plane of the outbound trajectory.

Another reason for requiring flexibility in the selection of the emergency return trajectory was shown when the integrated radiation dose for emergency return trajectories was investigated. The dose encountered by the vehicle on a return initiated at an altitude of 4000 miles by a velocity increment of 6000 ft/sec was found to be unacceptably high. The vehicle's trace through the radiation belts was shifted sufficiently to lower the dose by reasonable levels by reducing the velocity increment to a value of either 2000 or 4000 ft/sec.

The above discussion considered one class of emergency return trajectories: those initiated by a single velocity increment at the specific angle required to achieve satisfactory re-entry conditions. Another possible class of returns, which has not been studied, is the two-impulse case. For these returns, the first velocity increment would be applied at the initial altitude, as before. This velocity increment would be directed toward the earth in a radial direction. The resulting velocity vector would establish an ellipse with any arbitrary perigee altitude. The second velocity increment would be applied horizontally at apogee of this return ellipse. Its magnitude and direction would be selected to achieve the required re-entry path angle at an altitude of 400,000 feet.

This two-impulse maneuver would be well suited to changing the inclination of the plane of the final return ellipse if this is deemed necessary in order to change the re-entry latitude and azimuth.

The detailed, numerical studies reported above were made for altitudes up to 100,000 miles using, as a model, an isolated earth. The results obtained up to this altitude can be extrapolated to somewhat higher altitudes with reasonable accuracy. Emergency return trajectories initiated at altitudes greater than 150,000 or 200,000 miles would require use of the "n-body" trajectory computation program since the gravitational attractions of the Moon and sun will influence the vehicle's motion during the return trajectory.

One aspect of emergency return trajectories originating on the outbound transfer trajectory, which has not been investigated, is whether there is a maximum practical or effective altitude limit for initiation from a near-nominal trajectory. This limit might arise from the fact that return to the earth might be equally fast if the vehicle were allowed to continue on its trajectory around the Moon and return to earth without the addition of the velocity increment.

This study of emergency return trajectories would be directly applicable to a similar study for the major portion of the same free-flight phase of a continuous circumlunar mission defined by the same insertion velocity. The possible small difference in the lunar approach trajectory would result in small differences during the early portion of the outbound trajectory.

1.4.2 Return During Lunar Orbit Phase

Specific emergency return trajectories have not been computed for exit from the lunar orbit prior to the nominal final revolution. Considerations of returns from these non-nominal revolutions is discussed in Section 1.6.

The velocity increment available for initiating these returns is limited to the velocity potential remaining on-board after the lunar orbit has been established. This remaining velocity potential should be large enough to allow a reserve increment above that required for dis-orbit on the nominal revolution as mentioned in Section 1.6.

1.4.3 Returns From the Earth-Bound Trajectory

This section is included to complete the coverage of all phases of the lunar orbit and return mission. At this point in the mission the major portion of the on-board velocity potential has been used in the maneuvers of entry and exit from lunar orbit. The remaining velocity capability is the allowance for return mid-course trajectory corrections, plus the possibly unused reserve allowance for the exit maneuver.

Since the nominal total flight times for return are of the order of 3 to 3.5 days, the vehicle's velocity could be increased, if propulsion were available, to shorten the total flight time to values of possibly 2 to 2.5 days if the emergency occurred almost immediately after leaving the lunar orbit. Decreasing the total return flight time to values of the order of 1 to 1.5 days is impossible because of the very large velocity increments required.

The return trajectory described in Section 1.6.2 shows that the vehicle spends the first 9 to 12 hours at approximately the Moon's distance from the earth as it escapes from the Moon's gravitational field and starts to fall toward the earth. Thus, the portion of the trajectory which approximates an ellipse relative to the earth extends only during the last 80 hours of the flight.

If the emergency arose very far from the Moon on return, it probably would not be practical to attempt this type of maneuver in view of the relatively small time saving possible. In addition, changing the return flight time results in re-entry occurring at a different longitude. If this changed longitude did not correspond to an alternate landing site, the recovery of the vehicle and crew would be made considerably more difficult.

1.5 LUNAR SATELLITE ORBIT

The objective of the lunar orbiting phase of the APOLLO mission is the reconnaissance of the near face of the moon to provide information required for the selection of an eventual landing site.

According to Reference 1* studies have been made to tentatively select likely areas for consideration as landing sites. Three criteria were established to determine suitable areas:

- A minimum area of approximately 30 miles diameter with a relatively smooth floor, free from clefts, crater cones, etc.
- A location removed from the Moon's equatorial regions.

The first of these criteria will sort out those regions which obviously have uneven surfaces and thus be undesirable. The second criterion is predicated on the large temperature fluctuations at points near the equator during a lunar day.

Nine landing areas, shown in Figure I-1-21 which is copied from Reference 1, are considered feasible on the basis of the above criteria. Four of these areas are located in the northeast quadrant of the visible face of the Moon, and a total of 7 out of 9 are located more than 30 degrees from the lunar equator.

Based on the above distribution of potentially-interesting landing sites, and also to maximize the amount of surface area which is visible for any given orbiting duration, it would be desirable to establish the lunar satellite orbit at an inclination of at least 45 degrees relative to the quarter.

The lunar orbit, into which the APOLLO vehicle is to be placed for a limited number of revolutions, is delimited in the NASA APOLLO work statement. The satellite orbit is to be an ellipse with a maximum altitude, beyond the Moon, of 1000 to 2000 n mi and a minimum altitude, over the near side of 50 to 100 n mi. The orbital periods, assuming an isolated Moon with an inverse-square gravitational field, are shown in Figure I-1-22. Reading the abscissa as apolune altitude, the lower curves show the periods for ellipses with perilune altitudes of 50 and 100 n mi. These orbital periods vary from about 3.5 to 5.5 hours for apolune altitudes from 1000 to 2000 n mi. These periods correspond to frequencies of 7 to 4.5 revolutions per day.

Reference 1. "Notes on Possible Lunar Landing Sites" received from NASA STG Jan., 1960.

[redacted]

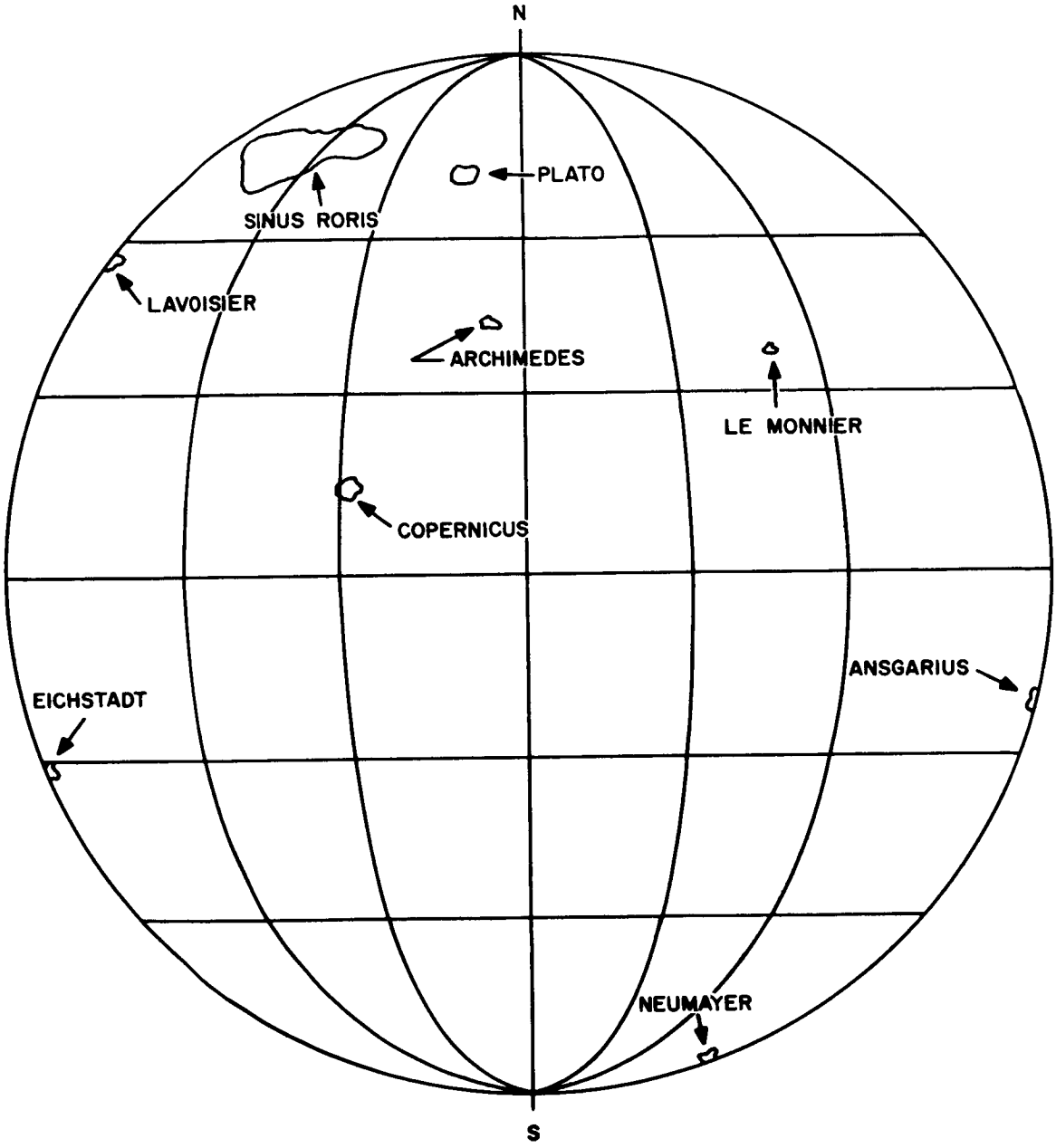


Figure I-1-21. Possible lunar landing areas

[redacted]

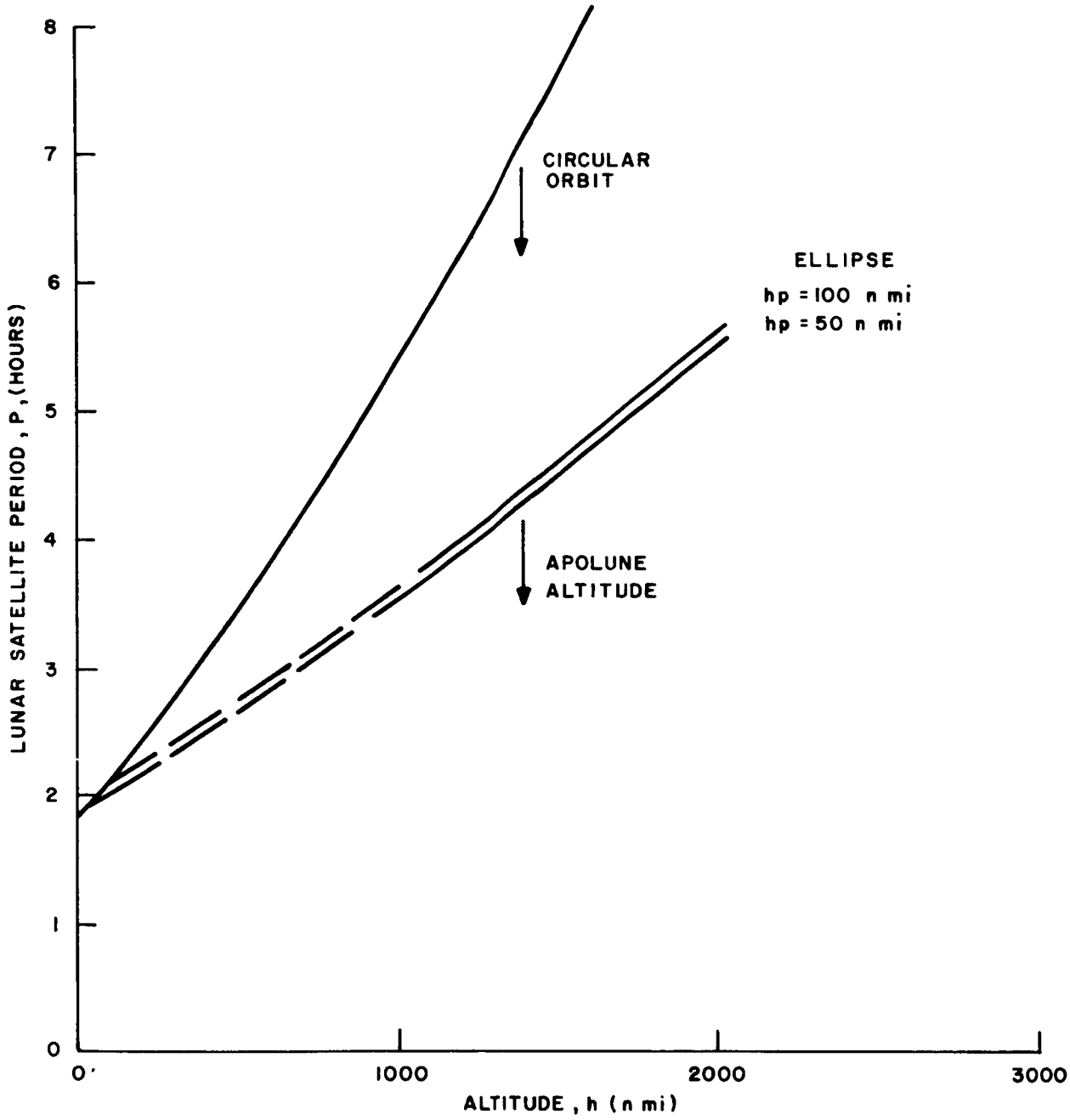


Figure I-1-22. Lunar satellite period

The corresponding periods for circular orbits are indicated by the upper curve. These vary from 1.8 hours for zero altitude to 7.6 hours for a circular orbit at an altitude of 1500 n mi.

1.5.1 Lunar Orbit Entry Increments

Since the approach velocity of the APOLLO vehicle is hyperbolic relative to the Moon a velocity increment must be subtracted to place the vehicle into a lunar satellite orbit. The magnitude of the entry increment required is a function of the insertion velocity at the earth (or outbound transit time), the altitude of the initial approach trajectory, and also of the desired final perilune altitude. The unbraked approach velocity is shown in Figure I-1-23 as a function of approach altitude for selected values of outbound transit trajectory insertion velocity. One of the interesting features is the magnification, at the Moon, of changes in the insertion velocity at the earth. For example a change of 100 ft/sec in the insertion velocity results in a change of about 425 ft/sec in the vehicle's velocity relative to Moon. This magnification factor is not constant, but varies slightly with altitude above the Moon.

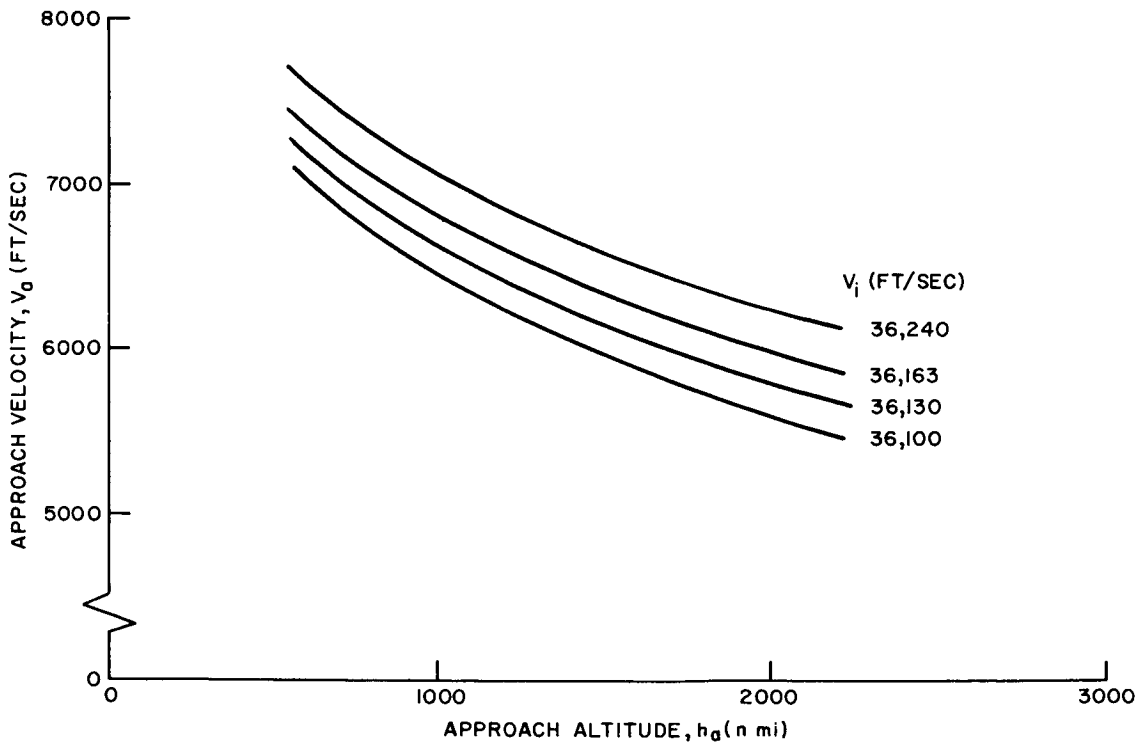


Figure I-1-23. Lunar approach velocity as a function of approach altitude

[redacted]

It has been assumed in this study that the velocity increment would be subtracted at perilune of the approach hyperbola in a direction which is colinear with the vehicle's velocity relative to the Moon.

It has been proposed in this study that the velocity increment used to establish the lunar orbit be subtracted in two separate thrusting phases. The first velocity change would change the initially-hyperbolic velocity to the value corresponding to a circular orbit at the initial approach altitude, and the second velocity decrease would establish the final, desired perilune altitude.

This two-impulse type of maneuver is attractive in the sense that the major axis of the ellipse can be established at an arbitrarily selected orientation in the orbital plane without the velocity penalty associated with a vector addition of the velocity increment. An analysis, which is beyond the scope of this study, may show an optimum orientation of the satellite ellipse for the subsequent orbit ejection maneuver on a specific revolution for the return to earth.

The magnitude of the velocity increments required for a family of transfer trajectories defined by the same insertion velocity is shown in Figure I-1-24. The initial increment, required to establish the circular orbit, is nearly constant at about 3000 ft/sec over a range of approach altitudes. The additional velocity increment required to establish elliptical orbits with perilune altitudes of 50 and 100 n mi vary from 550 to 850 ft/sec as the approach altitude varies from 1000 to 2000 n mi. The total increments for these apolune altitudes are about 3600 and 3800 ft/sec, respectively. Raising the perilune altitude from a value of 50 to 100 n mi is seen to reduce the velocity increment by only an average of 50 ft/sec.

Figure I-1-25 presents the total velocity increments required to establish lunar orbits with a 50 n mi perilune for the several of the families of transfer trajectories computed in this study.

Figure I-1-26 shows a cross-plot of the preceding graph to indicate the effect of insertion velocity at the earth on the velocity increment required to establish the lunar orbit. These velocity increments vary from 3300 to 3900 ft/sec for the cases studied.

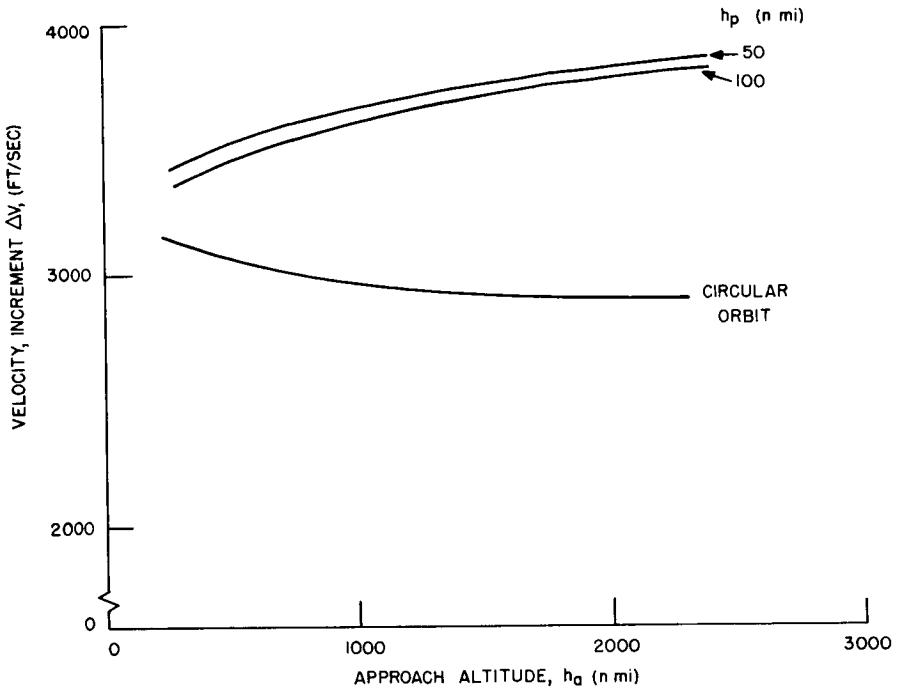


Figure I-1-24. Velocity increment required to establish lunar orbits for a sample insertion velocity

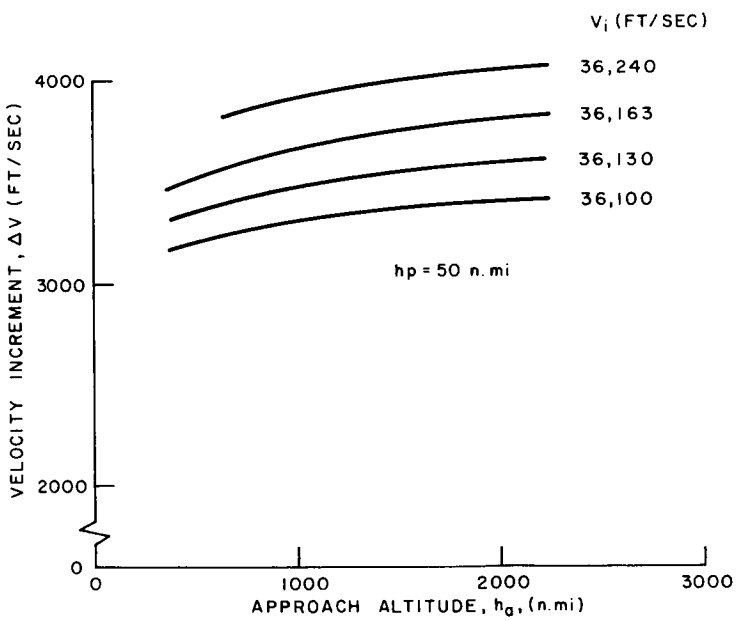


Figure I-1-25. Velocity increment required to establish a lunar orbit as a function of approach altitude

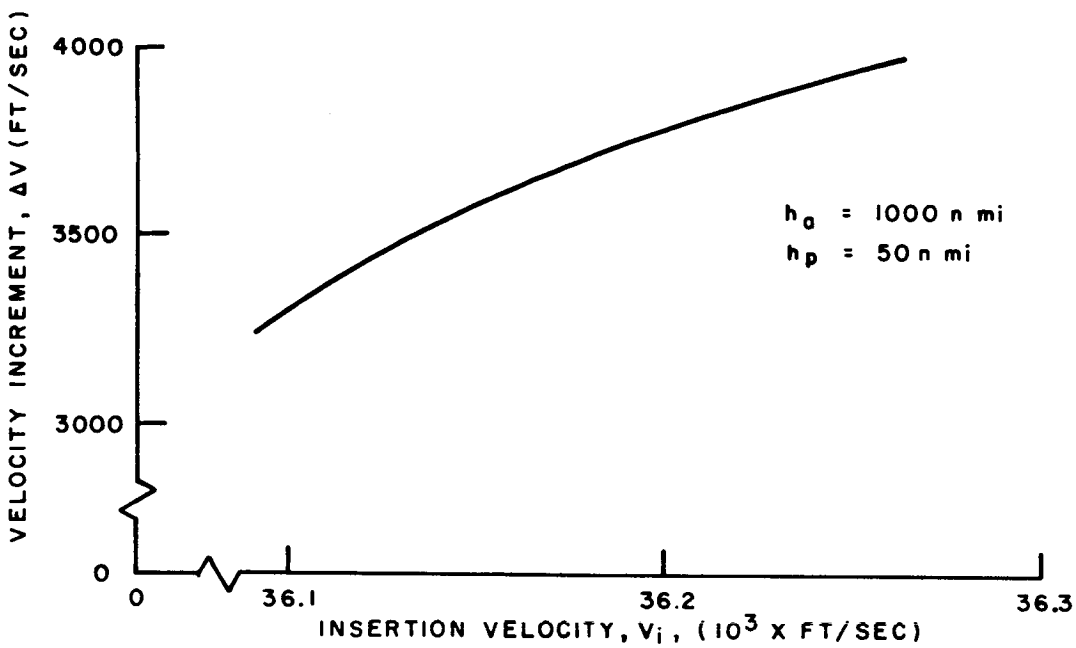


Figure I-1-26. Velocity increment required to establish a typical lunar orbit

The effect of limiting the magnitude of the velocity increment available to establish the lunar orbit is shown in Figure I-1-27. The shaded rectangle at the left of the graph indicates, for reference purposes, the lunar orbit parameters specified by NASA. An abritrary velocity increment of 2700 ft/sec is applied to three families of outbound trajectories which approach the Moon at altitudes ranging from 400 to 2000 n mi. The curve for an insertion velocity of 36,100 ft/sec shows that the resulting orbits vary from ellipses with apolune over the near face of the Moon, through a circular orbit at an altitude of about 800 n mi, and then to ellipses with apolune over the far side of the Moon. As the transfer trajectory energy is increased, the approach altitude which results in a circular orbit moves to a higher value, and finally the resulting ellipse has its apolune over the near face for all values of approach altitude.

1.5.2 Sample Lunar Orbit

For purposes of further analysis of the overall APOLLO mission, an elliptical lunar satellite orbit was established immediately from the sample outbound transfer

[redacted]

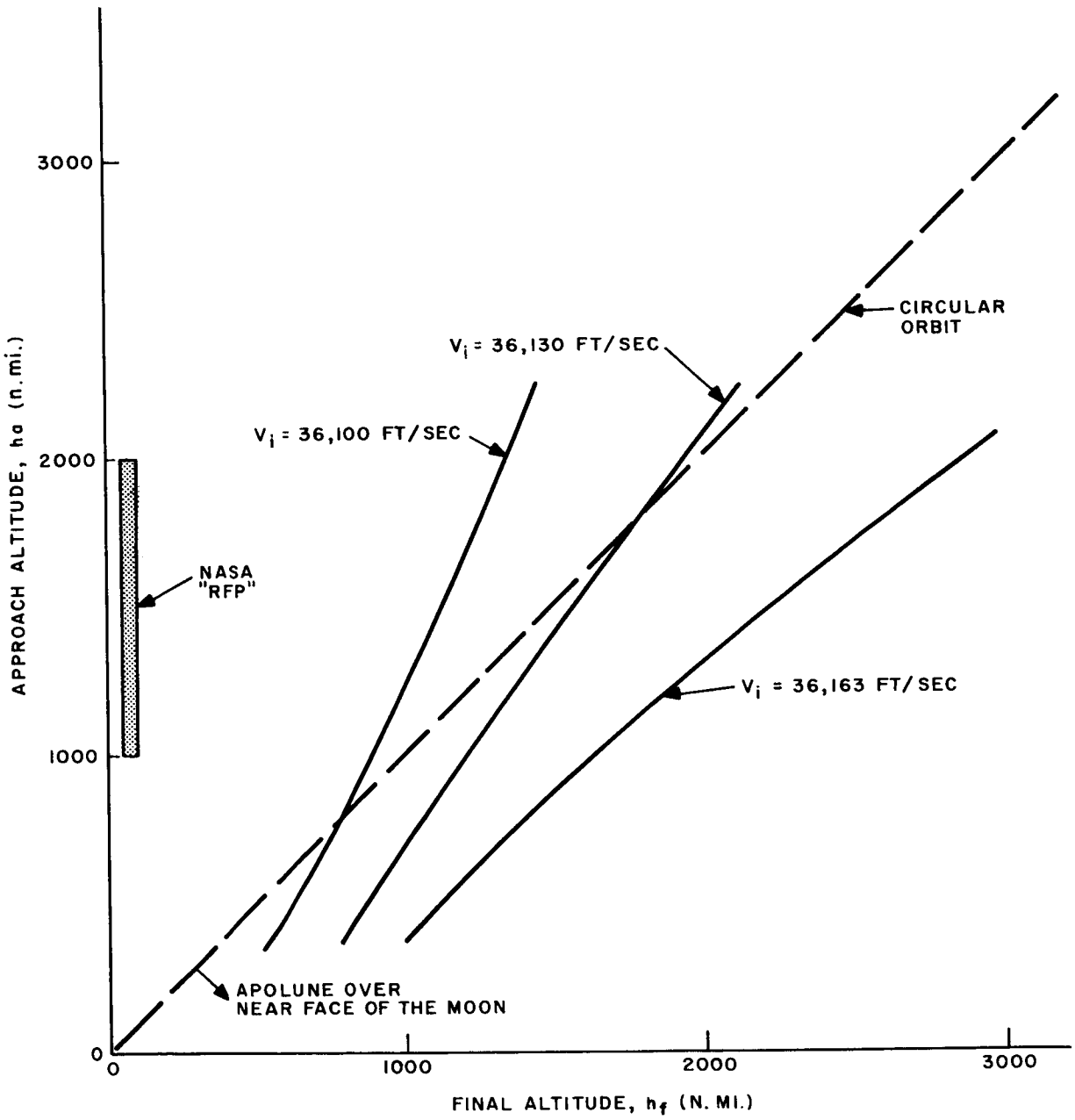


Figure I-1-27. Lunar orbits resulting from a velocity increment of 2700 ft/sec

[redacted]

trajectory described in Section 1.3.1. The lunar approach is at an altitude of 1122 n mi.

A velocity increment of 3315 ft/sec is used in one impulse to establish an orbit with a 50 n mi perilune. Apolune of the ellipse is located at perilune of the approach trajectory. The orbital period of the satellite, based on the initial value of the major axis, is 3.74 hours. The point of entry into the satellite orbit is located at an angle of 173 degrees around the leading limb from the center of the face of the Moon.

The x - y projection of the first revolution of the lunar satellite orbit is shown in Figure I-1-28. The positions of the sun and earth at insertion into orbit are indicated. The terminal portion of the outbound transfer trajectory, as well as the extension of that trajectory past the Moon, are included for reference. The y - z and x - z projections of the orbit are given in Figures I-1-29 and I-1-30. Dashed segments indicate when the orbit is beyond the Moon in that projection.

The lunar satellite orbit was computed with the General Electric "n-body" program using a Moon-centered, Encke method of computation. The computation was continued for a total orbiting duration of 6.25 days, equivalent to approximately 39 complete orbital revolutions, to "experimentally" investigate the behavior of the orbit under the perturbing action of the net gravitational attractions of the earth and the sun.

1.5.3 Orbital Perturbation

As the APOLLO vehicle travels around the Moon in its satellite orbit, it is subjected to net gravitational attractions of the earth and the sun which change in magnitude during each revolution. These accelerations act on the vehicle to perturb its path and cause the orbit to diverge from the Keplerian ellipse it would follow if the vehicle and the Moon were a pair of bodies isolated in space. The magnitude of these perturbations are a function of the ratio of the perturbing accelerations to the Moon's central force field acceleration at any point. The resulting effect of these accelerations is a function of the orientation of the perturbing body relative to the major axis of the satellite ellipse.

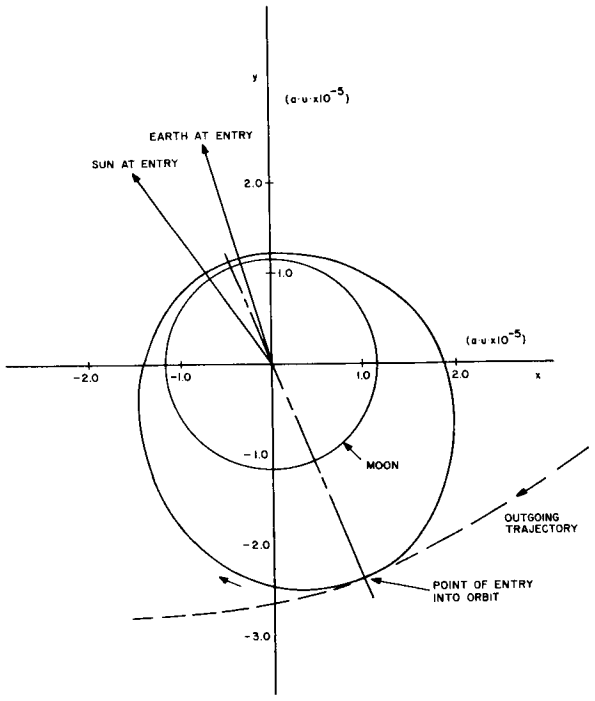


Figure I-1-28. Typical lunar satellite orbit (x-y projection)

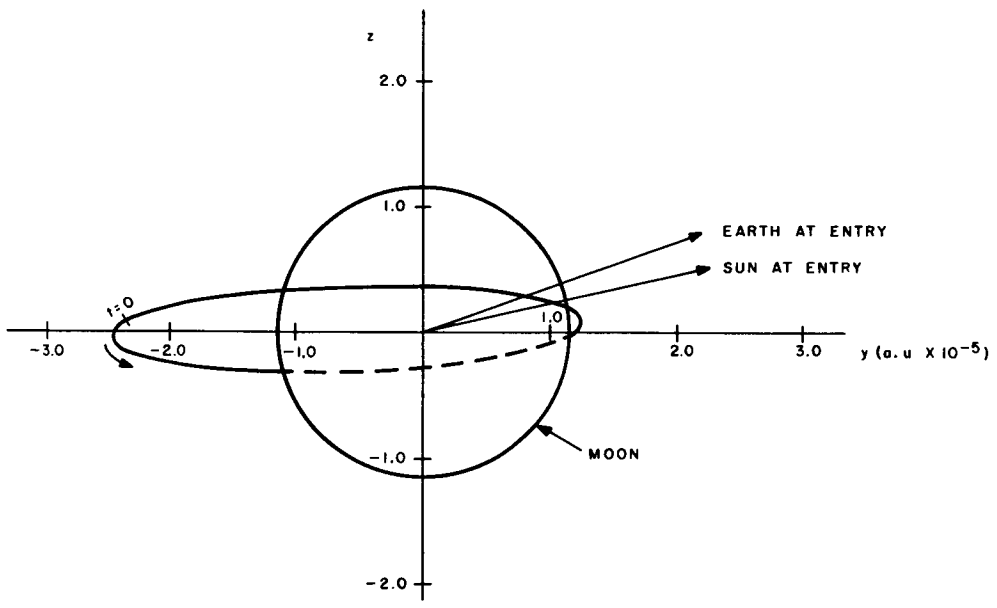


Figure I-1-29. Typical lunar satellite orbit (y-z projection)

[redacted]

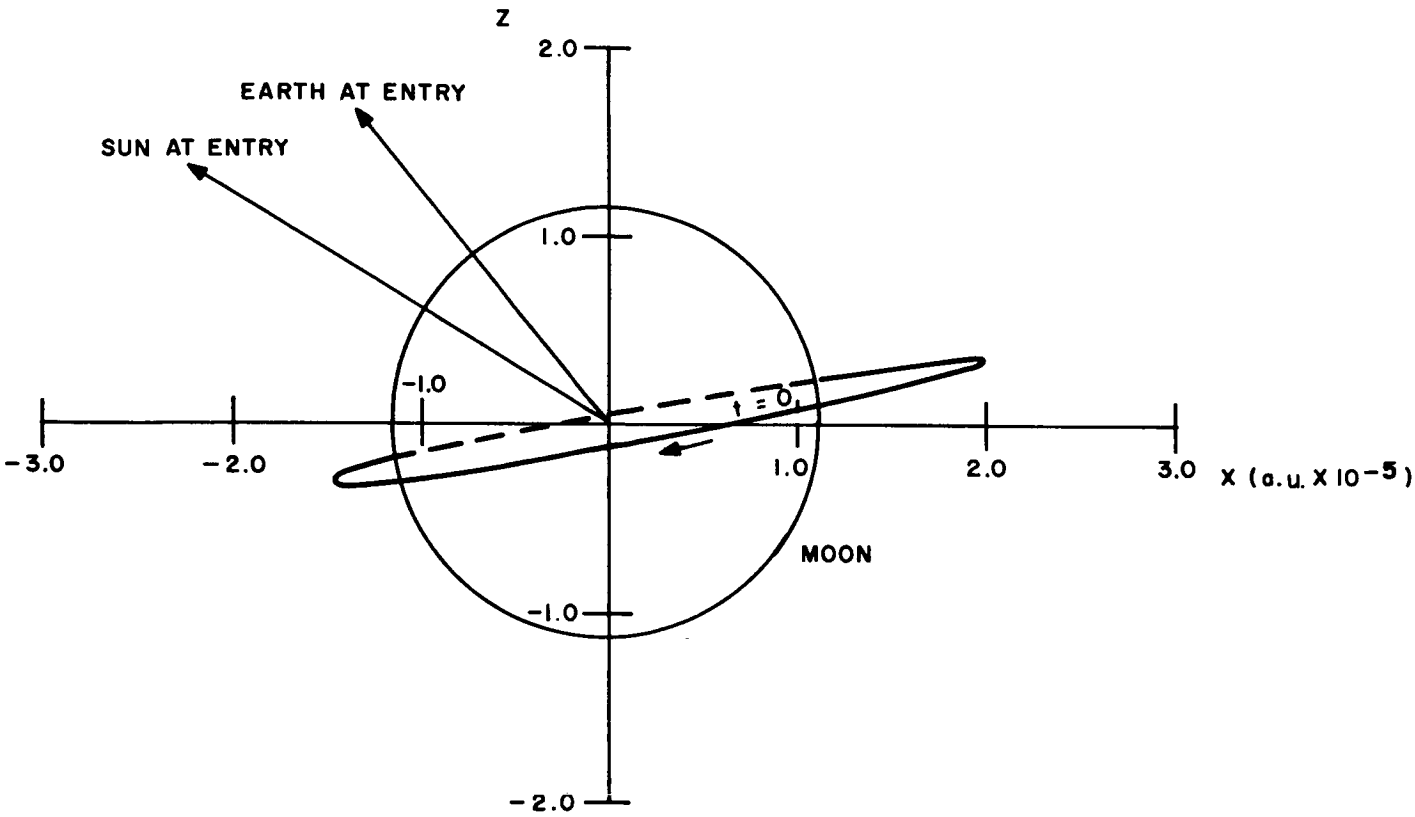


Figure I-1-30. Typical lunar satellite orbit (x-z projection)

The effects of the earth and the sun will differ because of the difference in their motion relative to the Moon. For orbiting durations of the order of one or two weeks, the sun can be considered as a perturbing center whose position relative to the lunar orbit is essentially fixed. The earth, on the other hand, is a perturbing body which rotates at a mean rate of about 13 degrees per day. Thus, in a seven day interval, the earth will be approximately 90 degrees from its initial direction.

The combined effect of the perturbations of the earth and sun can be shown by the time histories of the orbital parameters relative to the Moon. Figure I-1-31 indicates the long-period variation of the semi-major axis and eccentricity. These curves are approximate since the short period oscillations have been smoothed out. During the 150 hours of orbiting time which was computed, the semi-major axis causes the instantaneous mean motion, and therefore the orbital period to vary as a function of time. The amplitude of the short-period oscillation of the semi-major axis is seen to be of the same order of magnitude as that of the long-period oscillation as shown by the

[redacted]

[redacted]

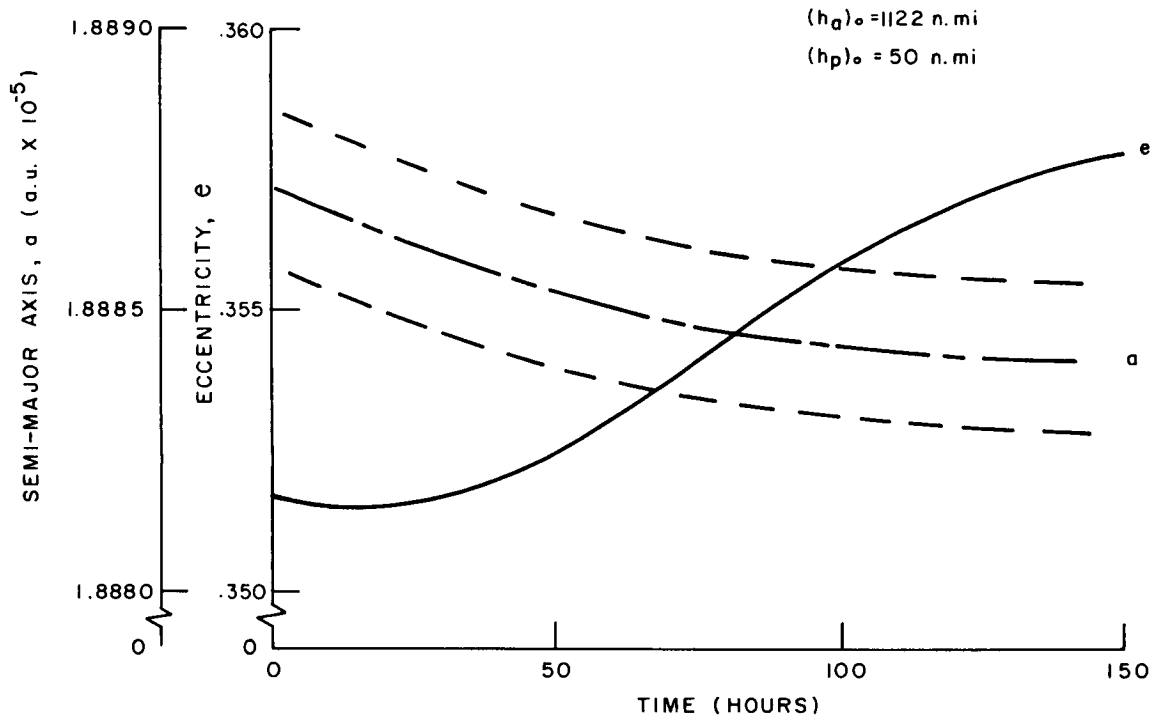


Figure I-1-31. Lunar Orbit Perturbations mean values

short-dashed envelope curves. The amplitude of the short-period oscillation of the eccentricity is small compared to the amplitude of the long-period variation.

Figure I-1-32 presents the short-period variations of semi-major axis and eccentricity during the first two orbital revolutions. The approximate locations of apolune and perilune are indicated by tics along the time scale. The osculating semi-major axis is seen to vary with a period of half the orbital period. The amplitudes at apolune and perilune are unequal due to the relative ratios of the perturbations at these points.

The eccentricity oscillates with a period nearly equal to the orbital period. This short-period oscillation is seen to be imposed on a long-period oscillation of large amplitude as shown in Figure I-1-31.

Figure I-1-33 presents comparable curves for the 17th and 18th orbital revolutions. The dis-orbit times used for the study of the return trajectory are between 70 and 70.5 hours. (Note that the scale for eccentricity is changed.)

[redacted]

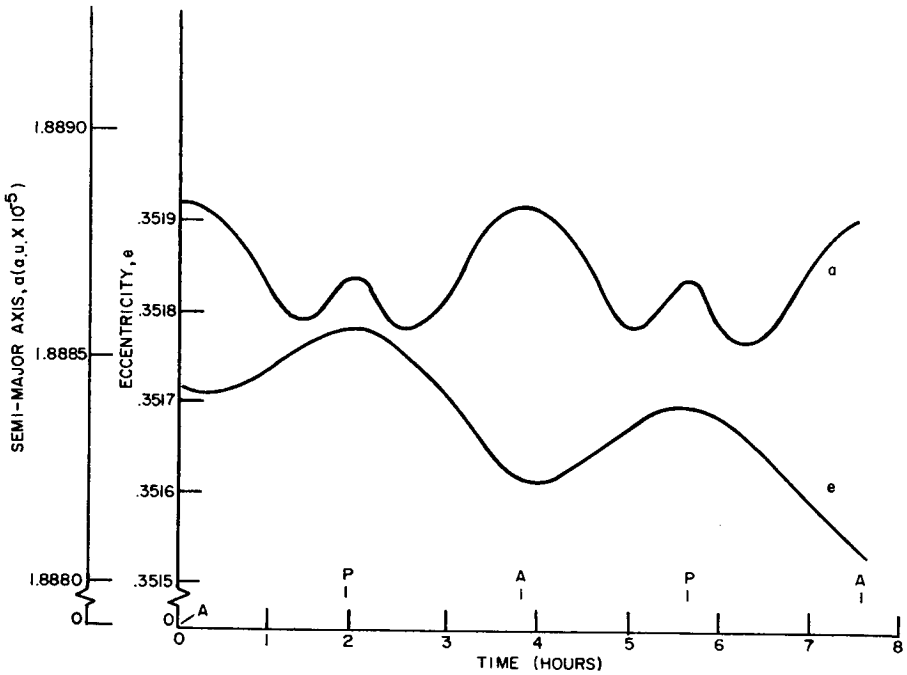


Figure I-1-32. Lunar orbit perturbations, (Revolutions #1, 2), $(h_a)_o = 1122$ nmi, $(h_p)_o = 50$ nmi

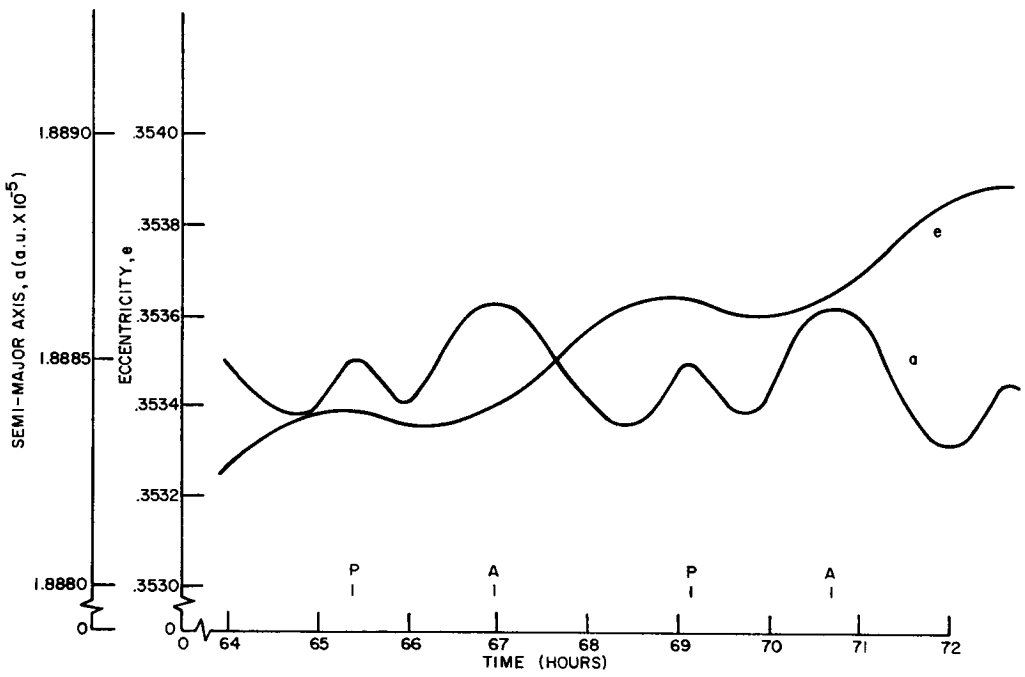


Figure I-1-33. Lunar orbit perturbations, (Revolutions #17, 18), $(h_a)_o = 1122$ nmi, $(h_p)_o = 50$ nmi

The osculating values of apolune and perilune altitudes are shown in Figure I-1-34. The initial values of apolune and perilune altitudes are 1122 and 50 n mi, respectively. Due to the increase in eccentricity shown earlier, these altitudes are changed to values of 1130 and 40 n mi, respectively at a time of 150 hours.

1.6 RETURN FROM LUNAR ORBIT

The establishment of a trajectory which returns the vehicle from its lunar orbit to a given set of re-entry latitude-longitude coordinates relative to the rotating earth requires the selection of the lunar orbit exit velocity increment as a function of position in orbit, taking into account the return flight time. The selection of the dis-orbit point should be made, not only for return at the end of the nominal orbiting duration, but also for other orbital revolutions for "emergency return" considerations in the event that the mission duration has to be shortened.

The geometry of the orbit ejection is shown in Figure I-1-35. The vehicle, represented by the point, P, is travelling clockwise in its relatively-unperturbed elliptical lunar orbit

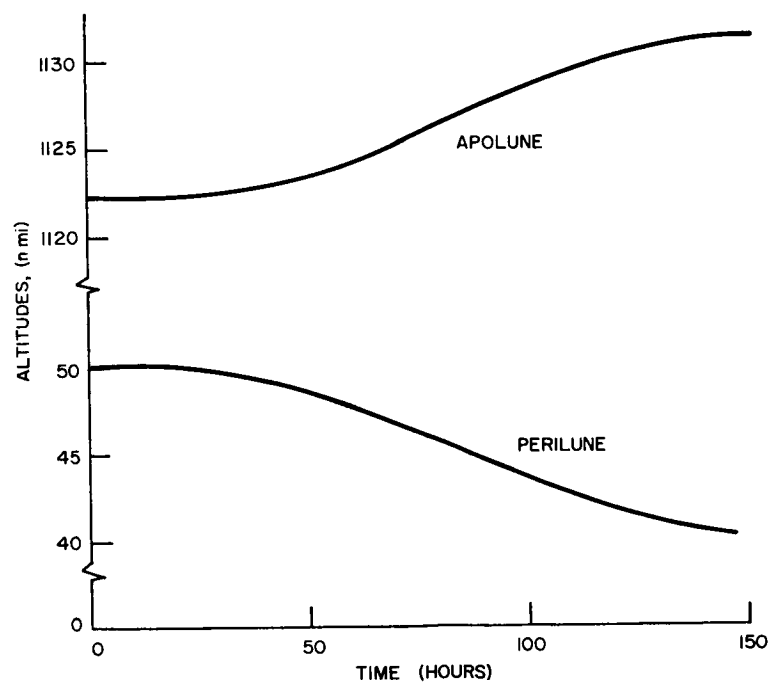


Figure I-1-34. Perturbations of apolune and perilune altitudes

[redacted]

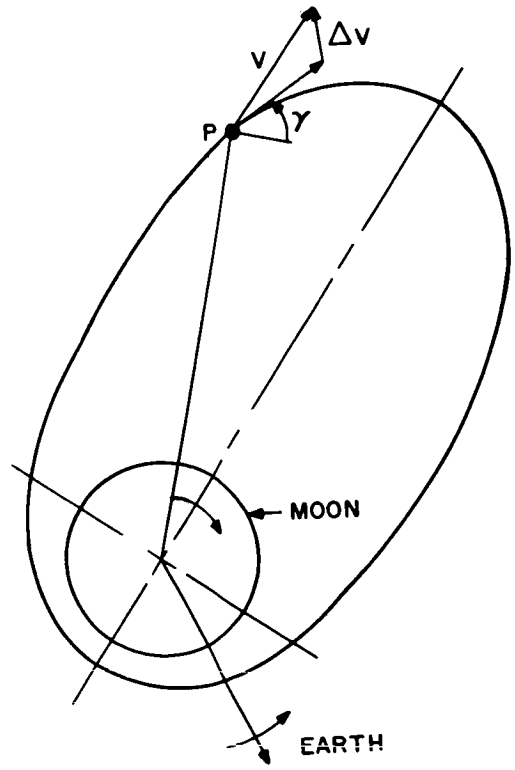


Figure I-1-35. Schematic of dis-orbit maneuver

with a period of 3 to 5 hours, in general. The earth, represented by its position vector, is rotating in a counterclockwise direction relative to the major axis of the ellipse whose orientation is effectively fixed relative to inertial space. The earth's mean angular velocity is 13.2 deg/day.

As the vehicle travels around the orbit, its velocity and position vectors relative to the Moon are continually changing. It would be possible to initiate a return trajectory to the earth from any point on the orbit by the vector addition of a velocity increment to produce the appropriate perigee radius. Returns initiated from certain portions of the ellipse, however, would require unacceptably large velocity increments. Such regions would occur in the vicinity of perigee of the orbit as sketched in Figure I-1-35.

The time of flight of the return trajectory must be selected so that the perigee (or equivalently, the re-entry point at an altitude of 400,000 feet) occurs at a longitude which is appropriate to the longitude of the recovery site. This criterion, required because of the rotating earth, would result in, typically, three sets of "acceptable"

[redacted]

return transfer trajectories to a specified re-entry longitude from a given orbital revolution. The sets could be generated by varying the magnitude of the incremental velocity at various points along the ellipse or by performing a vector addition of the velocity increment.

These sets of return trajectories would differ by approximately integer days, e.g. flight times of 1.5, 2.5, and 3.5 days.

These trajectories would be initiated at different positions along the orbit and require different velocity increments. Since the return flight time increases as the energy of the return trajectory is decreased (at least for paths which do not pass through apogee of the earth-centered conic before returning to perigee), the longer flight times would be desirable from consideration of the magnitude of the dis-orbit velocity increment required.

The variation of the velocity increment required to establish the return trajectory to earth as function of the position in the satellite is indicated schematically in Figure I-1-36. The values of the corresponding return flight times are indicated. This set would correspond to three trajectories returning to a particular longitude, and another set of three points displaced slightly would result in returns to a different longitude.

Return trajectories initiated at points along the left branch of the curve correspond to increasingly longer flight times because the vehicle would be at or near apogee of the earth-centered ellipse after "escaping" from the Moon's influence.

The above discussion pertains to any given orbital revolution. The magnitude of the velocity increment required, as well as the dis-orbit position, will change as a function of the revolution number. Since the orbital period of these lunar satellites can be between 3.5 and 5.5 hours (for the NASA-specified limits), the APOLLO vehicle can make from 5 to 7 revolutions per "sidereal" day of about 24.9 hours - since the Moon rotates in the same direction as the earth.

The flight time required to reach a particular longitude on the earth would have to be decreased by an increment approximately equal to the satellite period for each subsequent revolution. If the semi-major axis (the mean value under the influence of the

[redacted]

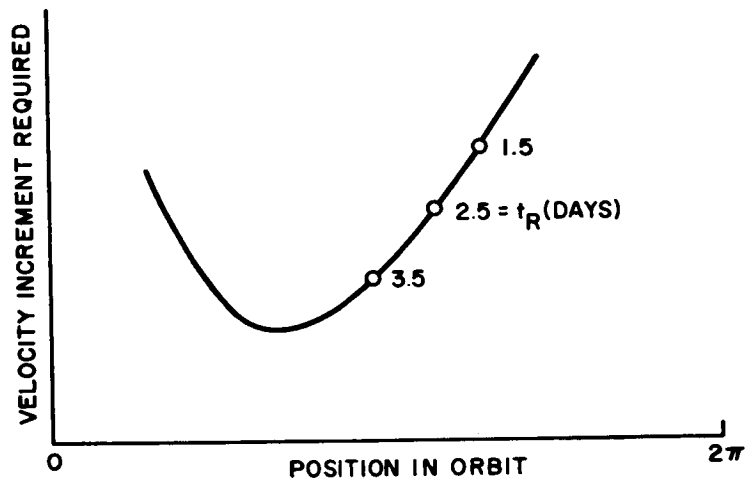


Figure I-1-36. Dis-orbit velocity increment vs position in orbit

perturbations of the earth and sun) is such that the vehicle makes an integer number of revolutions per "sidereal" day, then these velocity increments will be approximately repetitive.

The flight times for successive orbital revolutions would progress by this time increment until the return trajectory would be "equivalent" to one which requires one day more transit time. This cycle of return flight times is shown schematically in Figure I-1-37. There would, of course, be a family of curves similar to Figure I-1-36, each differing slightly because of the change in the angle between the satellite orbit's major axis and the direction to the earth for successive orbital revolutions. Following this reasoning, the magnitude of the velocity increment required to return the vehicle to a given longitude on the rotating earth would vary as sketched in Figure I-I-38. The velocity increments would follow a saw-tooth variation, as in the curve for return flight time, repeating the cycle each day.

The major factors in sizing the vehicle's velocity potential, and therefore the total on-board propellant capacity, are the velocity increments required to enter and leave the

[redacted]

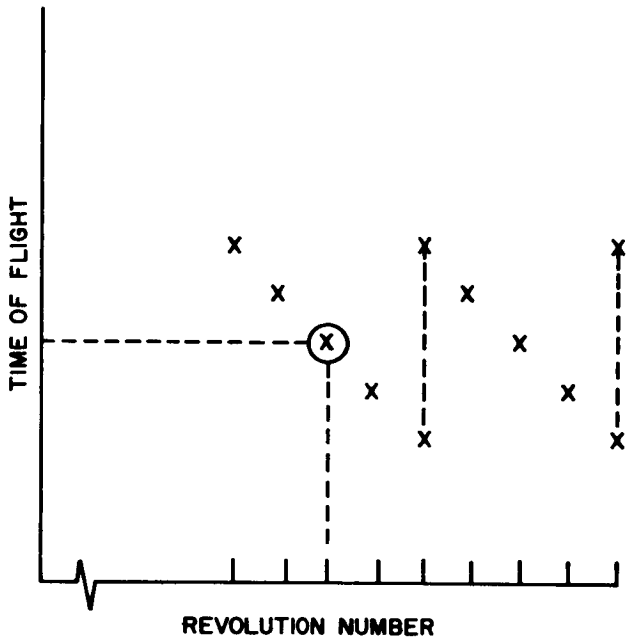


Figure I-1-37. Flight time required to return to a given longitude

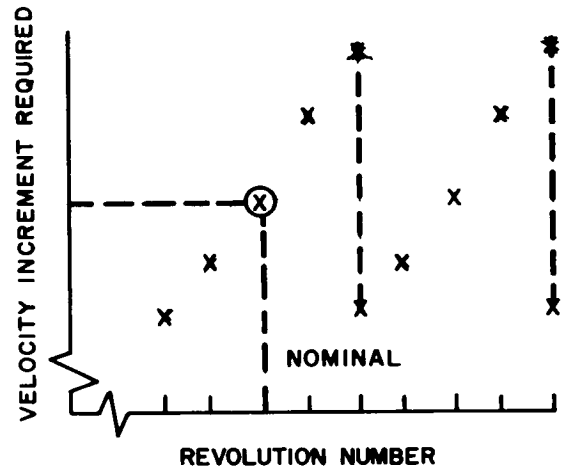


Figure I-1-38. Dis-orbit velocity increment required for various revolutions

lunar orbit. The velocity increment required to establish the orbit is determined by the insertion velocity of the outbound trajectory and the desired apolune and perilune altitudes of the satellite orbit. This increment will be of the order of 3300-3500 ft/sec for relatively slow transfer trajectories as shown in Section 1.5.1.

The amount of velocity potential, and hence of propellant capacity, to be reserved for the dis-orbit maneuver can be discussed, referring to Figure I-1-38. If the remaining propellant capacity is sized for the return from a particular nominal orbital revolution, a certain velocity increment will be required, as indicated by the point labelled "nominal". Returns to the pre-selected re-entry longitude will be possible on certain of the other orbital revolutions which require a smaller velocity increment, as shown. The remaining revolutions on that "day" however, could not be used since they require more velocity increment than available to return to the specified longitude.

Therefore, the on-board propulsion capability would either have to be sized to accomplish a return from any orbital revolution, or the re-entry longitude, and consequently

the landing site, would have to be allowed to move around the earth. This latter could be acceptable if a number of landing or recovery sites (at approximately the same latitude, but spaced in longitude) were provided.

This capability of returning from all orbital revolutions would be prompted by provisions for an "emergency return" in the event the mission time has to be shortened.

In any event, the system should be designed to insure the possibility of returning to the earth from more than one revolution of each day, by providing that the propellant capacity is greater than the minimum required by some reasonable amount.

1.6.1 Lunar Dis-Orbit Increments

In order to develop an understanding of the behavior of three-dimensional return trajectories, a 3-D patched-conic analysis was made. The program was simplified in the sense that the Moon-centered hyperbola was approximated by its asymtotes. Also, the earth-centered conic section was defined to be a parabola so that the perigee radius is determined solely by the angular momentum of the vehicle relative to the earth after the coordinate change is made. The inputs to the program are: the energy and perilune radius of the Moon-centered hyperbola, the latitude and longitude of its perilune, and the lunar longitude of the line of nodes. Computation of the velocity increment required to transfer from the satellite orbit to the hyperbola was not included in this study. The output of the program is the magnitude of the resulting perigee radius. For the sake of brevity, these intermediate results are not included.

By computing sets of curves for various energies the energy required to return to a perigee radius in the re-entry corridor can be determined for values of the other variables, such as the longitude of perilune of the hyperbola.

These energy levels of the vehicle relative to the Moon were used in generating return trajectories using one of many possible classes of lunar dis-orbit maneuvers.

The particular class of maneuver chosen for this exploratory, rather than exhaustive, study of return trajectories is the addition of the velocity increment in a direction which is colinear with the vehicle's instantaneous velocity vector relative to the Moon.

One instructive type of plot of the resulting perigee radius is shown in Figure I-1-39. The perigee radius is shown as a function of the vehicle's hyperbolic energy relative to the Moon. The family of curves corresponds to dis-orbits from various positions in the lunar satellite orbit during the 18th revolution, after a total orbiting duration of slightly more than 70 hours. The true anomaly of the dis-orbit points approach apolune as the curves progress downward. These points are located at true anomalies ranging from 90 to 150 degrees.

Several interesting features of the trajectories resulting from this particular type of dis-orbit maneuver can be determined. Trajectories orginated from orbital positions represented by the upper curves apparently have their initial velocity vector "aimed" at an angle such that small perigee radii are unattainable, regardless of the vehicle's energy with respect to the Moon. As the initial position is moved toward apolune, the angle of the initial velocity vector relative to the earth decreases until perigee radius is in the re-entry corridor (a band which is undistinguishable on this scale, centered at the radius of the earth). The curve which is tangent to the center of the corridor corresponds to the dis-orbit point for which perigee radius is insensitive to changes in the magnitude of the dis-orbit velocity increment. The energy level of these trajectories is such that moderately high velocity increments are required, however. Changing the energy of the vehicle at any point on the ellipse will affect the return flight time and thus move the longitude of perigee.

The most practical region for return trajectories is the line of intersections which lies to the left of the minimum of the curves. For these cases the energy is relatively low, and since the positions are approaching apolune of the orbit, the flight times are not too long. This is contrasted to trajectories defined by the corresponding intersections of the right hand branches of the curves which require larger velocity increments. The velocities are high enough to result in the vehicle passing through apogee of the earth-centered conic on the way to perigee depending on the direction of the asymtote, and thus can have very long return flight times.

Figure I-1-40 presents a summary of the parameters of the return transfer trajectory for various values of dis-orbit time. Dis-orbit times of 70.3 and 70.5 hours correspond

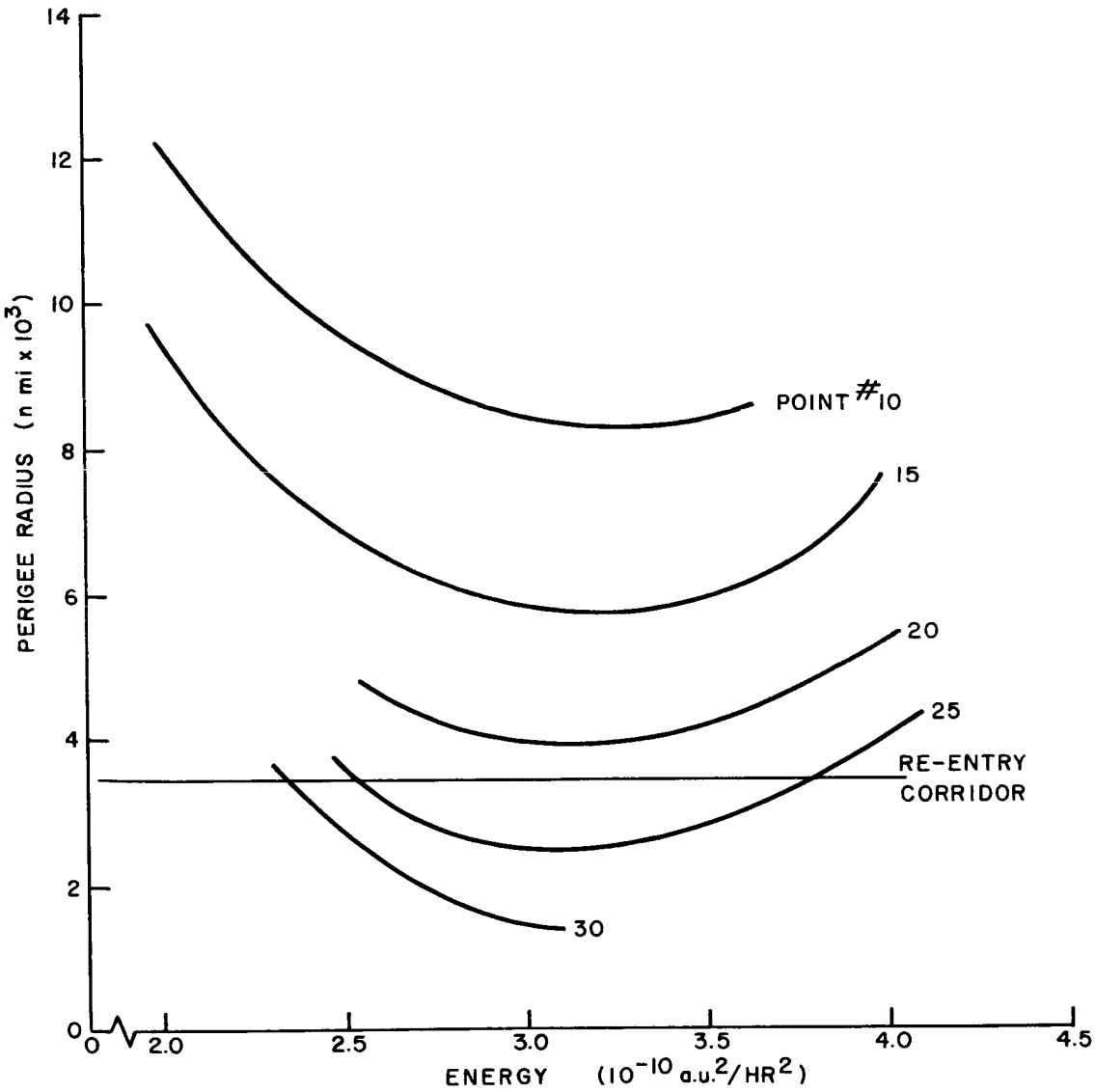


Figure I-1-39. Perigee radius of returns from lunar orbit (18th revolution)

[redacted]

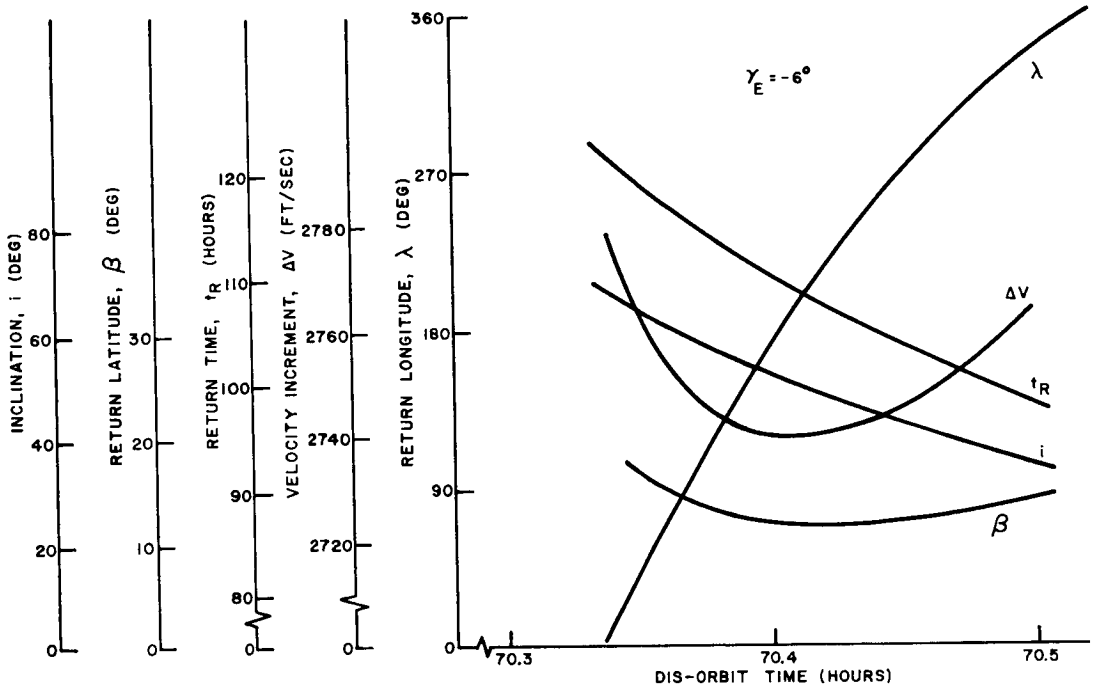


Figure I-1-40. Return from lunar orbit 18th revolution

to true anomalies in the orbit of 127 and 141 degrees respectively. The values of velocity increment, ΔV, at each position are adjusted to produce a re-entry path angle of -6 degrees which corresponds to a virtual or vacuum perigee altitude of about 30 n mi. This value corresponds to the mid-point of the allowable re-entry corridor for the APOLLO D-2 configuration.

The velocity increment is seen to have a minimum value of 2740 ft/sec at a dis-orbit time of 70.4 hours. The return flight time decreases monotonically with dis-orbit time in this interval, showing the effect discussed earlier.

The longitude of perigee for these trajectories was determined by considering the rotation of the earth during a typical total APOLLO mission composed of the outbound trajectory discussed in Section 1.3, the lunar satellite orbit discussed in Section 1.4, and the various return trajectories in this family. Knowing the launch or insertion longitude allows the return perigee longitude to be determined. It is seen that a complete sweep of 360 degrees of longitude can be achieved by a variation in dis-orbit time of only approximately 10 min. The velocity increments required to achieve the desired

[redacted]

perigee altitude vary by only 40 ft/sec for this sweep. The return flight times vary by almost exactly 24 hours in this time interval.

The other parameters shown in Figure I-1-40 are the inclination of the return trajectory plane relative to the equator and the latitude of perigee. The inclination of the plane is seen to vary between values of 65 and 30 degrees with dis-orbit time. The latitude of perigee, which occurs after the vehicle passes tangent latitude (southbound) for this set of return trajectories remains relatively constant at values between 10 and 15 degrees for this cycle of return longitudes.

In order to determine the initial dis-orbit parameters of a trajectory which returns to a point at a particular longitude — along the Atlantic Missile Range, for example, a portion of Figure I-1-40 was replotted. Figure I-1-41 shows the tangent latitude, perigee latitude and the mean impact latitude as a function of dis-orbit time. Figure I-1-42 presents the longitudes for the same points.

The mean impact point is defined as the nominal landing point for the D-2 vehicle re-entering the atmosphere at an angle of -6 degrees, a value mid-way between the allowable limits. With this path angle, the re-entry is initiated approximately 12 degrees or 700 n mi short of the vacuum perigee point. These two graphs can be combined to produce a locus of the various points on the return trajectory. Selecting a dis-orbit time of 70.478 hours results in a mean impact point located at 5 degrees south latitude and 23 degrees west longitude. The tangent latitude, and therefore the corresponding inclination of the orbital plane, is 35 degrees. The trace follows very nearly down the Atlantic Missile Range so that the vehicle could be tracked during the terminal phase of the return and during re-entry.

1.6.2 Sample Return Trajectory

The dis-orbit point and the early portion of the sample return trajectory initiated at the time given above are shown in Figures I-1-43, I-1-44, and I-1-45. The x-y projection shows the final orbital revolution and the directions to the earth and sun at dis-orbit. The dis-orbit point for this particular case is located almost diametrically opposite the earth. The asymtote of the Moon-centered escape hyperbola is approximately

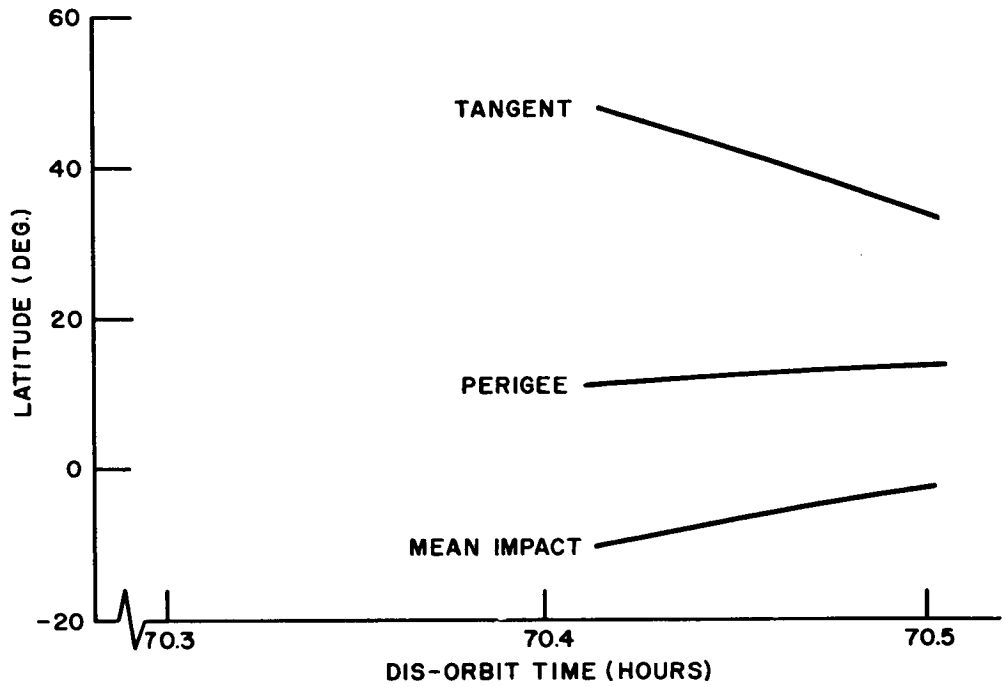


Figure I-1-41. Return from lunar orbit terminal latitudes

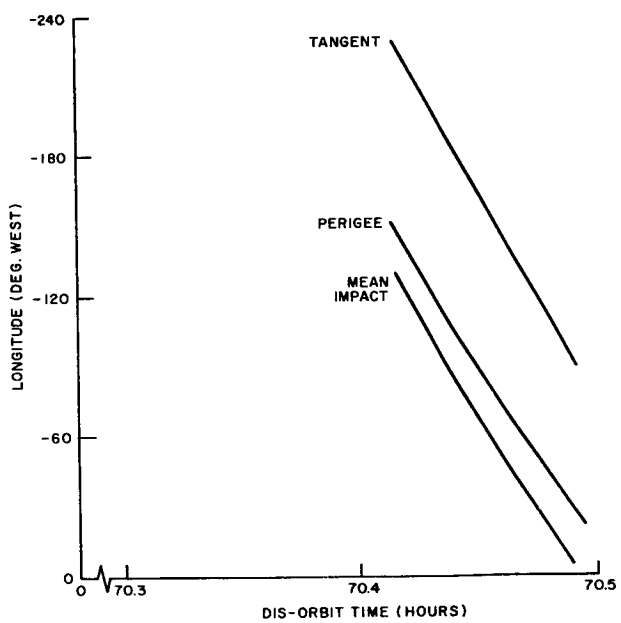


Figure I-1-42. Return from lunar orbit terminal longitudes

[redacted]

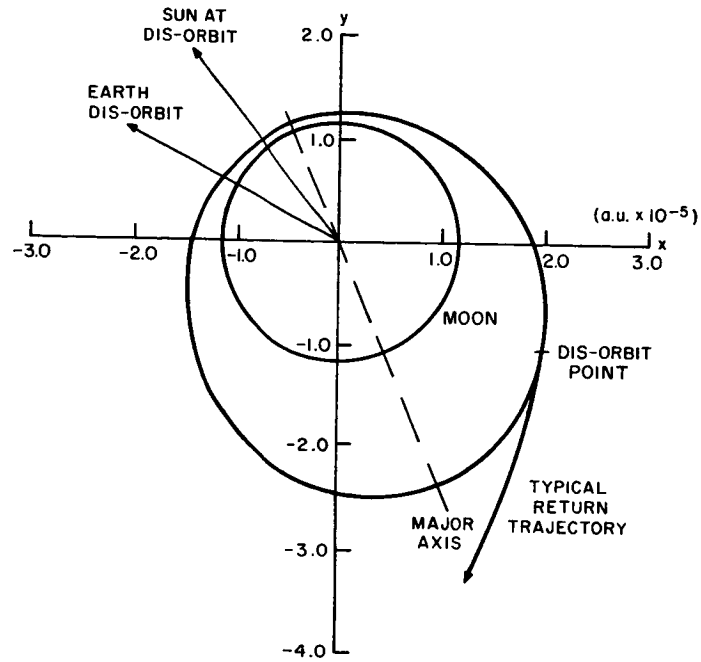


Figure I-1-43. Lunar satellite at dis-orbit (x - y projection)

perpendicular to the direction to the earth. The projections in the x-y and y-z planes look distored because of the inclination of the orbit to the coordinate axes.

The full return trajectory is plotted relative to an earth-centered coordinate system in Figures I-1-46, I-1-47 and I-1-48. The x-y projection shows the result of the vehicle's return velocity vector being directed nearly against the Moon's velocity vector since the vehicle initially moves back along the Moon's orbit. Within 4 hours of dis-orbit, the vehicle has started to fall toward the earth, reaching perigee after a flight time of almost 100 hours. The return trajectory is travelling with the rotation of the earth. Figures I-1-47 and I-1-48 show that the vehicle crosses through the equatorial plane about 11 hours before perigee, and reaches the maximum distance above the equatorial plane about 2 hours before perigee.

The complete earth surface trace of the return trajectory described above is shown as the solid line in Figure I-1-49. The lunar dis-orbit point occurs within 10 degrees of the longitude of Goldstone and then the trace moves westward. Thus, communication and tracking should be possible starting very shortly after the dis-orbit maneuver. The

[redacted]

[redacted]

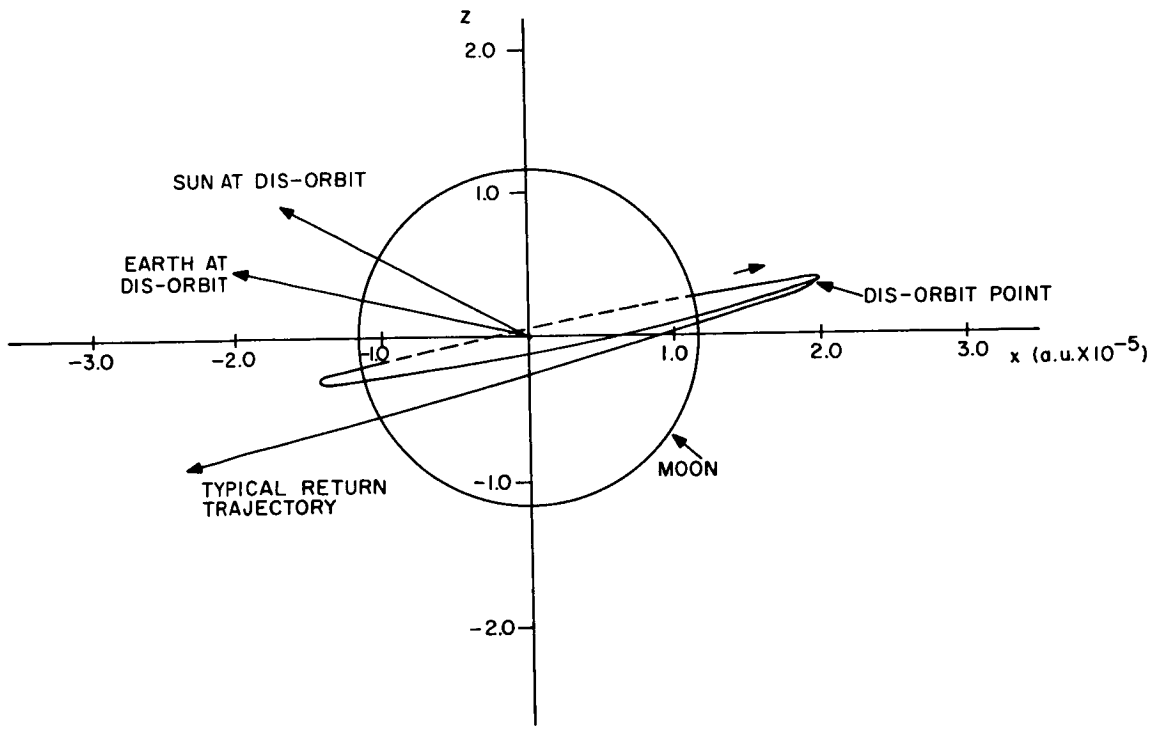


Figure I-1-44. Lunar satellite at dis-orbit (x - z projection)

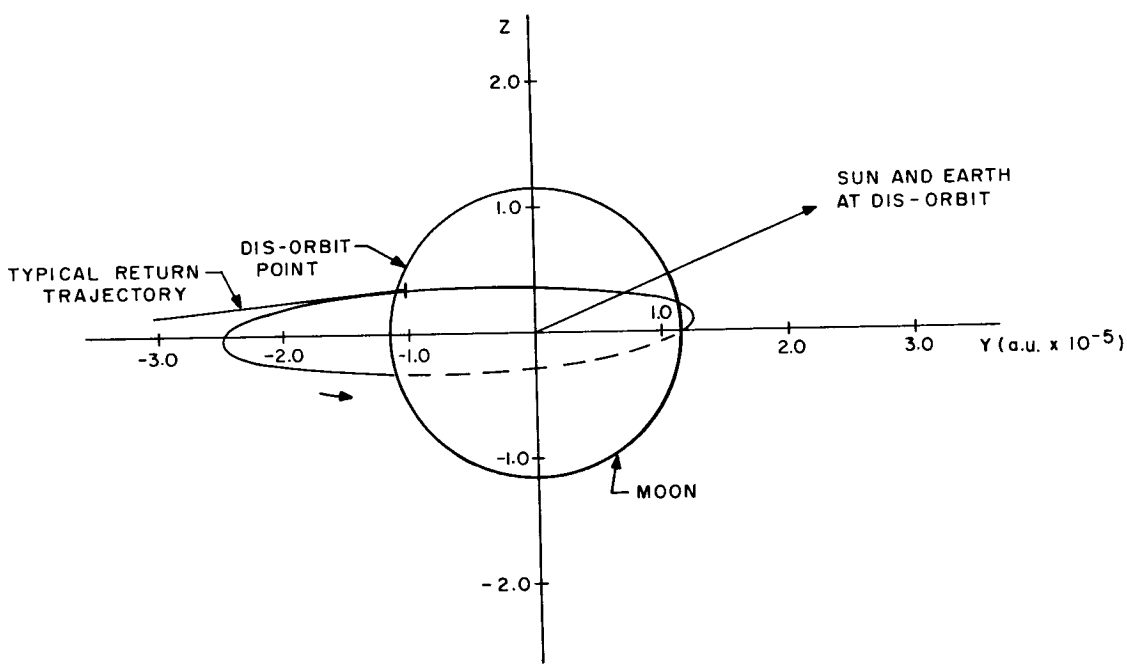


Figure I-1-45. Lunar satellite at dis-orbit (x - z projection)

[redacted]

[redacted]

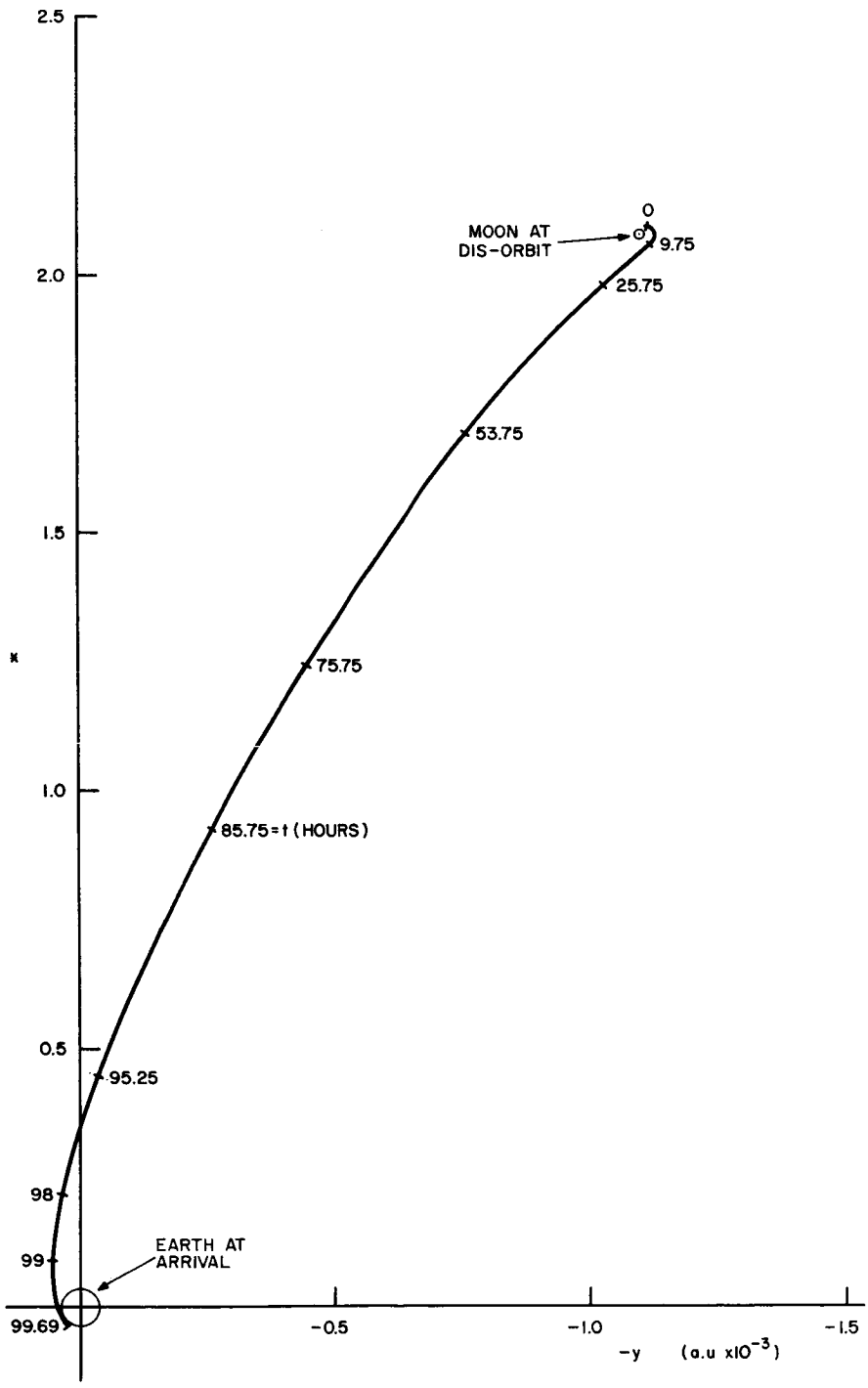


Figure I-1-46. Sample return trajectory (x - y projection)

[redacted]

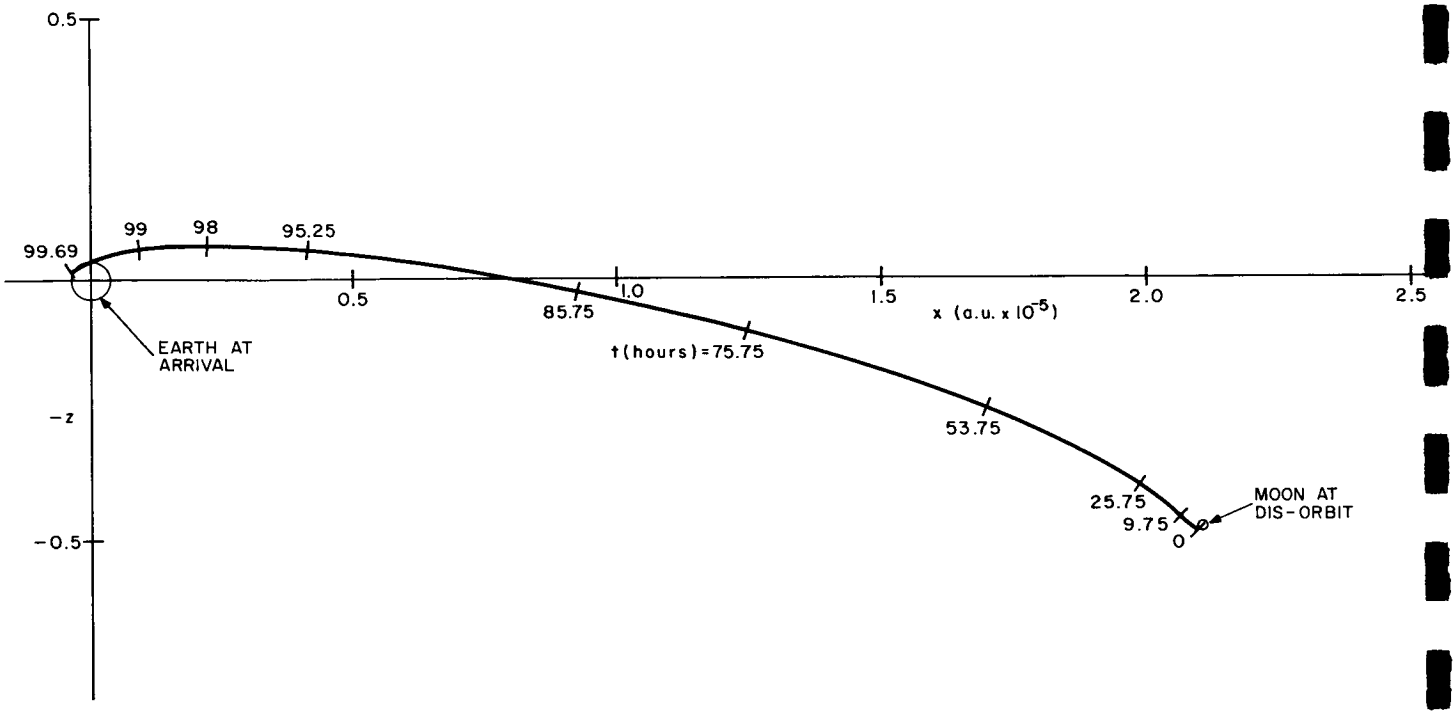


Figure I-1-47. Sample return trajectory (x - z projection)

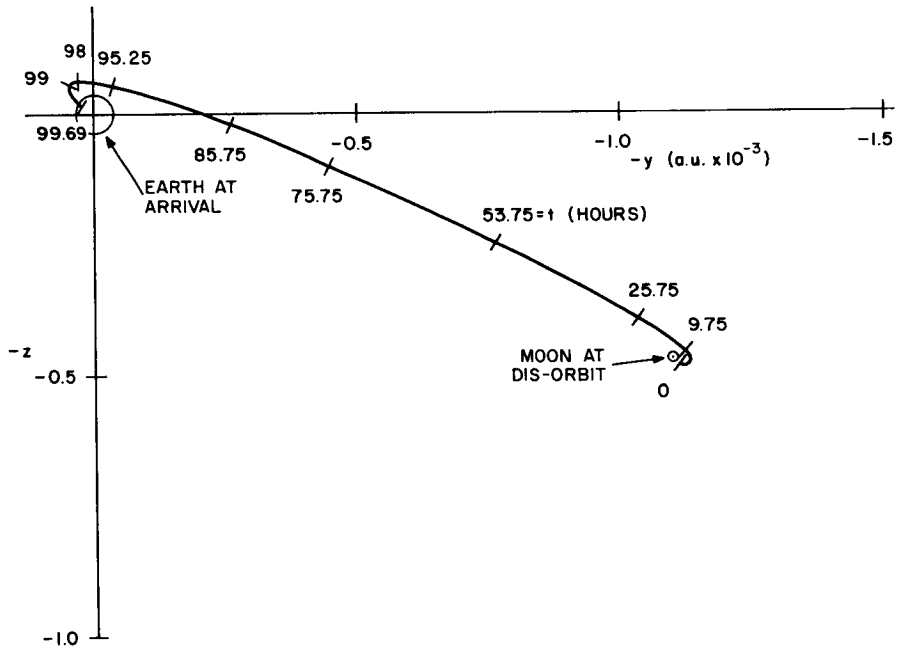


Figure I-1-48. Sample return trajectory (y - z projection)

[redacted]

terminal portion of the final approach reaches a longitude of about 155 degrees west, which is very near Hawaii. The trace then crosses the United States and lands at a nominal location of 5 degrees south latitude, 23 degrees west longitude as mentioned previously. The re-entry maneuver area available to the APOLLO D-2 vehicle configuration is indicated by the nearly-elliptical shaded area. This area is seen to contain Ascension Island so that it would be possible to fly the vehicle to a land landing if desired.

1.6.3 Re-Entry Maneuverability

In the context of the analysis of the complete APOLLO lunar orbit and return mission discussed above, it is proposed to expand the definition of re-entry maneuverability to include not only the classical concept of "flying" the vehicle while it is in the atmosphere, but also the selection of the geographic coordinates of the re-entry point and the trajectory azimuth. These can be considered as the microscopic and macroscopic aspects, respectively.

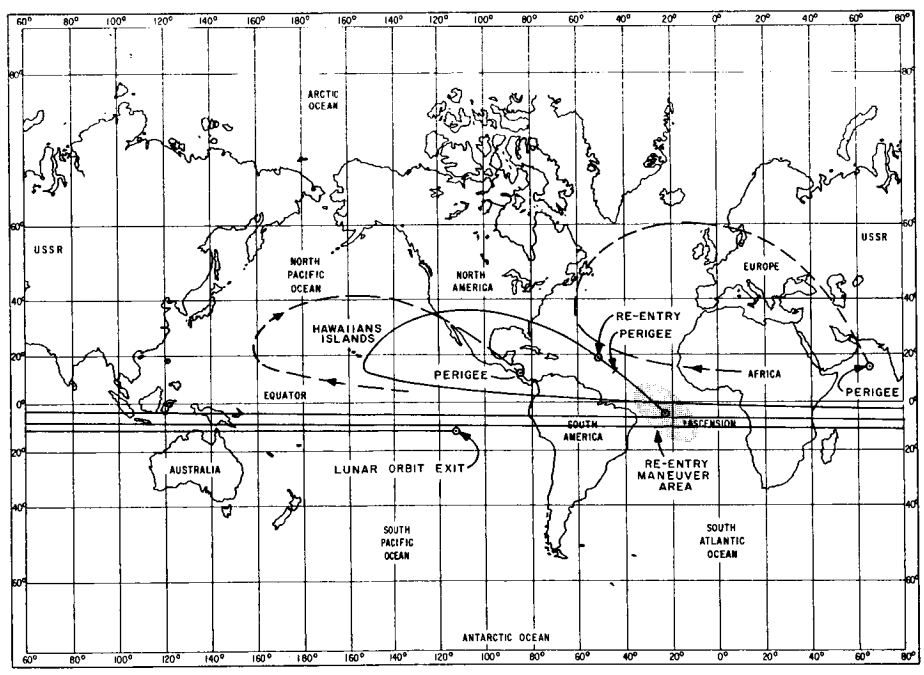


Figure I-1-49. Return trajectory traces

[redacted]

The return trajectory parameters of Figure I-1-40 and the two additional earth traces indicated by dashed lines in Figure I-1-49 illustrate the freedom in selection of the landing point longitude all the way around the earth by varying the dis-orbit time over an interval of only a few minutes.

As a result of the particular inclination of this sample lunar orbit and the colinear dis-orbit maneuver employed, the resulting perigee latitudes vary between 10 and 15 degrees north. By initiating the dis-orbit maneuvers during the portions of the lunar orbit immediately before or after the arc used in this study, the return trajectories would be modified and additional swaths of longitude coverage would be obtained. The variables which would be principally involved are the inclination of the return trajectory and the latitude of perigee. The magnitude of the velocity increment would be shifted to somewhat higher values.

Other, non-colinear techniques of the dis-orbit maneuver would give additional freedom or control of the return trajectory parameters. These two variations in the dis-orbit maneuver--the time of dis-orbit and the vector addition of the increment--will give additional selectivity and control of the inclination of the return trajectory plane and the latitude of perigee.

The motion of the vehicle in the atmosphere and its maneuverability are discussed in the following section.

1.7 RE-ENTRY TRAJECTORY STUDY

Of the missions which might ultimately be performed by the APOLLO vehicle, the most difficult from the standpoint of re-entry vehicle design, and hence deserving of primary consideration, is the re-entry at escape velocity following lunar return. Re-entry trajectory studies were conducted to ascertain the important parametric effects on the design and performance of suitable re-entry vehicles. The simple non-lifting or ballistic type of vehicle was quickly eliminated from further consideration because of the extreme guidance accuracy necessary to provide tolerable atmospheric re-entry and satisfactory landing areas. Reported here are some of the general results obtained for lifting vehicles and typical specific data obtained for the APOLLO D-2 (semi-ballistic) and R-3 (modified lenticular) configurations. Digital computer programs have been

employed to solve the basic equations of motion, treating the body as a point mass with instantaneous trim control. This representation gives the characteristic long-term flight path oscillations but neglects the short-period attitude responses. This can be shown to be an adequate assumption for vehicles of reasonable stability operating in sufficiently high density regimes. To establish maximum theoretical performance to measure energy management system capability against, the studies discussed here were performed as open-loop, or programmed control trajectories in the usual fashion. Closed-loop trajectories are reported in the energy management section of Chapter II of this Volume.

In our study of re-entry trajectories, we found that a pullout to zero path angle must be made subsequent to attaining the tolerable re-entry corridor. For very shallow angles, negative maximum lift coefficient is used and defines the capture boundary, while for steep angles maximum positive L/D is employed and defines the maximum tolerable peak deceleration boundary. The attainable ground area for landing point selection must reflect these extremes in re-entry angle. It can be seen that the minimum range which could be guaranteed is associated with the most shallow (overshoot) re-entry angle. The range limit thus imposed is easily defined on the basis of tolerable deceleration levels. The maximum guaranteed range, on the other hand, is associated with the steepest paths and could be considered to define the maximum range limit in this case, depending on the degree of skip permitted. The procedure which, in theory, would result in the greatest attainable range would be to permit the vehicle to skip to an altitude as high as 400 miles, the radiation boundary. Certain disadvantages are found with this procedure. Firstly, the performance of ablation materials under repeated pulses of this nature is questionable. Secondly, as shown by preliminary studies at Ames Research Center, relatively high trajectory range sensitivities are associated with the skipout, and since a considerable portion of the trajectory is flown in near-vacuum conditions where range changes cannot be made, energy management becomes a difficult problem. The path which provides the best range control is that where no skip at all is permitted, a constant altitude transition being performed after pullout to decelerate to suborbital equilibrium glide velocity. This approach was investigated in considerable detail by MSVD, but it was found that L/D had to be of the order of one or greater to be useful. A compromise approach is recommended whereby a small

skip is permitted such that the vehicle remains essentially within the atmosphere, although a short period of partial reaction control may be permissible.

As L/D increases, the maximum time spent under reaction control decreases; improved range control, energy management, and landing point selection then being possible.

1.7.1 Re-Entry Corridor

During the study, considerable effort has been devoted to defining the tolerable re-entry corridor as a basis for midcourse guidance accuracy requirements and re-entry vehicle design. The shallowest atmospheric re-entry (overshoot) which permits single pass capture is taken as the lower limit on re-entry angle. The vehicle is assumed to employ maximum negative lift during initial re-entry and pullout, the lift and weight forces precisely balancing centrifugal force in the limit case when the altitude rate reaches zero. The steepest tolerable re-entry (undershoot) occurs when the peak deceleration occuring in pullout equals the allowable value. Figure I-1-50 shows that for a given peak deceleration, the maximum re-entry angle is primarily dependent on L/D, while the minimum re-entry angle is established by $W/C_L A$. As a matter of interest, the positive lift capture limit is also shown. The use of negative lift during initial re-entry increases corridor width by about 0.6 degrees. It is also seen that L/D's greater than one do not greatly increase the angular corridor. The equivalent perigee altitude corridor can be found from Figure I-1-51, which shows virtual perigee for escape velocity re-entries. The effect of allowable peak deceleration and L/D on the angular corridor is shown by Figure I-1-52. The corridor is reduced by about 0.3 degrees in decreasing the peak deceleration from 10 to 8 g. The velocity at re-entry does not strongly influence the maximum re-entry angle, as shown by Figure I-1-53, but does affect the skip or minimum re-entry angle to some extent.

The effects of orbital inclination on re-entry angles are shown in Figure I-1-54 for re-entries from polar orbits and eastward from equatorial orbits. The inertial re-entry velocity is 36,000 ft/sec and the vehicle characteristics are $W/C_L A = 100$ lb/sq ft and L/D = 0.5. For a given maximum tolerable acceleration, re-entries from the equatorial orbit may be about 0.5 degrees steeper than from the polar orbit. This is because the relative velocities differ by about 1500 ft/sec, so that the aerodynamic forces developed

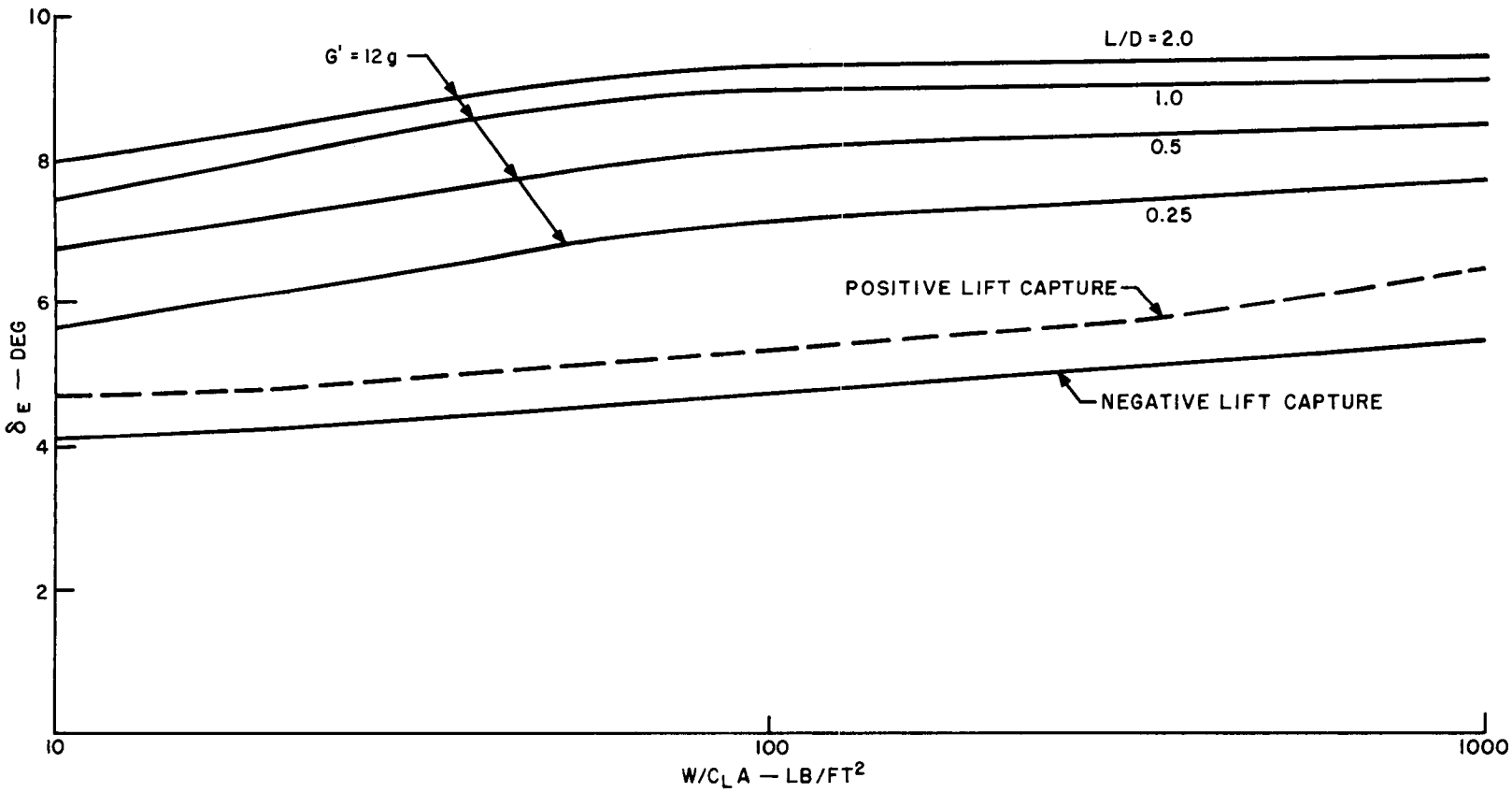


Figure I-1-50. Re-entry angular corridor
h_e = 400,000 ft,
V_e = escape velocity,
polar orbit

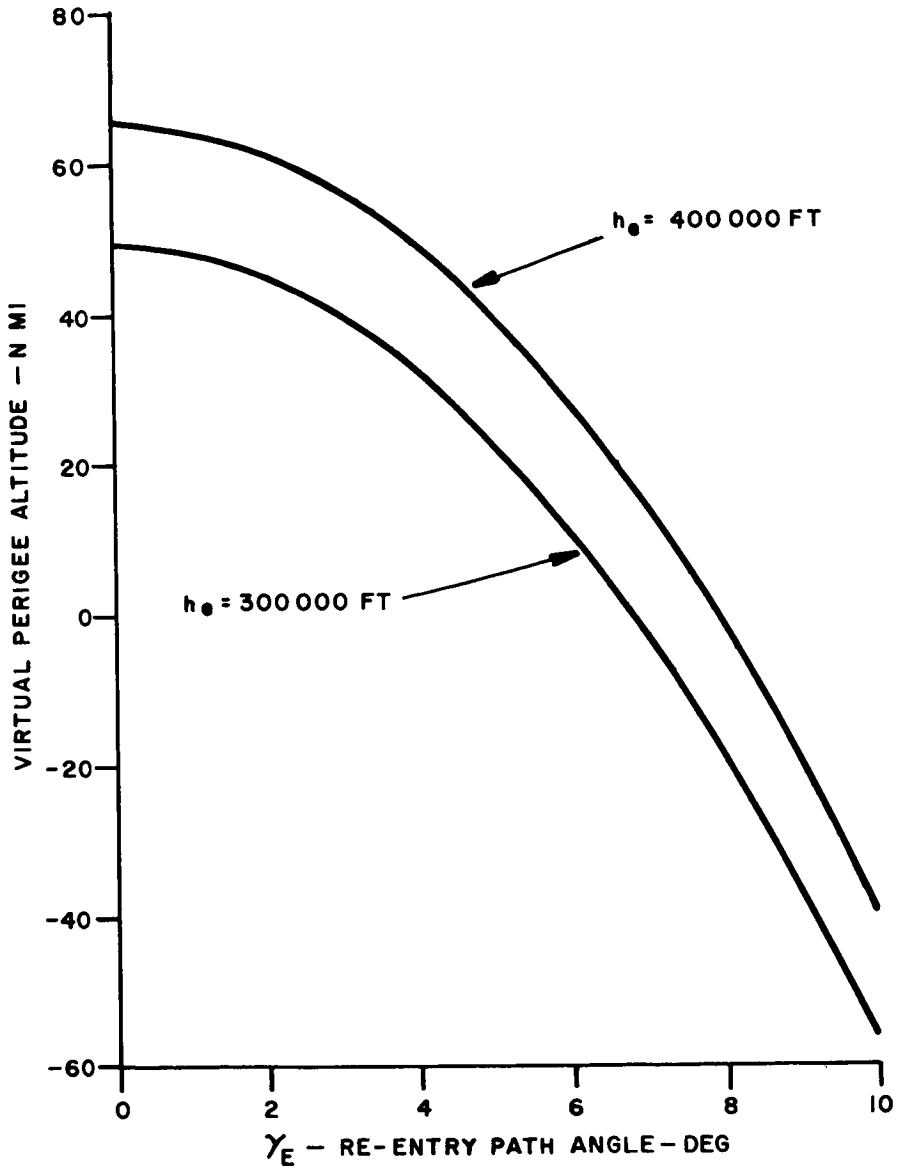


Figure I-1-51. Virtual perigee altitude vs re-entry path angle for escape velocity

[redacted]

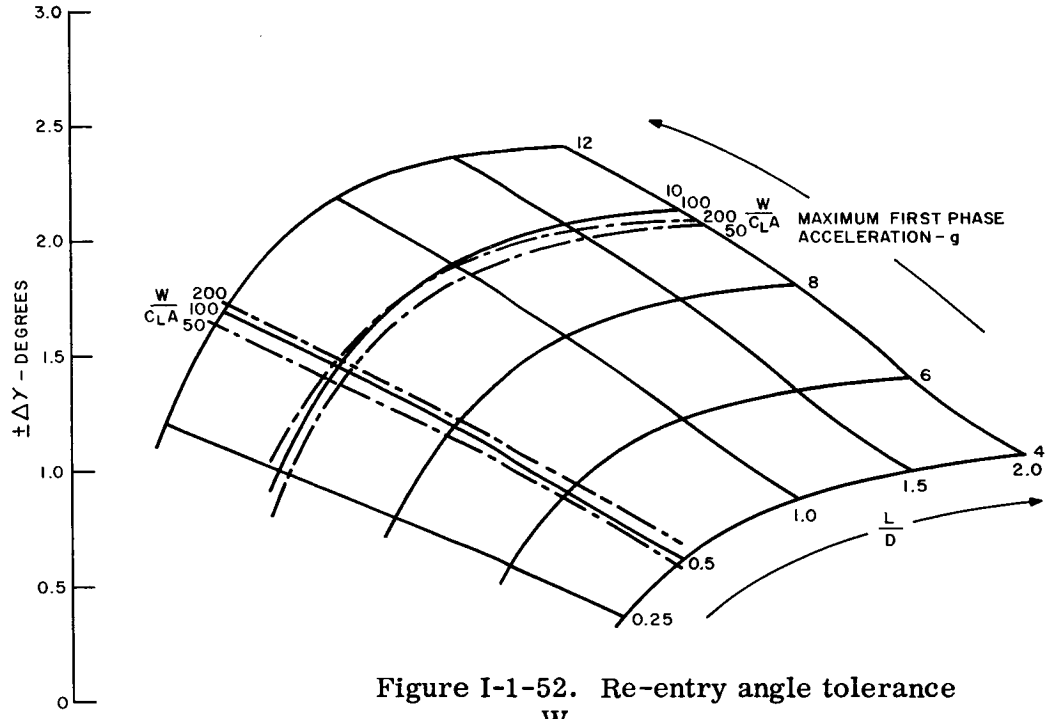


Figure I-1-52. Re-entry angle tolerance

$\frac{W}{C_{LA}}$ = 100 lb/sq ft

h_e = 400,000 ft

V_e = escape velocity

polar orbit

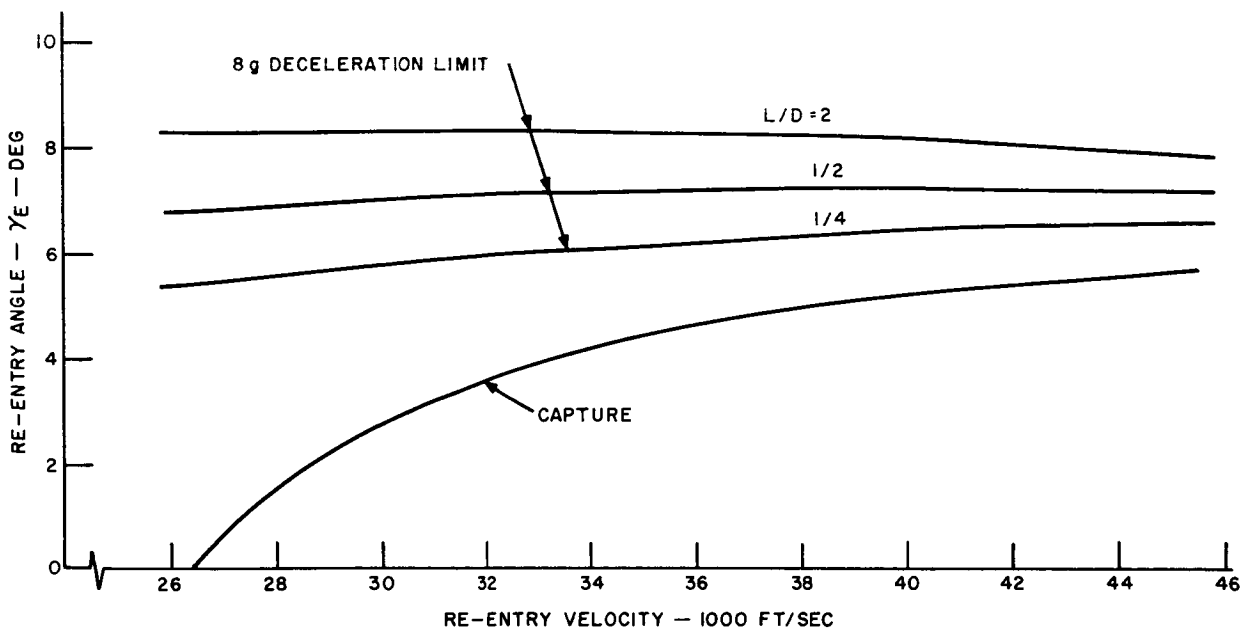


Figure I-1-53. Angular re-entry corridor for supercircular entry

h_e = 400,000 ft

W/C_{LA} = 100 lb/sq ft

polar orbit

[redacted]

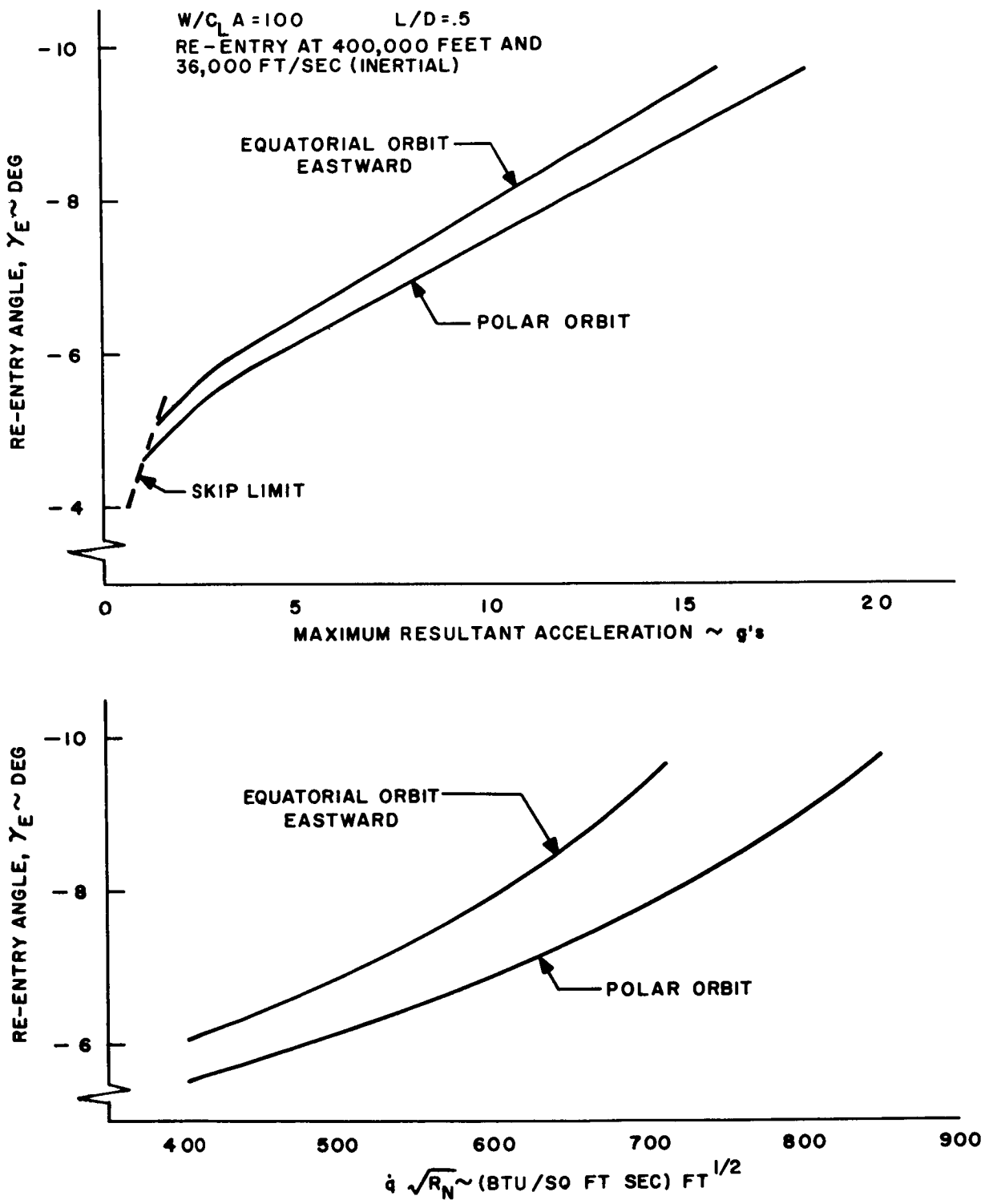


Figure I-1-54. Effect of orbital inclination on re-entry angle

at each condition of altitude and inertial velocity are lower for the eastward flight. However, the skip limit is also steeper so that no significant change occurs in the re-entry angle tolerance.

For a given re-entry angle the reduction in heat flux rate parameter $\dot{q}_s \sqrt{R_N}$ is appreciable. For a 12 g maximum re-entry, the maximum rate for the equatorial re-entry is 90 percent of the rate for the polar re-entry. This reduction is fairly constant throughout the range of maximum tolerable g's.

Studies of the effects on the re-entry corridor by modulating C_L during the re-entry to sustain flight at constant acceleration, were made for the modified lenticular vehicle. Since modulation can only be achieved by reducing the flight attitude, re-entries were assumed to begin ballisticly at an angle of attack of 90 degrees. For a re-entry angle of -7.51 degrees (which gives a maximum acceleration of 10 g's for constant $W/C_L A$ = 100 and L/D = .5) it was shown that a pull-out could not be effected without exceeding 10 g's. The flight was ballistic until 10 g's was reached, and then C_L was modulated to maintain that level. When C_L had reached its maximum value at an $\alpha \approx 50 - 55$, degrees, the flight path angle was still -4.4 degrees. Further reduction of attitude was not considered since heat protection for flight at low altitudes and super velocities has not been included in the vehicle design. Since the steepest re-entry at constant C_L for a specified maximum g is obtained by flying at $C_{L_{max}}$, the studies suggest that re-entries at lower C_L's and higher attitudes with modulation to limit g may not, for these types of vehicles, increase the corridor width.

1.7.2 Performance of D-2 and R-3 Configurations

Trajectory studies for the two most promising re-entry vehicle configurations for the APOLLO mission have been made in detail. Typical results such as altitude velocity profiles, time histories for shallow and steep re-entry, and attainable ground area for landing are presented.

At hypersonic flight conditions, where protection systems capable of handling high heat flux are required, the R-3 vehicle has been flown only at angles of attack greater than or equal to that for $C_{L_{max}}$. This positions the stagnation area on the large-radius

bottom surface and hence significantly reduces the amount of heat protection which is required. Both corridor width and maneuvering capability are penalized to some extent since they both increase by flying at low angles of attack near $(L/D)_{max}$. The reduction in vehicle weight, however, more than compensates for the performance losses. The D-2 is flown at $(L/D)_{max}$ in pullout without incurring a major heating penalty. The positions of the crew in the R-3 vehicle during re-entry are such that when flight is at $C_{L_{max}}$ (α = 50 degrees) the resultant acceleration vector is 5 degrees off the direction for eye-balls in. This was considered to be close enough that the resultant acceleration vector could be used as the maximum g criterion. The D-2 does not presently incorporate seat repositioning, and because of restraint limitations and required head-to-foot loads, the peak deceleration is limited to 8 g. The R-3 crew can tolerate and probably perform adequately at 10 to 12 g.

To establish ideal range correction capabilities, the following approach was taken: If the guidance system accuracy is considered equal to the re-entry angle tolerance, the maximum range which can be guaranteed from all re-entries is the maximum range attainable from the steepest tolerable re-entry while the minimum guaranteed range is the minimum range attainable from the shallowest re-entry. To obtain long ranges from the g-limit re-entries, it is necessary to allow the vehicle to climb-out from perigee to high altitudes in order to minimize the rate at which energy is dissipated by drag. To maintain full aerodynamic control capability during re-entry, minimum dynamic pressures on the order of 20 - 25 lb/sq ft are probably necessary. This would limit maximum altitude at orbital velocities to about 250,000 feet. However, with assistance from reaction controls higher flight altitudes and, hence, greater ranges can be realized. Under the assumption that control of this type is reasonable, a maximum altitude limit of about 280,000 feet was imposed for the digital computer studies. Effects of limiting the minimum dynamic pressure are discussed further in the energy management section of Chapter II of this Volume.

The procedure for evaluating maximum ranges from the g-limited re-entries is to continue the re-entry trajectory at positive $C_{L_{max}}$ and 0 degrees roll beyond pullout perigee and then to find the time, along this continuation, at which the vehicle could be rolled 180 degrees to produce a subsequent trajectory which would have an apogee on the suborbital equilibrium altitude-velocity profile for flight at $C_{L_{max}}$. The profiles

[redacted]

labeled (1) in Figure I-1-55 trajectory control procedure. The locus of apogees for inverted flight from various points along the 12 g re-entry trajectory lies entirely below the orbital velocity line and crosses the equilibrium profile ($W/C_L A = 83$) at a velocity of 25,250 ft/sec and an altitude of 279,000 feet. Range from re-entry to this point is 1125 n mi. Subsequent flight is at an L/D = .7 (α = 50 degrees) to V = 15,000 ft/sec where the angle of attack is reduced to approximately 32 degrees to give a maximum L/D = 1.0. Total range for the flight is 5360 nautical miles. The complete time history of the trajectory is shown in Figure I-1-56.

The locus of apogees from the 10 g re-entry lies above the orbital velocity line to altitudes at least as great as 300,000 feet. From an apogee at about 283,000 feet level flight was programmed by modulating C_L and C_D, until equilibrium conditions were reached. This was at a velocity of 25,175 ft/sec. Range to this point was 2265 n mi. Subsequent flight at L/D = .7 to V = 15,000 ft/sec and L/D = 1.0 to V = 4000 ft/sec gave a total range of 6360 n mi.

To obtain minimum range from the near skip limit re-entry, trajectories were calculated from various times along the re-entry path with flight at α = 90 degrees (maximum C_D). The profile labelled 4 represents a trajectory which has an average sustained deceleration of about 6.5 g's for about 1 minute which is tolerable for any condition of the eyeballs. To keep the descent from becoming too steep, a transition to $C_{L_{max}}$ is made between V = 15,000 ft/sec and V = 10,000 ft/sec. Total range for this flight is 1215 nautical miles. The complete time history is shown in Figure I-1-57. Figures I-1-58 and I-1-59 represent similar trajectories for the D-2 configuration, with the steep trajectory being limited to 8 g's during pullout.

To assess ideal cross-range maneuvering capability, trajectories were calculated from the g-limited re-entry paths to equilibrium flight conditions with flight at $C_{L_{max}}$ and roll angles of 135 degrees and 90 degrees. During equilibrium flight the roll angle was 45 degrees. From the near skip limit re-entry, a roll angle of 90 degrees was programmed (with $C_{L_{max}}$) to V = 15,000 ft/sec where it was changed to 45 degrees.

The re-entry ranges cited above are measured from the re-entry point located at an altitude of 400,000 ft. To use these trajectories in determining the maximum

[redacted]

TABLE I-1-I. R-3 RE-ENTRY VEHICLE

Profiles	Phase	L/D In Roll Plane	W/C_LA In Roll Plane	ϕ	Remarks
1	I	.7	83	0°	To appropriate pull-up conditions
	II	.7	83	180°	To equilibrium flight conditions (10g re-entry has a level flight sub-phase)
	III	.7	83	0°	Equilibrium flight to V = 15,000 ft/sec
	IV	1.0	119	0°	Equilibrium flight to V = 4,000 ft/sec
2	I	.7	83	0°	
	II	.7	83	135°	
	III	.7	83	45°	As above
	IV	1.0	119	45°	
3	I	.7	83	0°	
	II	.7	83	90°	As above
	III	.7	83	45°	
	IV	1.0	119	45°	
4	I	.7	83	0°	As above
	II-III	0		0°	Ballistic flight α = 90° to V = 15,000 ft/sec
	IVa	.0→.25	140	0°	To V = 10,000 ft/sec
	IVb	.25	140	0°	To V = 4,000 ft/sec
5	I	.7	83	180°	As above
	II-III	.7	83	90°	Ballistic flight α = 50° to V = 15,000 ft/sec
	IVa	.7→1.0	83→119	45°	To V = 10,000 ft/sec
	IVb	1.0	119	45°	To V = 4,000 ft/sec
6	I	.7	83	0°	As above
	II	.6	83	±90°	Ballistic flight α = 55° to equilibrium W/C_LA = 83
	III	.7	83	0°	Equilibrium flight to V = 4,000 ft/sec
	IV	.7	83	0°	

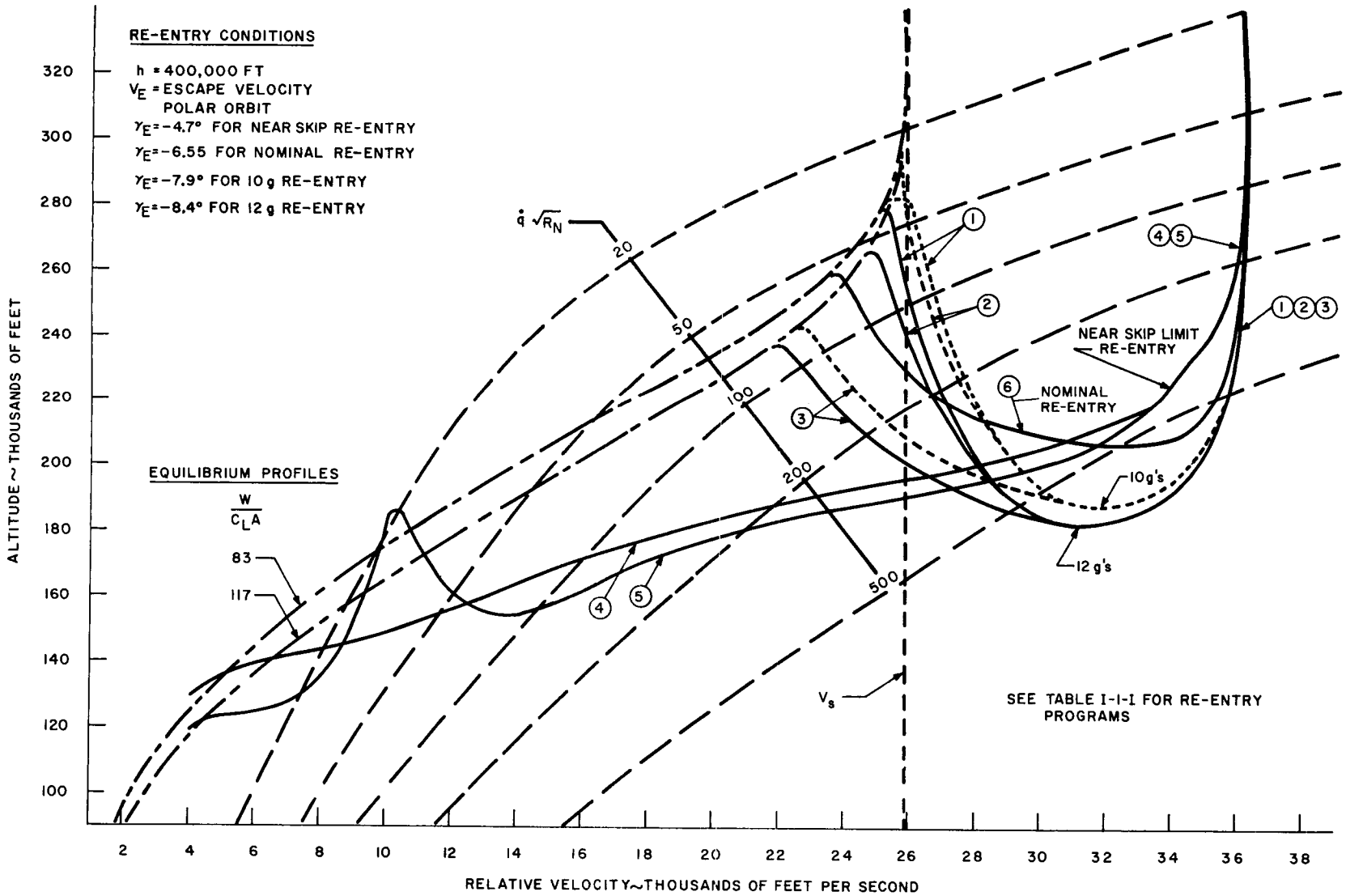


Figure I-1-55. Modified lenticular vehicle re-entry profiles

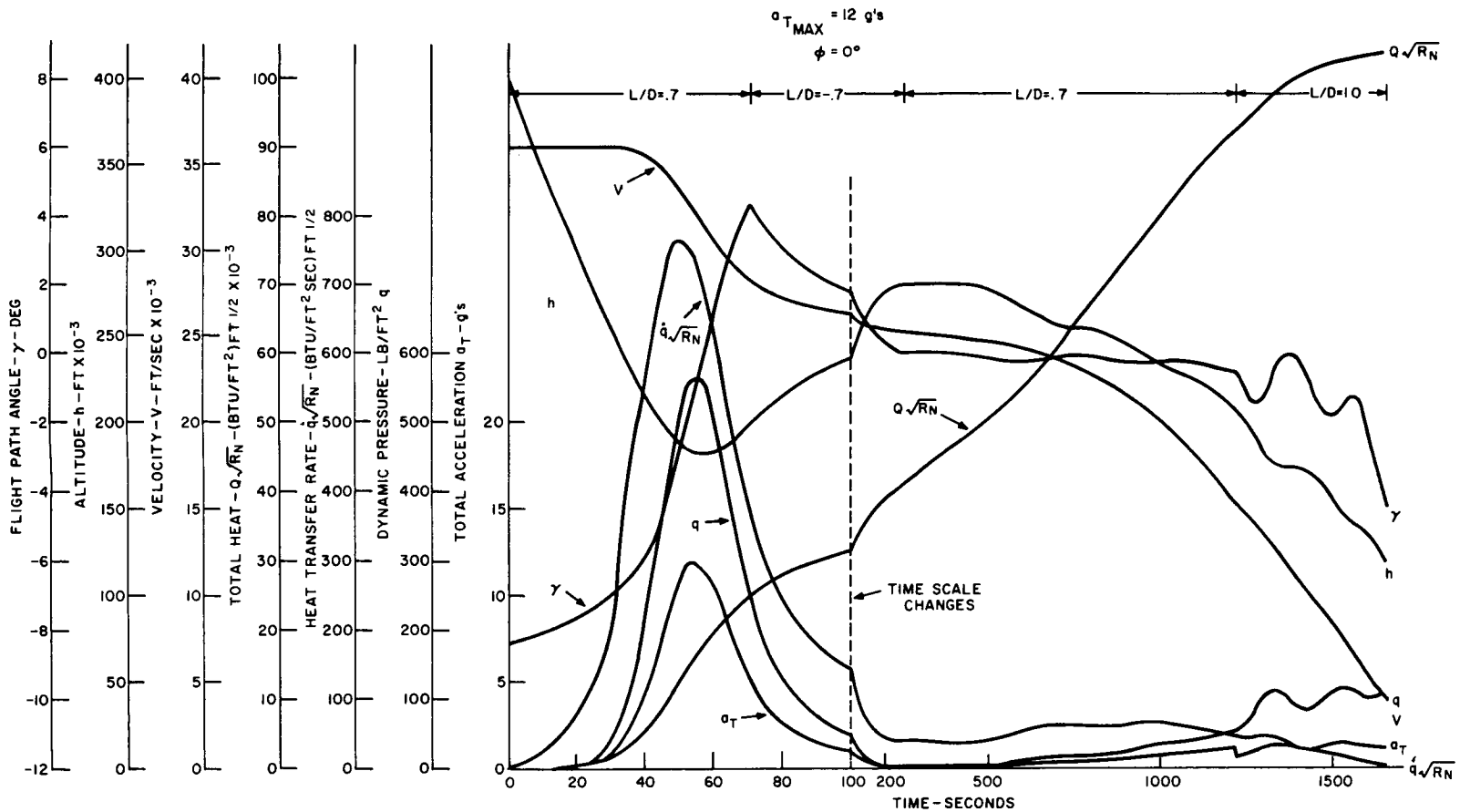


Figure I-1-56. Modified lenticular vehicle re-entry time history

[redacted]

[redacted]

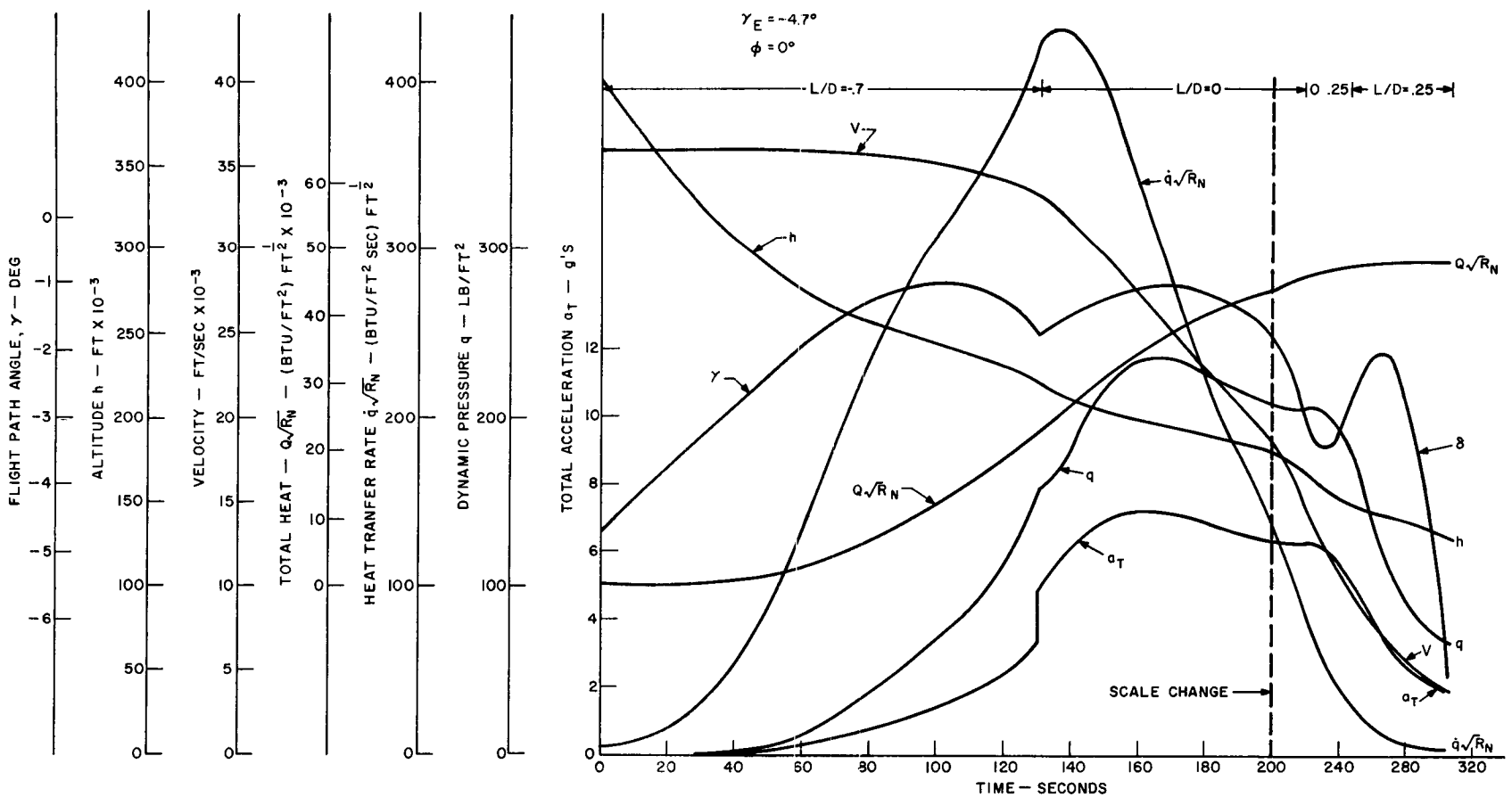


Figure I-1-57. Modified lenticular vehicle time history from near skip limit re-entry

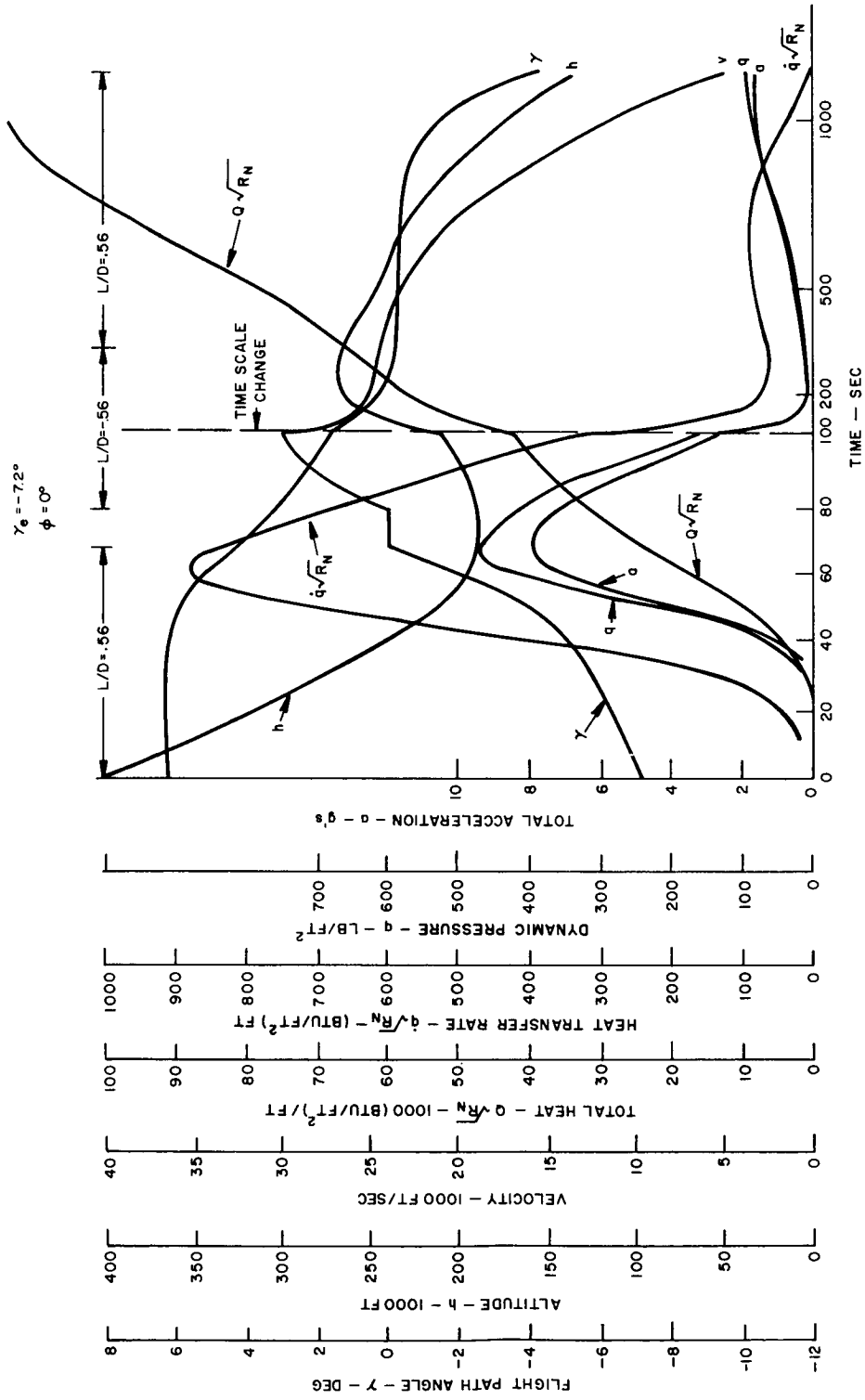


Figure I-1-58. APOLLO D-2 vehicle time history for 8g re-entry

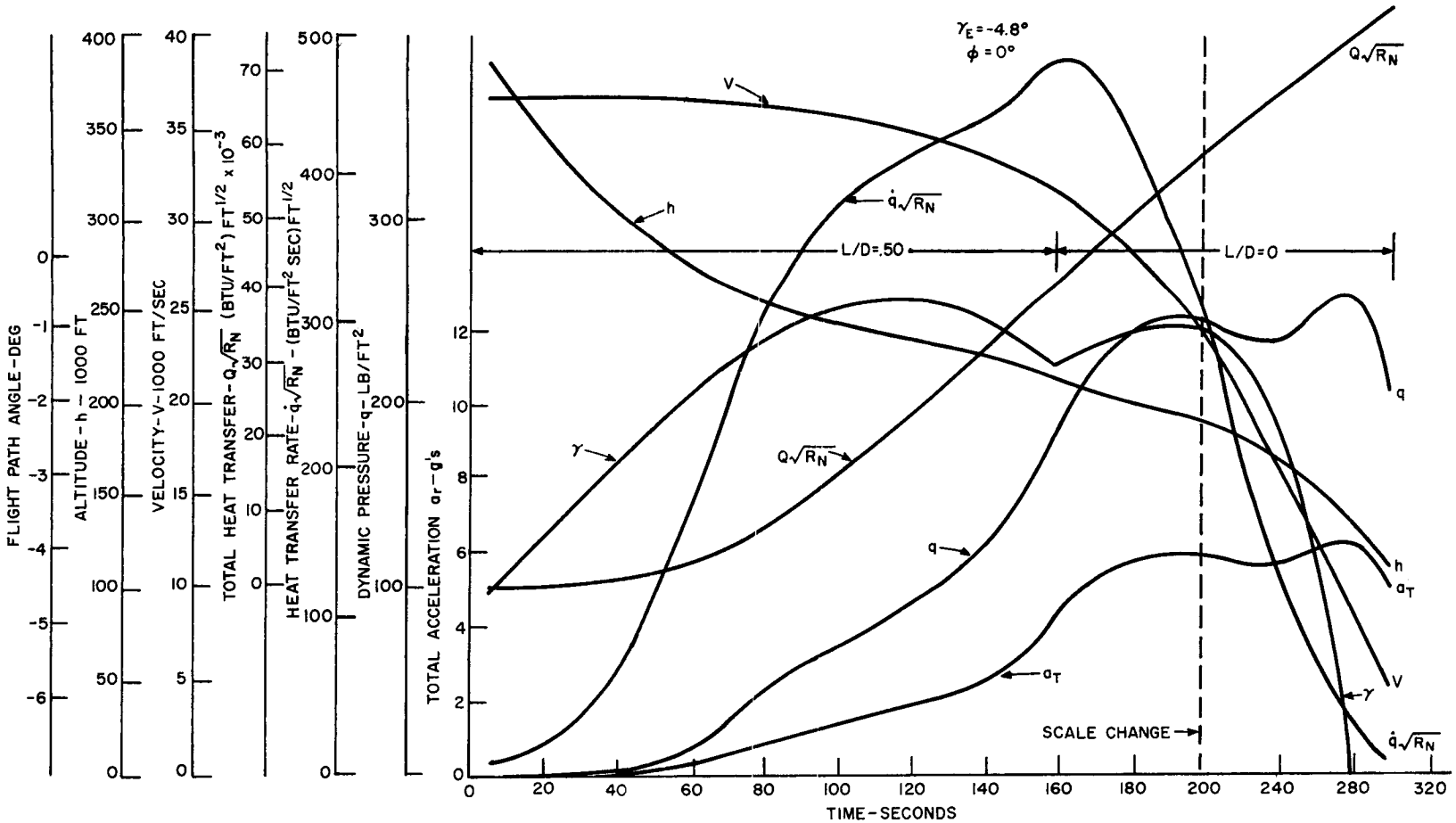


Figure I-1-59. APOLLO D-2 semi-ballistic vehicle time history from near skip limit

down-range maneuver capability and also connect them to free-flight approach trajectories they must be referenced to a common point. This has been taken as perigee of the vacuum approach ellipse. Since a return trajectory from the Moon is defined by an energy which is nearly parabolic, a ground range correction equal to twice the re-entry path angle is used to refer the touch-down point to vacuum perigee.

Figure I-1-60 shows the guaranteed ground area attainable which was determined from the aforementioned trajectory studies. For the APOLLO R-3 configuration, limiting the re-entry to 12 g's gives an area about 3730 n mi long and 1440 n mi wide. With a maximum of 10 g's these dimensions are increased to 4700 n mi and 1800 n mi at, however, a reduction of .50 degrees in allowable re-entry angle and 28 percent increase in the time during which reaction control may be required because dynamic pressure is less than 25 lb/sq ft. The APOLLO D-2 vehicle, because of its lower L/D of 0.57, has a lower but adequate attainable ground area, with a range correction capability of 1960 n mi and a maximum cross-range capability of ± 500 n mi.

1.7.3 Summary

Re-entry trajectory studies have been performed to determine the re-entry corridor and maneuverability for the APOLLO D-2 and R-3 configurations. The D-2 with a maximum peak deceleration of 8 g's has a total angular corridor of 2.4 degrees and an altitude corridor of 30 n mi. The R-3 with 10 g maximum peak deceleration has a total angular corridor of 3.2 degrees and an altitude corridor of 43 n mi. Considering maneuverability for landing point selection, the D-2 has, for 8 g's limit, a total range-change capability of 1960 n mi and a cross-range capability of ± 500 n mi. The R-3 with 10 g's limit has a range-change capability of 4700 n mi and a lateral capability of ± 900 n mi. To achieve these maneuverability values it is necessary to permit the vehicle to fly, for a short period during re-entry, in a region of limited aerodynamic control. With vehicles of higher L/D, full aerodynamic control could be maintained. Greater range could be obtained by allowing the vehicles to attain higher apogee altitudes but this is not recommended because of the extreme trajectory sensitivities and lack of continuous range control associated with this procedure. Further study of the short period motions of the vehicles on the trajectories is recommended. Finally, it

[redacted]

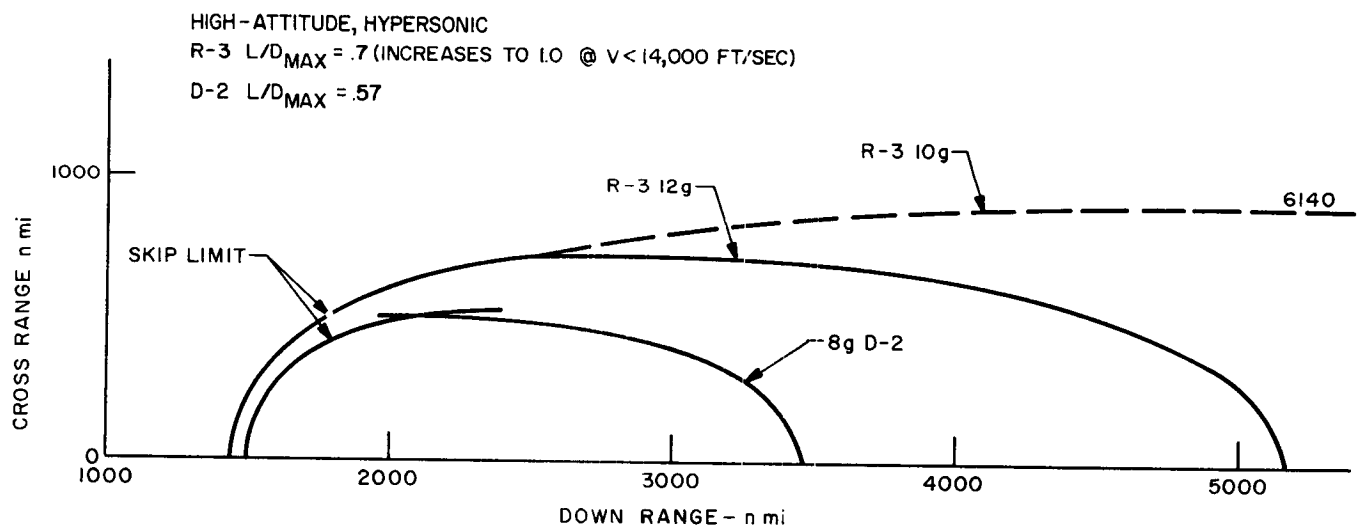


Figure I-1-60. Composite ground area attainable considering steep and shallow re-entries

[redacted]

should be pointed out that, while roll control alone, combined with c.g. offset for attitude generation, is theoretically capable of permitting trajectory control; the use of flaps for pitch control as assumed in this study adds considerable flexibility to the control system and is therefore recommended.

[redacted]

2.0 Earth-Orbit Development Flights

The orderly progression of APOLLO missions, leading eventually to a manned lunar landing and return, will include earth-orbital missions in the early phases of the development. These satellite orbits will progress from relatively circular, low-altitude orbits to elliptical orbits with apogee altitudes of the order 25,000 to 100,000 n mi.

The early circular orbits would be used for initial evaluation of the APOLLO vehicle and subsystem design and also for crew training. These orbits would also be used for experiments leading to capabilities in the areas of rendezvous, orbital assembly, refueling techniques, etc. In addition the dis-orbit, re-entry, and atmospheric maneuver to selected landing sites would be studied.

The altitudes of these circular orbits would be constrained at the lower limit by atmospheric drag perturbations on the vehicle's orbit and at the upper limit by the effective surface of the inner radiation belt. Thus orbital altitudes between values of approximately 150 and 400 miles could be used for these flights.

The satellite altitude for any specific mission would, of course, be a function of the booster payload capability, and also of the vehicle's re-entry trajectory relative to the selected landing site after the appropriate flight duration or the number of orbital revolutions. Sample earth-surface traces for earth satellite orbits are included. Figure I-2-1 presents the trace for a circular orbit an altitude of 150 n mi. Figure I-2-2 presents a similar trace for a circular orbit at an altitude of 300 n mi.

Later, elliptical orbits would be flown to more nearly simulate the outbound and return phases of the APOLLO lunar missions. The systems to be tested would include tracking, communications, and navigation including data acquisition. The midcourse propulsion system, which would be required to initiate re-entry after the appropriate number of revolutions by decreasing the vacuum perigee altitude of the final orbit, could also be used to possibly change the inclination of the satellite orbital plane.

[redacted]

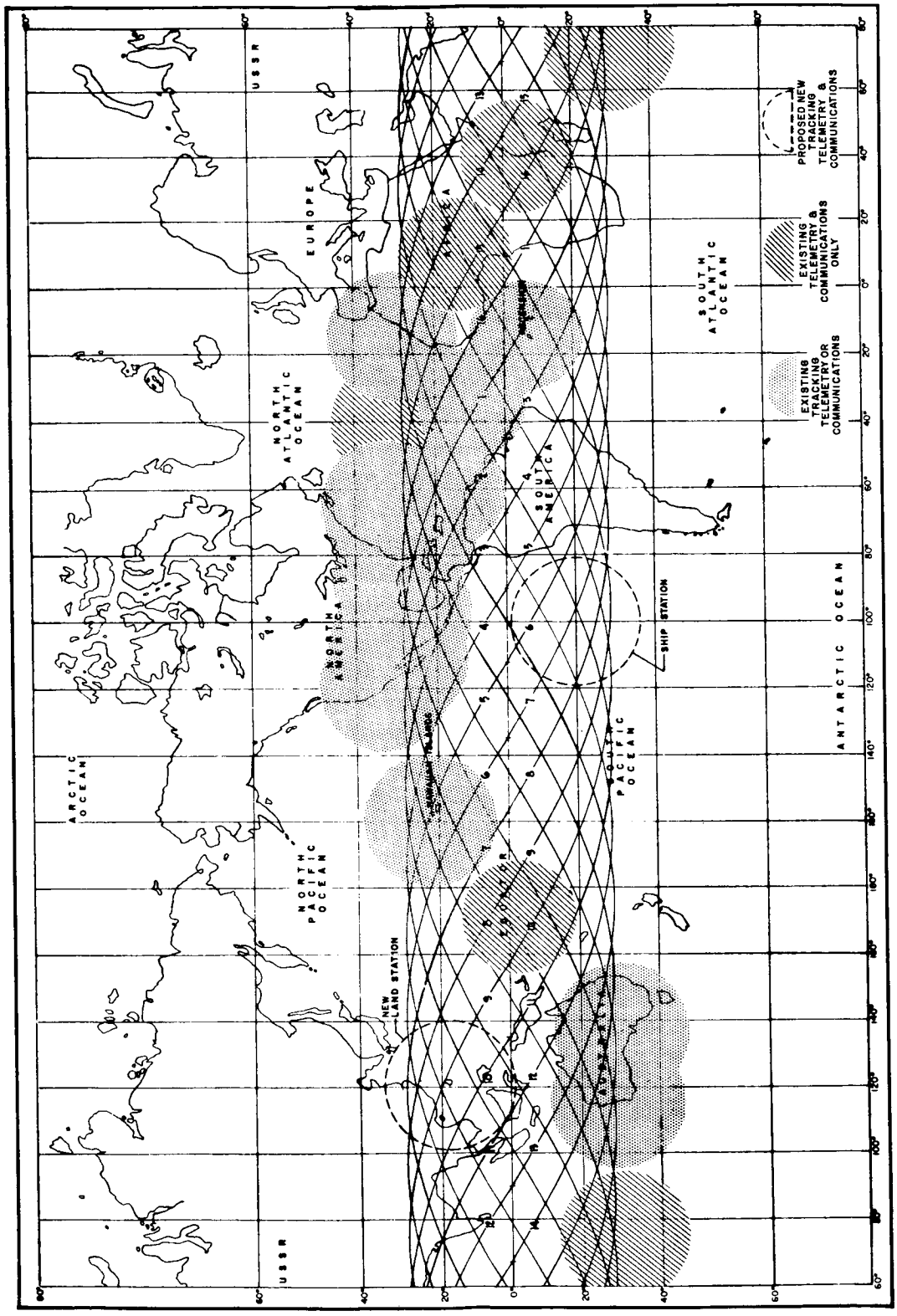


Figure I-2-1. Trace of a typical circular satellite orbit
(h = 150 n. mi.)

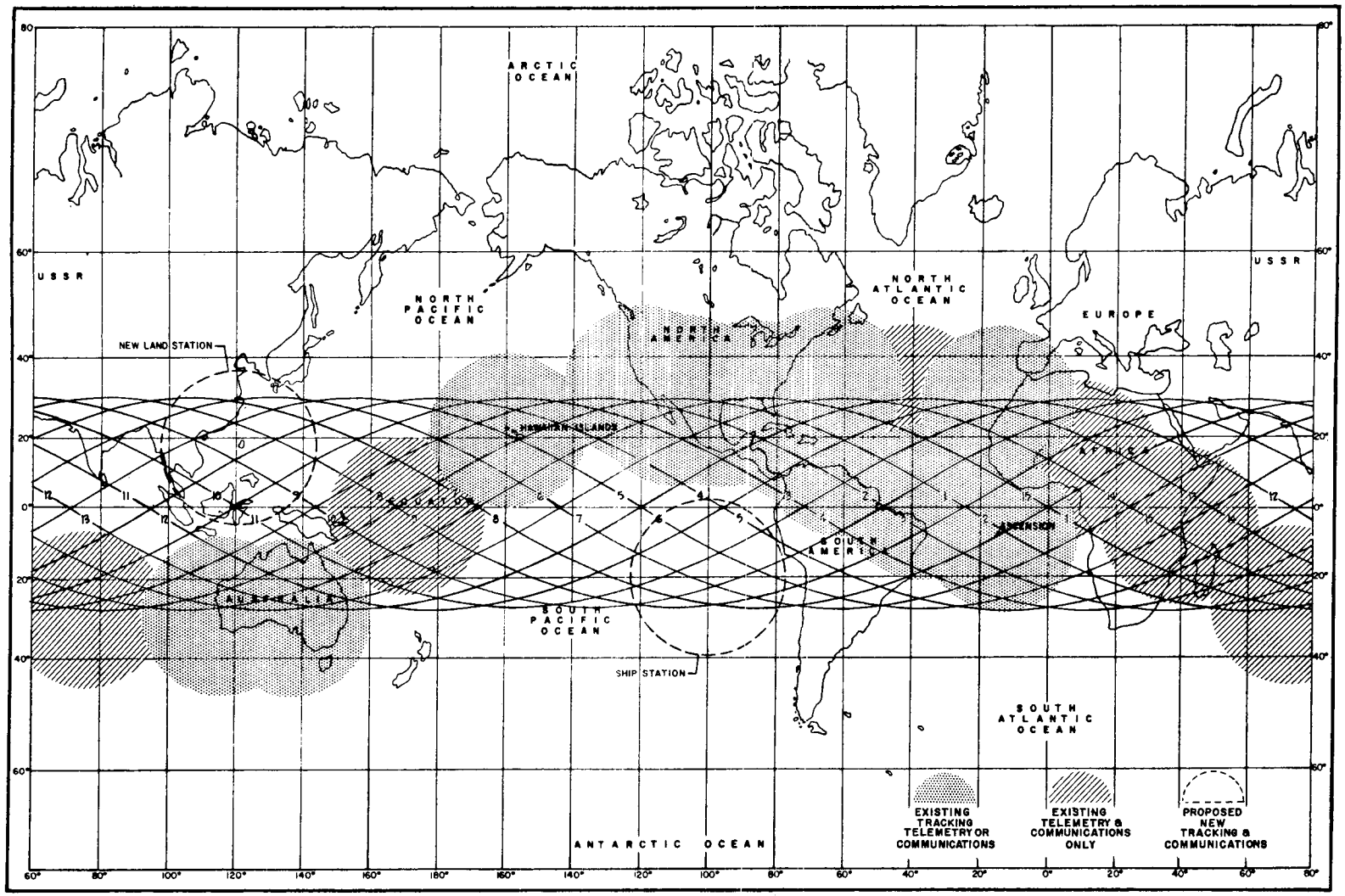


Figure I-2-2. Trace of a typical circular satellite orbit
(h = 300 n. mi.)

The factors which must be considered in the selection of the parameters of an elliptical orbit are the total mission time required, the geographical location of the landing site, perigee altitude, and the orbital inclination from radiation considerations.

The total mission time, which is limited to 14 days for the lunar mission, could be extended to longer durations for earth orbital missions. For a given perigee altitude, the apogee altitude of the ellipse must be selected to return the vehicle to the appropriate longitude for re-entry and landing.

For purposes of this exploratory study, the perigee altitude of the elliptical orbit has been selected as 150 n mi. This altitude, which is below the radiation belts, is high enough so that the vehicle's drag deceleration at perigee is acceptably low from the point of view of orbital perturbations.

Figure I-2-3 shows the period of elliptical satellite orbits as a function of apogee altitude for a constant perigee altitude of 150 n mi. As the apogee of the orbit is increased beyond an altitude of about 150,000 n mi, perturbations due to the gravitational attractions of the Moon and sun can become large enough to require "n-body" computations of the orbit for specific dates.

In determining the apogee altitudes required, the longitudes of the initial and final perigees have been approximated as Cape Canaveral and Edwards AFB. The satellite period required as a function of total duration time and number of revolutions results in the plot shown in Figure I-2-4. This curve presents families of apogee altitude points as a function of the number of revolutions. The points are connected by curves of total flight times between 2 and 14 days. The apogee altitudes range from about 22,000 miles for 4 revolutions in 2 days up to 160,000 miles for 1 revolution in 7 days.

The test objectives for a mission of a selected orbital duration would influence the choice of the apogee altitude. If the flight were set up to test the data acquisition equipment for the on-board guidance system for example, it might be advantageous to select an apogee altitude which is high enough to permit the realistic switching from earth-centered tracking to celestial tracking and back again a number of times. The apogee altitude for this mission would be the lowest value which is compatible with the test requirements. A mission to simulate the outbound and return mid-course guidance and

[redacted]

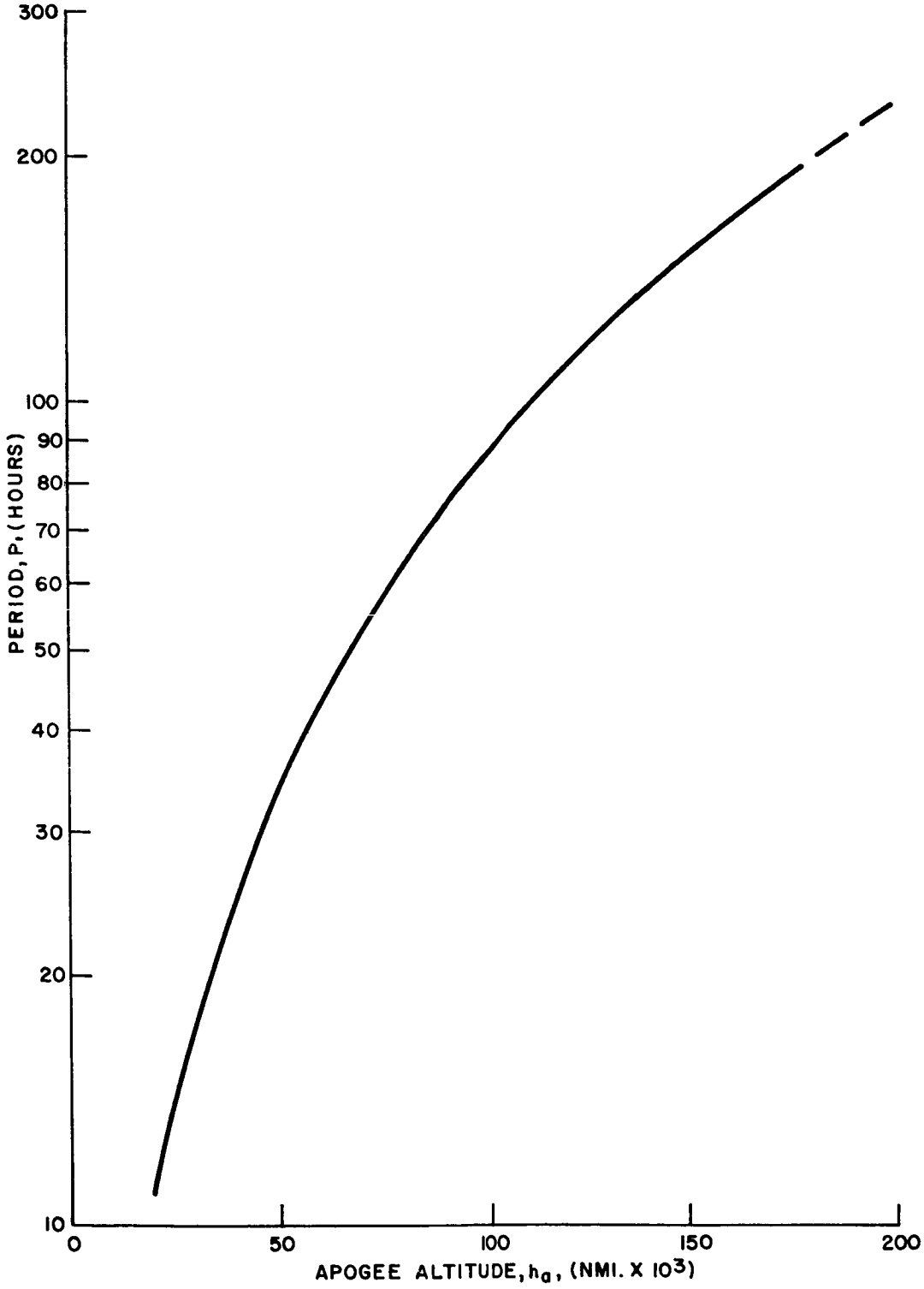


Figure I-2-3. Orbital period vs apogee altitude
(h_p = 150 n.mi.)

[redacted]

[redacted]

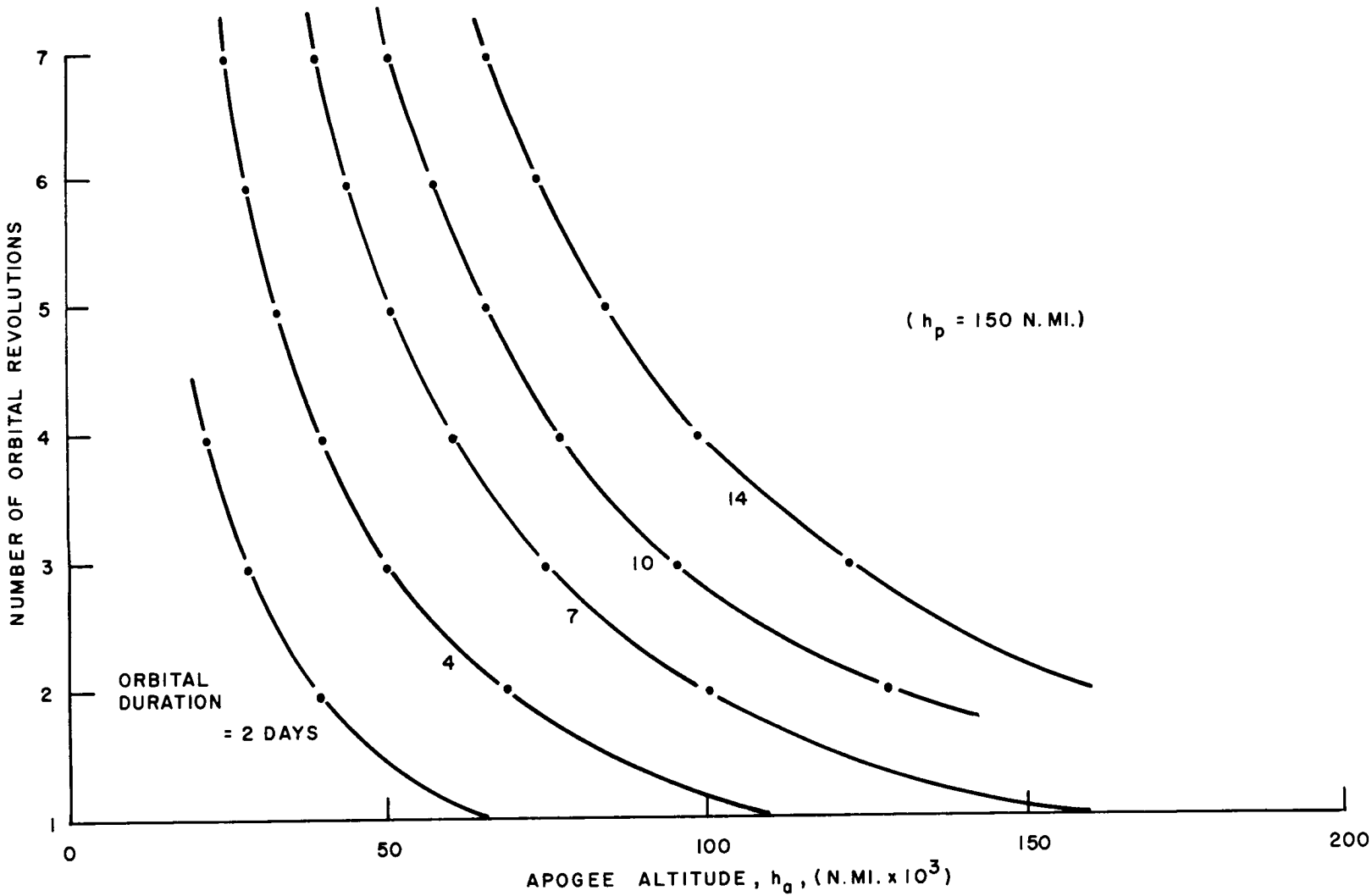


Figure I-2-4. Apogee altitude required for various orbital durations (h_p = 150 n.mi.)

[redacted]

propulsion systems, on the other hand, might make use of the highest practical apogee altitude.

The earth-surface trace of the first two revolutions of an elliptical satellite orbit are shown in Figure I-2-5. The apogee and perigee altitudes are 23,000 and 150 n mi, respectively.

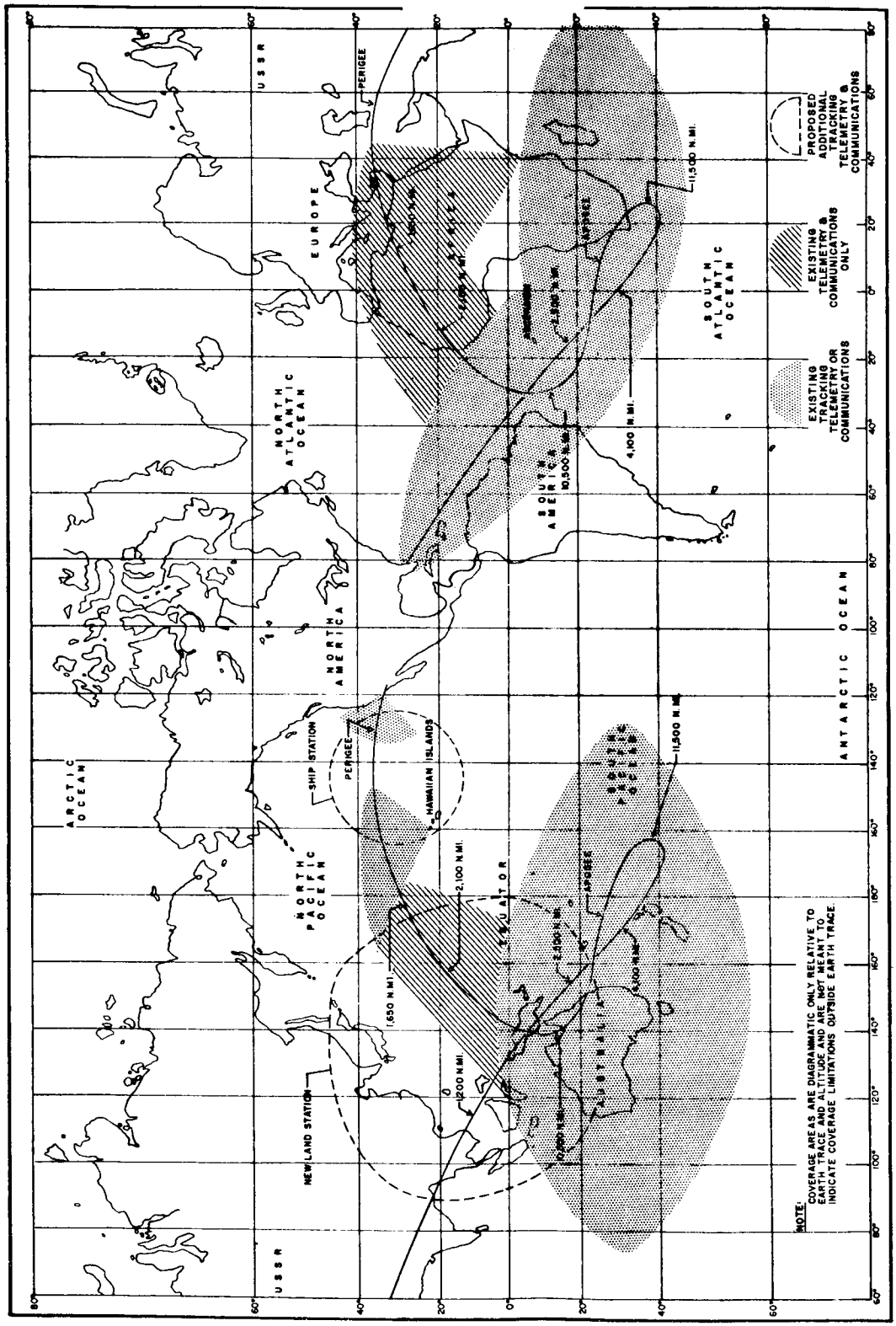


Figure I-2-5. Trace of a typical elliptical satellite orbit (h_a = 23,000 n. mi., h_p = 150 n. mi.)

[redacted]

CHAPTER II

NAVIGATION AND GUIDANCE

[redacted]

II NAVIGATION AND GUIDANCE

1.0 Introduction and Summary

The study effort reported herein details the General Electric Company's Missile and Space Vehicle Department's concept and preliminary design of a navigation, guidance and control system for fulfilling the APOLLO mission objectives.

The objectives of the Navigation, Guidance, and Control study have been to recommend, define, and substantiate a logical approach to a system which fulfills the requirements of APOLLO missions. These missions include lunar and earth orbital missions.

The major NASA guidelines which affect the overall Navigation, Guidance, and Control system design have been observed, and requirements resulting therefrom have been met. Within the framework of these guidelines, it was necessary to concentrate on preliminary concepts and equipment selection early in this study program to provide information to vehicle design personnel on size, weight, and desired location of components. Emphasis has been placed on meeting system requirements and synthesizing preliminary designs during the remainder of this study effort.

Guidance concepts were generated to meet the requirements for the various phases of the probable manned APOLLO missions. The phases considered are listed below:

Boost

Insertion

Outgoing Midcourse

Lunar Orbit Injection

Lunar Orbiting

Lunar Orbit Exit

Return Midcourse

Re-entry

Landing

Abort and Emergency Return for all phases

Rendezvous

Lunar Landing

Assembly in Orbit

Some phases have received more study effort than others and will be so indicated. The principal guidance requirements and most promising concepts are discussed. In general, further study effort, such as simulation study, is required to justify these concepts numerically.

To achieve primary on-board command, various spacecraft-based navigation equipments were investigated to implement the guidance concepts studied. These equipments include optical, inertial, and radio - both receiving and transmitting.

To utilize the crew to the fullest extent, the capability of the crew to perform useful functions was assumed not to be diminished in any way by their environment in space. In his primary role as a decision maker and monitor, the pilot is provided, by means of displays, with information as to the status of all equipment and with all information required to assess the desirability of executing an ensuing path or attitude change command. Using the displays, he is also able to monitor the kinematics of transition from one dynamic state to another, to interrupt the transition, or to allow it to continue at his discretion (with the exception of over-riding an automatically induced abort).

Consideration has also been given to providing for a capability to take navigational fixes. The crew plays a dominant role in this task. In some phase of the lunar mission, particularly on the return leg, the pilot may perform computation of the guidance problem to determine a safe return trajectory. Further study is required to assess whether the crew can compute the solution to the guidance problem in general. An excessive amount of fuel, by virtue of required mathematical simplification in order to reduce computational labor, may be required.

To the extent possible in this study, equipment has been designed to allow on-board maintenance and repair.

Probably the most limiting condition for the study is the high reliability requirement of such a mission. It is anticipated that such reliability can be demonstrated only by a fairly extensive test program using both simulated and actual flight conditions. To meet the projected time scale for the APOLLO mission, such a test program must be started soon with equipment that is either available now or in an advanced state of development.

The major elements of the Navigation and Control subsystem are:

1. An inertial reference used for monitoring the boost phase, for re-entry navigation and energy management, and as an inertial reference and measuring device during the application of velocity corrections during the midcourse and lunar phase.
2. Navigation sensors to make measurements of angles between earth, Moon and stars, and to measure the apparent diameter of the earth or Moon for stadiametric ranging.
3. A digital computer which processes the navigation fix data, calculates velocity corrections and performs other programming, data storage, computation and control functions.
4. An attitude control system to maintain the APOLLO vehicle at the proper attitude from Saturn separation to re-entry vehicle separation, and to make changes in vehicle attitude as required.
5. The re-entry vehicle flight control system to provide attitude stabilization and control of the re-entry vehicle from separation from APOLLO to the landing phase.

In the course of implementing the APOLLO navigation and control subsystem, a number of hardware approaches were investigated. The approach recommended currently is described in Section 4.0. Two others are described in the Appendices. The basic guidance concept is the same in all cases, the differences being primarily in the sensing

components. Details of all of these schemes were not available in time to be integrated into the vehicle design. Thus the system pictured in vehicle design drawings is the one described as system #1 in the Project APOLLO Data Book, dated March 14, 1961. This material is repeated in Appendix T-B.

In an effort to meet reliability goals and maintain development schedules, considerable effort was expended in determining hardware and hardware requirements to carry out the mission with "state-of-the-art" principals wherever possible. Considerable design and development effort, however, is required to evolve certain hardware required for the APOLLO mission. The Celestial Sextant and on-board computer are two pieces of hardware in this category.

The final design of the APOLLO hardware is influenced considerably by reliability considerations. As shown in the Program Plan presented in Volume IX of this report, a large scale effort on reliability during the initial stages of design is required.

KEY RESULTS AND CONCLUSIONS ARE LISTED BELOW

1. A navigation and guidance system to perform the lunar orbit, circumlunar, lunar landing, earth orbit and satellite rendezvous missions of the APOLLO vehicle is feasible for the APOLLO time schedule. The major developments required are:

   a. Vehicle-borne Computer

      The reliability of the Guidance and Control computer is currently considered the weak link in the subsystem. Improvement of computer reliability requires the following steps:

      1. A more detailed and comprehensive study of the tradeoffs between computer complexity and fuel requirements. The objective of this study would be to reduce computer complexity as much as possible. (For example, fuel required for midcourse corrections now constitutes about 3 percent of the vehicle weight. A computer simplification that increases this requirement to 4 percent would undoubtedly be worthwhile. A simplification which raises it to 6 percent would probably not be worthwhile.) This study will be initiated shortly with our simulation program.

2. Establish reliability of simple memory concepts (for example, solid state devices).

3. Develop "standard" logical blocks for maximum reliability.

4. Maximize the use of the crew in improving reliability by such techniques as fault location, repairing, inserting new inputs, bypassing defective parts, and modification of computer routing.

5. A re-evaluation of total guidance implementation to reduce computer size without affecting accuracy (i.e. eliminate unnecessary programming, more effective use of implicit or explicit computation).

b. Celestial Sextant

The basic problem here is the accuracy of the IR disc tracker. Demonstrated state-of-the-art accuracy is about 300 seconds. Accuracies better than 150 seconds represent a considerable development problem. Desired accuracy for APOLLO is 30 seconds. Problems involved in designing an IR sensor with this accuracy include:

1. Uniformity of the discontinuity at the edge of the disk.

2. Signal level of probable uniform zone vs. detector sensitivity.

The specification of measurement accuracy requirements involves study of the tradeoffs between sensing accuracy, computer complexity (data processing) and weight of propulsion.

The General Electric Company's Missile and Space Vehicle Department intends to continue studies in the above critical areas. The required simulation program is now well under way.

2. Our basic approach to the Guidance and Control problem utilizes a semi-automatic system in which man plays an important part. Our studies indicate that the crew will perform the following functions:

a. Monitoring of overall Guidance and Control subsystem operation by observing displays.

b. Make appropriate decisions based on all available information. For instance, decisions to make corrections, to abort under some conditions, to enter and leave lunar orbit.

c. Initiate appropriate course of action. For example, making gross orientation changes in vehicle and celestial sextant, and initiating corrections. The crew also will have the capability to take fixes and make computations and corrections manually in case of failure of the automatic equipment.

d. Maintenance, test and repairs of equipment.

3. It is preferable but not mandatory that the APOLLO guidance have no interface with the Saturn boost guidance.

2.0 Navigation and Guidance Concepts

2.1 INTRODUCTION

As discussed in the introduction in Section 1.0, it was necessary to take a gross look at the entire navigation and guidance problem early in the study in order that the hardware configuration be roughly defined. Subsequently, the detailed analysis was performed. Although, in some areas, the desired degree of analysis was not possible, a unified mission guidance concept was derived.

This section will cover the areas listed below. In some areas extensive analysis has been accomplished and is presented; in others conceptual discussions are presented.

- Mission Description
- Insertion Guidance
- Midcourse Guidance & Navigation
- Lunar Orbit Guidance and Navigation
- Re-entry
- Emergency Return
- Abort

2.2 MISSION DESCRIPTION

Figure II-2-1 shows a typical lunar mission, which includes establishing an orbit around the Moon, and a subsequent return flight to the earth. This mission has received primary emphasis during the course of the study. Reference to Figure II-2-1 will be made for the following description of the mission. Figure II-2-2 shows a simplified block diagram of the guidance and control subsystem.

The primary guidance and control subsystem is entirely self-contained and semi-automatic in operation. That is, certain functions which require either a large number of

[redacted]

[redacted]

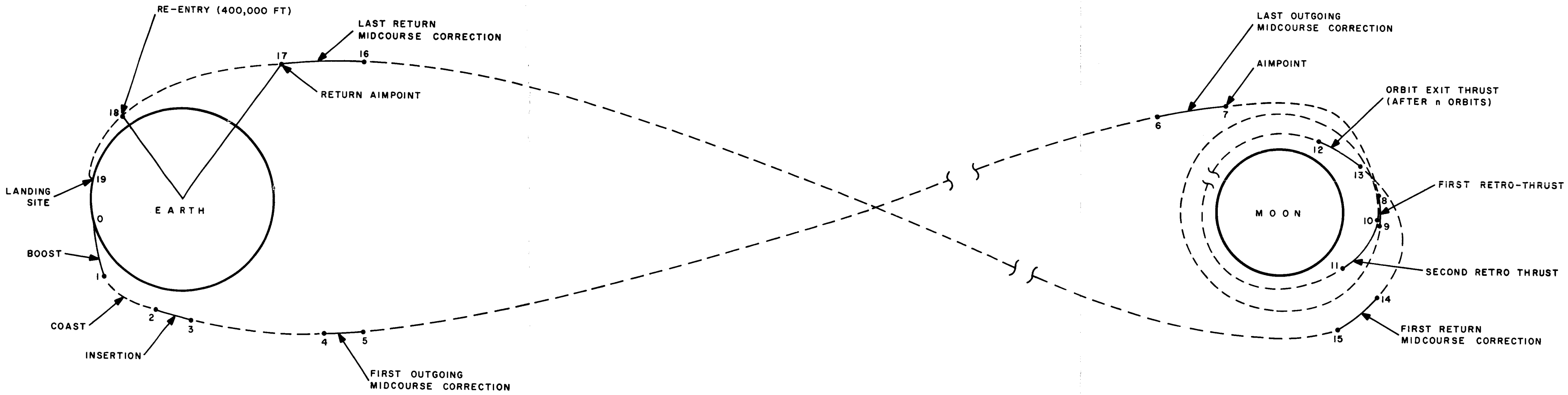


Figure II-2-1. Lunar orbiting mission profile

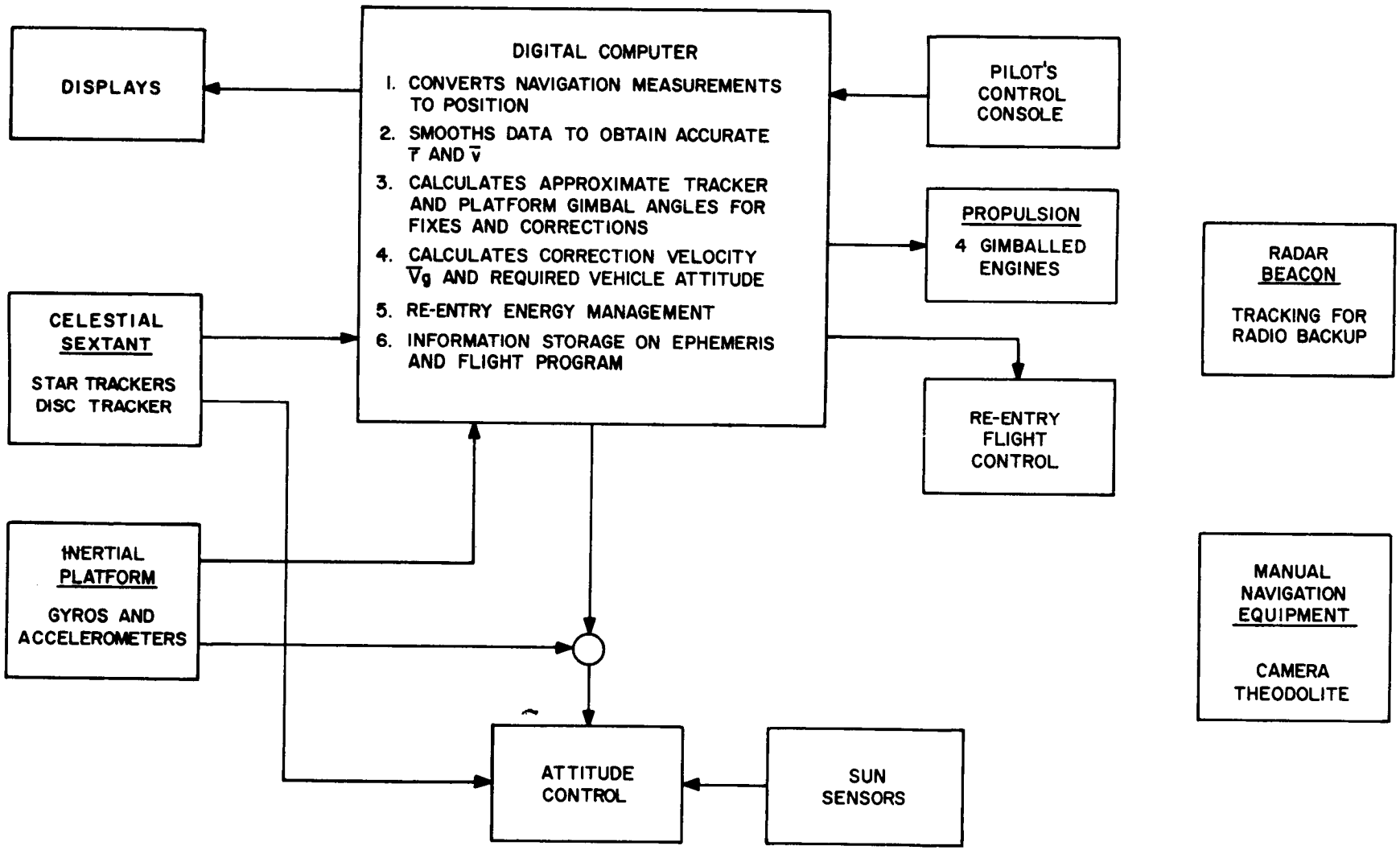


Figure II-2-2. Navigation and guidance subsystem

repetitive measurements or calculations, or which require precise control, are automatic. However, the crew maintains control of the mission through observation of displays and with the capability of initiation and override of all guidance and control functions.

The manner in which this equipment is used will also be indicated in the following discussions.

A three axis on-off attitude control system is utilized to align and hold the vehicle attitude in the proper direction to within ±1 degree while the low level (100 lb) fixed thrust units are used for midcourse corrections. This operation may be performed by the crew.

For the large gimballed thrust units (2,000 to 6,000 lbs) the on-off attitude control is used, either manually or automatically to rough align the thrust axis in the proper direction to within ±5 degrees.

While firing the large units gimbal deflections are commanded to keep the thrust axis aligned in the commanded direction.

The general APOLLO guidance problem is to first guide the APOLLO spacecraft with the Saturn C-2 booster from launch, point 0, to insertion, point 3. The nominal three-stage Saturn trajectory is chosen so that the final cut-off condition, at point 3, will establish a trajectory that will exactly hit a pre-selected point in space, at a pre-selected time. A nominal trajectory is chosen for the eight-engine Saturn booster and the seven-engine booster. In the event of a single engine failure at some time during the first stage a remaining nominal trajectory is found by interpolation. This is possible since the nominal trajectories and pitch profile for the two extreme cases are similar. (See Appendix T-C).

Inertial guidance will be used for the boost phase. As will be seen, an inertial measurement unit located in the APOLLO spacecraft will be employed for other phases of the mission. To achieve the required accuracy high quality inertial components are used. The spacecraft inertial system can monitor the powered flight phase to determine position and velocity.

A study of the boost guidance problem allows an assessment of the spacecraft computer requirements. In addition, a guidance approach which appears promising at this time is being developed. Appendix T-C discusses the concept in detail.

In general, a coast phase between points 1 and 2 at circular velocity may exist before the insertion phase begins. The coast period is a function of lunar declination at approach and can be as long as twenty-five minutes (See Chapter I of this volume). If no coast phase is required the insertion phase would commence immediately.

Navigational fixes may be taken during this period, with equipment to be described, if sufficient time is available to perform this task. An estimate of improvement in insertion accuracy resulting from corrections to the inertial unit from stellar fixes has been made. The improvement can be of the order of three to one.

The Saturn C-2 booster powers the spacecraft to insertion at point 3. The objective during this period is to establish the proper initial conditions to initiate a safe return circumlunar trajectory that would allow return to earth and re-enter at the proper point—and still meet the requirements of passing within 1000 to 2000 miles of the Moon. Although nominal trajectories may be established to meet these requirements further study is required to determine boost guidance capability to do likewise for perturbed flight conditions. The study of the remaining phases of flight assumes that safe return circumlunar initial conditions exist at point 3. Fixed times of flight will thus be established to the aimpoints at the Moon and earth.

It is shown that the insertion conditions may be established with a velocity accuracy of 25 ft/sec and a position accuracy of 1 nmi. These inaccuracies produce miss distances at the lunar aimpoint (point 7) of the order of several thousand miles. As a result of the large miss distances expected, midcourse corrections on the outgoing leg to the Moon will be required.

Following cut-off at insertion, angular rotations imparted by the separation process are eliminated by a three axis, on-off attitude control. The process can be controlled automatically or manually. The 4 gimballed inertial platform will provide the required reference. The first correction is shown to occur between points 4 and 5.

Orbit predictions begin soon after separation. At this time the spacecraft computer knows the spacecraft position and velocity at insertion. The equations of motion in earth-Moon space are integrated by the Encke method. The ephemerides of the sun and Moon are stored in the computer.

The automatic celestial trackers take measurements between points 3 and 4. Earth and Moon angular diameter measurements and direction of the vertical to these bodies are measured in a coordinate system established by star trackers. These measurements are processed by the on-board digital computer to obtain position and velocity estimates. Either instantaneous position information or information obtained through short term second order smoothing or long term maximum likelihood estimates is available. Velocity information is obtained by processing position data. No adequate instantaneous source of velocity information has been found.

Periodic updating from ground tracking information as relayed over the voice communication link will also be used, if available, although it is not required.

The APOLLO crew can also obtain navigational information by manual techniques. Independent computations are made and checked against the computer result.

Prime emphasis during this study effort has been placed on the celestial tracker design. Less emphasis has been placed on manual techniques such as navigational periscopes and photography.

Celestial trackers will be capable of measuring line-of-sight to the body being tracked with an estimated accuracy of 10 seconds (RMS). Further study is required to determine the accuracy of the photographic and periscope systems. Accuracies of these equipments are expected to be within several seconds of arc.

The large amount of data that is accumulated prior to the first correction point is processed by the on-board computer. By data smoothing techniques mentioned a good estimate of the actual trajectory being flown is obtained. The position and velocity will be known to within two miles and several ft/sec after four hours of data taking. Estimates of expected aimpoint miss components and miss components due to instrument errors are displayed to the pilot at his request. The pilot makes the decision whether to make a velocity correction or not.

The first velocity correction will be made within the first 10 hours after insertion. The correction is computed by a differential correction technique for a pre-established remaining time-of-flight to the aimpoint. The first velocity correction will be approximately 75 ft/sec and eliminates miss distance components at the aimpoint.

The correction is applied by first rotating the vehicle so that the desired engine is pointing in the commanded direction. The correction is then made by automatic control of the 0.5 to 2.0 g gimballed engine system through the enertial platform and computer (Points 4 to 5 on Figure II-2-1). Fixed low-thrust, ullage engines of 0.01 g's may also be used. Platform accelerometers control the thrust.

Following this correction the data processing cycle is repeated. Several more corrections may be required before reaching the aimpoint.

The last correction will be started at point 6. Nominally the spacecraft will arrive at this point at the correct scheduled time and position, but with a different velocity. This velocity deviation is corrected at this point. The aimpoint has been selected to be several hours from the distance-of-closest-approach at the Moon. This choice is made to allow adquate time to prepare to orbit the Moon. Adherence to reference conditions will also enable the spacecraft to continue on the safe return trajectory if the decision to orbit the Moon is cancelled. An additional advantage to returning to reference conditions is that stored data may be used to establish the required nominal thrust programs for both entering into orbit around the Moon—(points 8 and 10) and for subsequent disorbit at a precomputed time during any particular orbit vehicle in continuous orbit around the Moon—(point 12). Small perturbations from nominal conditions may be accounted for by using stored sensitivity coefficients.

Studies are reported herein for each of the foregoing phases in which explicit computations, i.e. not based on stored initial conditions, are made to establish the nominal thrust phases. A complete numerical evaluation of the concepts has not been attempted.

When the Moon is closely approached, position fixes will be measured relative to the Moon. Here too, the star tracker is used to establish the coordinate system and the horizon scanners measures angles relative to these axes and measures distance by an

angular measurement of the Moon's diameter. For a mission which orbits the Moon, perilune of the approach hyperbola is calculated and a guidance scheme is used which steers the spacecraft to a near-circular orbit about the Moon. In this orbit, position fixes are made and smoothed to establish the true elements of the orbit. A similar guidance scheme is initiated which further reduces the spacecraft velocity to achieve a 50 mile perilune orbit. Perilune may be established to an accuracy of several miles.

Studies have indicated that, for the launch date considered, the spacecraft orbit around the Moon is little perturbed by the sun and planets. Thus an orbit established around the Moon will be stable and, except for errors in establishing the orbit, no corrections are required to maintain the spacecraft on nominal schedule.

The lunar orbit exit thrust phase (point 12) may be initiated at a time precalculated before flight or by explicit computation—if the latter computation proves feasible. The explicit computation would entail a first estimate using "patched conic" approximations. If further refinement is necessary, the patched conic first estimates would be used for the n-body (actually four bodies-earth, Moon, sun and spacecraft) computation. Studies are reported which show the inaccuracies of two-body and patched conic approximations as compared to four-body return to earth reference trajectory. Promising preliminary results indicate that the on-board computations may be considerably simplified (recall that an explicit 4-body computation is presently being used) at the expense of increased fuel requirements. It also appears that computations may be performed by the crew to at least return the spacecraft from lunar orbit.

The expected errors at the orbit exit point, point 13, will be 10 ft/sec or less in velocity and several miles in position.

Although not completely determined as yet, it is expected that disorbit initial conditions during any orbit, can be established to return the spacecraft to the proper initial re-entry condition for near-ballistic flight to the landing target with no substantial penalty in fuel.

After the spacecraft leaves the Moon orbit, the return leg follows a similar midcourse correction scheme as the outgoing leg. The first correction indicated at points 14 and 15 can occur at 10 hours after disorbit and will be of the order of 35 ft/sec. The

[redacted]

spacecraft will be maintained on a nominal time schedule established when leaving the Moon. The spacecraft will arrive at a point 10,000 miles from the center of the earth (point 17) near nominal position and time. At this point a velocity correction (points 16 and 17) can be made to return the spacecraft to reference trajectory conditions.

From 10,000 miles to re-entry (point 18), the spacecraft will be prepared for the initiation of re-entry, approximately one-half hour away.

The spacecraft will arrive at the re-entry point within an error of 20 to 100 nmi down-range and 10 to 30 cross range. The navigation measurement system will establish, at the re-entry point, the position and velocity of the spacecraft to within 2 nmi and 20 ft/sec. The latter errors establish the initial condition errors for the re-entry navigation system. The former errors must be removed by the maneuvering capability of the re-entry vehicle.

The re-entry guidance system uses an inertial measurement unit which is aligned, prior to re-entry, reference to the stars. As a result of initial position and velocity errors and navigation system errors, the spacecraft can be landed with an accuracy of several miles

Re-entry guidance is performed by a stored program technique. This method stores nominal angle of attack for sub-orbital velocities and dynamic pressure profiles for super-orbital velocities, both as a function of velocity for flight path reference. Commands to seek constant dynamic pressure or angle of attack are generated. The commands are a function of range errors. Both an automatic and manual self-adaptive flight control mode of operation are provided. Pilot displays are included for energy management (points 18 to 19).

An important safety feature is the capability of prematurely terminating the mission at any point on the outward leg. For cases of equipment failure early in the boost phase, an abort procedure is used in which an aerodynamically stable portion of the spaceship is separated and allowed to fall back to earth, with the re-entry vehicle separating at the proper time before impact. For equipment failures later in flight, booster cut off and spaceship separation are prematurely initiated (if they have not already occurred) and an emergency return to earth procedure is followed. Calculations

[redacted]

of the return trajectory are based on two-body ellipse equations. Normally the calculation includes returning to specific landing points as well as observing the re-entry corridor restrictions. From some conditions it may be impossible to return to a specific landing point. Furthermore, computer malfunction may require a simplified return trajectory computation in which the corridor restrictions only are met.

2.3 INSERTION GUIDANCE

The question as to whether the APOLLO vehicle should be required to accomplish the insertion guidance is subject to discussion.

From the vehicle system point of view it is desirable that the APOLLO vehicle accomplish the insertion guidance. This is subject to consideration of other applications of the Saturn booster. Should another payload be used which does not have insertion guidance equipment, the Saturn would have to supply its own.

From the APOLLO vehicle point of view, it is desirable to avoid this interface problem with the Saturn booster.

However this may be, it appears possible for the APOLLO vehicle to carry out the in-insertion guidance. A preliminary study of the insertion problem has been carried out and is included in Appendix T-C.

In summary, a promising concept has been developed but considerable study is indicated.

2.4 MIDCOURSE AND LUNAR ORBITING

Midcourse navigation and guidance is required because of vehicle injection errors both from the earth and Moon. The purpose of this section is to discuss the favored approach and the reasons it is favored.

Both an implicit and an explicit approach were considered; the former requiring that the ship stay near a pre-established flight path with the various trajectory characteristics precomputed and stored and the latter requiring an on-board computation of the trajectory and its characteristics.

Upon considering the APOLLO mission it was decided that an explicit approach was the most compatible for the following reasons:

a. A consideration of the total trajectory information required for the implicit scheme to cover midcourse, lunar orbit and emergency return, indicated a very large quantity of stored data.

b. The possibility of large deviations from the nominal existed.

c. Velocity corrections are not truely impulsive, thus possibly requiring more than a impulsive linear differential correction approach.

A number of methods of calculating the on-board trajectory are available. These are:

The Encke Method
The Cowell Method
The Multiple Two Body Method
The Patched Conic Method
The Variation of Parameters Method

These methods are briefly discussed in Appendix T-E. A four-body Encke method was used in establishing the amount of on-board computer storage required.

A decision has not been made as to which method is preferable for the APOLLO mission.

The trajectory characteristics necessary to a space navigation and guidance scheme are the trajectory sensitivity coefficients. These are the first-order relationships between the selected independent and dependent variables between two points on the trajectory. (The first-order Taylor series expansion of the trajectory about a reference trajectory.) These serve as approximate relationships between the independent and dependent variables and are quite precise near the point about which the expansion was taken. Several methods are used to obtain these coefficients. One is to perturb the independent variables successively and compute the coefficients by differences. It is important to choose the size of perturbation such that non-linear effects and computational accuracy effects do not cause trouble.

Another method of computation is to solve the differential equations satisfied by the differential coefficients. This method is discussed in Appendix T-F.

The first scheme is now being used by the General Electric Co., however, the second appears to be more promising from the computational time point of view for APOLLO.

When near earth (after insertion) the on-board computations and data smoothing will be done in an earth-centered inertial coordinate system. When near the Moon, the on-board computations will be done in a Moon-centered coordinate system which is parallel to the earth-centered system. This change of coordinate systems removes the effect of the uncertainty of the Moon's position relative to the earth and assures utmost accuracy in obtaining the elliptic lunar orbit. If it proves to be desirable, the orbital information can be transformed to an orbital coordinate system for display purposes.

2.4.1 Data Smoothing

The APOLLO guidance and navigation system requires a knowledge of the actual position and velocity of the spacecraft. The instruments used to determine these parameters are star and planet trackers, whose inherent accuracy will not allow an accurate deterministic computation of position and velocity; hence a statistical calculation must be used. Redundant data is processed to establish an estimated position and velocity.

The purpose of this section is to discuss the proposed on-board data smoothing scheme and to present the reasons for its selection. Results from an IBM 7090 simulation of the data smoothing scheme are presented in Appendix T-G.

2.4.1.1 DATA PROCESSING

The selection of an on-board data processing scheme is based on a number of considerations.

First it must be able to operate during three of the regimes of the APOLLO mission. These are midcourse, lunar orbit, and emergency return.

Second, it must provide estimates of position, velocity, and their noise matrix, to be used to determine velocity corrections. These must be available for display at the pilot's command.

Third, it must be able to accept manually-read data (at a given time instant, insufficient to constrain position).

Fourth, it must be such that it can be handled by an on-board computer.

The maximum likelihood technique as applied to the Space navigation problem (See Appendix T-G) will meet all of the requirements listed. There is another approach which will also meet the requirements. This is the R. E. Kalman approach as applied to APOLLO by NASA Ames. These two approaches should be compared to determine which is best. The Maximum Likelihood Approach is the one which MSVD has investigated and is the subject of discussion.

A diagram of the Maximum Likelihood Approach is shown on Figure II-2-3. The entire operation is directed by a programmer which controls the sequence of events, primarily at the pilots' command.

It should be noted that the a priori insertion noise characteristics will be used in the data smoothing technique. In addition, all position and velocity estimates will reflect prior information; i.e., after insertion, the best estimate will be based on insertion noise characteristics. Estimates after a velocity correction will reflect the level of confidence already established by prior measurements. How this is handled is not shown on Figure II-2-3.

The pilot will act on the basis of information supplied by the display. Since the pilots' decisions will be based on information from the velocity correction computation, a discussion of his decisions will be covered in that section.

The Maximum Likelihood Approach as described by Figure II-2-3 establishes the most likely position and velocity at any point on the trajectory along with the confidence level of the estimate. (See Appendix T-G.) This is accomplished by establishing the most probable deviation from a point on a reference trajectory which is near the true trajectory of the spaceship. The data smoothing is handled assuming that deviations from the reference trajectory are linearly related. Although not shown on Figure II-2-3, it is possible to handle the situation in which a high degree of non-linearity is present by saving data and periodically correcting the reference trajectory, and reprocessing the data. Since the data smoothing scheme is never used over a greater than 5 hour segment.

CONFIDENTIAL

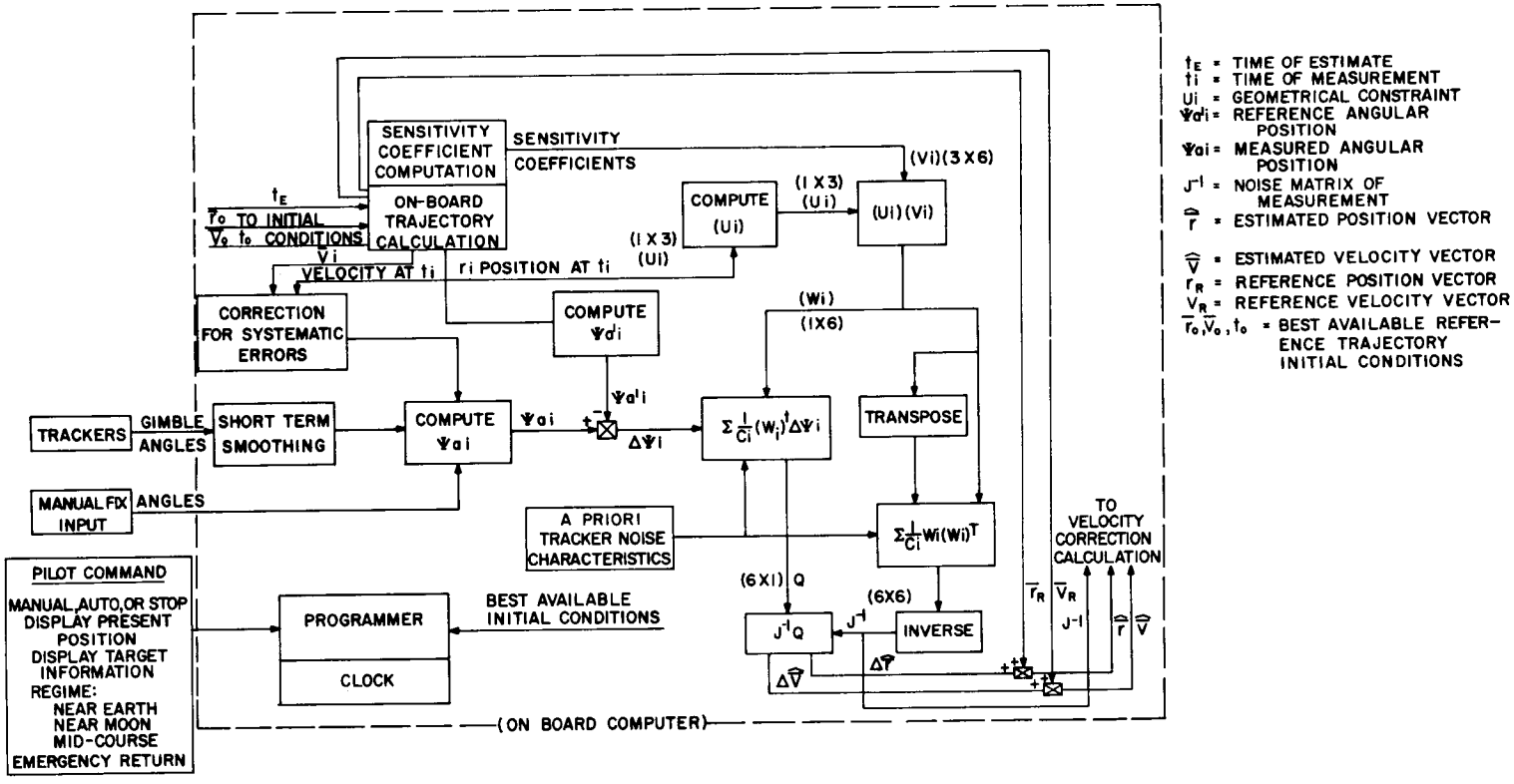


Figure II-2-3. Data smoothing block diagram

CONFIDENTIAL

of the trajectory, it is highly unlikely that non-linearity will affect it. A short study of coefficient linearity is presented in Appendix T-H.

When near earth, the coordinate system used is inertial earth centered, when near Moon, it is Moon centered. This reduces the effect of uncertainties in the Moon's position.

It is possible to incorporate into the processing scheme, as unknowns, certain selected trajectory constrants. Such action requires a sophisticated on-board trajectory calculation and increases the on-board computation time. This method is under study.

In addition, it appears that it may be possible to take into account certain bias'. This is also under study.

In the automatic tracking mode, the pilot would switch to automatic and select the regime. The programmer would then supply the location unit vector information of the bodies to be tracked. The acquisition of these bodies (not shown on Figure II-2-3) then takes place. The programmer will then initiate the tracker gimbal angle read-out at an appropriate rate. The short term smoothing indicated on Figure II-2-3 will be used, provided the trackers can be sampled at a fixed rate and very rapidly. This technique will fit a second order curve to the tracker angle data over a short period of time. The implementation of this method is very simple and is described in General Electric Co. Report R 59-ELC-35. The advantage of the short term smoothing is that a large amount of data can be accumulated in a short time under circumstances which allow a rather simple second order preliminary smoothing. It should be recognized that the output of the short term smoothing is insufficient by itself to establish an estimate of position and velocity.unless supplementary calculations are made. It accomplishes an interim improvment in the data.

The short term smoothing is sampled at time ti, and ai for this measurement is calculated. This is the sum of the true angular reading and noise. At the time $\downarrow$ ai is improvement, the systematic system errors are accounted for.

The reference trajectory position i is computed by integrating the trajectory calculation from t_o to t_i; to being the time of the reference trajectory initial conditions. ψ ai, the tracker reading from the reference trajectory, is calculated from γ i. The difference $\psi_{ai} - \psi_{ai} = \Delta\psi_i$ is the measured perturbation from the reference trajectory. The trajectory sensitivity coefficients are calculated between the time t_i and time at which it is desired that the estimate be made. These are identified as Vi on Figure II-2-3. The Maximum Likelihood estimation of $\Delta\hat{\bar{r}}$ and $\Delta\hat{\bar{v}}$ now proceeds. This involves the two summations shown on Figure II-2-3. The summations continue until it is desired that an estimate be made. When the pilot requests an estimate, the operation inverting J takes place and the product $J^{-1}Q$ is computed. This is the estimate. The J^{-1} is the (6 x 6) noise matrix of this estimate. The position Rr and velocity Vr of the reference trajectory are computed and added to the estimated Δ_r and Δ_r to produce $_r$ and $_v$ This information along with J^{-1}, the noise matrix, is now available for a miss distance computation from which the pilot can establish whether more data is required or whether a velocity correction should be made.

Manual data can be inserted in a manner similar to the automatic. It should be noted that the sensitivity coefficients will not be calculated for every observation but periodically. For instance, for the initial simulation runs, it was calculated every 6 minutes. There is every reason to believe that this time interval can be increased.

2.4.2 Trajectory Correction

Correction of the APOLLO trajectory is required either to overcome errors due to guidance and propulsion inaccuracy or to change the trajectory for mission accomplishment or emergency return.

The selection of the trajectory correction scheme is based on a number of considerations similar to those for the data processing.

First, it must be able to operate during three of the regimes of the APOLLO mission. These are midcourse, lunar orbit and emergency return.

Second, it must establish the vector velocity correction required to hit the target, along with the probability characteristics of doing so; these to be displayed to the pilot for decision at his discretion.

Third, it must be such that it can be handled by an on-board computer.

The purpose of this section is to discuss the proposed on-board correction scheme.

For midcourse, a differential correction scheme assuming eimpulsive thrusting will be used. If future studies so indicate, this approach will be modified to account for the finite thrusting times during midcourse velocity correction. This can be as high as 2 to 3 minutes under some circumstances.

The differential correction scheme assuming impulsive thrusting will most probably not be used going into, changing, and leaving the lunar orbit because of the thrusting times are quite large and the orbit quite sensitive. The method used will be a combination of precomputed data and the on-board trajectory computation. The position and velocity determination scheme is the same as the midcourse scheme. (This is currently being run with the simulation program.)

The initial conditions for the return trajectory will be stored. This information will be used as the first guess for an iterative differential correction scheme which takes into account finite thrusting times.

Emergency return will be handled in a fashion similar to lunar ejection.

2.4.3 Midcourse Detailed Discussion (See Figure II-2-3 II-2-4)

At the request of the pilot, the position and velocity estimates along with the noise matrix is made available as previously discussed. Using these initial conditions (r , F at t_E) a trajectory to the target is computed. The miss distance is then computed by comparing the target position at the desired time of arrival to the trajectory at the same time. A velocity correction is computed using a (3 x 3) matrix of sensitivity (Coefficients relating miss distance to velocity). The velocity correction is then added to the original velocity to determine if the target was in fact hit. If not, an additional correction is computed. This continues until the miss distance is acceptable. The number of inerations have proven to be small during simulation. The velocity correction (ΔV_c), velocity of arrival, and miss distance are displayed to the pilot.

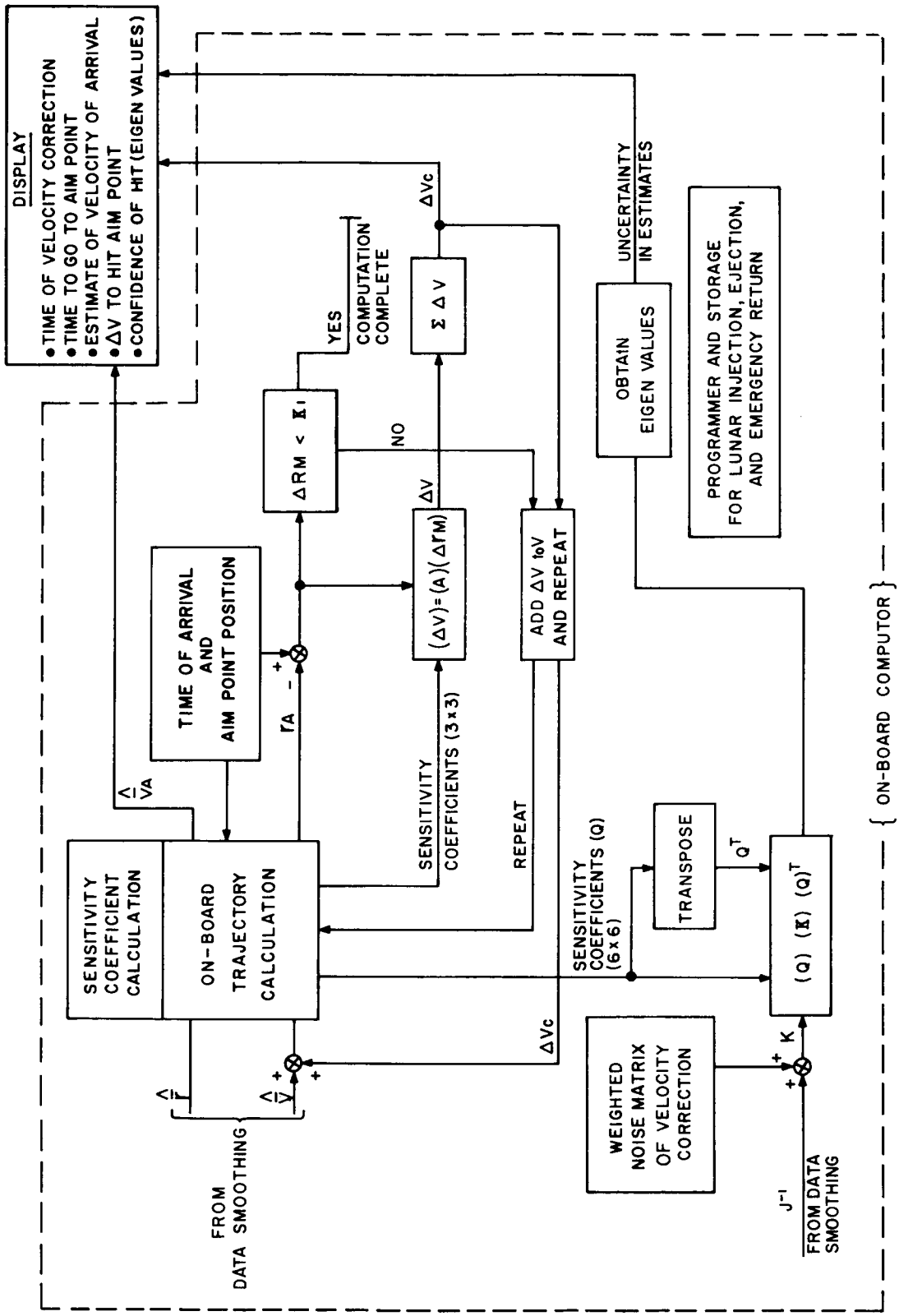


Figure II-2-4. Trajectory correction

In addition to the above, the noise matrix eigenvalues at the target are also displayed so that the pilot may establish his position uncertainty. Figure II-2-4 shows how this is computed. The pilot can then decide whether he desires to make the correction or not. His decision will be based on miss distance, magnitude of the velocity correction and position uncertainty at the target.

The criteria for the decision the pilot makes under various conditions are as follows:

A. The magnitude of the velocity correction must be K_1 times the minimum possible correction and K_2 times the uncertainty (3σ) in making the correction.

B. The target miss distance being corrected must be K_3 times the uncertainty (3σ) of the position estimate at the target.

C. The target miss distance must be greater than K_4 miles.

(Notes: The values of K_1, K_2, K_3, and K_4 will be determined from simulation runs.)

The various situations and actions are listed below. Some of the situations will probably not present themselves because of the nature of the mission.

Situation	Action
A and B and not C	No action required
B and C and not A	Choose a later correction time and recompute correction
A and C and not B	Take more data to increase estimate accuracy
A and B and C	Make correction at proper time
A and not B and not C	No action required
B and not A and not C	No action required
C and not A and not B	Take more data to increase estimate accuracy
Not A and not B and not C	No action required

2.4.5 Lunar Orbit Detailed Discussion

The lunar terminal phase of the APOLLO mission consists of the following three phases:

1. Injection of the vehicle into a suitable orbit around the Moon from the hyperbolic approach trajectory.
2. Orbiting around the Moon for a period of time up to a maximum time of seven days.
3. Ejection from the lunar orbit into an appropriate return trajectory.

2.4.5.1 INJECTION INTO LUNAR ORBIT

As the APOLLO vehicle approaches the Moon, it is essentially on a free-fall trajectory which will return the vehicle to the earth if only small midcourse guidance corrections are made. It is at this time in the mission that the astronauts on board must decide whether or not to go into orbit around the Moon. If the decision is made to go into orbit, the conservative approach will be to go into a near-circular orbit first and modify the orbit, in one or more steps, so that the final minimum altitude of 50 nmi is achieved.

The objective is to achieve an elliptical orbit with a maximum altitude above the surface of the Moon of 1000 to 2000 nmi and a minimum altitude of 50 to 100 nmi. Such orbits have periods in the range of three to five hours. The minimum altitude must be in the vicinity of the portion of the surface of the Moon which is closest to the earth. In order to minimize fuel consumption for injection into an orbit with a minimum altitude of 50 nmi, the point of closest approach to the surface of the Moon should be 1000 nmi on the farthest side of the Moon with respect to the earth. At the point of closest approach, retro-thrust is applied in order to reduce the vehicle velocity to that required for a circular orbit about the Moon with an altitude of 1000 nmi. Measurements with respect to the Moon can then be made and additional thrust applications at a selected Apolune point will appropriately reduce the velocity at that point so that the corresponding minimum altitude point will move toward the desired 50 nmi perilune. There is theoretically no fuel weight penalty associated with this incremental approach to the desired elliptical orbit. If additional requirements are placed on the orientation of the plane of the orbit with respect to the Moon, the latitude of the point of closest approach should be controlled by means of the midcourse guidance corrections and the inclination of the plane

about the radius vector at the point of closest approach can be controlled by the orientation of the injection velocity increment with a very slight fuel penalty for angles up to 10 degrees.

2.4.4.1.1 Concepts—Injection Phase

The point of closest approach to the Moon is the most efficient point for injection into an orbit around the Moon. This is true since a given velocity change results in the greatest energy change when the velocity is maximum and this occurs at the point of closest approach.

Injection into the final elliptical orbit should be made in two or more steps for the following reasons:

1. Lower energy lunar orbits can be achieved which will require less fuel than does the ultimate elliptical orbit. For instance, a circular 1000-nmi altitude orbit requires a velocity decrement of about 3000 ft/sec whereas the orbit with a 1000 nmi apolune altitude and a 50 nmi perilune altitude requires about 3600 ft/sec or 600 ft/sec more for a representative family of outbound trajectories. Thus the decision to consume the additional fuel required to go into and out of the ultimate elliptical orbit can be made after the vehicle is in a lower energy orbit around the Moon.
2. Approaching the minimum perilune altitude in steps will permit better control of perilune altitude since resulting orbital elements can be more accurately measured prior to the time at which final corrections are made.
3. If the transfer includes an initial circular orbit, the line of apsides can be arbitrarily selected if desired.
4. This is the conservative approach from the safety point of view.

If the inclination of the resulting orbit about the radius vector to the point of closest approach must be adjusted in order to meet observational or ejection requirements, this must be done at the time of injection to minimize fuel consumption. Small plane changes can be accomplished by proper retro-thrust orientation with essentially no fuel penalty for angles up to 10 degrees.

As the vehicle approaches the Moon, measurements with respect to the Moon can be made with increasing accuracy. The basic information which is required is as follows:

1. Range information, i,e., the distance between the vehicle and the center of the Moon.
2. Line-of-sight motion information, i.e., the motion in space of the line connecting the vehicle and the center of the Moon.

Chapter I of this report shows how velocity decrement requirements vary with apolune and perilune altitudes. This information is required when making the decision as to whether to make a midcourse correction or use the additional fuel which would be required when going into a specified elliptical orbit at a higher altitude because of midcourse guidance errors. In these circumstances, it may be desirable to change the perilune altitude. The minimum velocity increment requirement for a given transfer trajectory applies to the Case in which the injection altitude is 1000 nmi and the perilune altitude is 100 nmi. However, Figure I-1.152 in Chapter I of this volume indicates that the variations in velocity decrement with altitude of closest approach and perilune altitude are relatively small for the range of approach altitudes considered in this study.

Studies have shown that the simple two-body equations can be used for predicting the altitude of the point of closest approach and the corresponding time of arrival. Table II-2-I shows the error in both of these quantities when the two-body model is used in place of the n-body model for making these calculations. This also indicates that there is sufficient time left after accurate predictions become available to prepare for injection.

TABLE II-2-I

Times at which measurements are made*	Error in the altitude of the point of closest approach	Error in the time of arrival at the point of closest approach
3 hr, 2.5 hr, 2 hr	9.7 nmi	71 sec
2 hr, 1.5 hr, 1 hr	1.34 nmi	25 sec

*The times are determined by subtracting the time at which the measurement is made from the approximate time of arrival at the point of closest approach.

[redacted]

2.4.4.1.2 Guidance Concept

In accordance with the operational sequence previously discussed in this section, it is necessary to first transfer from the hyperbolic approach trajectory to a circular orbit about the Moon. This involves reducing the vehicle velocity by the order of 2300 ft/sec. The time required to accomplish this velocity change is a function of the thrust level used for injection and is of the order of 2 minutes or more. There are no critical requirements placed on the orbital elements and time is not a prime factor.

The proposed guidance scheme is explained by reference to Figure II-2-5. With finite thrusting times, the velocity correction required for injection into the circular orbit must terminate when the vehicle's altitude and vector velocity bear the proper relation. The point on the hyperbola at which thrusting begins is precalculated in order to minimize guidance errors. The initial conditions are:

Initial position: X_o, Y_o

Initial velocity: U_o, V_o

The end conditions at the point at which thrust is to be terminated are:

Final position:

$$X_1 = R \sin \Theta$$

$$Y_1 = R \cos \Theta$$

Final velocity:

$$U_1 = V \cos \Theta$$

$$V_1 = -V \sin \Theta \qquad \text{where } V = \sqrt{\frac{\mu m}{R}}$$

and μm = gravitational constant of the Moon.

The normal end point is at $\Theta = 0$. However, the angle Θ is allowed to vary in order to satisfy the circular velocity constraint. Since the vehicle is essentially at a constant

[redacted]

altitude during the thrusting period, gravity terms can be represented as follows:

$$g_x = -\frac{\mu \hat{m} x}{R^3}$$

$$g_y = -\frac{\mu \hat{m} y}{R^3}$$

The equations of motion are as follows:

$$\ddot{x} = a_x + g_x$$

$$\ddot{y} = a_y + g_y$$

where a_x and a_y are the components of acceleration resulting from thrust. Therefore,

$$\ddot{x} + \frac{\mu m x}{R^3} = a_x$$

$$\ddot{y} + \frac{\mu m u}{R^3} = a_y$$

Since the thrust and vehicle mass are essentially constant during the thrusting period, it is reasonable to assume that a_x and a_y are constant. Then, by integrating these expressions from the point at which thrust is initiated to the point at which thrust is terminated, the following results are obtained:

$$\underline{r} = \underline{i}\ R \sin \Theta + \underline{j}\ R \cos \Theta$$

$$= \left(\underline{r}_o - \underline{a} \frac{R^3}{\mu m} \right) \cos \sqrt{\frac{\mu m}{R^3}}\ t_1 + v_o \sqrt{\frac{R^3}{\mu m}} \sin \sqrt{\frac{\mu m}{R^3}}\ t_1$$

$$\underline{v} = \underline{i}\ V \cos \Theta - \underline{j}\ V \sin \Theta$$

$$= \sqrt{\frac{\mu m}{R^3}} \left(\underline{r}_o - \underline{a} \frac{R^3}{\mu m} \right) \sin \sqrt{\frac{\mu m}{R^3}}\ t_1 + \underline{v}_o \sin \sqrt{\frac{\mu m}{R^3}}\ t_1$$

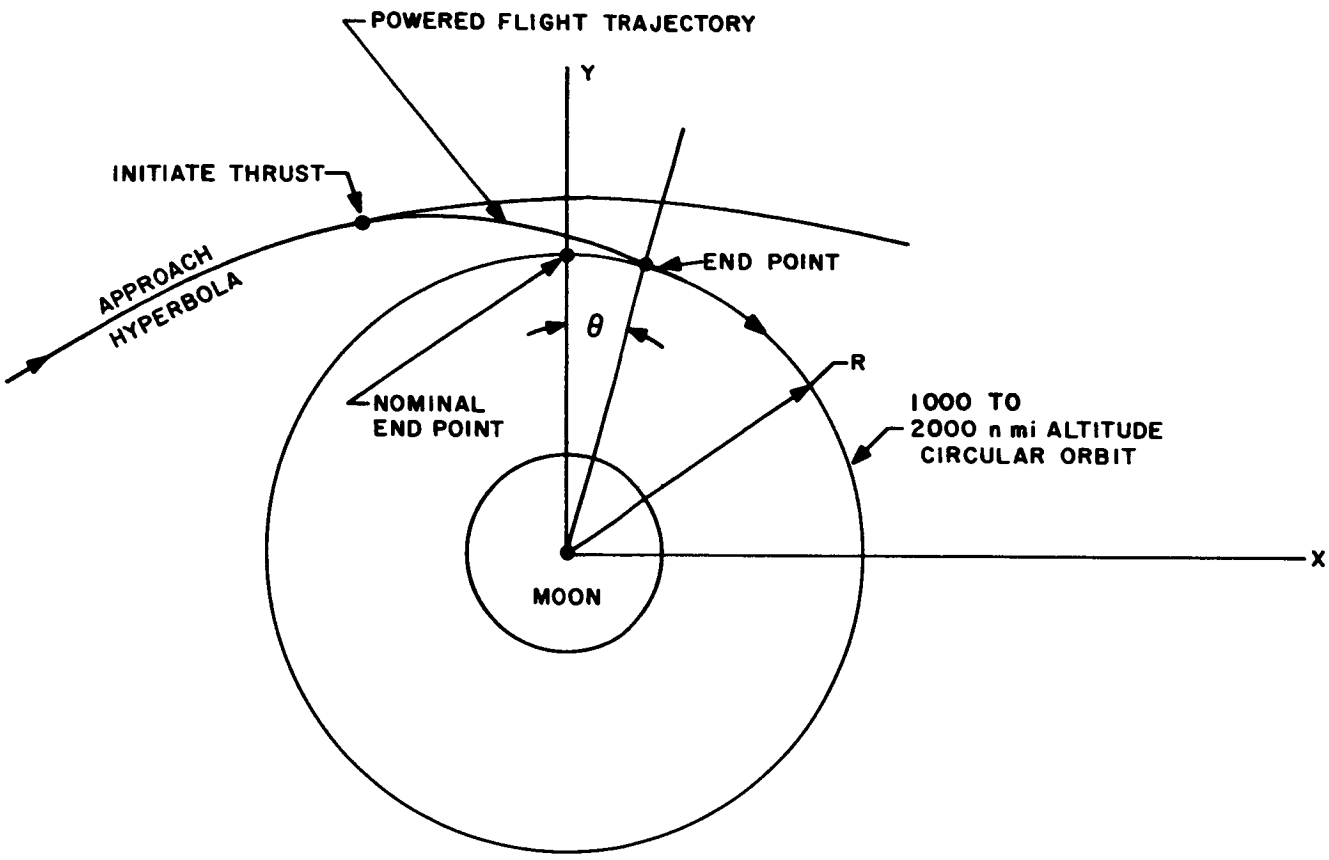


Figure II-2-5. Injection into lunar orbit

These equations can be solved prior to the actual time of initiation of thrust using the expected value of thrust level. Thus, when the starting point is reached, the required direction of the thrust vector has been calculated so that the vehicle can be propelled by the engines in the proper direction. The accelerometers on the inertial platform supply data to the computer so that the position and velocity for the present up-to-date initial conditions using measured acceleration levels and yield t_1, Θ, a_x and a_y. The steering and the duration of the thrusting is based on these results. This same guidance scheme is used when additional velocity corrections are made to achieve the final elliptical orbit with the 50 to 100 nmi perilune altitude.

2.4.4.2 ORBITING AROUND THE MOON

When retro-thrust has been applied at the point of closest approach, the vehicle enters an orbit around the Moon. The elements of the resulting orbit in space must be determined so that information will be available for making orbit modification and orbit ejection calculations.

During injection, the accelerometers on the inertial platform provide data to the computer so that velocity and position information with respect to the Moon is continuously computed. When thrust is terminated, the vehicle will move around the Moon in a Keplerian orbit except for perturbations due the earth, sun and the Moon's oblateness. The studies discussed in Appendix T-J indicate that the orbital elements of the type of nominal orbit being considered for this mission (1000 nmi apolune altitude, 50 nmi perilune altitude) are very stable with time and position in orbit. It is therefore reasonable to use the simple two-body elliptical orbit model for obtaining the approximate orbital elements based on the initial position and velocity vectors at the time of thrust termination. Since the orbit is stable, the measured position fixes obtained over a period of time can then be used to determine the mean parameters of the ellipse. When these parameters are found, position and velocity at any future instant of time can be predicted quite accurately for at least several future orbits. During the period of time when the vehicle is in orbit around the Moon, data taking and smoothing is carried out in the same manner as that used in the midcourse phases except that the earth is not included in the measurements. While in the vicinity of the Moon, the trajectory information for display purposes will be printed out in orbital components relative to the Moon-centered inertial frame.

Simulation studies now under way will provide information on the effects of measurement errors on position and velocity, and the amount of data which is required to determine the orbital parameters.

2.4.4.3 EJECTION FROM LUNAR ORBIT

When the elements of the orbit have been established, it is probable that any orbit which would be considered as nominal is sufficiently stable so that vector position and vector velocity with respect to time can be predicted accurately for at least several future orbits. Ejection time is important since the vehicle must arrive at the earth at the proper time. The velocity increment required for ejection should also be minimized in order to minimize fuel consumption. Accuracy of ejection is also important since return midcourse velocity corrections must also be minimized because of fuel consumption.

2.4.4.3.1 Studies

In order to investigate the problem of ejection from the lunar orbit, studies were initiated using a planar patched conic model for making trajectory calculations in lunar space. This study is discussed in more detail in Appendix T-W. The following results were obtained from this study:

1. Errors in the magnitude of the velocity increment of the order of ± 2.5 ft/sec at the ejection point are sufficient to result in terminal conditions which exceed the required re-entry corridor limits.
2. Errors in the direction of the velocity increment of the order of less than ± 1 degree are sufficient to result in terminal conditions which exceed the required re-entry corridor limits. This result, along with the first result listed above, shows that velocity corrections on the return midcourse leg must be used to compensate for errors introduced at the time of ejection from the lunar orbit.
3. The minimum velocity increment for safe re-entry is not necessarily compatible with the flight time required to land at a specific location on the earth. The otpimization involves an extensive linear programming problem.
4. The angle between the earth-moon line and the major axis of the earth-centered conic is quite insensitive to initial conditions. This fact can be used to ease the problem involved in returning to a specific location on the rotating earth.
5. The velocity increment required for ejection is reduced, on the average, as the time the vehicle remains in orbit increases.
6. Sensitivity coefficients obtained are close in value to sensitivity coefficients obtained from the n-body trajectory program.

Additional studies comparing the three-dimensional patched conic model with n-body trajectory results are also progressing. These studies are described in Appendix T-Y.

The results of these trajectory studies could materially influence the design of the on-board computer. If the errors resulting from the use of simplified models are of the same order of magnitude as the instrumentation errors, the complexity of the on-board

computer can be reduced with essentially no fuel penalty resulting. Even if this possibility does not materialize, there are additional possibilities for simplifying manual on-board calculations without a sacrifice in accuracy which would be great enough to prevent a safe return to the earth.

A possible targeting scheme for ejection from lunar orbit involves the use of pre-computed tables to provide approximate information on ejection requirements to meet certain constraining conditions such as flight time to a specific location on the rotating earth and maximum permissible velocity increment. These initial conditions can be manually introduced into the computer which can then calculate more precise results by means of iteration methods. This is in contrast to requiring the on-board computer to solve an extensive linear programming problem. The guidance scheme used in the orbital ejection phase is essentially the same as that described for injection into the lunar orbit.

2.5 RE-ENTRY ENERGY MANAGEMENT AND GUIDANCE

One of the objectives of the APOLLO mission is that the vehicle return to some pre-selected landing area. In order to accomplish this task within some acceptable error limit, it is necessary to have on board the vehicle an energy management system which serves to close control loops to effect range and cross range maneuvering.

Maneuvering capability is required as a result of tolerances in the range from re-entry to the landing area, tolerances in the re-entry path angle, atmospheric variations and variations in aerodynamic characteristics. The problem considered here was to investigate the range maneuvering capability for various re-entry path angles. If the vehicle re-enters with a steep path angle, the nominal range capability is reduced, whereas for shallow re-entry path angles, the nominal range capability is increased. Therefore, to achieve a single pre-selected landing area, the vehicle must be capable of extending range beyond nominal for steep angles and effecting ranges less than nominal for shallow angles.

For a single nominal landing area, the permissible re-entry path angle tolerance is a function of the vehicle's aerodynamic characteristics. That is, the capability of the vehicle must be compatible with the accuracy of the midcourse guidance system. If

this compatibility does not exist, then the landing area must be selected as a function of the re-entry path angle which, of course, implies multiple landing points.

Another point worth mentioning is that the required range from re-entry to landing area is established once the vehicle has been selected. Therefore, given a vehicle and a landing area, the midcourse guidance system must not only establish the correct re-entry angle, but must also control the re-entry position. For vehicles flying trajectories perpendicular to the earth's axis, range is controlled by varying in-orbit time of flight; whereas for trajectories which are coplaner with the earth's axis, cross range is controlled by varying time of flight.

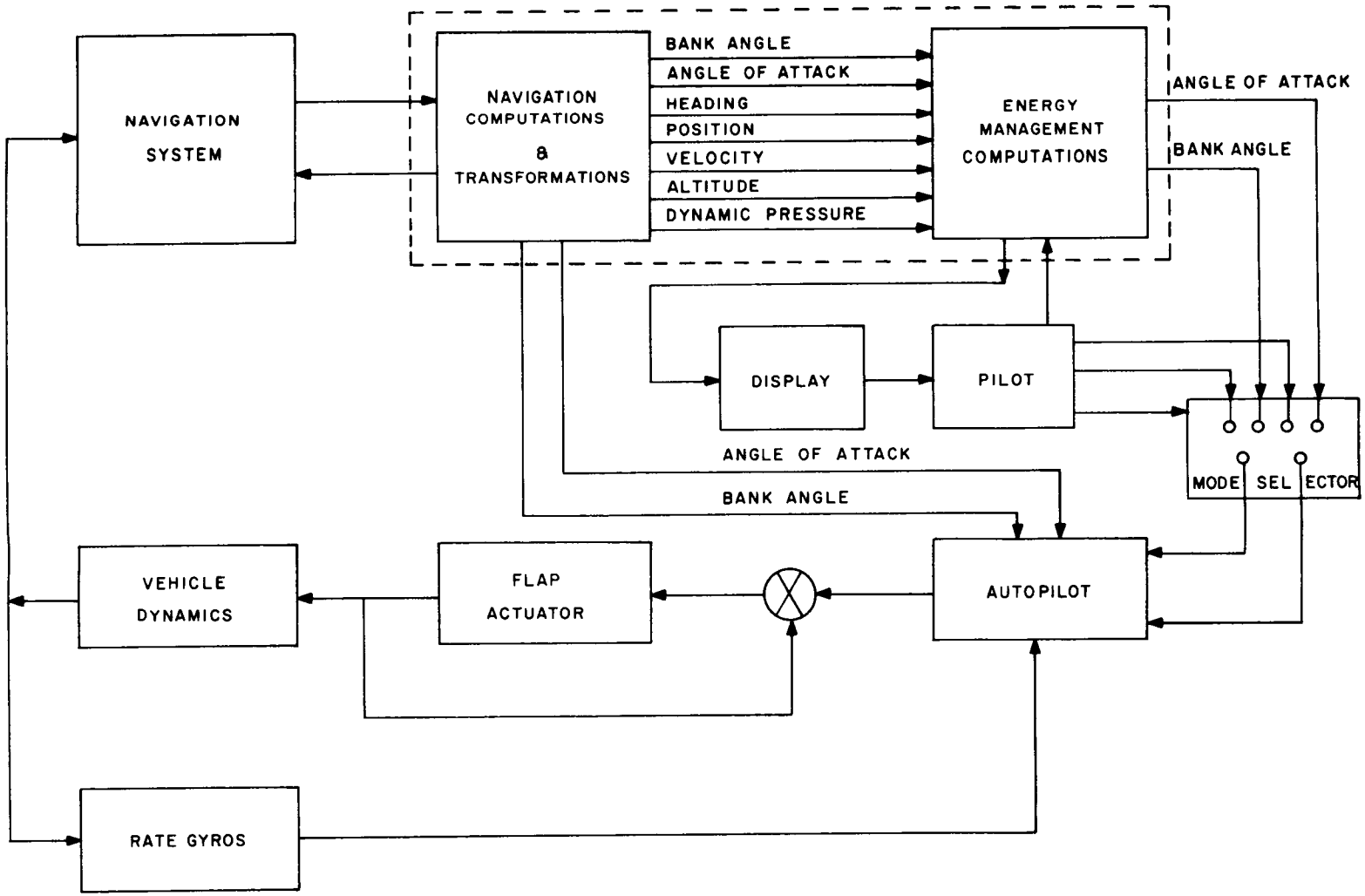
Effort has been directed towards obtaining a path control concept. The results of this investigation indicate that the attitude control loop should operate to achieve commanded levels of dynamic pressure where dynamic pressure indication is obtained from platform accelerometers.

2.5.1 Re-Entry Guidance Approach

The problem of developing guidance concepts for the APOLLO mission during operation within the earth's atmosphere was initially divided into two specific tasks. The first task was concerned with developing methods for controlling the vehicle to fly preselected flight paths which involves basically the selection of the parameter(s) used to close the feedback loop. The second task involves the determination of the maneuvering capability and the flight paths which must be flown in order to achieve maximum maneuvering capability.

The development of the energy management system for the APOLLO vehicle encompasses the combining of these two tasks into a single system wherein the range and cross range-to-go, which is obtained from the navigation system, results in the selection of the proper flight path. The selected flight path is then commanded by the pilot control or the automatic path control system. The path control commands will be accomplished through control of the angle of attack which causes variations in the lift force. Thus, except in an emergency manual mode, the commands will be sent to a flight control system (autopilot). The general form of the re-entry control system is shown in Figure II-2-6.

[redacted]



[redacted]

Figure II-2-6. Re-entry system block diagram

2.5.2 Re-Entry System Description

The block diagram of Figure II-2-6 shows the major subsystems required by the energy management system, and the flow of information between the subsystems. The navigation system makes use of the inertial platform and supporting equipment included in the midcourse guidance system. The computer is intended to be part of the digital computer used in other phases of the flight.

Since the vehicle's position and heading must be known, the navigation system is required to measure the velocity changes and vehicle attitude angles. The computer will process this information and transform the position, velocity and heading into earth coordinates as required for the energy management computations. The computer is also required to continuously compute the vehicle's bank angle, angle of attack, altitude and dynamic pressure. The energy management section of the computer will compute the appropriate angle of attack and bank angle command necessary to fly that path which will satisfy both range control and vehicle safety requirements. The computer must be digital since many of the parameters computed vary over a wide dynamic range, while demanding a high degree of accuracy such as range to the target and altitude velocity path control.

The outputs of the energy management section are of two distinct forms. One is direct attitude commands (bank angle and angle of attack) to the autopilot. These commands are channeled through the pilot-controlled mode selector which permits the pilot override capabilities, if necessary. Other outputs will be used to drive indicators and generate graphical displays. Indicators in the form of flashing lights or the like, inform the pilot of malfunctions while needle-dial-type indicators monitor the variations in the important parameters such as heating rates, g-loading, altitude rates and will indicate when corrective action must be taken by the pilot. Graphical displays presented on oscilloscopes can indicate, for example, vehicle's predicted landing point, and the various target positions with respect to this landing point. Upon this display, the vehicle's maneuvering capability is superimposed, thus permitting the pilot to select the landing site. Also, the trajectory of the vehicle in the atmosphere can be presented to the pilot via an altitude-velocity display upon which safe flight corridors are superimposed. Should the vehicle approach the limits of this corridor or exhibit any unusual

phenomena, the pilot will be aided in determining if corrective action is required to maintain the safety of the vehicle. If so, then he can over-ride the system via the mode selection switch and command the proper angle of attack/or bank angle commands.

The energy management computations for angle of attack and banking will include the necessary prediction or lead compensation and damping required for the long term path control stabilization. The autopilot will contain the compensation and damping required for the short term vehicle stabilization.

The error signal for the flap actuator is generated within the autopilot by comparing the angle of attack and bank angle commands with the actual angles as computed in the navigation computation section. The rate gyros provided attitude rate information for the autopilots.

2.5.3 Selection of Safe Flight Paths

The selection of flight paths which are considered safe for the vehicle and passengers was constrained to lie within bounds of a maximum g limit of 8 g's and the equilibrium maximum lift coefficient line.

If the vehicle trajectory goes above this line, it cannot generate sufficient negative lift to counteract the centrifugal force, and thus, it will perform a skip maneuver. The General Electric Co. study has adopted the limitation that this situation is to be avoided since the vehicle will skip out of the atmosphere and all aerodynamic control will be lost. The range traversed, and time spent out of the atmosphere are extremely sensitive to the path angle and velocity at which the vehicle leaves the atmosphere; the conditions for the next re-entry are determined solely by the conditions at the previous skip.

It was found that a low L/D vehicle ($L/D \approx 1/4$) re-entering at a relatively steep angle will exhibit large range variations for small re-entry angle variations when the skip maneuver carries the vehicle above the maximum lift coefficient line. For variations in re-entry angle about a 7.5-degree re-entry, there is approximately 900 nmi of range change per degree variation in re-entry angle. The one important fact is that the majority of this range difference is acquired while the vehicle is in the stall condition while above the maximum lift coefficient line, which means the range difference is

acquired when aerodynamic control is non-existent. Also, as the re-entry angle becomes shallower and the lift-to-drag ratio increases this difference in range becomes much larger per degree of re-entry angle variation, while the majority of the range difference is still acquired when aerodynamic control does not exist.

Unlike sub-orbital speed flight where holding a constant angle of attack results in a damped oscillation about the equilibrium glide trajectory, flight at super orbital speed requires precise, constantly varying, angle of attack control in order to fly any prescribed path. If one were to hold a constant angle of attack, the vehicle will either climb or dive depending upon initial conditions. That is, super orbital flight presents an unstable situation with respect to angle of attack control.

2.5.4 Path Control System

To achieve path control, then, it is necessary to have available information representing altitude, altitude rate, path angle, density or dynamic pressure, any one of which can be used in a control system to effect flight along some prescribed path.

Since ionization prevents the use of a radar altimeter, and also since angle of attack control presumes knowledge of the local vertical, an inertial guidance system is required and represents the only known method for obtaining any of the aforementioned parameters. If path angle is known accurately it follows that altitude rate is also accurate since the velocity computation is good. In this case, the altitude information should be pretty good, but it may have some constant offset error.

We could, thus, use an altitude command control system using altitude rate for damping wherein a nominal altitude program is followed. Range control would be obtained by commanding variations from the nominal altitude. Flying at different attitude results in a different dynamic pressure thus causing variations in the rate of change of velocity.

However, this approach has its disadvantages in that the vertical channel of an inertial guidance system is unstable, in which case the altitude rate and altitude computations may assume rather large errors. Therefore, other methods of path control should be considered.

It seems much more desirable to control path with a parameter which is measured rather than one which is computed. For example, dynamic pressure appears to be an excellent control parameter since it causes drag forces or g loading, generates lift and is a function of altitude or density and velocity. Controlling with dynamic pressure would make the system independent of altitude density variations which is quite desirable.

Because the flight path angles are quite small, an accelerometer whose input axis is horizontal and in the direction of flight will be a good indication of the deceleration due to drag, which can be expressed as the product of a constant times q C_D where q is dynamic pressure and C_D is drag coefficient. Division of the accelerometer output by C_D times a constant yields a measurement of q. The coefficient of drag can vary with angle of attack, Mach number and Reynolds number. For blunt shaped bodies, there is negligible variation in C_D with Mach number, less than 2 for the D-2 configuration, down to Mach 1. Reynolds number effects C_D only above 200,000 feet, but its effect varies also with angle of attack, to an altitude of 280,000 feet, the C_D varies by a factor of two for relatively blunt vehicles, thus it becomes necessary to store a number of C_D vs angle of attack curves, for use at the various altitudes up to maximum of about 260,000 feet. During the range control phase, the vehicle is not expected to go above this altitude. About 75 percent change in C_D can be expected over this altitude range so that storage of 5 curves, each containing a 15 percent difference in C_D results in maximum errors of 7.5 when computing q which is near the maximum dynamic pressure measurement error limit. Further studies into error allotments will have to be made before the number of stored curves is determined.

Narrow, point shaped vehicles and flared vehicles exhibit variations in C_D due to Mach number variations as well as with Reynolds numbers, but only below hypersonic velocities.

It appears then that we should control the flight path by using the accelerometer output as the controlled variable. The desired paths and path variation commands will be expressed in terms of desired accelerometer outputs and the error signal plus the rate of change of the error will control angle of attack or pitch angle. These variations in path angles are required in order to change the expected vehicle range to equal the

range to go to the target. The amount of path variation (or change in accelerometer output) is a function of both the velocity and instantaneous value of q. This information may have to be stored in the control system, via a computer.

The horizontal accelerometer referred to above is actually non-existence as such in the APOLLO re-entry vehicle since the IMU is stabilized in inertial space. Therefore, it is necessary to transform the output from the three IMU mounted accelerometers either to the local horizontal or on to the velocity vector itself.

This type of control system, without the range loop closed around it was programmed on the analog computer. Variations in paths were commanded and the system works quite well. However, at superorbital velocities, if the commands required ascending to altitudes just below the equilibrium maximum lift coefficient line, the path angle often becomes so large that even when maximum negative lift is commanded, the vehicle is committed to go outside the flight corridor, making control impossible until such time as the vehicle returns to the sensible atmosphere, which may be some considerable time later. However, for any given value of dynamic pressure, q, and velocity, there is a maximum time rate of change of q, ($\dot{q}_m$), below which proper angle of attack commands can control the vehicle within the flight corridor; but above which, maximum negative lift is not enough to maintain the vehicle in the flight corridor.

Both q and velocity and could be programmed into the control system such that when the $\dot{q}_m$ is approached, maximum negative lift would be commanded as an override regardless of any other command. A stored program system could be used where the rate of change of the accelerometer output (actually a measure of q) is limited possibly as a function of the error in q and velocity or, as a function of q and velocity. Another possibility is a predicitve control system wherein we would predict that unless the angle of attack is changed now, path control capability will be lost.

Using a q control system with the proper limits on $\dot{q}$, the range loop was closed by varying the q commanded as a function of the range error. The range error is the difference between the range to go to the target and the range the vehicle could go if it flew the stored nominal q vs. velocity trajectory. This system is capable of controlling the vehicle in a stable manner from the initial re-entry pullout point down to a velocity as low as 12,000 ft/sec, the lower limit for which the computer was scaled.

Any time below a velocity of approximately 24,000 ft/sec, the control system could be switched over to an angle of attack program vs. velocity, since each value of q in the q vs. velocity program has a corresponding value of angle of attack.

2.5.5 Modes of Operation

The re-entry energy management modes of operation are:

1. Initial re-entry to pullout.
2. Super orbital flight employing constant q flight paths.
3. Suborbital flight employing constant angle of attack, equilibrium glide flight paths.
4. Terminal flight employing constant vertical approach phase.

At initial re-entry, approximately 400,000 feet, a decision is made whether to fly positive or negative lift by rolling to either 0 to 180 degrees. The decision is based on comparison of actual path angle to stored reference nominal path angle. If actual path angle is less than nominal, 180 degree roll is command to assure capture; if path angle is greater, 0 degree roll is commanded to reduce effects of g loading and heating. The nominal angle is not necessarily in the center of the re-entry angle corridor tolerances.

At pullout, vehicle range and cross range control is initiated in order to navigate the vehicle to the desired landing area. Constant q flight paths and flown according to the range errors until below orbital velocity.

Below orbital velocity constant angle of attack equilibrium glide paths are flown for correcting range errors and maintaining the vehicle headed toward the landing area.

When equilibrium glide conditions no longer exist, terminal velocity conditions result, where the path angle is a function of L/D, or angle of attack. Varying angle of attack varies the path angle and therefore the range capabilities. Around 2000 ft/sec, the retardation devices are operated and the vehicle makes a safe descent to the desired landing area.

2.5.6 CONTROL SYSTEM OPERATION

The diagram shown in Figure II-2-7 presents in detail the operation of and the information required by the energy management system. This diagram is, for the most part, an elaboration on the operations to be performed in the boxes labeled "NAVIGATION COMPUTATIONS AND TRANSFORMATIONS" and "ENERGY MANAGEMENT COMPUTATIONS", shown in the system block diagram of Figure II-2-6. An analysis of the energy management system and supporting performance data is given in Appendix T-J.

In the following discussion, the computer subsections are defined as:

(1) Navigation Computer: That part of the computer which computes the motion of the vehicle in space from the forces acting on the vehicle.

(2) Energy Management Computer: That part of the computer which determines the range errors, and the flight paths the vehicle must fly in order to accomplish the energy management and guidance mission.

1. Summary of Information Required

   A. From the Platform:

      (1) Vehicle accelerations in inertial coordinate system from platform mounted accelerometers

      (2) Attitude angle pick offs.

      (3) Altitude rate

2. Navigation Computer Requirements

   A. Inputs:

      (1) Vehicle inertial accelerations from accelerometers on platform

      (2) Attitude (vehicle) angles - from platform pick offs.

   B. Outputs:

      (1) Total acceleration along velocity vector

      (2) Total velocity magnitude

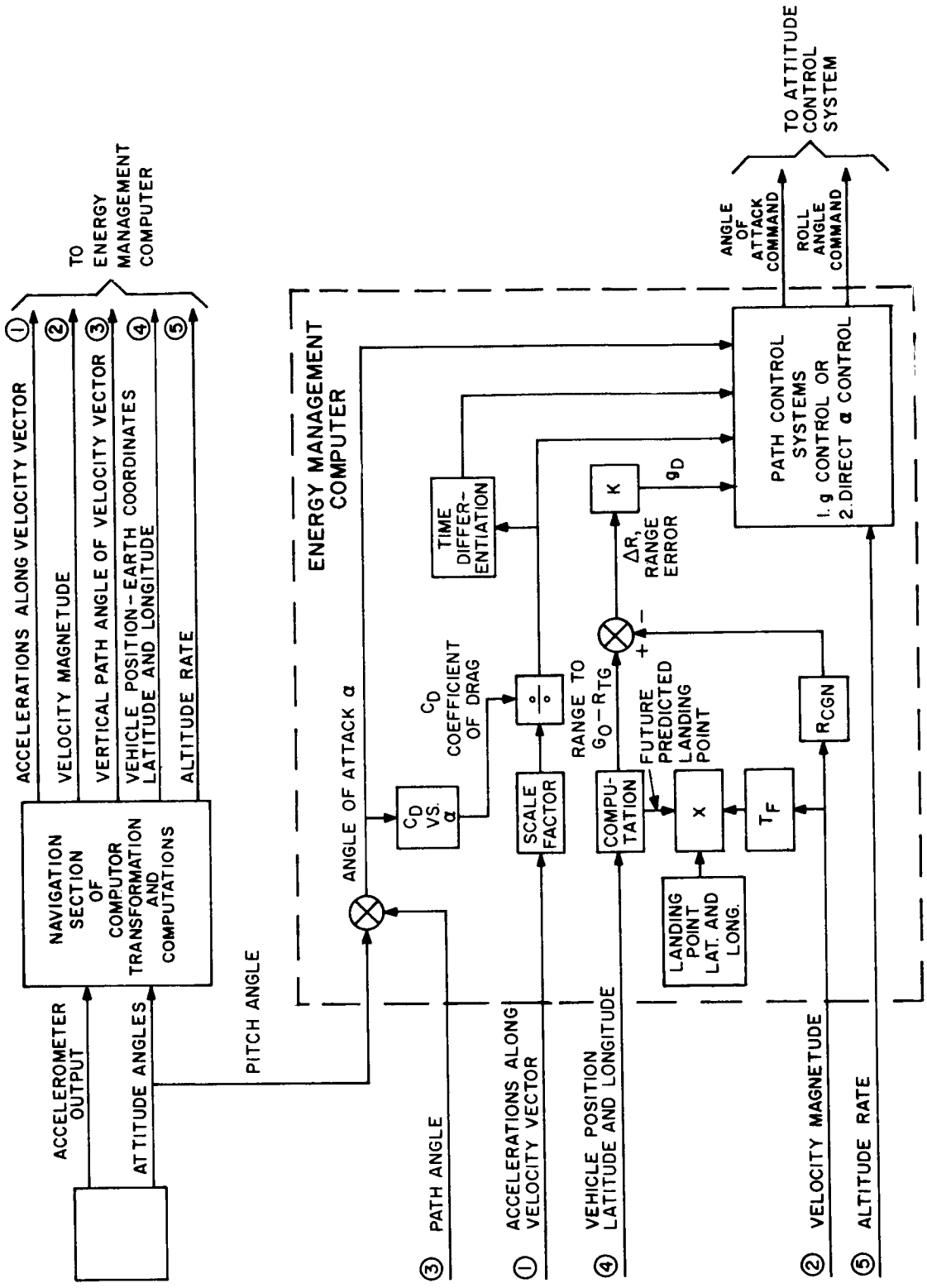


Figure II-2-7. Operation and information for the energy management system

(3) Vertical path angle (of velocity vector)

(4) Vehicle Position in earth coordinates; latitude and longitude

3. Energy Management Computer

A. Inputs:

(1) All of outputs of 2.B. above

(2) Vehicle attitude angles

2.5.6.1 Operations Performed by Energy Management Computer

From stored landing point latitude, multiplication of the landing point velocity by stored time of flight bias, (time of flight bias is a function of velocity.) The predicted landing point is obtained, and is compared with present vehicle position in latitude and longitude to determine range to go R_{TG} and cross range R_{XTG} to go to target. A stored program, as a function of velocity, yields the range the vehicle can go R_{CGN} flying a nominal flight path program.

Range error ΔR is

$$\Delta R = R_{TG} - R_{CGN}$$

Cross range error ΔR_X is

$$\Delta R_X = R_{XTG}$$

The ΔR information goes to either the super-orbital q control system, or the sub orbital direct angle of attack control system, as does ΔR_X.

Dynamic pressure measurement, q_m, comes from the division of the total accelerometer measured accelerations along the velocity vector by the coefficient of drag, C_D. The C_D is a stored function of angle of attack, α, which is in turn the difference (or sum) of vertical path angle of velocity vector and vehicle pitch attitude angle. The vertical path angle is computed in the Navigation Computer, the pitch angle is a gimbal pick-off signal. Now, q_m is compared with the product of ΔR times a gain constant (where this product is the desired dynamic pressure q_m) to yield a q difference,

Δq. q_m is differentiated with respect to time to get $\dot{q}$. The inputs $\dot{q}$, q and q_m are used by the q control system to command either roll angles, or angles of attack, depending on the magnitude of ΔR_X.

During sub-orbital flight, the Δ R is multiplied by a gain constant, and is added with the altitude rate measurement to generate angles of attack, or roll directly; once agian ΔR_X determining which is commanded. Altitude rate is computed in the Navigation Computer.

The feedbacks are through the vehicle attitude changes in angle of attack and roll, and the resultant effect on changes of the vehicles' flight trajectory.

2.6 ABORT AND EMERGENCY RETURN

The function of the Guidance and Control subsystem during abort varies depending on the phase of the mission in which the abort accurs. Prior to second stage burnout a ballistic re-entry is used so the only G & C requirement is re-entry vehicle attitude control. Up to orbital velocity the re-entry inertial navigation system is used to guide the vehicle to a landing at Ascension Island. Beyond orbital velocity the problem will be handled in much the same manner as Lunar dis-orbit. Initial conditions will be stored for the iterative determination of the vector ΔV necessary to achieve a safe re-entry at the proper location. Chapter I, the trajectory portion of this volume discusses the various trajectory characteristics.

2.7 DIGITAL COMPUTER SIMULATION OF THE APOLLO MISSION

Four considerations entered into the decision to simulate with an IBM 7090 digital computer on the APOLLO navigation and guidance problem. They were:

1. The complexity of the problem
2. The need for optimization from a weight point of view
3. The fact that success is measured only by consideration of the complete mission
4. The large number of trajectories when emergency return is considered

The programming of the simulation is currently in process at MSVD. At the time this was written, the data processing and velocity correction portions are beginning to produce data. It is expected that during the next three months (May, June, and July, 1961) a very large amount of very pertinent information will be generated. (See Below).

2.7.1 General Description

The program will consist of the following basic building block subroutines: (see Figure II-2-8, Simulation Block Diagram)

A true trajectory generation subroutine

An "on-board" trajectory and sensitivity coefficent generation subroutine

A data processing subroutine

A velocity correction subroutine

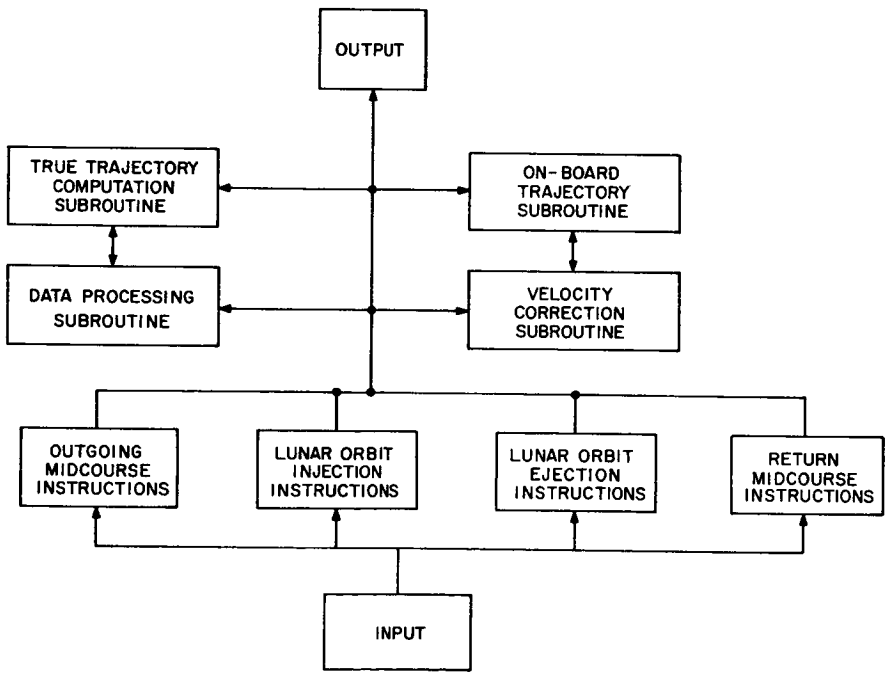


Figure II-2-8. Simulation block diagram

For the specific APOLLO mission the program will contain an instruction subroutine to accomplish the following:

Take positional data during midcourse to the Moon and accomplish velocity corrections to hit a target at or near the Moon.

Inject into lunar orbit.

Take positional data during lunar orbit and apply velocity corrections.

Determine initial conditions for return trajectory velocity correction iteration.

Establish return trajectory velocity correction and inject to earth.

Take data during return midcourse and accomplish velocity corrections to hit a target near the earth which will allow successful re-entry and landing.

Allow emergency return at the appropriate time and carry out proper navigation and guidance.

Briefly, the program is set-up to produce a true (or actual) trajectory, which the space craft is following, which can be used to judge the effectiveness of the guidance and navigation scheme. The navigation and guidance scheme consists of a data taking and processing scheme to establish an estimate of the actual spacecraft position and velocity, and a velocity correction scheme which will project the estimated trajectory (as determined by on-board trajectory computation) to a target and establish the necessary velocity correction to hit the target.

The above schemes are married by instruction logic to accomplish the mission.

The basic approach to the guidance and navigation utilize linear perturbation techniques although means are incorporated to overcome an extremely large amount of non-linearity.

2.7.2 Uses

A fuller appreciation of the simulation can be obtained by noting some of the uses it can be put to:

Establish the accuracy requirements of the various trackers used to determine spacecraft position and velocity.

Establish the tracker read-out rate and total tracking time.

Establish the bodies to be tracked at various parts of a mission.

Establish the frequency of trajectory sensitivity coefficient calculation.

Establish the statistical characteristics of the mission, i.e., establish the probability of spacecraft location at a given point.

And grossly, to establish a trade-off between tracker and data processing characteristics with the overall mission objectives.

Establish the total velocity correction energy required to accomplish the mission considering injection errors, tracker errors, and on-board computation errors. To trade-off velocity correction with simplicity of on-board navigation and guidance scheme.

Establish the limits of the Navigation and Guidance Scheme with regards to the linear perturbation technique and velocity correction energy.

Provide a practical way of proving out a navigation and guidance scheme.

Provide the information for a first approximation of the on-board computer requirements.

Provide a means of studying the effects of finite thrusting times.

Provide a means of studying the effect of thrusting accuracy on the mission.

Provide a means of determining the portions of the mission to which a man can best contribute his skill.

2.7.3 More Details on Building Block Subroutines

The true trajectory subroutine will be the basic n-body program which is currently in use at MSVD. This provides the best indication of the true trajectory. It should be recognized that this subroutine would not be part of an on-board navigation and guidance scheme and is only necessary for a simulation. The input initial conditions supplied to this subroutine are the true initial conditions.

As the mission progresses, and velocity corrections are made, the initial conditions are modified to account for the true correction made.

The on-board trajectory and sensitivity coefficient subroutine (Program) is not yet established. It is desired that the computation procedure be as simple as possible. Unfortunately, simplicity varies inversly as accuracy.

Under consideration at present are a streamlined n-body program, a multiple two body program and a patched conic program.

The streamlined n-body program is basically the MSVD n-body program previously mentioned but with portions of it removed and modified for operation during a Lunar mission only. Should this streamlined n-body program not require an unduly large on-board computer, it will be the on-board subroutine; otherwise, the less accurate patched conic will be used. The effects of its decreased accuracy will be determined from this simulation.

The multiple two body program is still in the check-out stage and will be considered seriously at a later date when it is working.

The initial conditions for the on-board trajectory subroutine is always the best estimate available. For instance, at launch, the initial conditions are the readout of the launch guidance.

As the mission progresses, and velocity corrections are made, the initial conditions are modified to account for new position and velocity estimates and estimated velocity corrections.

The data processing subroutine is a simplified Maximum Likelihood approach which on the basis of large number of tracker angle data, will determine the most likely position and velocity deviation from the reference trajectory computed on-board, based on estimated initial conditions. The noise characteristics of the various tracker errors must be known in order to produce an intelligent position and velocity estimate. The present approach assumes no correlation between data points. In the limit, as the number of samples become very large the estimated position will converge on the true position.

The data processing scheme requires trajectory sensitivity coefficients between the point the data is taken and the point at which the estimate is to be made; therefore there exists the possibility of non-linearity of coefficients affecting the estimate. This can be overcome by improving the reference trajectory initial conditions as improved position and velocity estimates become available.

The simulated data processing scheme is set-up to handle the three basic types of tracker data i.e., (1) planet-planet (2) star-planet (3) planet diameter.

A by-product of the scheme is the noise moment matrix of the estimate. This is used to decide when enough data has been taken.

The velocity correction subroutine computes the required velocity increment to hit a specified target. For input, it receives the estimate of position and velocity. It then computes a trajectory and establishes the miss distance based on constant time of arrival. A velocity correction is then computed (using sensitivity coefficients) which will reduce the miss to an acceptable value. Since the sensitivity coefficients are not truly constant, the velocity correction may not be correct. To determine if it is, the correction is added to the original velocity estimate and a new trajectory is computed. Should the miss distance not be reduced to an acceptable value a new correction is computed. This iteration continues until the miss distance is acceptable.

2.7.4 More Details on Mission Instruction Subroutines

The mission instructions cover the following phases of operation:

Injection

Outgoing Midcourse

Lunar Orbit Injection

Lunar Orbit Improvement

Return Initial Conditions

Return Injection

Return Midcourse

Emergency Return

Input for the injection phase is the initial conditions of the true trajectory and estimated initial conditions. These two differ by the inability of instrumentation to read the true conditions at injection. The true initial conditions differ from those conditions (nominal) which will result in hitting the target by the inability of injection guidance to produce this so called nominal.

Input for the midcourse phase are the data processing instructions (bodies to be tracked, tracking sequence, start time, tracking rate, number of data points, frequency of iteration, frequency of coefficient matrix change, time at which estimate is to be made and estimation time interval.)

At the completion of position and velocity estimate, the velocity correction phase projects a trajectory to the target based on constant time of arrival and determines if miss distance is large enough to worry about. If not, no action is taken. If it is, a velocity correction is computed which will reduce the miss to an acceptable value. This correction is then added to the trajectory and a new set of initial conditions at the point of correction are determined.

The above procedure is repeated until the vehicle is well within the sphere of influence of the Moon where the on-board trajectory computation is accurate, i.e., there is adequate assurance of hitting the predicted point. The input to the lunar orbit injection phase is the inclination and radius of the desired circular orbit. A velocity correction is computed which will produce this orbit. At target arrival, this correction is added to trajectory conditions to produce initial conditions for the circular lunar orbit.

Next follows a series of position and velocity estimates each followed by a velocity correction to produce the final specified elliptic lunar orbit. The major axis orientation, inclination, and perilune are input to this phase.

The input to the return initial condition computation is the orbit number it is desired to return from. The initial conditions to be used for computation of the return trajectory are then calculated.

The return injection velocity correction is then established by calculating a trajectory to the vicinity of the earth and then projecting from this point a path which represents the nominal re-entry capabilities. The miss distance from the target is then used in an iterative procedure which converges on the exact return time and velocity correction. The propulsion energy and travel time must be considered in this computation. The velocity correction is then carried out.

The return midcourse is then handled simultaneously to the outgoing midcourse.

Sample results of the data smoothing portion of the simulation may be found in Appendix T-L.

3.0 NAVIGATION AND CONTROL SYSTEM DESIGN

3.1 INTRODUCTION

This section describes the implementation and operation of a navigation and control subsystem capable of accomplishing the APOLLO lunar orbit and return mission. This system is semi-automatic in its operation. That is, certain functions which require either a large number of repetitive measurements or calculations, or which require precise control, are automatic. However the crew maintains complete control of the mission through observation of displays and with the capability of initiation and override of all N & C functions. A later section will describe an all manual backup system. The primary N&C system is entirely self-contained and requires no information from the ground. However, as further backup, provision is made for receipt and utilization of radio information from ground bases, which can be used in the event of failure of the on-board equipment.

Studies of APOLLO N&C requirements indicate that, in order to achieve the mission with close to optimum utilization of propellant, it is necessary to take large numbers of navigation fixes, and then perform some sort of data optimization process in order to minimize errors in position and velocity determination. For this reason it is our opinion that the semi-automatic system described below is necessary. It was necessary to take this approach because of the severe weight problem in the APOLLO vehicle design. However in following this approach the resulting N&C subsystem becomes relatively complicated. Thus it would be very desirable to make further studies to investigate the effects of simplifications in N&C on propulsion requirements in order to arrive at a more optimum tradeoff between weight and complexity.

The major elements of the N&C subsystem are:

1. An inertial reference used for monitoring the boost phase, for re-entry navigation and energy management, and as an inertial reference and measuring device during the application of velocity corrections during the midcourse and lunar phases.

2. Navigation sensors to make measurements of angles between earth, Moon and stars, and to measure the apparent diameter of the earth or Moon for stadiametric ranging.

3. A digital computer which processes the navigation fix data, calculates velocity corrections and performs other programming, data storage, computation and control functions.

4. An attitude control system to maintain the APOLLO vehicle at the proper attitude from Saturn separation to re-entry vehicle separation, and to make changes in vehicle attitude as required.

5. The re-entry vehicle flight control system to provide energy management and attitude stabilization and control of the re-entry vehicle from separation from APOLLO to the space craft through the landing phase.

In the course of attempting to implement the APOLLO N&C subsystem, a number of hardware approaches were investigated. The approach which we currently favor is described in this section. Two others are described in Appendices. The basic guidance concept is the same in all cases, the differences being primarily in the sensing components. Details of all of these schemes were not available in time to be integrated into the vehicle design. Thus the system pictured in vehicle design drawings is the one described as system #1 in the Project APOLLO Data Book, dated March 14, 1961, which material is repeated in Appendix T-B.

The recommended implementation approach described in this section is essentially the same as the one shown on the vehicle drawings, the only difference being the substitution of the celestial sextant for the tracker platform containing three astrotrackers. The celestial sextant is a considerably simpler device in that it requires only two precision pickoffs instead of six, and it provides its own inertial reference by locking on a pair of stars, thus eliminating the need for slaving it to the inertial platform. This allows the inertial platform to be shut down when not in use, thus decreasing its probability of failure during the mission.

Since the celestial sextant uses IR sensors for determining the vertical to the earth and Moon, it must be located external to the vehicle. This also increases the flexibility

with regard to vehicle orientation while making navigation measurements. Preliminary reliability studies indicate that the celestial sextant can be built to be a very reliable device. However some thought has also been given to the possibility of designing it in conjunction with an air-lock so it can be brought inside the vehicle in case of failure. The use of a dome to make celestial measurements also presents problems which are discussed in Appendix T-V. In this appendix a unique solution to the optical distortion problem is given. The recommended system was selected over the alternative described in Appendix Volume II only because it is based on more conventional hardware and it is therefore felt that it has a better chance of leading to qualified hardware at an early date.

The implementation approach described in Appendix Volume II is designed around the electrostatic gyro currently being developed by the General Electric Co. This approach has a number of significant advantages. The equipment is small, light weight and potentially very reliable. Since the electrostatic gyro has the dual capability of providing a precise inertial reference and an accurate angle readout, no additional precision angle readouts are required. The optical system described as part of the space sextant in Appendix II-A has the capability of making measurements to the earth, Moon and stars. Thus it is recommended that study and development of this approach also be continued as a possible future improvement. This approach will become much more attractive as experience is gained in the application of the electrostatic gyro to other programs.

3.2 SYSTEM DESCRIPTION

3.2.1 Axis Notation

Figure II-3-1 shows the location of N&C components in the D-2 APOLLO configuration. The celestial sextant is not shown but would be rigidly mounted externally near the forward end of the vehicle. The control axes are defined as +Z (yaw) parallel to the solar collector; + X (roll) rearward along the longitudinal axis of the vehicle; and + Y (pitch) perpendicular to the XZ plane to form a right handed coordinate system.

The celestial sextant axes are defined as C, the optical axis of the Polaris Star Tracker, K, the optical axis of the Kochab Star Tracker, A & B correspond to the vehicle roll and

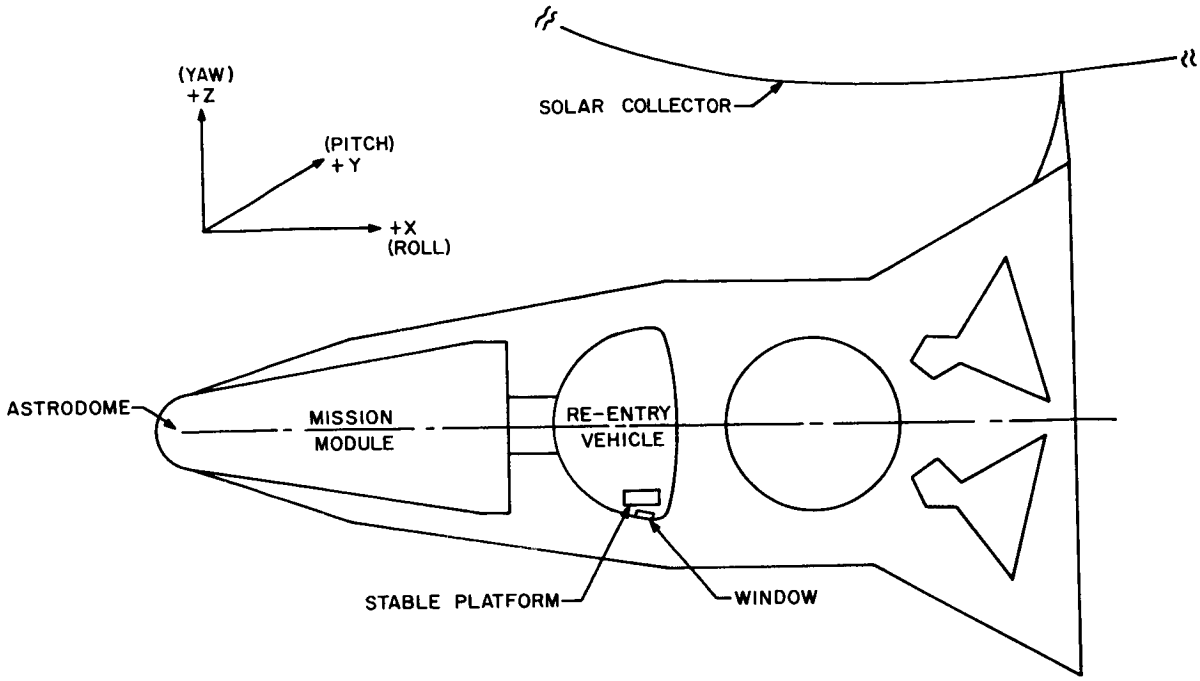


Figure II-3-1. Location of N&C components in the D-2 APOLLO

pitch axes when the tracker gimbals are zeroed (the C axis then corresponds to the vehicle yaw axis). The horizon scanner azimuth axis is D collinear with C, E is the horizon scanner elevation axis, and F the horizon scanner scan axis. These axes are shown in Figure II-3-2.

The stable platform axes are primed and lettered similarly to the celestial sextant axes to signify identical zeroed conditions. They are defined as A" outer roll, A' inner roll, B' pitch, C' yaw or Polaris optical axis, and K' Kochab optical axis. These axes are shown in Figure II-3-3.

3.2.2 Prelaunch

During prelaunch the stable platform is erected in pitch and roll by torquing the respective gyros with the pitch and roll accelerometers. The block diagram is shown in Figure II-3-4. The circuit elements will be somewhat determined by the missile motion due to wind loads. However, the accelerometers have a threshold of 0.1 cm/sec^2 and will cause the platform to level within 25 seconds of arc.

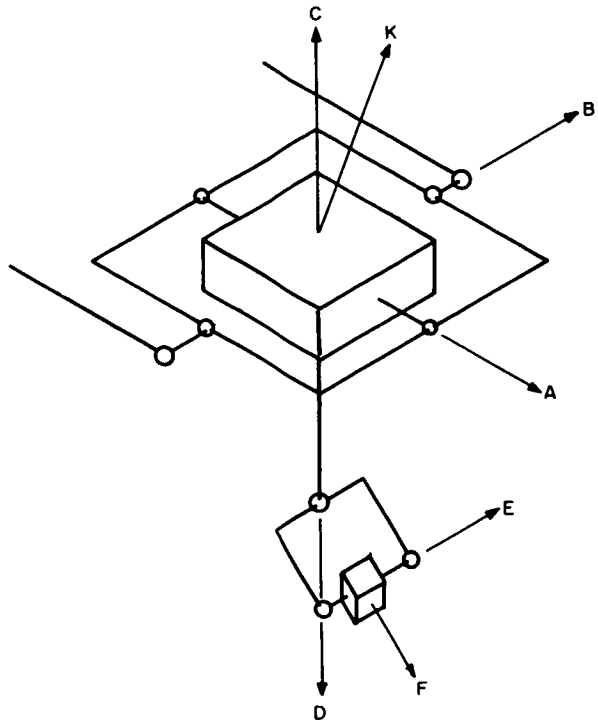


Figure II-3-2. Space sextant coordinates

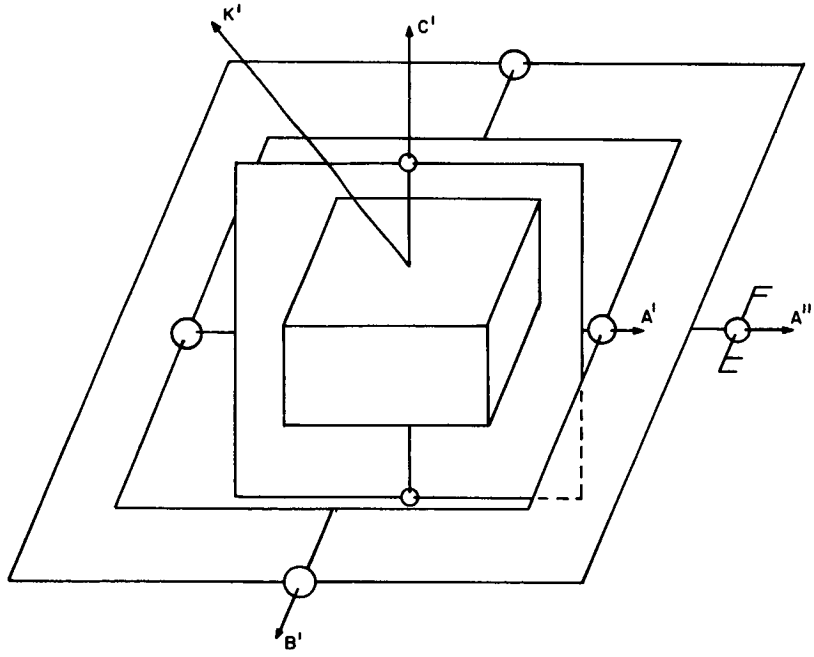


Figure II-3-3. Stable platform coordinates

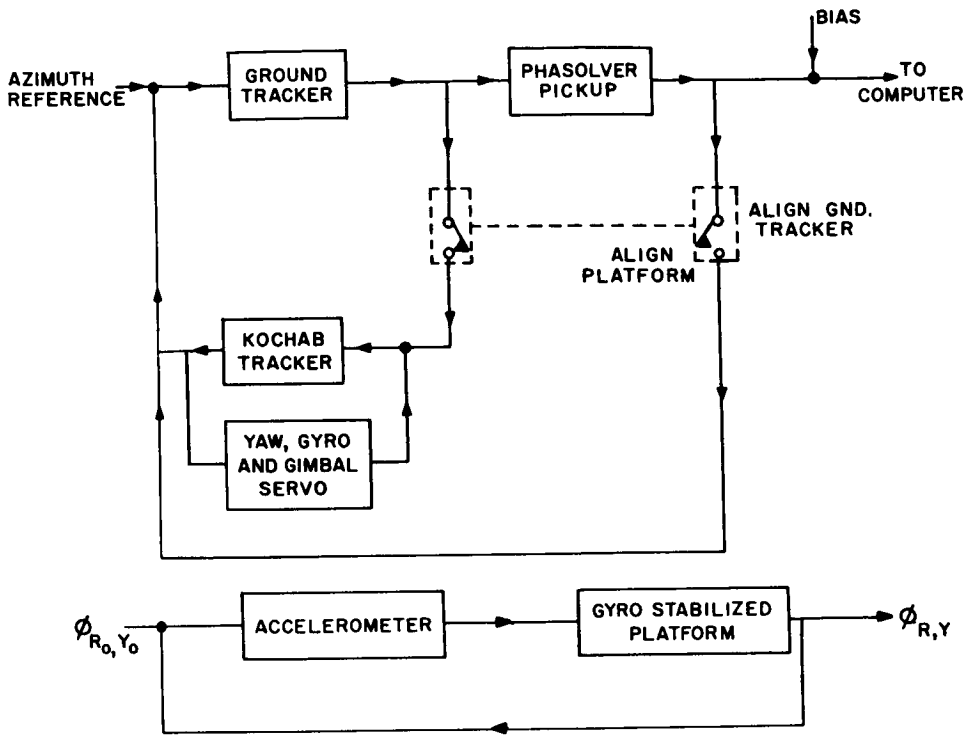


Figure II-3-4. Ground alignment

The platform will not be "zeroed" in azimuth but rather its error in azimuth from zero is measured by the ground alignment unit and digitally fed to the missile computer. Having a star tracker on the platform stable element greatly simplifies the initial platform alignment before launch, and eliminates the need for any precision internal alignment equipment inside the vehicle.

The azimuth block diagram is also shown in Figure II-3-4. The ground tracker heading, with its output phasolver nulled, is first surveyed by conventional techniques. In the platform alignment mode, the ground alignment unit then tracks the motion of the platform as the missile sways.

For purposes of initial alignment, the inertial platform stable element has a mirror with a light rigidly attached in such a manner as to direct the line of sight of star tracker K' to the ground. This unit would be attached and aligned in the factory but removed after the booster "cut off" on the Moon missions. See Figure II-3-5.

Figure II-3-6 shows the geometry of the azimuth error angle generated by the platform due to missile sway perpendicular to the azimuth axis. This angle is read directly by

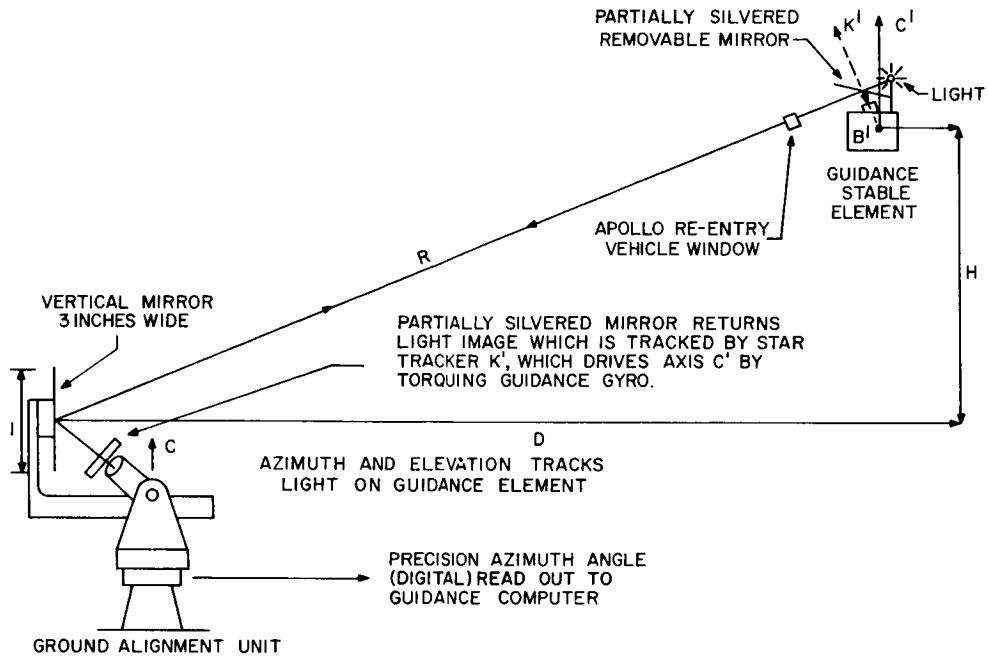


Figure II-3-5. Azimuth alignment scheme

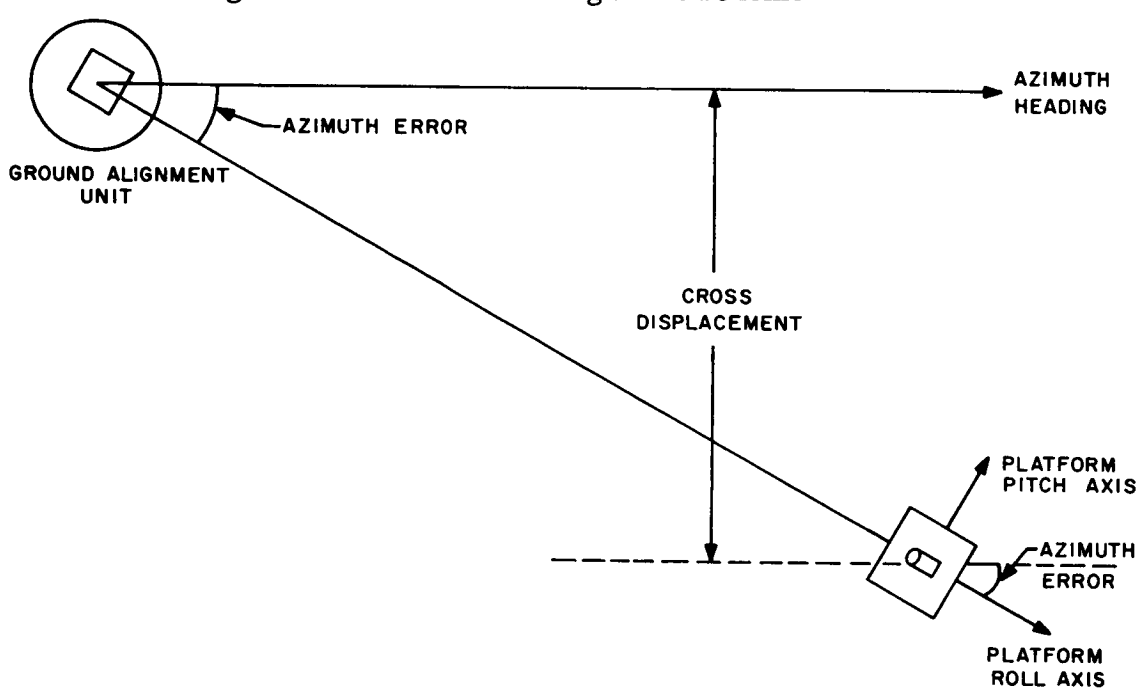


Figure II-3-6. Azimuth error due to missile sway

the ground tracker phasolver. Bias is added to correct for ground tracker misalignment from the base line and the target-to-base-line heading angle. The sum of both angles is fed to the on-board computer.

The ground tracker tilts about its elevation axis to track the image of the light in its tilting mirror when the missile sways along its azimuth axis. A mirror 12 inches by 3 inches is estimated to be adequate to keep the light source in view for a 5 foot vehicle sway at the stable platform station. It is expected that azimuth may be aligned to within 25 arc seconds using this method.

Prior to lift off, the ephemerides of Polaris and Kochab (see Appendix T-AA for further discussion of navigation star selection), the sun, earth, Moon and expected occultations are stored in the on-board computer. The appropriate boost guidance initial conditions are preset. Nominal as well as the abort and emergency escape problem are run in the computer and their answers checked. All pre-flight checks appropriate to the guidance, navigation and control systems are successfully completed.

During boost the APOLLO inertial system monitors the flight, continually calculating position and velocity to be used for the first midcourse correction and to provide initial conditions for the midcourse guidance system.

A detailed description of the inertial platform is given later in this section

3.2.3 Abort

In abort situations prior to second stage separation (velocities less than 18,000 ft/sec) the re-entry vehicle returns ballistically and no guidance is employed. Attitude Control is employed to damp separation transients and maintain a stable attitude orientation for return.

At greater velocities, up to escape, midcourse propulsion is available to allow some degree of return target selection up to complete circumnavigating the earth. The problem is similar to a normal midcourse velocity correction and re-entry.

3.2.4 Outgoing Midcourse

After separation the attitude control system automatically stops the separation angular rates and orients the vehicle to null the platform gimbal angles. The crew then orients the vehicle to the proper attitude for navigation measurements and solar collector operations as described in the following paragraphs. The attitude control system is an on-off jet system. A single loop block diagram of the system operation during this phase is shown in Figure II-3-7. The deadband can be changed from 1 degree to 5 degrees by a simple gain change in the amplifier. The attitude reference is changed for various phases of operation, as described below. Attitude references include the inertial platform, the celestial sextant and a two-axis suntracker. Further details on the design of the attitude control system are given in Appendix T-N. The attitude control propulsion is described in Volume IV, Section 6.0.

We now proceed with a description of vehicle reorientation at separation. An Astro-Navigation Functional Block Diagram is shown in Figure II-3-8. The vehicle orientation with respect to the inertial platform is displayed and compared to the computer presentation of earth orientation in the same coordinates. The vehicle is thus reoriented by manual attitude control to present its minus roll axis and consequently the celestial sextant to the earth.

The celestial sextant gimbals are uncaged. By comparison of the celestial sextant gimbal pickoffs to the ephemerides of Polaris and Kochab in vehicle coordinates, the celestial sextant is torqued manually or automatically about its pitch and roll axes until Polaris falls within the field of view of the Polaris tracker, and about its azimuth axis until Kochab falls within the field of view of its Kochab tracker. A vehicle limit cycle of + and ± 1 degree is the only significant uncertainty in the stellar ephemerides at this time. Since the trackers have a field of view of 3 degrees, they cannot fail to acquire their respective stars.

Now that an additional inertial reference has been achieved the stabilized platform can be shut down and caged. This procedure is performed to extend the reliable operating life of the platform. The vehicle inertial reference is now transferred to the celestial sextant star tracker gimbals.

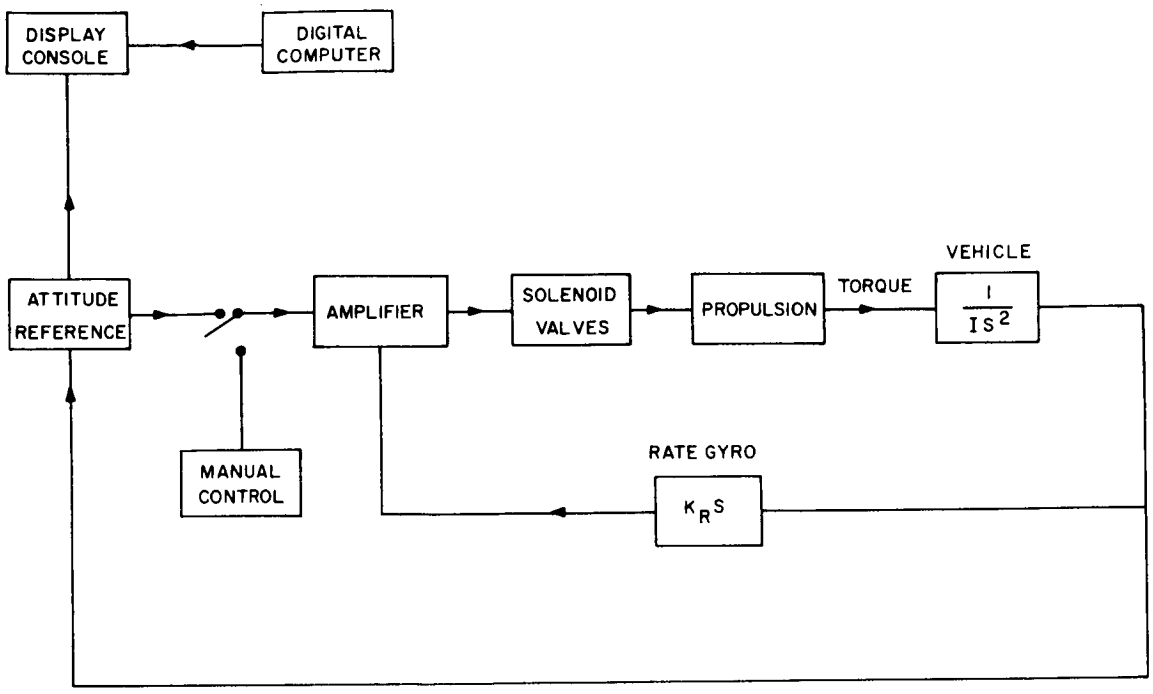


Figure II-3-7. Attitude control block diagram

The solar collector is deployed and the vehicle yaw axis is pointed towards the sun by rotating the vehicle about pitch and roll in a direction dictated by the sun azimuth and elevation information displayed from the guidance computer. Manual attitude signals can at all times be superimposed upon the autopilot reference.

At this time pitch and roll control is transferred to a two axis sun sensor mounted on the solar collector. Yaw control is maintained by the celestial sextant astrotrackers. The astrotracker gimbal angles must be resolved to provide proper signals for attitude stabilization. Attitude control in this phase is automatic.

Navigation measurements are made by measuring the direction of the verticals to the sun and Moon, and their apparent diameter at close ranges. These measurements are made by the celestial sextant which is described in detail later in this section. This device consists of a shaft which is inertially stabilized by two startrackers and a pair of IR sensors which rotate in azimuth and elevation with respect to the shaft. The IR sensors are manually controlled in azimuth and elevation to roughly align them to either the earth or Moon. The sensors then lock on and automatically track the vertical. The azimuth and elevation angles are read out by digital pickoffs and give the

data to compute the orientation of the vertical in inertial coordinates. Measurements can also be made of the apparent diameter of the body to be used for stadiometric ranging.

The horizon scanner furnishes digital azimuth and elevation of the Moon or earth along with the angular diameter, to the guidance computer. The random error is about 25 arc seconds (3σ). Bias error from the target body itself is compensated in the computer. From these six pieces of navigational information, any three may be used to establish a position fix. The selection and optimization of fix data is covered under Navigation and Guidance Concepts, Section 2.0. In flight this optimized mode is selected by the guidance computer which also smooths and processes the data to obtain velocity, predicted target miss distance and the trajectory parameters. These computations will be made in earth centered coordinates in the vicinity of the earth and in Moon centered coordinates in the vicinity of the Moon.

Two hours of data accumulation in this manner is considered typical before positional and velocity estimates improve sufficiently to warrant a velocity correction. These velocity corrections are measured directly by the platform mounted accelerometers. Before making the correction the platform is first realigned to the navigation stars.

To perform this alignment the re-entry vehicle window must be aligned to the reference stars. This can be done by nulling the celestial sextant astrotracker pitch, roll and yaw gimbals by rotating the vehicle about the sextant. The solar collector orientation to the sun is lost by this maneuver since the yaw axis now points towards Polaris.

The stable platform is now energized and erected to null its gimbal pickoffs to the vehicle attitude. The star Polaris should now be within the field of view of the platform Polaris tracker. Platform pitch and roll alignment is therefore referenced to the Polaris tracker. Once Polaris is acquired and locked on, the platform is torqued in yaw until Kochab falls in the field of view of the Kochab tracker. Yaw alignment for the platform is now referenced to the Kochab tracker. The inertial reference can be aligned in this manner to 0.01 degree.

The required velocity correction will be displayed by the guidance computer in the appropriate stellar reference frame. Once the stabilized platform has acquired the

stellar reference it will be freed, and the celestial sextant caged to the space vehicle since subsequent maneuvering for velocity correction may place the vehicle body between the sextant and one of its optical targets, causing it to lose acquisition.

The vehicle is oriented by manual attitude control to align the thrust direction of one engine along the calculated velocity-to-be-gained vector. On command from the crew the engine fires. The velocity correction is then controlled automatically by means of thrust vector control as described in Appendix T-M. Referring to Figure II-3-8, the computer uses the signals from the three accelerometers on the IMU to compute components of the velocity change normal to and along the calculated velocity-to-be-gained vector. Signals from the computer are used to control the engine gimbal angles so as to null the velocity components normal to $\overline{V}_g$. When the component along $\overline{V}_g$ equals the calculated value, the engine is cut off.

At the cessation of velocity correction, the procedure for transferring inertial reference to the celestial sextant, acquiring the sun with the solar collector and acquiring navigational fix information is the same as that following separation. A minimum of two such midcourse corrections is required.

3.2.5 Lunar Orbit

As the vehicle approaches the Moon, at about 30 lunar radii, the computer will switch to Moon centered coordinates. A greater proportion of the navigational fixes will then be taken on the Moon itself. Navigational data will be taken at this time to prepare for retro-thrust for capture by the Moon. At a lunar attitude between 1000 2000 n mi, this thrust is applied. No specific difference exists in procedure from that indicated for midcourse velocity correction.

Upon application of the velocity correction the space vehicle is in an approximately circular orbit about the Moon. Note is here taken that while circumnavigating the Moon in any except specific polar orbits, the vehicle will pass through the Moon's shadow. The sun sensors will thus lose their reference and pitch and roll vehicle control must be referenced to the celestial sextant.

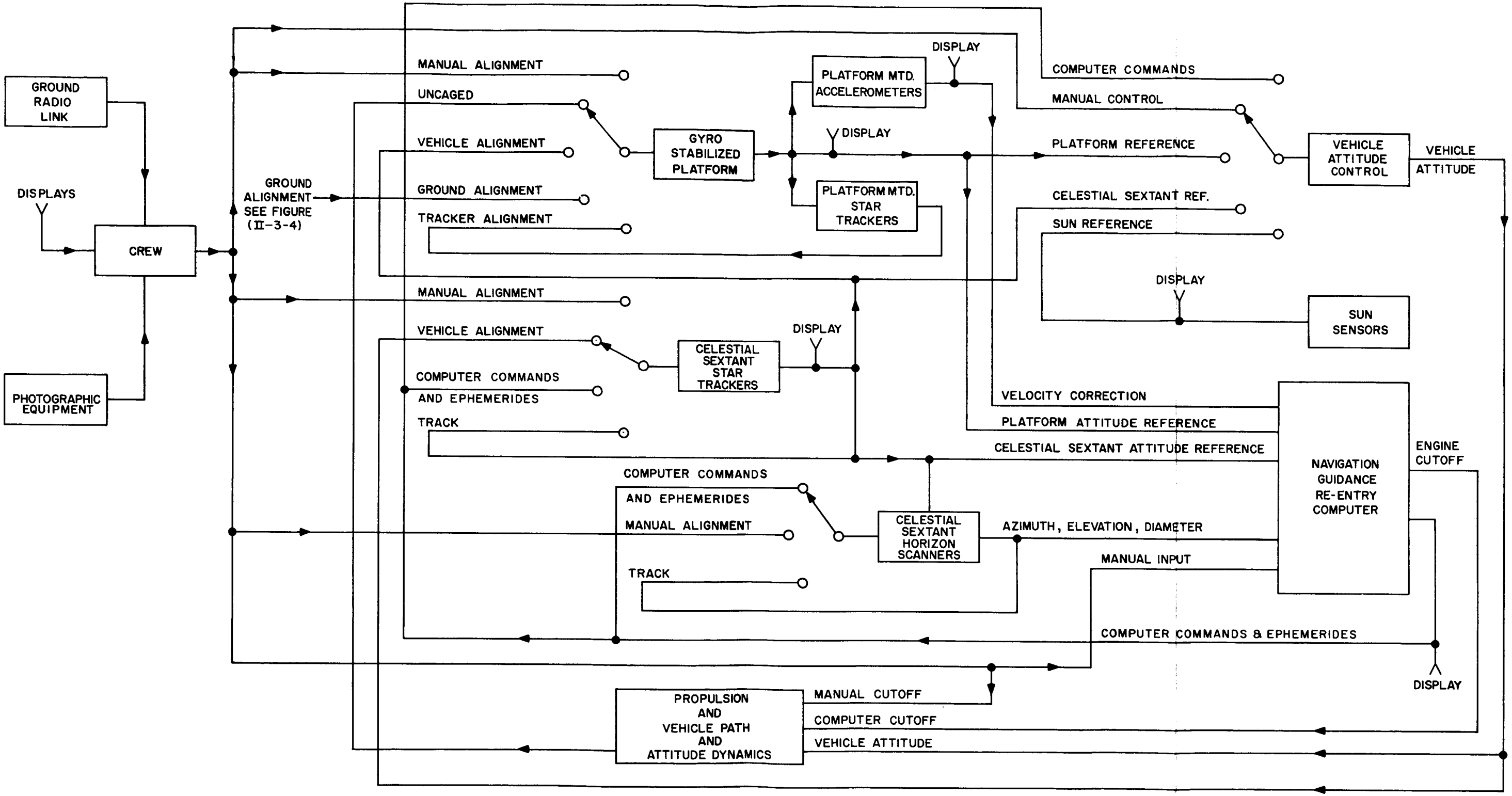


Figure II-3-8. Astro navigational functional block diagram

[redacted]

During this circular parking orbit it is expected that greatest attention will be given towards assimilating navigational information to accomplish the precision transfer to the low altitude perilune trajectory. This again is accomplished by taking fixes on the Moon with the celestial sextant.

During circumnavigation of the Moon the vehicle may lose its inertial reference due to Polaris or Kochab being occulted by the lunar surface. Approximate lunar latitudes as a function of vehicle altitude for the worst case occurrence are listed below.

Altitude	Latitude for Polaris	Latitude for Kochab
50 n mi	45.5°	29.5°
100 n mi	37°	22°
1000 n mi	-32°	-17°

It is, of course, noteworthy that the very occurrence of occultations of these and other stars can offer precise navigational fix information. These data may be taken by the crew while observing the times of their occurrence through the astrodome and manually feeding the information to the computer for processing. If the occurrences are timed to 0.1 second, the position estimate can conceivably be accurate to 500 feet.

When navigational reference to the stars is lost, attitude control signals must be obtained from the stable platform. Due to the variety of possible sequences of maneuvers in the vicinity of the Moon, a general description of occurrences is not presented. With the celestial sextant suspended sufficiently forward and away from the body of the space vehicle, navigational information can be taken continuously without violating the solar collector orientation up to the time of stellar reference occultation. At this time the inertial reference for the horizon scanners is of course lost. Conceivably, another visible stellar reference may be acquired, and fix taking resumed. Another possibility is the addition of two more star trackers to the space sextant. These would normally provide redundant information, but would allow operation to continue if one pair of stars was occulted.

After the circular orbit is sufficiently verified, the computer determines the optimum time and position along the path to retrothrust to a low altitude sweep across the lunar

[redacted]

surface. If the platform had been used to provide attitude reference prior to this time, every effort should be made to realign it to its stellar reference as near as possible to the time of making the velocity correction. The engines are aimed from the platform in the direction of the retrothrust as indicated by the computer. Due to the precision required for this maneuver it is expected that the computer will both initiate and terminate the thrust. The crew will monitor the velocity adjustment in preparation for initiating an emergency cutoff should there be a failure in the automatic equipment. The perilune altitude at this time varies 1 mile per foot per second of velocity for a 50 mile lunar perilune. The thrust level available is variable between approximately 5 and 60 ft/sec^2. Thus from 0.8 to 10 second overthrust, depending upon thrust level employed, will cause the vehicle to impact the Moon.

Once the correction is made, navigational fixes are again taken and smoothed to determine the orbital elements. The same altitude constraints are observed for the solar collector orientation and stellar acquisition for the celestial sextant. Further maneuvering will be required for scientific observation and mapping of the Moon. Optimum trajectories for emergency escape will be computed at regular intervals.

When the mission in the lunar vicinity is completed a return trajectory is determined by the computer. It is not expected that this can be accomplished for any arbitrary orbit. Rather the proper time must be taken to return to earth for specific landing sites.

The vehicle attitude is then commanded on the basis of the velocity correction direction determined by the computer. Navigating by the stable platform, the ejection is made and cutoff automatically as in previous procedures.

Navigational fixes, smoothing and several midcourse corrections are made during the return to earth. Beyond 30 lunar radii from the Moon, the computational center of coordinates is switched from the Moon to the earth.

During the period starting one or two hours from the earth final preparations are made for re-entry into the earth's atmosphere. After the final midcourse correction the stable platform is again updated to its stellar reference. Adequate time is taken

for recharging the re-entry vehicle fuel cells by the solar collector. The space vehicle is oriented perpendicular to its flight path, an attitude computed by the guidance computer, and the explosive separation process executed.

Between expulsion of the mission module and propulsion packages and entry into the sensible atmosphere, no path control capability exists. The re-entry vehicle is committed to re-enter a specific corridor. Attitude control is maintained by the re-entry vehicle attitude control system. After re-entry, control of angle of attack is achieved by means of a flap and actuator. The direction of the lift force resulting from the angle of attack is controlled by rolling the vehicle with the roll attitude jets. This system is described further in Appendix T-O.

After re-entry into the atmosphere the re-entry energy management and navigation system operates to control the vehicle so as to result in a safe re-entry and landing at the desired site. This system is described in detail in Appendix T-K. Briefly its operation is as follows:

During super-orbital velocity the stabilized platform provides attitude angles and the three path velocity components to the computer. From this, the computer determines acceleration, velocity, path angles, position and altitudes. By comparison of the computed path angle with a stored reference path angle, an angle of attack is computed and displayed by the computer. Positive or negative angle of attack is achieved by manipulation of the flap in a 0 or 180 degree roll attitude. The vehicle is controlled to the nominal path angle profile. The vehicle flies through the re-entry corridor in this manner avoiding the limit of skipping from the atmosphere and excessive g loading and heating. Vehicle range and cross range are controlled to acquire the desired landing area.

3.3 ON-BOARD DIGITAL COMPUTER

3.3.1 Introduction

Under normal operating conditions on the APOLLO mission, the crew will have complete control of the flight and will make all decisions regarding its progress. Thus under normal circumstances completely automatic operation is not required. However,

it appears that, under certain emergency conditions in which the crew is incapacitated, it will be necessary to complete the mission with automatic operation. For this reason the computer described herein has been designed to navigate and guide the APOLLO vehicle from launch to landing completely automatically. Also, in case of abort, it has the capability of guiding the vehicle to a safe landing at any of several pre-selected sites. Other functions to be performed by the computer are to provide information for various guidance displays, to make decisions regarding fuel management or the need to abort and to assist in the telemetry link. In addition, the crew must be able to manually insert data and commands into the computer.

The operation of the computer is generalized by discussing the five operational phases: preflight, launch, midcourse, lunar orbit and re-entry. The table of Figure II-3-9 enumerates the computations and control functions to be performed during each phase of the overall mission. In addition the number of words of computer memory required for each function is given. The block diagram of Figure II-3-10 indicates the organizational relationship of the computer with respect to the operations to be performed.

The initial phase of the computer operation is preflight and is combined with the launch phase since the computations are the same. Such computations as gravity corrections, position and velocity are performed during these phases. This portion of the mission requires a high solution rate for accurate determination of position and velocity. The incremental portion of the computer is ideally suited for this purpose.

At cutoff of the rocket booster, the computer automatically switches to the programs associated with midcourse guidance. The number of computer instructions and the constants used are determined by the method of guidance.

At a predetermined periapsis radius from the Moon, the computer program transfers to a routine associated with the Moon injection and orbit phase. In essence these programs will not differ substantially from those associated with midcourse guidance.

Upon completion of the lunar orbiting and lunar ejection, the computer switches to the midcourse guidance computations until the re-entry point is reached. The computer and its associated incremental portion computes latitude, longitude, path angle, heading, selection of appropriate landing site, and energy management.

[redacted]

Mission Phase	Words of Memory Required	Notes
1. PRE-LAUNCH PHASE	Unknown at this time	This is a separate computer program which will not be included in the flight program
A. Test Navigation Equipment		
B. Instrument Calibration		
C. Program & Computer Checkout		
2. LAUNCH PHASE		
A. Monitor Nav. Inst. to Determine r, v, λ, L	100	D.D.A. using 35 integrators
B. Abort Re-entry & Target Selection	300	In addition to return re-entry program which is used as sub-routine
3. MIDCOURSE GUIDANCE		
A. Command of Star-Trackers & Navigation Instruments	900	
B. Navigation Measurements Conversion to Coordinate System	1100	
C. Data Processing		
D. Computation of Velocity Correction (Encke Math Mod)	2000	
E. Computation of Differential Corrections	300	
F. Stellar Catalog	–	Part of storage in 3D
G. Command of Velocity Corrections & Vehicle Controls	100	Uses most of 6B 1, 2
H. Computation of Distance of Closest Approach & Displays	100	Uses part of 3D
I. Transition to Insertion	100	
J. Abort	250	Same as 2B. Uses part of 3D

Figure II-3-9. APOLLO computer computational requirements

[redacted]

Mission Phase	Words of Memory Required	Notes
4. INSERTION		
A. Lunar Mission	100	Alternate Mission
B. Earth Orbit		
1. Establish Crude Orbit (Powered Flight Guidance)		
2. Orbit Correct		
5. LUNAR ORBIT		
A. Command of Radar Altimeter & Attitude	350	
B. Measurement of Orbital Parameters	250	
C. Computation of Orbital Corrections	100	Uses part of 3D
D. Command Guidance While Establishing Orbit	100	In add. to 6B. 1,2
6. LUNAR EXIT		
A. Targeting		
1. Patched Conic Computation for Exit Initial Condition Using Stored Data for Initial Guess	150	Plus part of 3D
2. Iteration to Find Exit Velocity for Earth Target	50	Same as 3E plus 3D
3. Use of Patched Conic Computations for Initial Values in "N" Body Computations	–	Same as 3D
4. Iteration and Differential Corrections		Same as 3E
B. Guidance 2 body (Q Tube) Guidance		
1. Q Guidance During Powered Exit to Achieve N Body Computed Nominal	300	

Figure II-3-9. APOLLO computer computational requirements (cont.)

Mission Phase	Words of Memory Required	Notes
2. Vehicle Commands	100	Also uses D.D.A.
3. Transistion to Return Midcourse Guidance	50	
7. ESTABLISH INITIAL RE-ENTRY CONDITIONS		
A. Command Altimeters & Platform Equipment and Compute Fix	200	Uses part of 5A
B. Compute Coordinate Transformations	100	
C. Command Separation Sequence & Attitude Control	75	
8. RE-ENTRY		
A. Target Confirmation and Selection	100	These are tables (λ, L)
B. Navigation Computations to Determine h, v, λ, L, Path Angle, q	50	Load DDA
C. Energy Management (Explicit - Stored Data)	1100	
D. Landing Commands (Parachute, etc.)	100	
9. AUXILARY		
A. Displays	100	
B. Input Interrupt	25	
C. Diagnostic & Check Subroutines	300	
10. ALTERNATE MISSIONS		
A. Rendezvous in Orbit		
B. Lunar Landing	Not Determined	
C. Earth Orbits		
Total	9,050	

Figure II-3-9. APOLLO computer computational requirements (cont.)

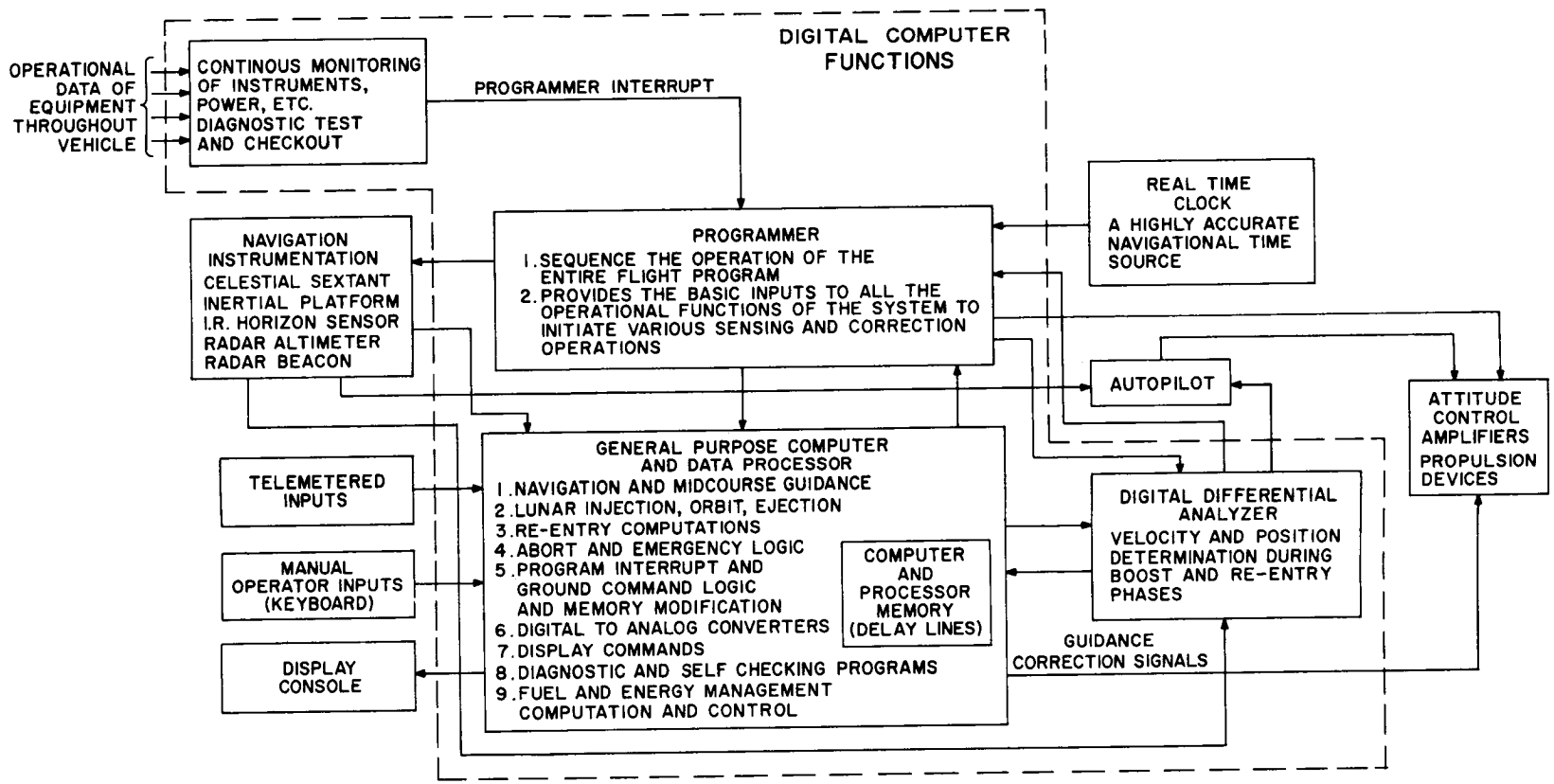


Figure II-3-10. Computer organization

The calculations during launch and re-entry involving real time integration of accelerometer outputs and trigonometric functions are best done by incremental methods. The other phases of the mission each require one or more computer programs composed of a number of program subroutines such as Runge-Kutta integration and data smoothing. This is best performed by a stored program general purpose computer (GPC). Therefore, a hybrid computer of the general purpose class with a small incremental analyzer (DDA) is definitely indicated for the APOLLO mission.

The computer selected for APOLLO is designed around a delay line memory. In the following sections the computer requirements are discussed followed by a description of a computer with a delay line memory which will fulfill these requirements. The many advantages of delay lines are discussed in detail.

An alternative computer design using a wired fixed program-core memory with a 512 word erasable core memory is also described. The unique feature of the proposed wired fixed program core memory is that only one core is needed for each instruction word.

3.3.2 Design of a Computer Using a Delay Time Memory

As discussed in the introduction, the APOLLO computer will consist of a digital differential analyzer and a general purpose computer. The general purpose computer solves problems in a manner similar to that used by an operator and desk calculator. The basic operation is to obtain two pieces of data from storage, perform a required operation on them, and transfer the result back into storage or use it as an output command. Whole numbers are transferred between the operational units, and the machine may accordingly be referred to as being of the absolute type.

The digital differential analyzer (DDA) operates through an arrangement of switching and storage elements whose behavior is analogous to that of a system governed by the equations being solved. The information transferred between operational units in a DDA consists of single increments of the variables. Therefore this type of machine is often referred to as an incremental analyzer.

3.3.2.1 COMPUTER REQUIREMENTS

The computer requirements of memory, precision and speed of operation are discussed in the following paragraphs.

3.3.2.1.1 Memory

Figure II-3-9 gives the breakdown of memory locations required. The total memory required is approximately 9,000 words. By judicious organization of instruction words the memory may be reduced to approximately 7,000 words. This may be readily accomplished by using two instructions per word.

3.3.2.1.2 Precision

Computer simulation studies of midcourse guidance indicate that if single precision computations are to be used, a computer data word length of 43 bits (41 + sign + parity check bit) is needed. Moreover because of the range of the variables in the guidance equations, floating point arithmetic is desirable. If a shorter word is used, double precision is needed in order not to introduce truncation errors. if possible, double precision operations are evidently to be avoided because their program execution is time consuming and is wasteful of valuable memory space.

A 43 bit word can be organized to contain two instructions. A data word can be organized thus:

Parity Bit	Sign Bit	Nantissa 33 bits	Exponent 8 bits

3.3.2.2 SPEED

Computer studies of midcourse guidance and re-entry energy management have shown that the computer must be able to perform between 1,200 - 1,500 multiplications per second and about 15,000 additions and logical operations per second.

With the speeds mentioned above, the solution of re-entry energy management will be sufficiently fast and corrections for midcourse navigation using single precision may be computed in approximately 30 minutes. A computation of distance of closest approach to the celestial body will take approximately 3 minutes for extreme distances.

With a computer word of 43 bits, a serial machine must have a bit rate of one megacycle to reach these computational speeds.

These speeds are very well within the capabilities of random access parallel core memories. In fact, the present speeds of core memories are considerably greater than are actually required for the APOLLO computer. It is desired to optimize the memory with respect to speed since the power consumption of the computer rises greatly with speed.

The fastest currently available serial memories of the drum and disk type operate at about 500 KC. Sonic delay line memories are the only currently available large capacity serial memories that operate at bit rates in excess of 1 Mc. Delay line memories operating at a 1 Mc bit rate in 2000 μsec length are currently available and are extremely reliable and flexible in operational requirements.

3.3.2.3 GENERAL PURPOSE DELAY LINE SERIAL COMPUTER ORGANIZATION

The block diagram of Figure II-3-11 depicts the general organization of the serial delay line computer. This diagram is a further breakdown of the general purpose computer and data processor, the digital differential analyzer, and a part of the programmer shown in Figure II-3-10.

The basic serial control of the computer is governed by the operation of the phase decoder network.

The phase decoder network divides the computer operation into four sequential phases:

1. Phase 00
   a. The address in the instruction counter is placed into the memory address register.
2. Phase 01
   a. The instruction whose address is in the memory address register is placed into the instruction register.

[redacted]

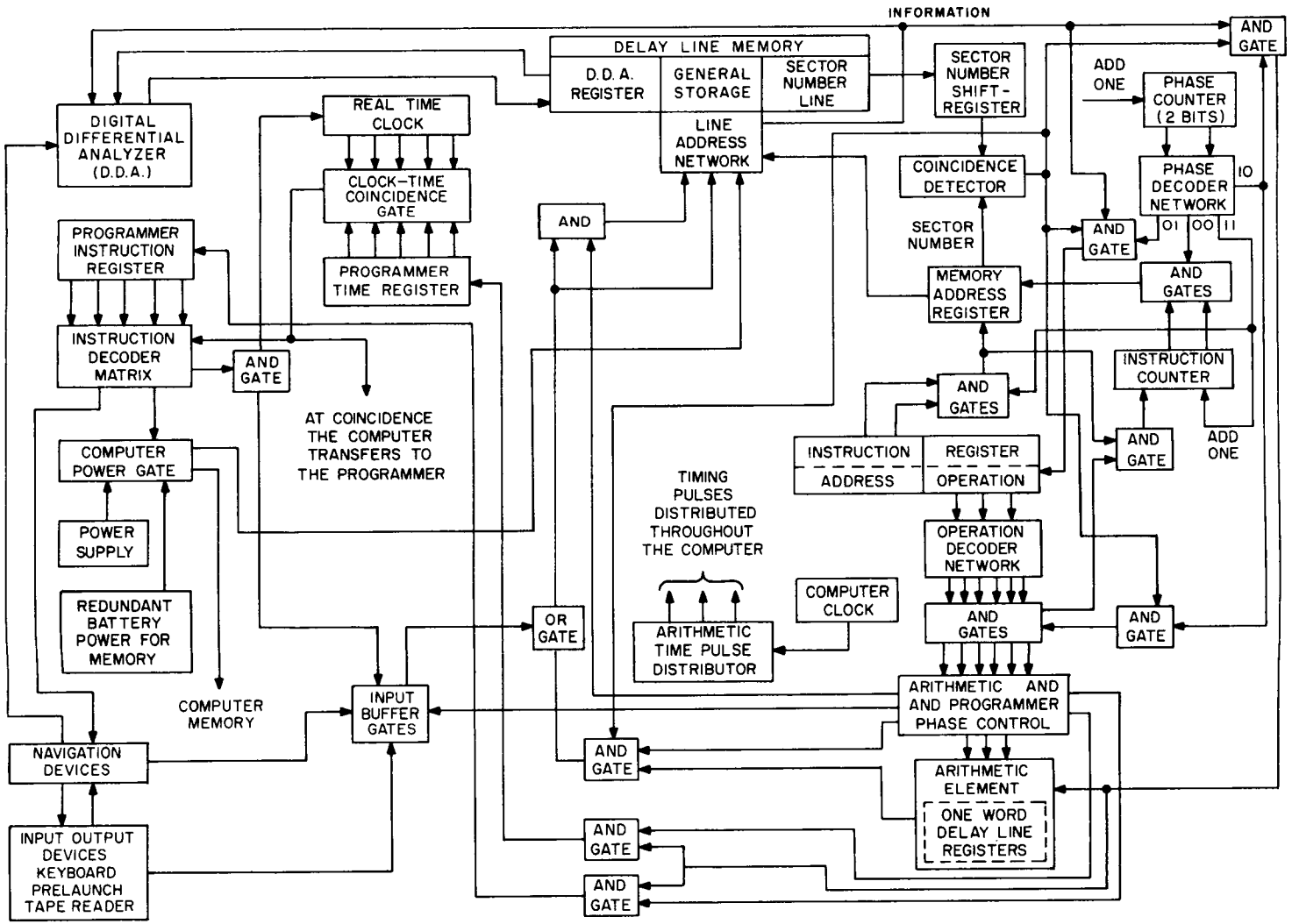


Figure II-3-11. Block diagram of computer with delay lines

[redacted]

3. Phase 10

   a. The operation specified by the operation register portion of the instruction register is performed.

4. Phase 11

   a. The instruction counter is augmented by one or replaced by the address of a transfer instruction.

Inasmuch as this is a synchronous serial computer, the timing pulses are conveniently obtained from a polyphase computer clock. The arithmetic time pulse distributor provides the necessary timing through the computer.

The functions of the various blocks shown in Figure II-3-11 are:

Phase Counter: This counter provides inputs to the Phase Decoder Network. The counter is augmented by one as each successive phase of the serial computer cycle is completed.

Phase Decoder Network: This network is a block of combinational logic which decodes the 2 bit input from the Phase Counter. Its outputs are discrete levels on one of 4 lines. The outputs control the computers through its serial sequence.

Instruction Counter: This is a register that contains the address of the current instruction being executed during phases, 00, 01, 10 of the computer operational cycle. During phase 11, the counter is augmented by one if the next instruction in the program is normal or the contents of this counter may be replaced by another address which causes the control to transfer to a new portion of the program. The address of the instruction in this counter is loaded in the Memory Address Register during phase 00 of the computer cycle.

Sector Number Line: This line is a special portion of the memory which contains information specifying the location in time of the serial contents of all of the delay lines in the memory. The information is serial binary and specifies the sector address of the words in memory just following it in time. Its output is continuously read into the Sector Number Shift Register.

Sector Number Shift Register: This register contains the address of the sector of the delay lines which is to arrive next.

Coincidence Detector: The Coincidence Detector is a serial detector which compares the sector bits of the address present in the Memory Address Register to the bits in the Sector Number Shift Register. When the numbers are equal, a signal is initiated which allows information to be gated to and from the memory.

Memory Address Register: This register consists of two parts. The first part is the delay line select field and the second part is the sector numbers select field. The first field is decoded by combinational logic and enables one of the delay lines in the memory to gate information at the proper time. The second field is the portion that is compared with the contents of the Sector Number Shift Register by the Coincidence Detector. The memory address register is loaded by the instruction counter when an instruction is being obtained from memory. After the instruction is decoded, the Memory Address Register is loaded with the address field of the Instruction register to obtain the operand (Phase 11).

Instruction Register: This register consists of two fields. One field usually contains the address of the operand while the second field contains coded information specifying the particular instructions (e.g. add, multiply etc.). The instruction register is loaded from the memory during phase 01 of the computer cycle.

Operation Decoder Network: The Operation Decoder Network is a combinational logic matrix which decodes the operation field of the instruction register. Its outputs are discrete signals which control the arithmetic and programmer phase control.

Arithmetic and Programmer Phase Control: This control element receives the outputs of the Operation Decoder Network during phase 10 of the computer cycle as its inputs. Its outputs are signals which gate the register and logic of the arithmetic element in sequence to perform the operation specified by its inputs. It also controls the programmer registers and loads the DDA registers.

Arithmetic Element: This is composed of several registers (one and two word length delay lines) and additional associated logical elements such as serial adders and subtractors. The multiplication and division operations contain separate channels in order to facilitate program optimization.

Arithmetic Time Pulse Distributor: The time pulse distributor is essentially a binary scaler with combinational logic. It has lines which are energized at each bit time and lines which are energized at definite word times. The bit and word time lines synchronize the arithmetic and programmer phase control as well as the arithmetic elements. Reshaping control pulses for the delay line drivers also are generated by the time pulse distributor.

Delay Line Memory: The memory contains general storage consisting of instructions, constants and working storage as well as the arithmetic and DDA registers.

Computer Clock: This is the master clock of the computer system. Its function is to generate accurate time pulses which are used by the time pulse distributor to synchronize the entire computer.

Real Time Clock: The real time clock is the primary mission time reference used in determining ephemeris, target location and other time bounded functions. The primary frequency source may be a gas clock if such accuracy is required or the input may be an accurately scaled division of the master computer clock providing that the master clock itself is accurate enough. If a scaled master clock is used, pulses will be obtained from the time pulse distributor. The actual register for this clock would be a delay line. (scaled to the delay line length) A scaled master clock pulse is presented together with the serial real time information to a serial adder. The sum is then reinserted in the delay line. In such a manner the delay line can be made to operate as a binary scaler up to 2,000 bits in length.

Clock Time Coincidence Gate: It compares the time from the real time clock with the time information present in the programmer time register and upon coincidence allows appropriate functions to be activated.

Programmer Time Register: This register contains the time at which the next sequential programmer event is to occur. The number present in this register is compared with the serial real time clock word.

Programmer Instruction Register: This register contains the coded programmer instruction which is to be executed at a signal from the time coincidence gate.

Instruction Decoder Matrix: This is a block of combinational logic which decodes the programmer instruction word in a manner similar to the decoding of the computer operation field by the operation decoder network.

Input Buffer Gates: These gates are the means by which the computer communicates with the input-output equipment. The gates are controlled by the computer when a number of predetermined logical conditions are met.

Input-Output Devices: The means by which the computer communicates with the outside world are implemented by the input-output devices. The initial program is placed into the memory through this unit as well as inputs that the astronauts may enter in flight. The primary results of computations which must ultimately control the vehicle are also buffered by this unit prior to being displayed.

Navigation Devices: The primary instrumentation inputs.

Digital Differential Analyzers (DDA): The DDA is in essence a separate computer which is specialized to compute various functions in real time. The action of the DDA is quite analogous to the operation of an analogue computer with the exception that the results can have digital precision. The register for the DDA are in the main delay line memory. The DDA has the capability of being addressed by the computer proper (GPC) and is capable of updating various registers with the results obtained by the GPC. Updating techniques eliminate the major source of error of DDA's.

Computer Power Gate: This gate is a solid state switch which places the redundant power supply into use in the event of a power transient from the primary supply.

Power Supply: The prime source of power for the computer.

Redundant Battery Power Supply: Since a delay line memory is a volatile memory, this system of power provides a means for insuring against the loss of information through temporary power failure or transients.

3.3.2.4 SELECTION OF ARITHMETIC UNIT

By proper selection of the arithmetic unit for a serial computer using a 2 millisecond delay line memory, the efficiency of the computer can be made comparable to a random access parallel machine. Efficiency is here defined as the ratio of time that the arithmetic unit is performing useful work to the total computer operating time.

By judicious selection of sector interlace, optimum programming techniques can be utilized. The basic principle of optimum programming for serial non-random access computers is to reduce the access time of the memory to a minimum. The future position in time of a word location can be known a-priori as well as the synchronous operation time of most of the computer operations (such as add, shift, compare, etc.). The position of the desired work can therefore be program-selected to be ready for access as soon as the arithmetic unit is ready for a new operation. The major difficulty is that for an operation such as multiplication and division to be optimum, fixed times of solution are not required. Rather, the solution rate is dependent upon the structure of the operands.

Program optimization can still be accomplished by using separate independent arithmetic elements. The advantage of independent arithmetic elements is obvious when one considers that the time for the proposed computer to perform a multiplication takes on the average 17 word times. While the variable length operations are being performed the computer can be performing other operations such as additions and other fixed word operations. It is evident that such a technique greatly increases the efficiency and solution rate of the computer.

3.3.2.5 SELECTION OF MEMORY

The hardware design of any digital computer is almost entirely determined by the choice of memory. The memory determines all the associated logic and is, generally speaking, the limiting factor on speed, power and reliability. When considering a memory which must ultimately store on the order of 400,000 bits of information, almost all discrete bit storage devices are eliminated automatically. It is fairly obvious that devices such as flip-flops, magnetic shift registers and similar devices cannot be used reasonably to store large numbers of bits. The only discrete redundant bit storage which could be considered at all for this storage requirement is coincident current core planes. Because of the relatively high power requirements and complexity of the core planes and drive circuitry associated with them, coincident core current memories have been ruled out as the primary choice of memory for the APOLLO computer. The major advantage of core storage is the speed. Analysis of the APOLLO requirements indicates the ultimate speed capability of cores is not needed.

To efficiently accommodate the amount of information required, it is necessary to go to a bulk storage type of device. The term bulk storage refers to that class of devices such as magnetic tapes, magnetic discs, and magnetic drums where individual storage devices are not needed for individual bits. Rather, a large number of bits are stored either in a magnetic film or in some other equivalent medium.

Magnetic tapes have not been considered for this application, because of their extremely low speed.

The next general class of bulk storage devices is represented by magnetic discs and magnetic drums. The major limitation of these devices stems from the fact that in these memories a single device stores the complete program. In case of mechanical failure of the drum, or disc, all storage capability is lost. It is thus obvious that a limited number of independent storage devices, when properly utilized, must offer higher potential mission reliability than a single storage device.

Magnetic discs offer several advantages over magnetic drums. Because of the physical construction of a disc, and the relative placement of the read and write heads, it is much easier to build a disc than a drum. Further advantages can be obtained by using a flexible disc and determining the disc head spacing by airflow as is done in a disc currently being produced. However the major reason for deciding not to use magnetic disc or drum storage is their relatively low reliability. Some of the potential failure modes of the magnetic disc and drum include failures in: high power drive circuitry, bearings, motors etc.

The memory chosen for the APOLLO computer is of the redundant sonic delay type. Justification for this selection follows in the description of the operation of a delay line memory.

3.3.2.6 DELAY LINE OPERATION

During the past four years sonic delay lines have progressed from a laboratory curiosity to the state where they are highly efficient devices for use not only in land-based systems but also in airborne, missile, and satellite environments. Continued development has led to lines which have very low temperature coefficients and are packaged

in such a manner that they are not vulnerable to shock and vibration environments. The simplicity of a delay line memory makes the potential reliability very high. The basic storage medium is a piece of metallic wire in which information is stored in the form of a propagating sound wave. The input and output transducers may be either magnetostrictive or piezoelectric. To date the majority of the work has been done with magnetostrictive transducers.

Several systems being designed and built by the General Electric Company utilize this type of sonic delay line. In addition to these systems a number of other systems are being built by others. At least two different satellite systems have used or are intending to use sonic delay lines in their electronic systems. An example of the commercial application of sonic delay line is found in the computer presently being produced by Packard-Bell. This system was described in December, 1960 at the Eastern Joint Computer Conference. Other computers such as Univac also use sonic delay lines for at least some functions. Sonic delay lines have been built and operated while storing information in environments which have extended as high as 3,000 cycles at 20G.

Figure II-3-12 illustrates the response of a sonic delay line as the width of the driving pulse is varied. The transducer doubly differentiates the signal. It is thus seen that when a step function is applied, a waveform approximating 1 cycle of a sine wave is received at the output. This signal has a phase which is dependent upon the polarity of the driving signal. The lower part of the figure shows the pulse shortened to an optimum length in which case a double amplitude signal is obtained at the output. When sonic delay lines are operated as digital memories the transducers are so designed that this double amplitude pulse occurs when driven with the pulse width appropriate for the recording frequency being used.

Figure II-3-13 illustrates typical waveforms which would be found in a sonic delay line storing digital information. At the top are marks which indicate clock pulse time. The second line indicates an arbitrary logic input. The third line indicates the type waveform which is actually applied to the input line transducer. It may be seen that pulses approximately 40 percent of the bit period are used. A positive pulse is used for a

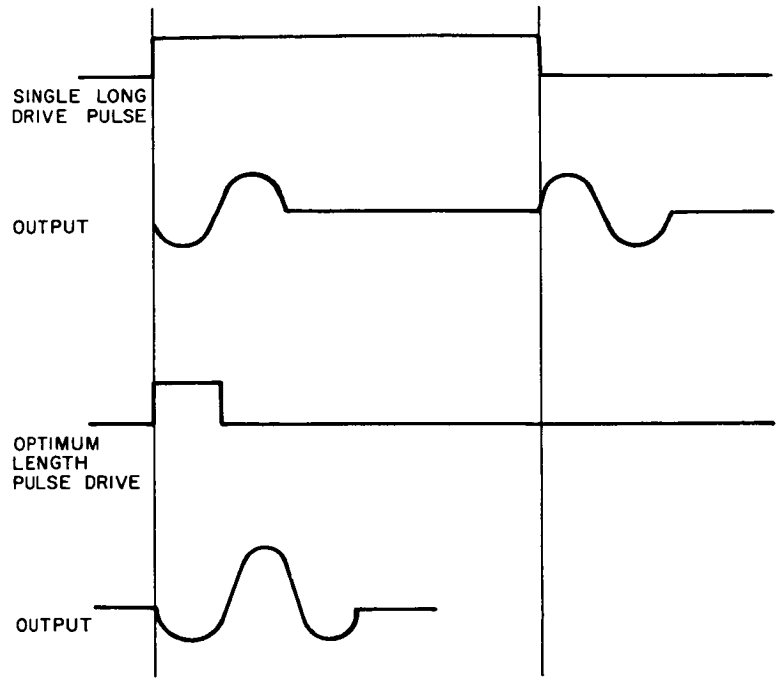
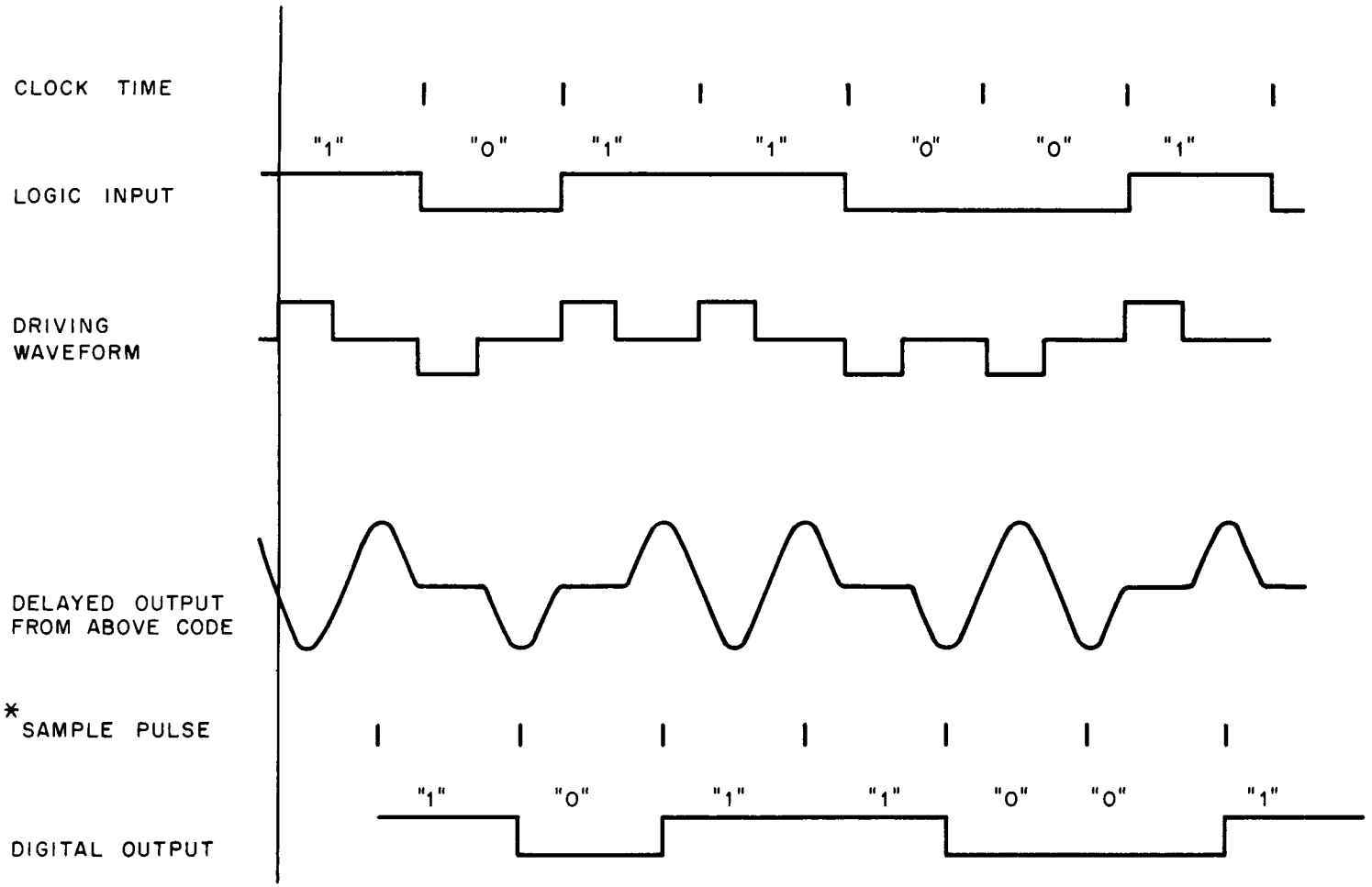


Figure II-3-12. Delay line characteristics

binary "1". A negative pulse is used for a binary "0". The leading and trailing edges are doubly differentiated as described in the previous figure. The output waveform which results is shown in the next line. When a "0" follows a "1", or the inverse of this case, small flat portions appear. This is due to a cancellation of the negative overshoot due to the "1" by the positive overshoot due to the start of a "0".

As shown, sample pulses occur at an appropriate time and a "1" or "0" is read out. In order to show the output transducer waveform in the appropriate time phase with the driving waveform, it is necessary to show a displaced sample pulse. In reality the length of the line would be adjusted such that the output waveform could be sampled at the regular clock time. This output is inserted into a flip-flop, the output of which is similar to any other flip-flop, which is synchronously set by clock pulses.

There are several other modes for driving this type of delay line, including a non-return to zero technique. It is also possible to record only "1"s as pulses and not to



*SAMPLE PULSES ARE SHOWN DELAYED 270 DEGREES TO PERMIT LINE OUTPUT TO BE SHOWN IN PHASE WITH THE DRIVING WAVEFORM. ACTUAL SAMPLE PULSE OCCUR IN PHASE WITH THE CLOCK.

Figure II-3-13. Delay line waveforms

record "0"s. Both of these modes of operation suffer from a common deficiency. It is very hard to maintain an acceptable signal-to-noise ratio. In the case of the non-return to zero technique it is necessary to have amplifiers following the output transducer with adjustable gain and to follow these amplifiers with adjustable threshold levels into a Schmidt circuit. In the case of recording only "1's, it is necessary to set a threshold and call any pulse above this threshold a "1". Unfortunately, the output amplitude may vary considerably with temperature as may the gain of the output amplifiers. This technique leads to a marginal system for other than those operated in a closely controlled temperature environment. The technique described above, however, does not suffer from these limitations, as the threshold is set at zero.

Since a one or a zero is always recorded, the noise must exceed the signal amplitude actually recorded in order to produce an error. The noise actually received consists to a large extent of reflections on the line.

It varies in amplitude with the signal, thus the signal will never drop into the noise and, as there is no threshold which must be set above noise and below the top of the signal, a very reliable system results.

The immediate apparent limitation to a delay line memory is the fact that if a power loss is suffered, even for a few microseconds the contents of the memory will be altered or even possibly totally destroyed. This limitation is very simply eliminated by the use of a small internal battery that can supply power to the delay lines in the event of a short duration power failure. It is quite simple to make this battery redundant as well as the isolation diodes which switch the redundant batteries into use in the event of a power failure. The emergency batteries are used only at loss of primary power and are automatically switched into use by the diode isolation.

The other apparent disadvantage to a delay line memory stems from the fact discussed above, that power must be continuously applied. It might appear, that if a core or drum type of memory is used instead of the delay line type, that a power saving may be realized during periods of computer idleness. Examination of the APOLLO computer requirements indicates that the computer idle time is rather small. Furthermore, any idle periods, defined as the times when the computer is not involved in primary mission calculations, would be very well spent in the running of checking routines. If it is

desired however to let the computer remain idle, the only power that would necessarily be consumed would be by the memory and real time clock. If a comparable drum memory is used, the total watt-hours consumed by the computer would tend to be greater rather than less than that used by an equivalent delay line memory.

3.3.3 Reliability

Navigation digital computers with drum memories used in commercial aircraft have been built with a mean time to failure of over 500 hours. For the APOLLO mission, the digital computer should have a mean time to failure of at least 10,000 hours to achieve a reliability of 0.97. Redundant logic design techniques have been developed at the General Electric Company for space application on Signal Corps contracts. Using these techniques, improvements factors of 100 to 1000 or more are feasible with present day components. This type of design coupled with the use of high reliability components such as those required for the Minuteman missile insure a high confidence level for proper performance of equipment.

Redundant logic design uses majority logic to vote on the various outputs of duplicated computers or elements. The majority connections are made in numerous places throughout the duplicated computers. Techniques have been developed allowing the majority elements themselves to be made redundant. Proper system design allows the reliability of the total system to be almost entirely dependent upon the failure rates of the majority elements which can be made quite low.

The technique advocated here triplicates the basic computer and uses 2 out of 3 majority elements. The maximum number of independent failures that may be sustained without system failure is equal to the number of elements into which the system is divided and voted upon with a majority element.

It is thus apparent that to use as many majority elements as possible results in the most reliable system. The limiting factor below which further division is fruitless arises when the failure rate of the sections being voted upon approaches the failure rate of the majority element itself.

The majority technique described above is a special member of a class of redundancy techniques which are known as passive. In a passive redundant system the redundant

elements always remain in the system physically and, in the case of an electronic system, electrically. In an active redundant system, the redundant elements while usually present physically, are not, in the case of an electronic system, electrically placed in the circuit until some mechanism detects that a failure has occured and that a repair is necessary.

Major limitations to active redundancy systems are the complexity of the failure detection circuitry and the fact that the system operation is affected or stopped until the failure has repaired. For the APOLLO mission, an interruption of computer service for even a few minutes is undesirable especially during high speed operations such as required by re-entry. Active systems by themselves are not recommended for the APOLLO mission. However, a hybrid redundant system could ultimately offer an ultra-reliable system. The system would be active in the sense that failures in redundant elements are actively sensed by special purpose checking elements or test programs.

Defective elements are replaced by spares (The computer will consist of essentially many similar modules of which a limited number of each element are carried as spares.). Computer operation is not affected or interupted by replacement because of the passive redundancy logic design.

For manned flights, one of the astronauts would monitor failures and physically make the replacement. If the crew is unable to make repairs due to severe physical conditions such as imposed by launch or re-entry, the passive system is the prime redundant mode.

3.3.4 Alternative Design-Computer Program Using Wired Proqram

A part of the following material on the wire core program and memory was taken from MIT Report R-23, "A Recoverable Interplanetary Space Probe."

3.3.4.1 GENERAL

Control of a digital computer can be accomplished by one of two methods:

Control of the computer arithmetic through a stored program.

Control of the computer arithmetic through a wired-in form of program.

The stored program method uses stored numbers which are coded to indicate the operations desired by the computer. Under the control of a sequencing device, the coded instructions are selected, decoded, and outputs are developed for use in the internal logic circuitry of the computer. The stored-program method of computer control is flexible inasmuch as computer operation can be readily changed by modification of the stored numbers or words whereas in a wired program, this cannot be done.

For this reason, the stored-program method is normally used in general-purpose-type machines where maximum operational flexibility is a prime requisite.

For more specialized applications, where flexibility is secondary and simplicity of hardware is of prime importance, a wired-in type of program control is usually employed. Examples of applications of this type are missile guidance and launch computers. This type of program control is also applicable for the APOLLO guidance computer. The wired-in type of program control is simple and does not require as much hardware as the stored-program method. Its main disadvantage is that it is not readily changeable. In cases where the functional operations are completely known and not subject to constant modification, then the wired program control results in considerable savings in hardware. However, additional programmed functions can be incorporated by merely adding additional core elements.

A wired program control offers considerable savings in hardware and is more amenable to redundancy techniques. Redundancy can be applied without excessively increasing hardware complexity. In addition, the basic program information is carried by physical wiring and is not susceptible to circuit parameter changes.

3.3.4.2 MAGNETIC MATRIX AND CORE-WIRED PROGRAM CONTROL

The wired-program technique uses a magnetic-core array for both the selection of the program step and the generation of an instruction code.

The proposed wire-program technique is a form of a linear selection switch and is also known as the Diamond switch or Rajchman selection switch. The technique makes use of magnetic cores as saturable pulse transformers. The cores are small metallic ribbon cores having a fairly square hysteresis loop.

The operation of the device can be illustrated by referring to the magnetic core-wired program shown in Figure II-3-14. The program step in which the computer is operating is indicated by a program counter consisting of transistor flip-flop stages. The numerical value in the counter is advanced by one count for every program count pulse received.

Outputs from the flip-flops are indicated by horizontal lines from each counter stage. Both the "0" and "1" outputs from the flip-flops are used to control the cores. The slanted mark at the intersection of the line and core indicates that a particular drive line threads that core. The direction of the slant indicates the direction of threading.

To generate an output, the core must be set or triggered. Initially, all cores are at a point on their hysteresis loops representing a "0". Initiation of the program transmits a current pulse over the set drive line. The set drive line, which threads all cores, attempts to drive all cores to a "1". Simultaneously with the application of the set drive current, an inhibit pulse (I) is supplied to the power NOR gates which have been previously conditioned by the output lines from the program counter. Depending upon the number that is in the program counter, certain inhibit drive lines from each counter will be activated. As an example, if the counter contains all "0"s, indicating program step zero, the bottom line from each counter stage is impulsed. The inhibit windings on each core are wound in opposition to the set winding, and an application of inhibit pulses on these lines will prevent all cores but one from being set to a "1". Examining the inhibit lines threading the cores, and given the counter condition as stated (all stages at "0"), it is evident that the core labeled "0" will be the only one set to the "1" condition. An increment of time later (less than a clock pulse interval), the reset drive line is pulsed, and the core previously set to a "1" is reset to "0". Now all cores are in readiness for selection of the next program step.

Thus far, only how a single core is selected during a given program step has been considered. The generation of appropriate control pulses is obtained by means of additional windings threading the core as indicated by the horizontal or sense lines in Figure II-3-14. As before, a threaded core is shown by means of a slant line at the intersection of a core and wire. For example, core "0" is threaded by all three indicated sense lines, while core "1" is threaded by only one sense line.

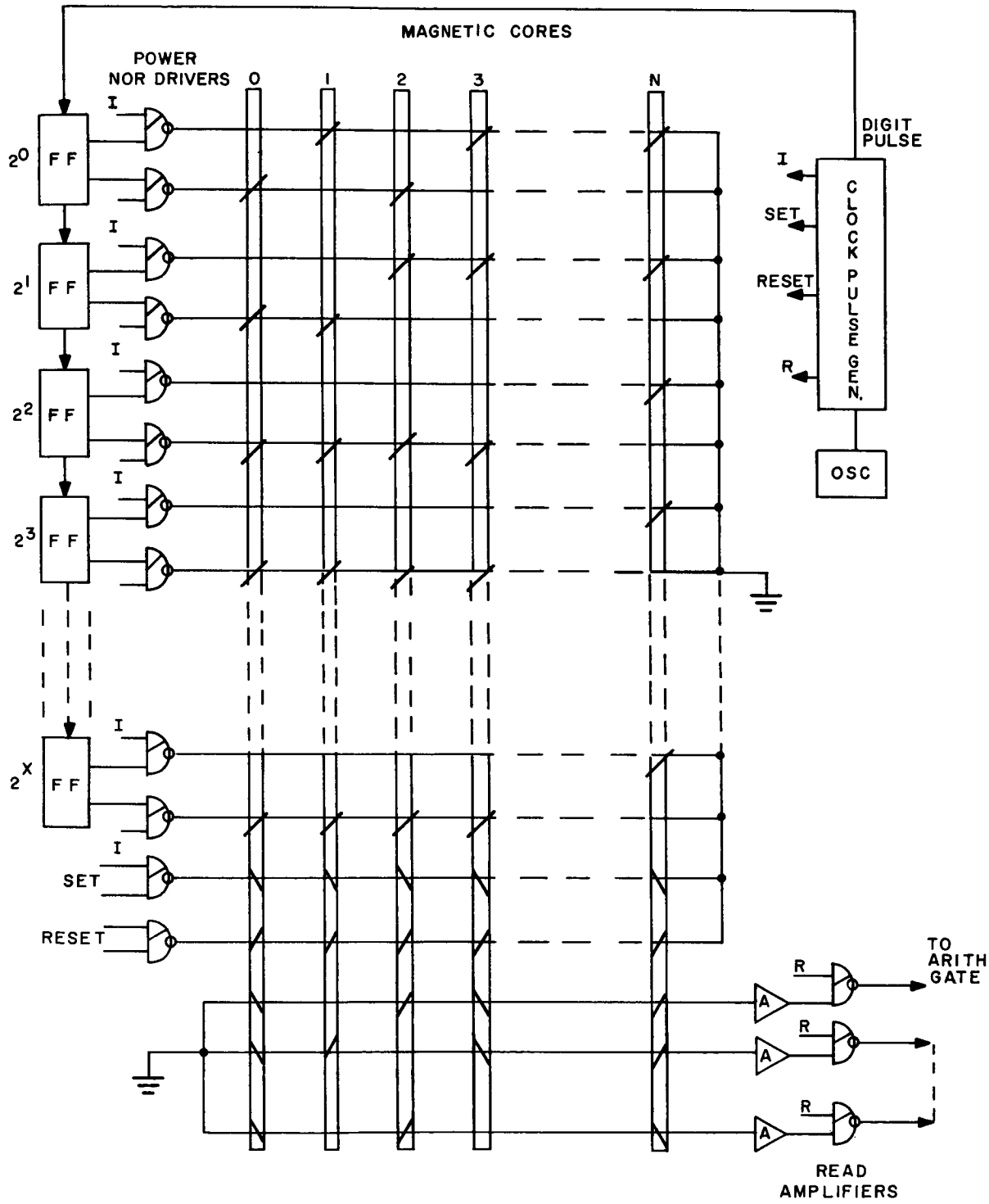


Figure II-3-14. Magnetic core control

At the time the reset drive pulse is applied and the set core is being reset to zero, the changing flux induces an emf in each sense line that is threading the resetting core. Since the remaining cores have not been set to the "1" condition, no output is generated from these cores. Thus, depending on how many sense lines thread a selected core, individual patterns of "1" s and "0" s can be produced for each core.

The core array described can be considered as a coding switch, rather than a memory, in the conventional computer sense. In the above method of core selection and switching, a core requires the logical minimum of 2^N inhibitor widenings for selecting one out of 2^N cores. In switching a core, a complete word of parallel multiple-digit code determined by its threshold wires is obtained from a single-core element. A core per bit is not required. Therefore, a single core element in the array can be regarded as a full program word. A 100-word instruction store should be theoretically implemented with only 100 cores. A pictorial diagram of the magnetic-core control showing the drive and sense windings is shown in Figure II-3-15.

The advantage of the magnetic matrix scheme over the diode matrix is that the requirement of a large number of diodes is eliminated. Additional cores are used in lieu of diodes. The tradeoff is N cores for 2N diodes. The magnetic-matrix arrangement is also more amenable to redundancy techniques. It is even possible, because of the inherent high reliability of the cores, themselves, redundancy need be applied only to the program counters, drive amplifiers, and sense-read amplifiers and not to the cores themselves, although this is not advocated until more quantitative core reliability data becomes available.

Some of the practical considerations of the magnetic-matrix, wired core programming techniques are:

> When being reset, the outputs from the unselected cores are not equal to zero because of the finite slope of their hysteresis loops in the saturated region. A very small output is produced from each unselected core, and since the sense windings through each core are effectively connected in series, the small emf's generated by each core are cumulative. This represents the disturbed core output, and the ratio of desired to disturbed output on a given core is approximately

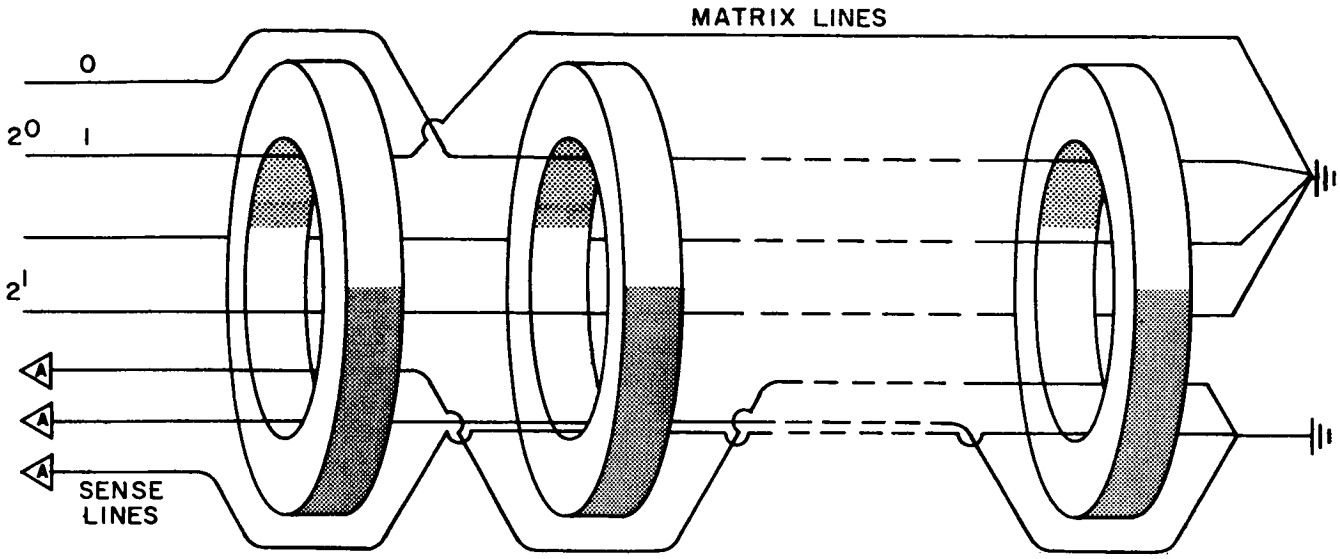


Figure II-3-15. Pictorial diagram of magnetic core control

50 to 1. However, by reversing the direction of the sense winding on every other core, effective cancellation of the disturbance can be accomplished.

Good environmental tolerances are obtained with this wired core technique, as compared to the coincident current types which must be held to about 5°C for reliable operation.

In the proposed selection process the only requirement is that the inhibited cores be held well into the "0" saturation region during the selection process. Thus the inhibit mmf can be greater than the set mmf. In contrast, the coincident current core arrays require holding the drive currents to vary close tolerances.

3.3.4.3 LOGIC DESCRIPTION - MICROWATT CIRCUITS

During the launch and boost phase and re-entry the guidance system requires high computation rates from the computer. The high speed computation requirements coupled with the capacity requirements dictate at least a 1 mc clock frequency. If conventional low power transistor redundant logic circuitry were used for the APOLLO guidance

computer, total power dissipation of the logic and control circuits would be approximately 460 watts.

A new approach has been investigated by the General Electric Company to provide the logic circuits which operate at high speeds at significantly reduced power levels. Using the proposed microwatt circuits the total power dissipated by the APOLLO guidance computer would be 400 watts.

The ability to operate with considerably less total power supplied per stage means that there will be less heating of the internal components for a given packing density. This is extremely important when dealing with high packing densities as would be used in the APOLLO guidance computer. A very low transistor dissipation means that there is virtually no heat rise in the transistor, so that the transistor is essentially operated in a storage condition. Failure rate data on transistors indicates that the failure rate approximately halves for each 10°C reduction in the ambient temperature of the device. Failure rates of other components show a similar reduction as the operating temperature is dropped. Hence, operation at lower power levels leads directly to an improvement in overall equipment reliability.

The basic technique used is that of providing a clock cycle with a low duty factor. This is accomplished by logic circuitry operating on extremely narrow pulse signals and remaining dormant for the remainder of the clock pulse period (Figure II-3-16). The reduction in power is directly proportional to the pulse duty factor.

Laboratory breadboards of the proposed computer logic circuitry were built and given preliminary evaluation. The particular design incorporated 1/10 microsecond clock pulses. With a clock frequency of 10 KC each stage consumed approximately 10 microwatts average power. Extrapolating this to a 1 mc clock and with redundancy, a 2 milliwatt average power per stage is required. Conventional transistor logic circuitry consumes from 10 to 400 milliwatts per stage while the low voltage DC microminiature logic circuitry proposed for the delay line computer consumes about 3 milliwatts per stage.

The circuits incorporate presently available 2N706 high speed silicon transistors. Interconnections and logic functions are performed by especially wound transformers

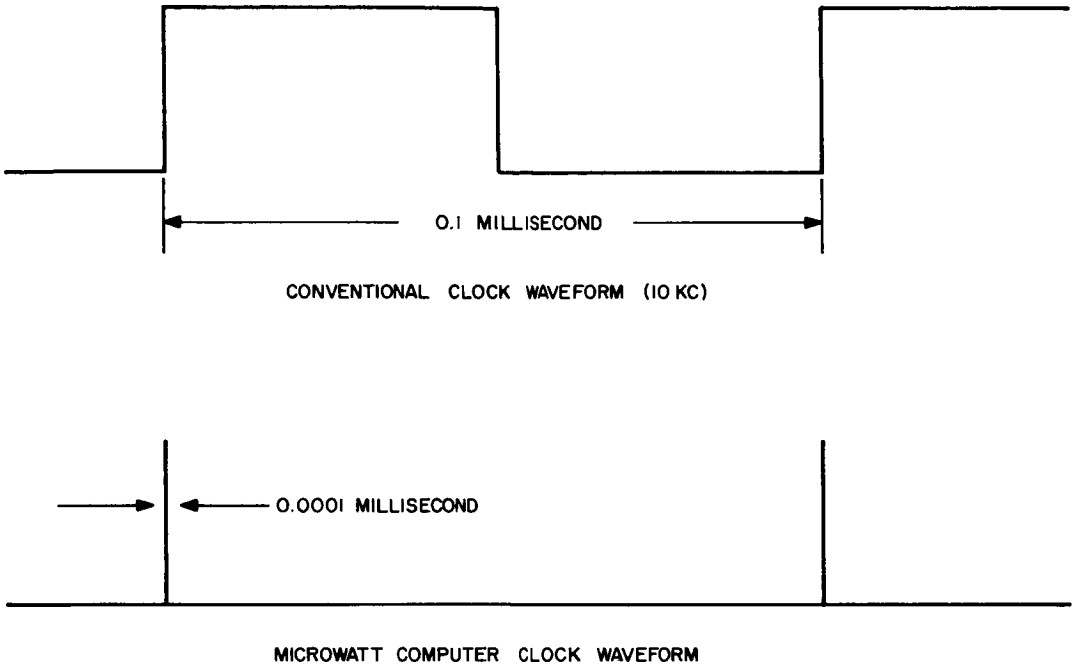


Figure II-3-16. Clock waveshapes

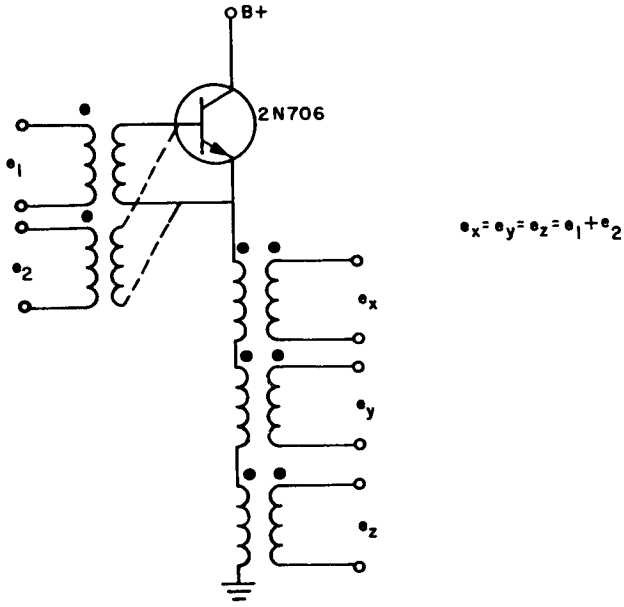
with small ferrite cup cores (see Figure II-3-17). The breadboard units were found to be relatively insensitive to supply voltage variations.

Present state of the art makes it possible to fabricate micro-energy logic circuits with a low duty cycle that operate at about 1/10 of the power that is required for standard logic circuits. The characteristics of such a microwatt computer are ideally suited for space application and would satisfy the requirements for the APOLLO guidance computer, as power supply requirements would be reduced.

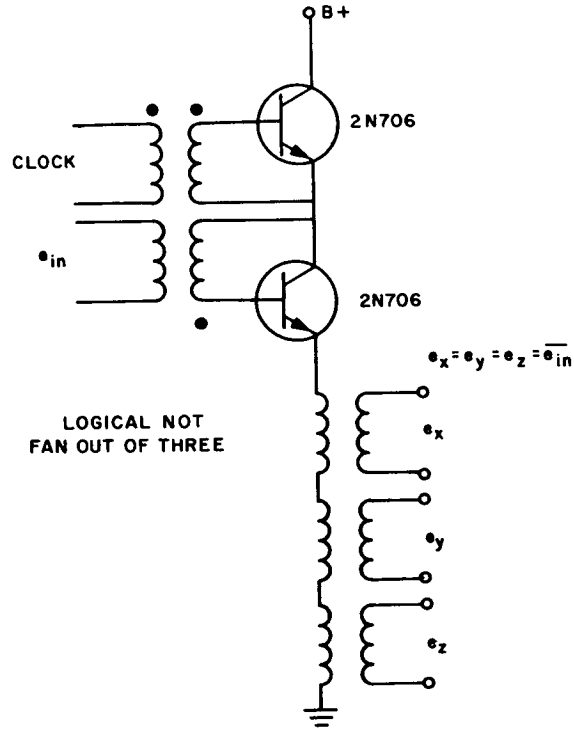
3.3.4.4 CORE WORKING MEMORY

A computer which uses a wired fixed program technique for instruction and constant storage requires a means for storing a quantity of computed data either temporarily or indefinitely. The device which provides this erasable storage, along with the associated driving and sensing equipment, is called the working memory.

The working memory described is used to store computed data or intermediate results. A given stored number or word is stored at a given address, which is a spatial



TWO INPUT OR WITH FAN OUT OF THREE



LOGICAL NOT
FAN OUT OF THREE

Figure II-3-17. Transformer logic circuitry

location in the memory. The address at which a word is stored is determined by the number stored in an address register at the time of inserting a word. A set of magnetic cores, which is located at an address is used to store a word. One core is used to store each bit of the stored word. The remanent states of these cores correspond to the binary ones and zeroes of the word stored in that address.

The working memory for the APOLLO guidance computer will be a random access memory. Arbitrary addresses may be put into the address register to select a word.

The two major types of random-access magnetic memories that were considered are, the coincident-current memory (CCM) and the linear-selection memory (LSM).

The significant difference between the two systems is that the CCM employs a coincidence of two half select currents at each addressed core, and the LSM employs a single full select current to select a specified core.

During read and write, each memory core in a CCM must react to full select currents and ignore partial select currents. Since the switching threshold current of ferrite cores in common use is roughly 60 or 65 percent of minimum full select current I_S, the drive currents must be held within close limits.

In effect, the cores must discriminate, within a fine tolerance between two levels of drive current. The memory cores in an LSM must discriminate only between the presence or absence of a full pulse.

The working storage requirements for the APOLLO guidance computer will be 512 words with each word made up of 43 bits. A linear select type memory will be used for the computer because it permits greatly relaxed current tolerances and core specifications as well as component count reduction.

The technique of linear word selection permits a great variety of memory configuration. The single plane of bit lines as shown in Figure II-3-14 is selected for the APOLLO guidance computer based on the size of the memory as well as for ease of packaging.

Each memory core is threaded by a drive winding and a bit plane winding. The word selection elements are arrayed either in a plane parallel to, and corresponding to the bit planes, or in a plane not corresponding to the bit planes (Figure II-3-18). Each row of selection elements in this plane is usually wired by a row winding, and each column by a column winding. A drive winding passes from each work selection element directly through all bit planes, and threads only the line of cores storing the set of bits comprising a single word.

The addressing of a given word, either for reading or for writing is accomplished by addressing the selection element associated with the memory core comprising the word. In the design described, word lines are driven directly through transistors. Great savings are achieved in bulk, power consumption, and operating time and design is very much simplified, compared to other techniques involving cores as word selection elements.

The memory cores are arrayed in bit primes, which use two transistors per address. Each row includes a read and write transistor which operates as the switch and driver. In this way the entire 43 bits associated with each word is read in parallel to the buffer register. Provisions are made to read the word back to the cores if it is desired to retain the stored information. New data is applied to the memory by energizing the write driver associated with the required address location and enters through the buffer register. Information is also written in parallel to the selected address. A one or zero is written depending upon the state of the flip-flops in the buffer register.

If it is desired to shorten the read time, the drive current amplitude can be increased thus decreasing the memory core switching time and increasing the induced voltage output. Read current is essentially an independent parameter, and may have any value greater than the minimum full select current I_m, depending upon the read time desired.

The advantages of word selection without cores are several. There is no degradation of current tolerances between the current driver and the memory core. No time is wasted in resetting flux left in a selection core or a pulse transformer after a read/write cycle is over. Cycle times therefore, can be as fast as the memory core allows. There are no swamping resistors to consume the power capability of the current drives, and the allowable number of bits per word is consequently higher than with other linear

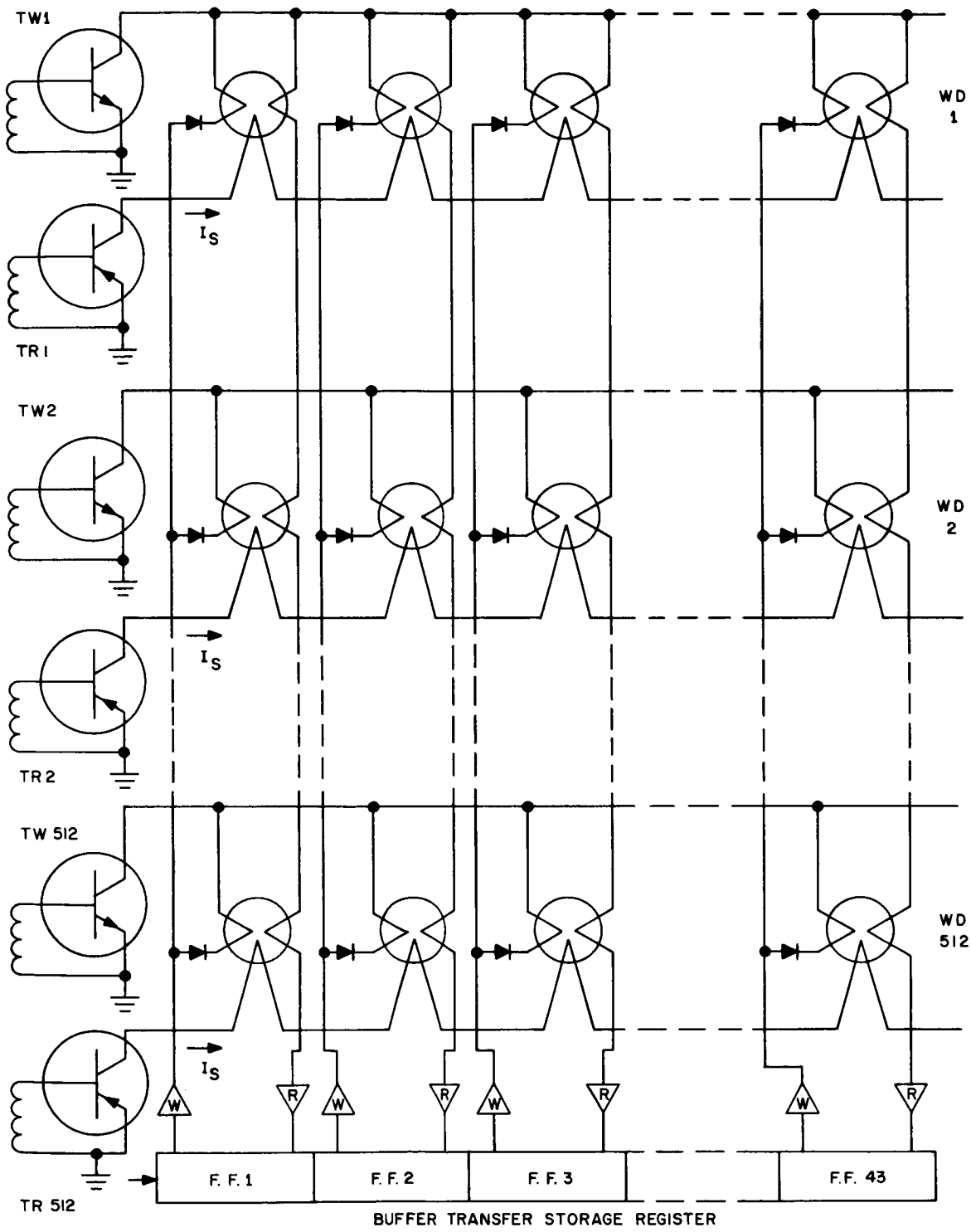


Figure II-3-18. Linear selection memory

selection memories. The number of words that can be stored in a single memory may also be increased, since stray inductance in the driver lines is minimized which will allow for future growth possibilities.

3.3.4.5 CONSTRUCTION

In general, modular construction will be used throughout for ease of maintenance, testing and replacement. The module size will be approximately 12 inches long x 6 inches wide x 1 inch thick and will contain the transistor logic, control circuits, and amplifiers encapsulated in cubical form. Each cube containing two transistors and associated circuitry occupies approximately 0.3 cubic inch. The magnetic program cores are mounted on similar modules.

A sketch of the proposed alternate APOLLO Guidance Computer configurations is shown in Figure II-3-19. In the sketch the transistor circuits are placed in modules which are mounted like the leaves of a book. This allows access to each module and greatly aids testing and troubleshooting.

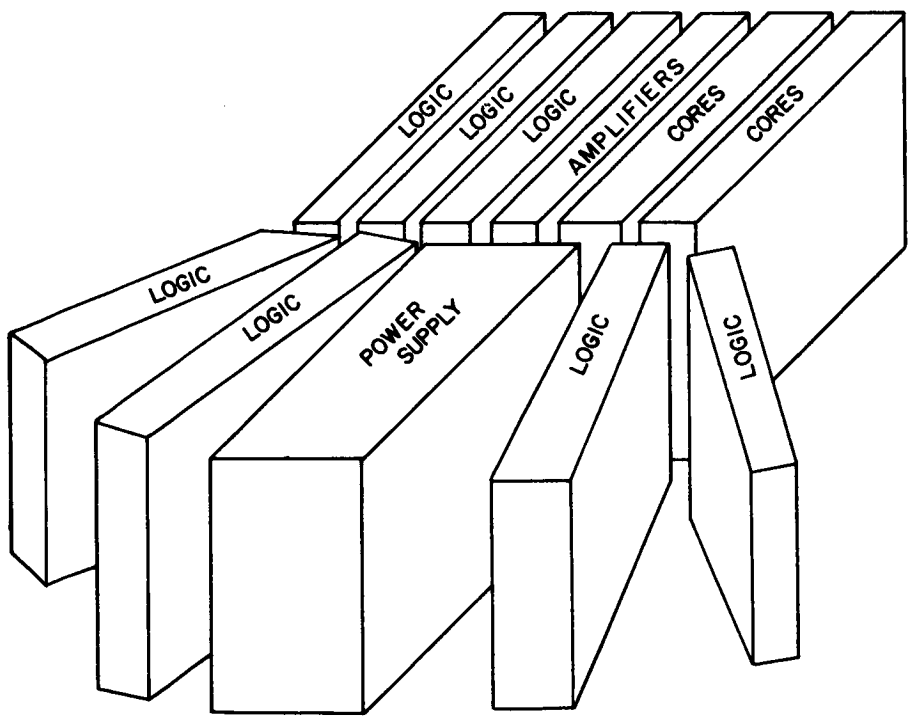


Figure II-3-19. Alternate APOLLO computer configuration

A block diagram is shown in Figure II-3-20 of a computer utilizing a one core per word wired core fixed program memory and a 512 word magnetic core erasable memory. This type of memory is amenable to a varying clock frequency which can reduce power requirements whenever high computational speed is not needed.

3.3.4.6 THIN FILM MEMORY ELEMENT - A FUTURE POSSIBILITY

An alternative method that may be used for random-access memory that allows for future performance possibilities and higher solution rates is an ultra-thin film of ferro-magnetic material. The proposed delay line memory and ferro core memory are based on present state of the art components, while reliable thin magnetic films would be available for use by 1963.

The use of these thin films results in significant advantages in both the operation and physical configuration of the memory. Nonmagnetostrictive ferro-magnetic films of less than a critical thickness tend to switch remanent states by rotational switching and at speeds much higher than obtainable with ferrite cores. The memory element and all associated drive and sense conductors can be fabricated by a series of vacuum depositions.

The choice of magnetic thin films as memory elements for future growth appears reasonable considering the characteristics of the films in performing the memory function. When the films are switched in a rotational mode, they have two stable magnetic states which require no standby power to maintain. Thus, like ferrite cores they provide a memory function which is not dependent upon the continuity of the power supply. Moreover, they exhibit very high-speed switching capabilities measured in the range of a few-nanoseconds to fractional-nanosecond. In addition, temperature problems with films are likely to be appreciably less severe than in memories using ferrites. The extremely small volume of the film, its high ratio of surface area to volume, and its good thermal conductivity contribute to lower heat generation and lower temperature rise in a film than in a ferrite, despite the higher repetition rate at which the film is switched.

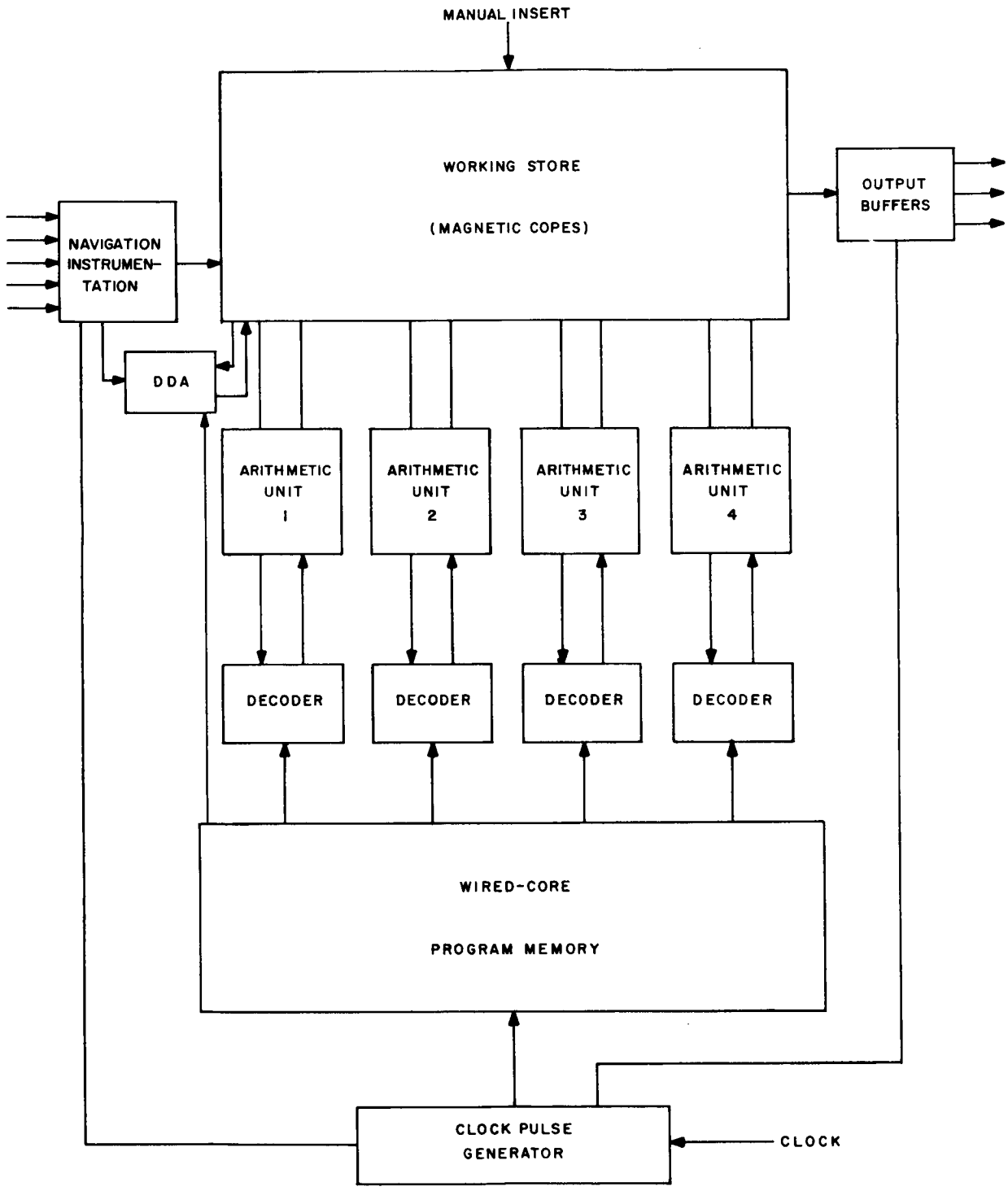


Figure II-3-20. APOLLO computer block diagram

A memory organization which is most suitable for use with thin film elements is a word organized memory. The memory sites are crossed with three windings: (1) The bit driver lines, (2) the word driver lines, and (3) the sense lines.

The word line provides a transverse field to the memory sites and the sense line links the easy-axis field as indicated by Figure II-3-21. During the read interval in such a memory, a current pulse is passed along the word-drive line, affecting a rotation of the magnetization of each bit of that word toward the transversal direction; the sense of rotation is different for a stored 0 from that for a stored 1 resulting in opposite polarity signals on the sense line depending upon the value of the stored binary digit. Current may or may not be forced to flow in the sense lines during the read interval. If word drive current is forced to flow in a direction to switch to the 0 binary state, then the voltage appearing on the sense line for a stored 1 will be comparatively large and of positive polarity, while the voltage for a stored 0 will be comparatively small and of opposite polarity. The distinction between a 0 and 1 is then made on the basis of the polarity of the signal, essentially independent of its amplitude; moreover, only one bit per sense line experiences any change of field and thus there is no inductively coupled noise. If, on the other hand, current in the sense drives lines is maintained at zero during the read interval, the sensed voltage is unequal in magnitude and of opposite sign for a stored 0 compared with stored 1.

If small enough transverse fields are used, a reversible rotation results and readout is non-destructive, i.e., the magnetization of the film returns to its original position after the word-driver pulse disappears and the stored information is not lost. This is a desirable feature in many instances; its practical utilization is dependent upon the adequacy of the amplitude of the sensed signal, which is associated with the somewhat limited drive fields required to avoid destruction of stored information, and on the ease of controlling the drive-pulse amplitude.

Thin magnetic film memory has been chosen as another alternative memory for the APOLLO alternate computer because of its high speed capabilities and future growth possibilities.

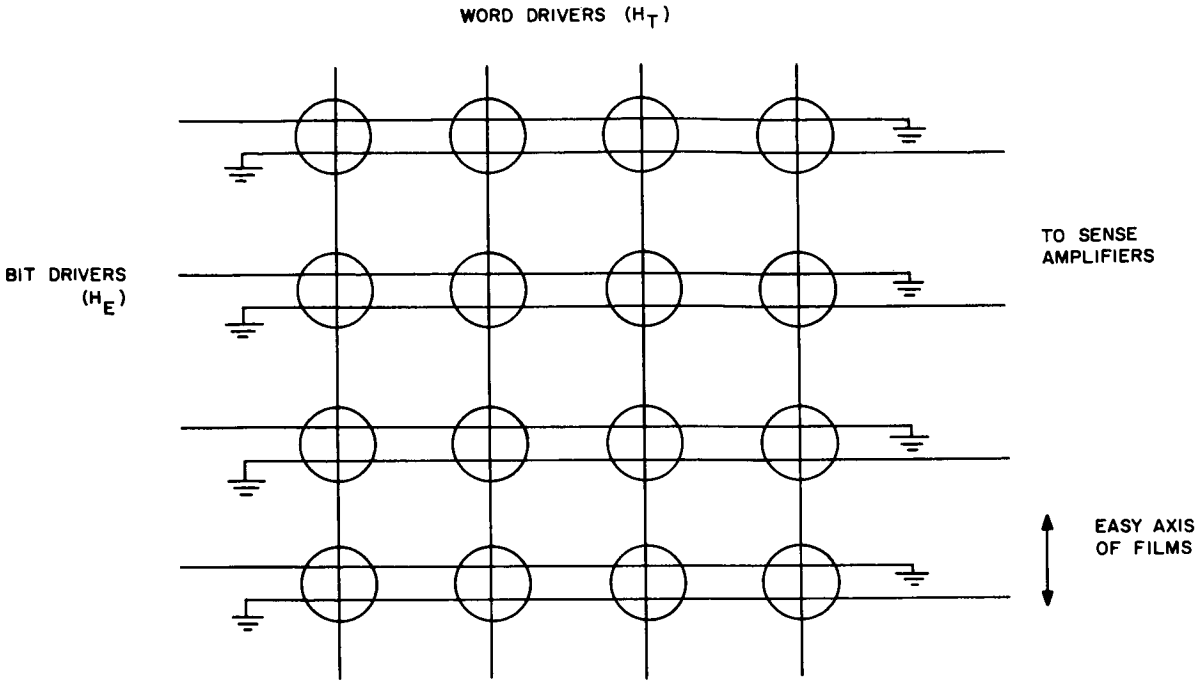


Figure II-3-21. Orientation of drive and sense lines in word-organized memory array

3.3.5 Summary

A general purpose computer incorporating a digital differential analyzer and a memory composed of delay lines has been proposed as the first choice for the APOLLO mission. An alternate computer utilizing a wired fixed program core memory together with an erasable core storage memory has been proposed.

For higher reliability, the stationary core and delay line memories far exceed a mechanical rotating drum. Conventional core memories, using one core per bit, are inherently more complex than delay lines in terms of selection and read/write control unless the proposed alternate design of one core per word is used. The penalty in power consumption and complexity of a conventional core memory would exceed the unrequired computational speed advantages. A delay line memory for the APOLLO computer can readily be fabricated using present day knowledge and techniques whereas further development is necessary for large fixed programs using a one core per word memory.

Accordingly, for the APOLLO mission, the use of a delay line memory is considered optimum.

In Table II-3-I are summarized the essential characteristics of the two proposed computers. As stated previously these are based on completely automatic operation of the N & C subsystem.

If completely automatic operation is not required, the computer can be simplified by eliminating those programming and decision-making functions which can be done by the crew. It is estimated that the memory required could then be reduced to about 6000 words, by judicious organization of instruction words. In this case the computer weight would be approximately 65 lb. without redundancy, and approximately 200 lb. with triple redundancy.

[redacted]

TABLE II-3-I. CHARACTERISTICS

	Computer Using Delay Line	Computer Using Core Storage and Wired Program
Type	Serial Binary General Purpose with incremental analyzer	Serial Parallel Binary General Purpose with incremental analyzer
Memory Storage Capacity	7,000 words of 43 bits per word of erasable delay line memory	6,500 words of 43 bits per word using magnetic core fixed program 512 words of 43 bits per word erasable core storage
Physical Characteristics-Weight	(Non-Redundant) 80 lb (Redundant) 250 lb	(Non-Redundant) 65 lb (Redundant) 205 lb
- Power	150 watts 460 watts	125 watts 400 watts
- Volume	1.50 cu ft 3.75 cu ft	1.25 cu ft 3.50 cu ft
Clock Rate	1 megacycle	1 megacycle
Speed (Single address arithmetic operations/sec exclusive of access time)	15,000 additions per sec 1,300 multiplys per sec	29,000 additions per sec 1,500 multiplys per sec
Circuits	Silicon transistors microminiaturization and silicon diode, nor and flip flop circuits	Silicon transistor microwatt and silicon diode, nor and flip flop logic
Availability	Memory - 6 months Total Computer - 24 months.	Memory - 12 months Total Computer - 30 months.

[redacted]

3.4 INERTIAL-STELLAR STABLE PLATFORM

This assembly consists basically of a stellar-gyroscopically stabilized inner gimbal member supported in three additional gimbals. The inner member (azimuth gimbal) contains the three gyroscopes, two fixed star trackers (three output axes), and three accelerometers that comprise the sensing elements for the assembly to function as a stellar-inertial reference and sensor. See Figures II-3-22 and II-3-23.

3.4.1 Platform Design

The designation of the various gimbals is as follows, reading from the inside out:

(1) Azimuth Gimbal (stable element with sensors)

(2) Inner Roll Gimbal

(3) Pitch Gimbal

(4) Outer Roll Gimbal

3.4.1.1 AZIMUTH GIMBAL ASSEMBLY DESIGN

The azimuth gimbal assembly consists of a magnesium structure machined to receive the trackers, gyros, accelerometers and other components. Integrally mounted blanket heaters assure the nominal operating temperature of the sensing elements. The azimuth-inner roll gimbal bearing is of large diameter design to permit in-flight changing of the sensor portion of the azimuth gimbal. It can be removed for replacement of gyros, or tracker photo-detector or replaced by an entire sensing element.

3.4.1.2 AZIMUTH AXIS

The axis of freedom between the azimuth gimbal and the inner roll gimbal is called the azimuth axis. The azimuth gimbal will have unlimited travel about the azimuth axis. Components mounted on the azimuth axis will be:

a) One pancake resolver for gyro error signal resolution

b) One pick-off resolver

c) Two slip ring assemblies

d) One d-c servo motor

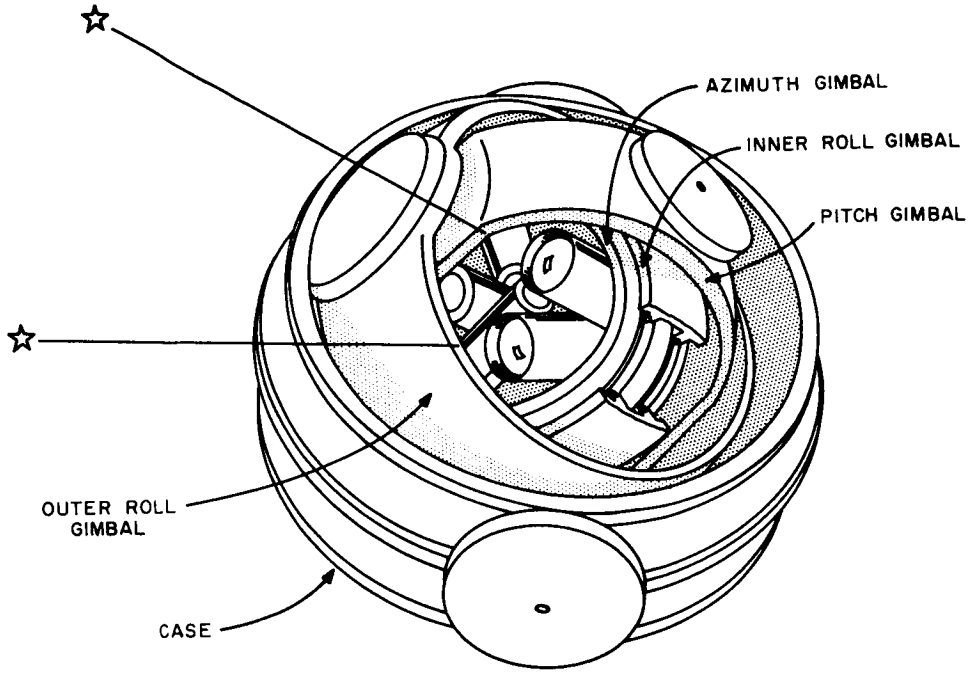


Figure II-3-22. Inertial-stellar platform

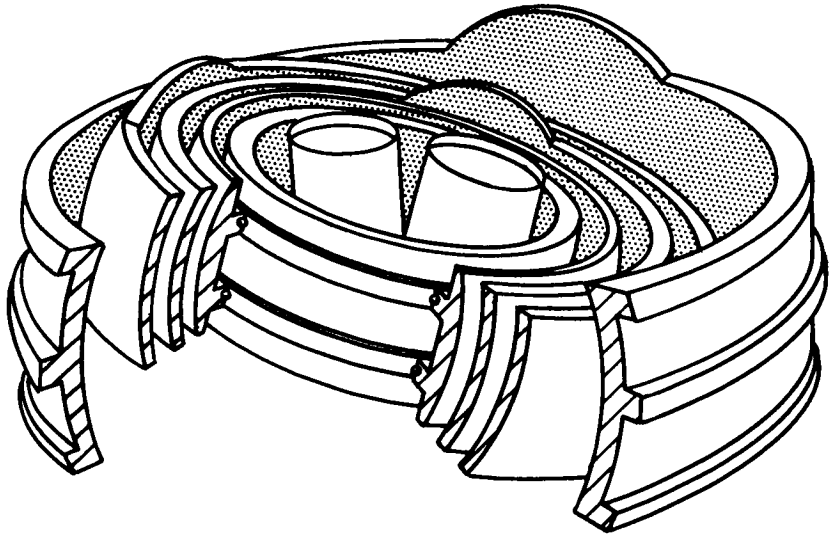


Figure II-2-23. Schematic of platform gimbal arrangement at orthogonal axis alignment

Most of the components are presently used on flyable Polaris Guidance equipment. They can be mechanically adapted to fit the requirements of the APOLLO guidance platform.

3.4.1.3 INNER ROLL GIMBAL

The inner roll gimbal will be an aluminum structure having a truncated spherical shape and a cross section resulting from the nesting of two such concentric shapes. This design was selected to result in a rigid structure whose amplification factor will be low. While it is recognized that the Mk 2 Polaris Platform gimbals are true spheres, the particular design mentioned above was chosen because of having star tracker "look out" and the ease with which such a system, as opposed to the Mk 2 Polaris System, may be serviced and maintained. The inner roll gimbal will mount two angular differentiating accelerometers (ADA) whose sensitive axes will be along the inner roll and pitch axes respectively.

3.4.1.4 INNER ROLL AXIS

The inner roll axis is the axis of freedom between the inner roll gimbal and the pitch gimbal. The inner roll gimbal will have an angular freedom of at least ±25 degrees. The following components will be mounted on the inner roll axis:

2 pancake resolvers

2 4-point contact bearings

1 d-c servo motor

2 slip ring assemblies (32 circuits)

3.4.1.5 PITCH GIMBAL

The pitch gimbal will have a shape and construction similar to the inner roll gimbal. It too will be made from aluminum.

3.4.1.6 PITCH AXIS

The pitch axis is the axis of freedom, permitting unlimited rotation, of the pitch gimbal inside the outer roll gimbal. Components mounted on the pitch axis are:

1) Two pancake resolvers for gain control of the outer pitch servo and angle pick-off
2) Two slip ring assemblies (32 circuits)
3) One interrupted slip ring segment providing SPST switch function at plus and minus 75 degrees roll angles
4) Two 4-point contact bearings

3.4.1.7 OUTER ROLL GIMBAL

The outer roll gimbal will be of a construction similar to the pitch and inner roll gimbals. It will be designed to mount one ADA whose sensing axis is along the outer roll axis.

3.4.1.8 OUTER ROLL AXIS

The outer roll axis permits unlimited rotational freedom between the outer roll gimbal and the case. The outer roll axis components are as follows:

1) One pancake resolver for angle pick-off
2) One d-c servo motor
3) Two slip ring assemblies
4) One interrupted slip ring segment providing SPST switch function at the 180 degree outer pitch gimbal position
5) Two four-point contact bearings

3.4.1.9 HEAT TRANSFER

The platform will be encased in a pressurized housing that is pressure isolated from the environment of the re-entry vehicle so that a loss of pressure there would not disturb the transfer of heat by forced convection to its case.

3.4.2 Components

3.4.2.1 GYROS

With the emphasis on early design dates and high reliability to weight ratios, it is desirable to provide a system design that utilizes each piece of equipment to its greatest extent.

Since no presently available gyro will maintain the stability required of an angular reference for the length of the missions, star trackers must be used to eliminate long time drift. Having accepted the star trackers, the remaining requirements for the gyro are compatible with existing gyros, with the exception of longer time of operation.

It was therefore desirable to choose a gyro that will also allow the inertial system to measure large angles to an accuracy consistent with the field of view of the star trackers, such a feature allowing considerable simplification of the guidance equipment.

To meet this angle measuring capability, two methods are available; a completely free gyro, with fairly accurate read-out, or the fully floated single axis gyro with a precision torquer. There is considerable data on the latter, while the former is under development and may permit a body mounted type inertial system. With emphasis on the reliability and minimum design time, the single axis, precision torqued gyro has been selected and is the heart of the system described here. It allows the movement of the system reference coordinates from ground alignment to reference stars or from one pair of stars to another pair by simply constant current, measured time, torquing of the gyros.

The requirements in drift rate are available in several miniature gyros:

.225 degree per hour - constraint

.225 degree per hour per g.

.030 degree per hour per g^2

Design can begin using existing ball bearing spin axis gyros which are adequate. However, at later dates, this type of gyro will be available in about the same size with a

self-acting air spin bearing which will considerably increase the reliability and life of the gyro and will be substituted for the ball bearing units.

To date, the air spin bearings have had anisoelastic problems, but the following excerpt from a report on work related to a smaller, higher performance gyro than required here indicate the solution to the anisoelastic problem.

3.4.2.1.1 Spin Axis Gas Bearing

In order to design and optimize the parameters of this bearing, the Reynolds equation was transformed to a difference equation and using iterative procedures, the equations were solved on an IBM 704 computer. This procedure has been developed for ONR and has been confirmed by experimental data. The results obtained from the computer studies were used to derive these bearing characteristics such as the compressibility number, attitude angle, gas pressure, clearance, $L/_D$ ratio, etc., which are required for a 50 g - isoelastic system. The resulting spring constant versus load is shown in Figure II-3-24 from which one of the anisoelastic drift coefficients was computed. In no case, was it greater than .004 degree/hr/g^2. When the attitude angle was included in the calculation, the drift coefficient increased to a value of .008 degree/hr/g^2.

3.4.2.2 ACCELEROMETERS

The three accelerometers will be mounted with their sensing axes alinged with platform axes A'B'C'. They will be the miniature restrained pendulum, pulse torqued type, so that the net pulse count is the velocity gained. There is not expected to be any problems in obtaining units to the required specification of:

0.1 cm/sec^2 max. threshold error

75 parts per million linearity error.

3.4.2.3 ANGULAR DIFFERENTIATING ACCELEROMETER

Four ADA's are used on the gimbal structure to provide feedback to stabilize the various gimbal servos. Basically ADA, is a damped torsion pendulum with a tachometer type pick-off. The spring-mass resonant frequency is 0.7 rad/sec and above that, the device generates a signal proportional to its angular velocity in space. The tachometer gradient is .03 volts per rad. per sec. The size of the device is 0.75 in. diameter by 0.75 in. length.

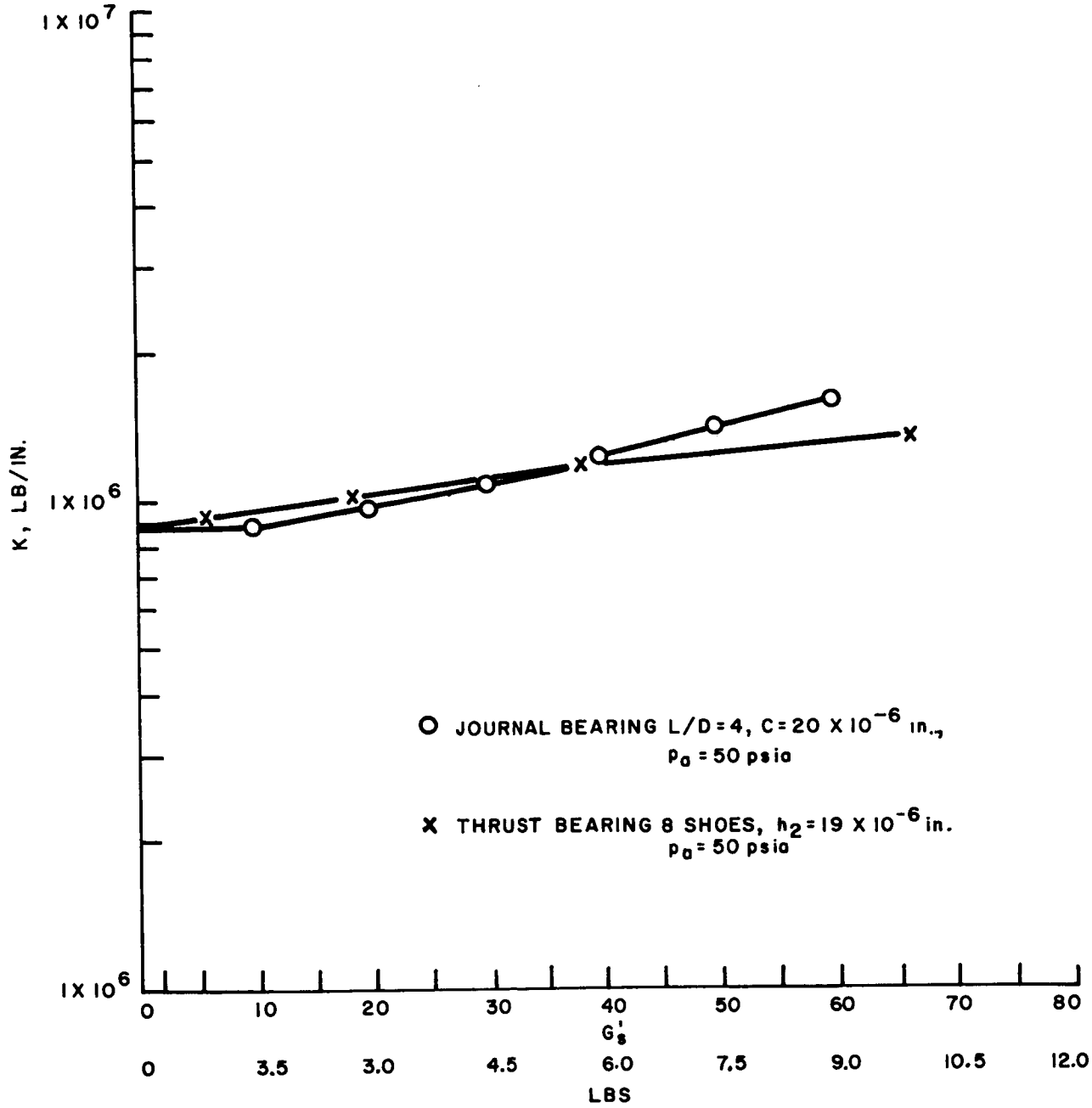


Figure II-3-24. Spring constant versus load

An ADA whose sensing axis is parallel to the azimuth axis is used to stabilize the azimuth gimbal servo. Two ADA's are mounted on the inner roll gimbal. One with its sensing axis parallel to the inner roll axis, thus stabilizing the inner roll servo. The other, with its axis parallel to the pitch axis, stabilizes the pitch gimbal servo. A fourth ADA mounted on the outer roll gimbal stabilizes the outer roll servo.

3.4.2.4 STAR-TRACKERS

The star-trackers used on the inertial platform are similar to those used on the celestial sextant. A general discussion of star-trackers is given in Appendix I- . The platform trackers will be arranged within the other instruments to most efficiently utilize the space.

3.4.2.5 GIMBAL MOUNTED ELECTRONICS

The pre-amplifiers and adjusting devices associated with the inertial and tracker equipment will be mounted on the azimuth gimbal. Circuitry and components, whose performance and reliability have been proven on the Polaris Program, will be utilized. Due, however, to the packaging requirements of the azimuth gimbal, the shapes of the various electronic modules must be adapted for this application.

Woven wire blanket type heaters, cemented to the gimbal and gyro mounts, temperature sensors, and a small vane-axial blower are also utilized. These comprise the part of the temperature control system which is gimbal mounted. These are all Polaris proven components.

3.4.2.6 RESOLVERS

Gyro error signal resolver Kearfott P/N 26370-001 (Ref. BuWep 2104021) will be used. This resolver is a proven component used in the Polaris guidance equipment. It is a pancake type resolver having a maximum error of 1 degree when operating at rated frequency of 800 cps and 25 v. and has a 1:1 transformation ratio.

3.4.2.7 FOUR-POINT CONTACT BEARINGS

With the exception of the inner axis bearings, the bearings to be used are special four-point contact bearings developed for the Polaris Platform. Low frictional torque levels,

for this size bearing have been achieved (15 in-oz at 500 lb.thrust load). Four-point contact bearings provide the stiffness of a duplex pair in the size and weight of a single bearing.

Specification:

1) Number of balls: 23 - 5/32 Dia
2) O.D.: 2.0625 in.
3) I.D.: 1.3387 in.
4) Width: .2500 in.
5) Contact Angle: 20 degree ±4 degree
6) Material: Stainless Steel
7) Max. axial load capacity: 600 lb
8) Max. radial capacity: 800 lb
9) Bearing "Built-in" Preload: 5-10 lb
10) Tolerances: ABEC 7

The large diameter inner gimbal bearing is of the type used successfully on the Hermes platform.

3.4.2.8 SLIP RING ASSEMBLIES

The slip ring assemblies to be used in this platform for inter-gimbal electrical connections, are presently being qualified for use in the Polaris Guidance System. They are small enough to fit within the stub shafts. Assemblies of identical design were used without failure in the Mk 3C Guidance equipment feasibility study when an Inertial Platform was successfully operated at accelerations of 7 g.

Specifications:

1) No. of rings: 32 ring pairs on each axis
2) Operating life: 2000 hr or 1,000,000 oscillating cycles

3) Total life (operating and storage): 5 years

4) Speed: 20 rpm max.

5) Brushes: min. of 2 contacts per slip ring

6) Current and voltage rating: 1 Amp. max. 150 v peak

7) Ambient pressure: 12 psia to 75 psia

8) Noise: 7 milliohms rms with rotor oscillating at 2 cps, 3 degree ±1 degree double amplitude test at 120F

9) Vibration: 10 g's, 5 - 2000 cps in any direction

10) Shock: 30 g's - 3 milliseconds
20 g's - 11 milliseconds in any direction

3.4.2.9 DC SERVO MOTOR (TORQUER)

The d-c torquers used on this platform are also components of the Polaris Guidance System. These torquers have encapsulated windings, permanent magnet field poles and have proven to be very reliable under all environmental conditions.

The exception is the inner torquer which will be of sectionalized design and must be developed.

Typical Specification:

1) OD: 5.125 in.

2) ID: 3.500 in.

3) Width: .562 in.

4) Torque Constant: .305 lb-ft/amp

5) Back EMF Constant: .413 V/rad/sec

6) Inductive time constant: .0005 sec

7) Armature resistance at 25C = 24 ohms

3.5 CELESTIAL SEXTANT

The celestial sextant unit will be designed for space environment.

There are some unknown factors in designing for space environment at normal temperatures, but the General Electric Space Sciences Laboratory at the MSVD will provide the necessary testing to assure a reliable design. The unit has its star trackers, which are similar to those used on the inertial-stellar platform except for environment, mounted in such a manner that high accuracy is required only between the tracker axis and the various horizon scanner axes.

The only precision bearing required are those of the scanning unit themselves and Axes D and E on Figure II-3-2.

It will employ a-c torquers, and resolver pick-offs and will have ±360 degrees of freedom on Axis C and either A or B, depending on the mounting location on the vehicle. The remaining axis will have a limited motion of ± 45 degrees.

The scanner pointing axes D and E will have precision pick-offs of the Phasolver type, which will read the direction of the earth or Moon center directly from the stellar coordinates. These axes will have ±360 degrees of freedom.

The scanning lens units rotate about their sensors so that no motion on the sextant requires slip rings. The motion about axis F is ±180 degrees.

The sextant will be withdrawn into the vehicle for maintenance by the crew. The unit is expected to have dimensions of about 15 inches length by about 10 inches diameter and weight 30 lb exclusive of the air lock. A sketch of the concept is shown in Figures II-3-25 and II-3-26.

3.5.1 Startrackers

A concept of the star-tracker assembly for the celestial sextant is shown in Figure II-3-27. The housing is fabricated from a solid block of aluminum for stability and to minimize the effects of temperature. Two holes 2 inches in diameter and 5 inches long are bored in the piece, one longitudinally and the other at an angle of 14 degrees,

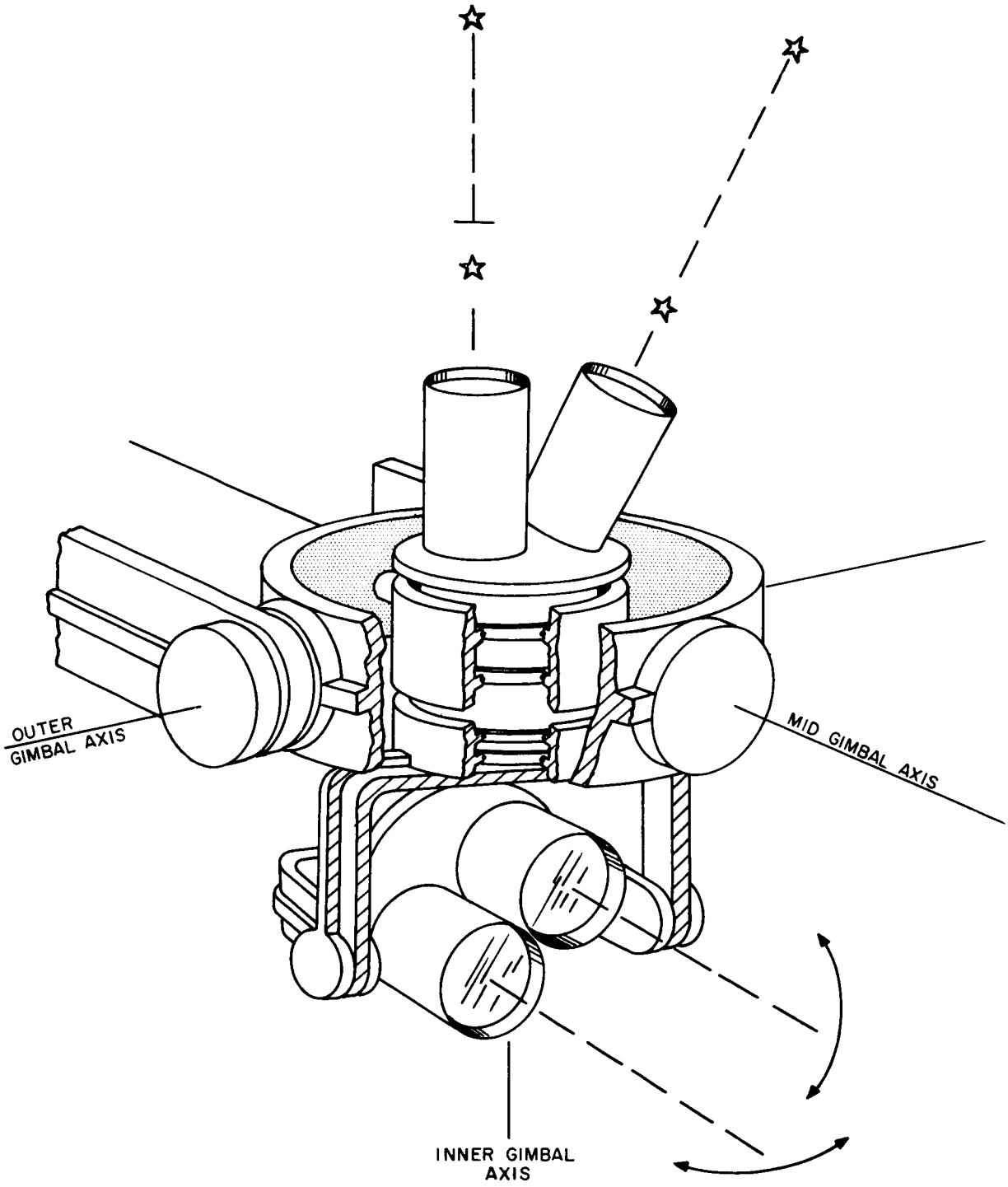


Figure II-3-25. Celestial-sextant

[redacted]

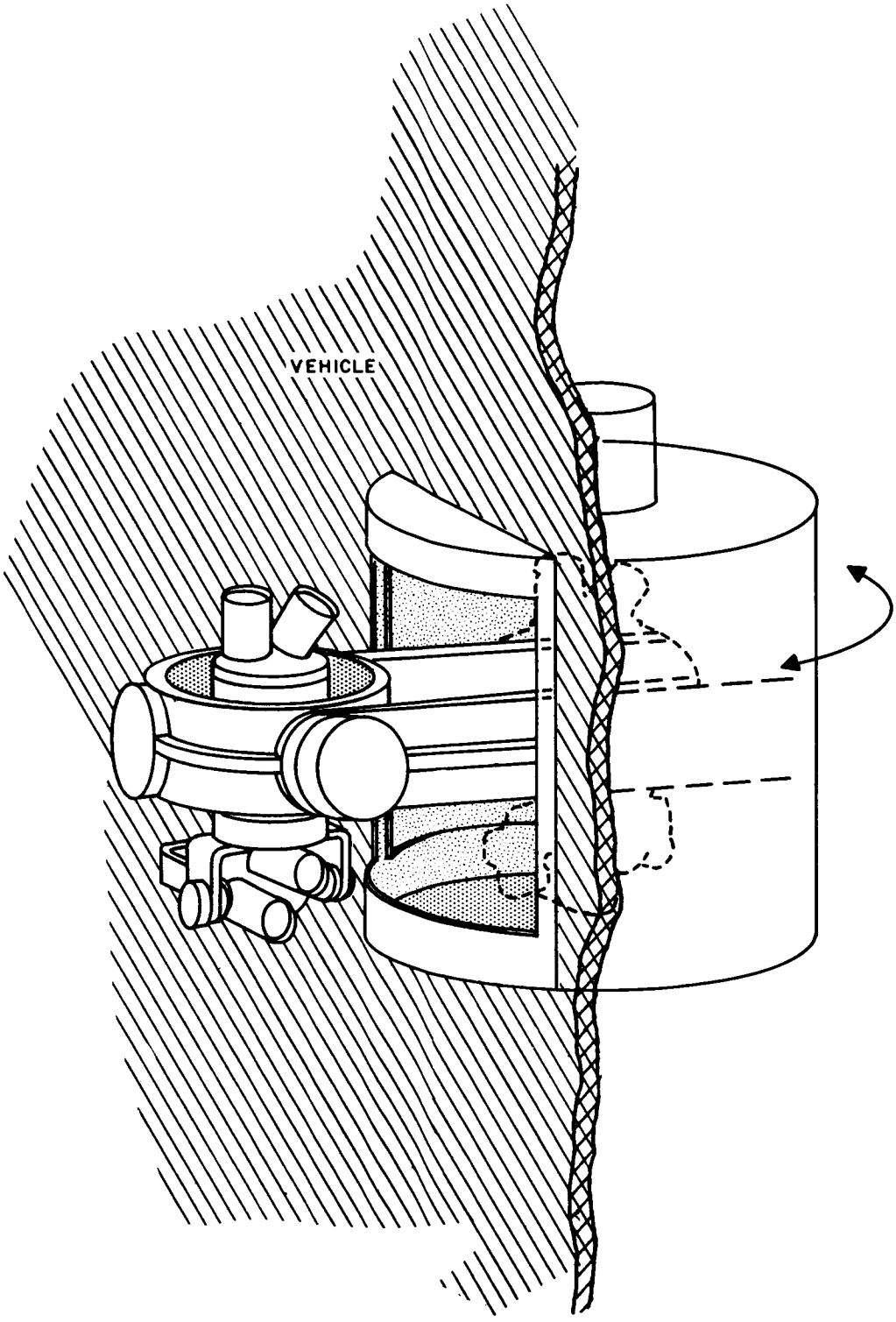


Figure II-3-26. Celestial sextant mounting arrangement

[redacted]

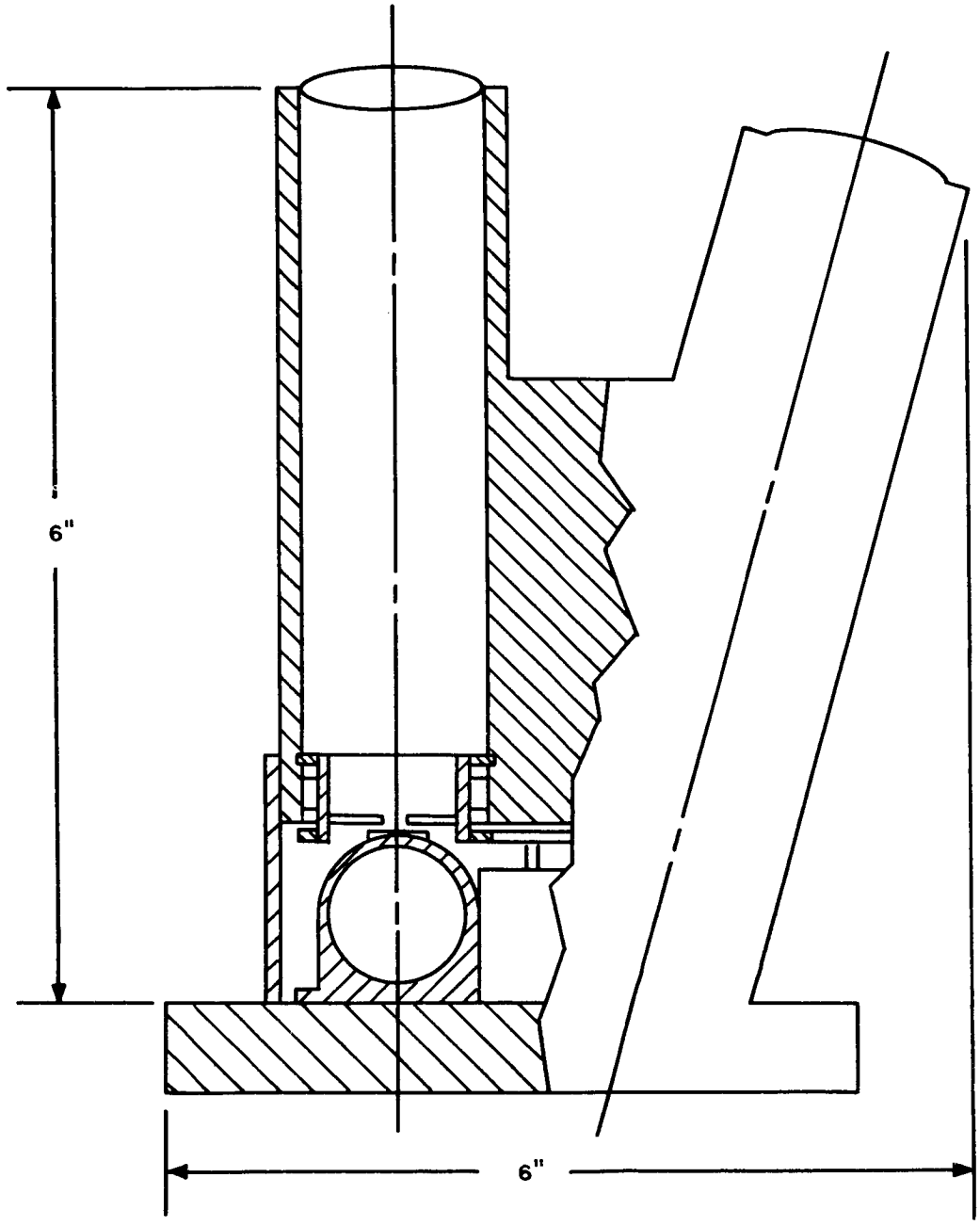


Figure II-3-27. Star-tracker assembly

45 minutes from the first. This is the exact angle between Polaris and Kochab which are the two stars chosen for navigation. The optical elements are at one end of the tubes and mechanical modulators and photomultipliers at the other. A modified Cassegrain lens system could be used to decrease the length of the assembly if it is decided later that this change would benefit the overall design.

The modulators will be supported by lightly preloaded angular-contact ball bearings for accurate rotation about the optical axis. Special materials such as sapphire balls will be used to prevent seizing.

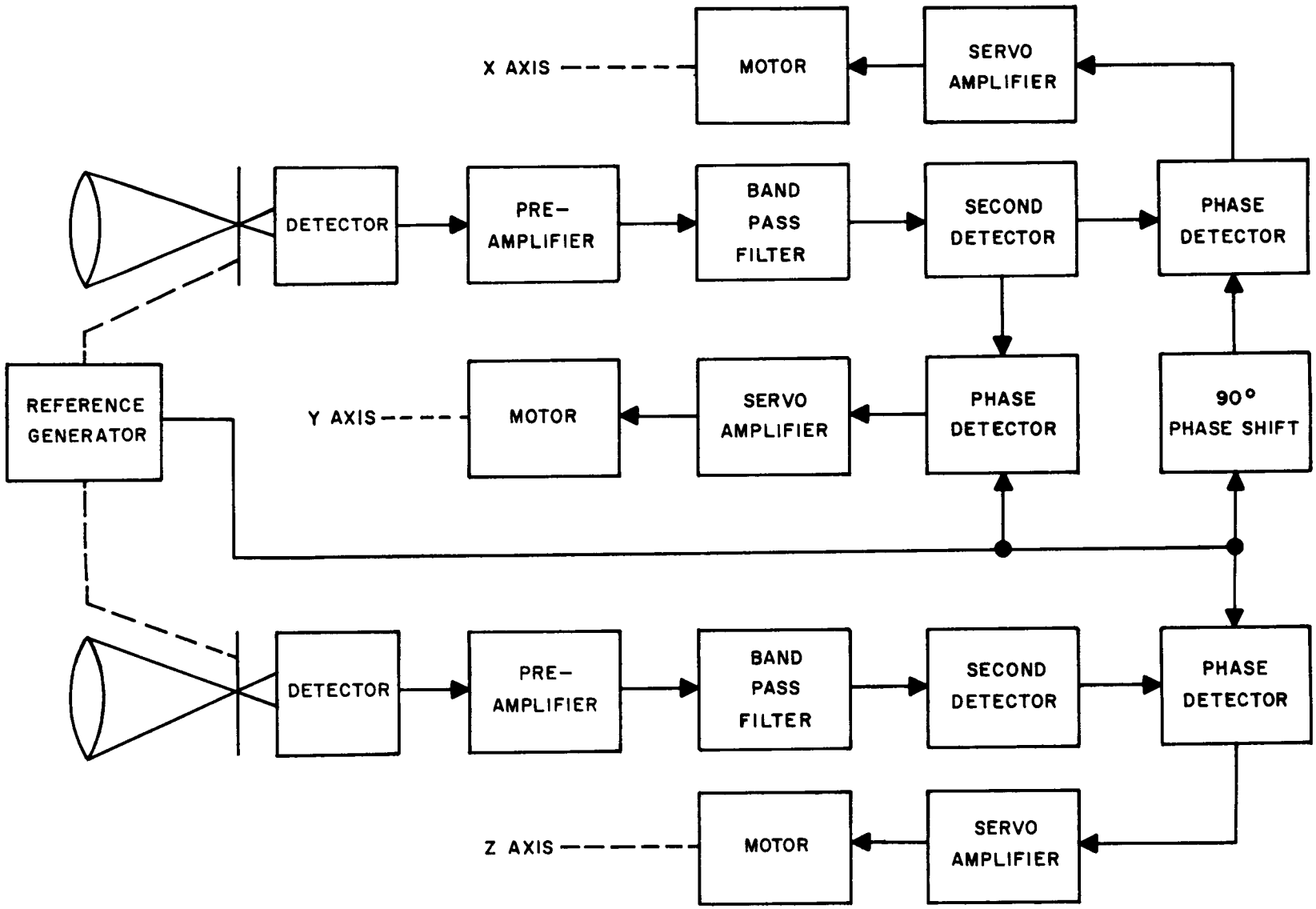
Tests will be made at the General Electric Co. in the near future on this type of bearing designed for a similar application. The modulators will be driven by phase-locked synchronous motors in which the rotor runs at a constant speed and maintains a fixed relationship to the stator field. The phase of the stator voltage will indicate the position of the modulator, and the star image will be driven to a null on the optical axis by driving the modulation of the off axis image to zero by comparison to two electrical signals in quadrature developed from the reference generator (see Figure II-3-28.)

The RCA 7029 photomultiplier tube has been selected for this application since it has been developed to have more stable characteristics than the 1P21. Cathode sensitivity of the 7029 is 125 μ amperes/Lumen and gain is 3.2×10^5.

The signal to noise ratio of the star-tracker for space application will be very high (approximately 100 for an aperture of only 1 cm^2) since the sky background will be almost completely dark except for the stars and planets. With good mechanical design it should be possible to track a star accurately to the 3 to 5 seconds of arc diameter of the star's diffraction pattern (see Figure II-3-29) with a 2 inch aperture diameter. Since the accuracy of pointing toward the second star, Kochab, must be multiplied by 4 in determining the accuracy of the position of the vehicle in space it may be necessary to increase the aperture of this telescope.

The star Polaris has an intensity of 3.5×10^{-11} lumens/cm^2 and with a 2 inch aperture diameter and an optical transmissivity of 0.8, 5.6×10^{-10} lumens are available at the input to the photomultiplier. Since the gain factor of the 7029 is about 40 amperes/lumen, a 22 millivolt signal would be produced across a 1 megohm output impedance.

[redacted]



[redacted]

Figure II-3-28. Star-tracker block diagram

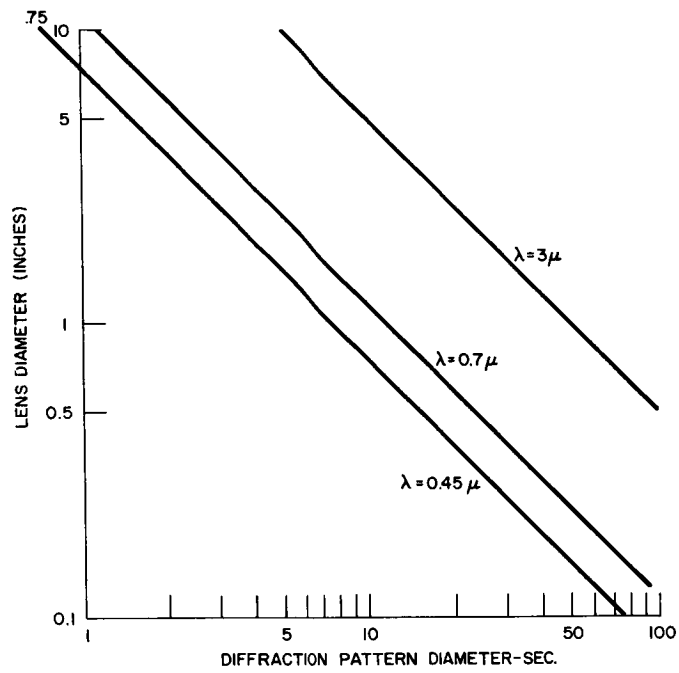


Figure II-3-29. Star diffraction pattern vs. lens aperture

This signal may then be amplified to produce an adequate tracking signal accurate to the star's diffraction pattern.

The star tracker should not point nearer than about 30 degrees from the sun. However, it may be possible to place baffles in the telescope so that a star may be tracked to within 15 degrees of the sun. Shielding from the sun may be required to control the temperature of the detector.

A shutter actuated by the level of the final anode current may be used to automatically cover the aperture to protect the detector from intense light sources.

3.5.2 IR Horizon Scanners

The proposed design for the horizon scanner is shown in Figure II-3-30. This consists of a paraboloidal gold plated reflector of approximately a 3-in. aperature, an inside out synchronous motor, and a plug in type, lens detector assembly, along with a module type pre-amplifier.

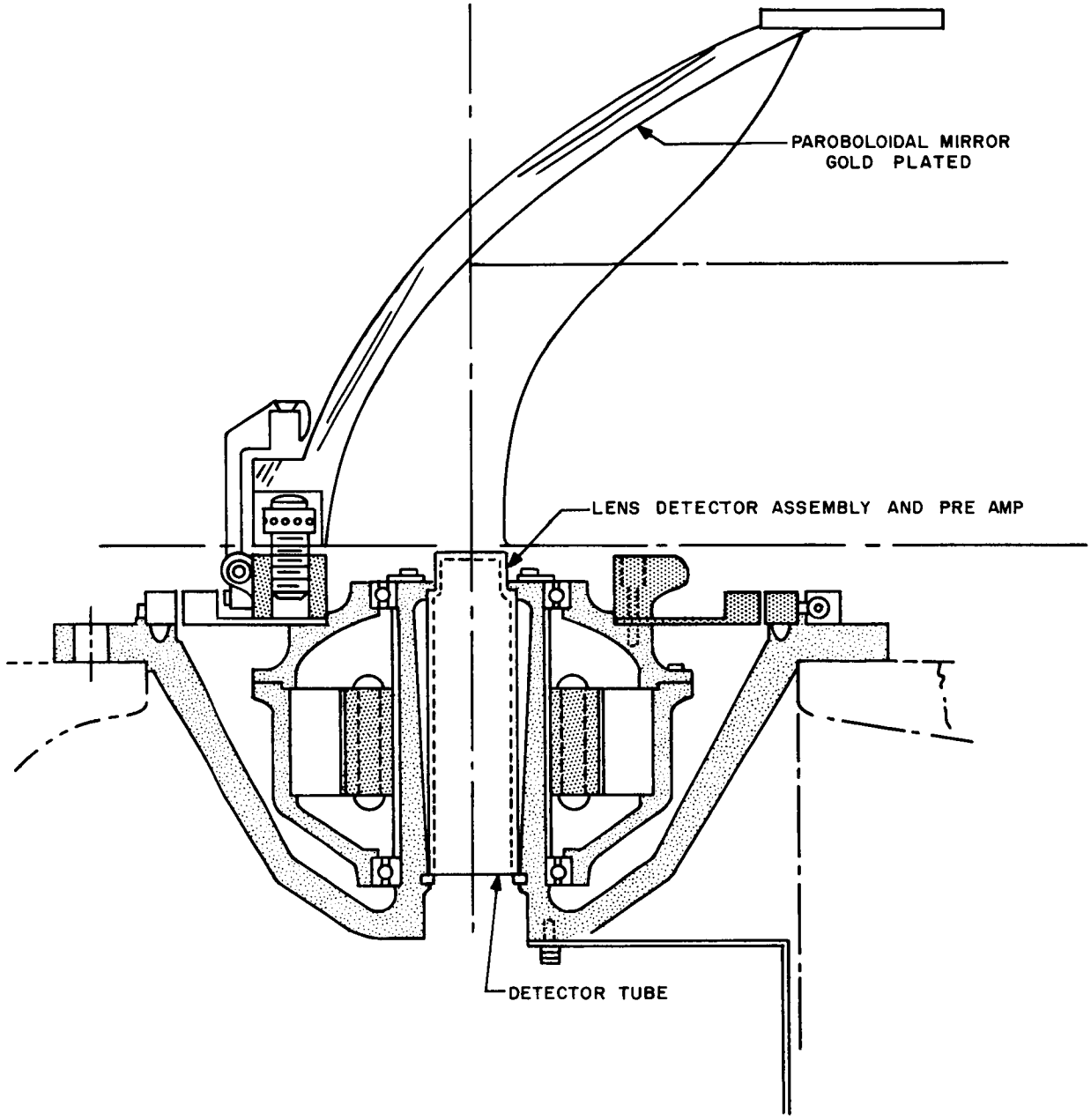


Figure II-3-30. Horizon scanner assembly

The advantages incorporated in the design are the following:

a. Unlubricated ceramic bearings will be used. These will be good for any high and low temperature which will be encountered.

b. The lens detector assembly can be a plug in unit and easily replaced.

c. The pre-amplifier will be a module type and easily replaceable. It can be mounted in the detector tube shown.

d. The parabolodial reflector is kinematically mounted, providing unlimited temperature profile.

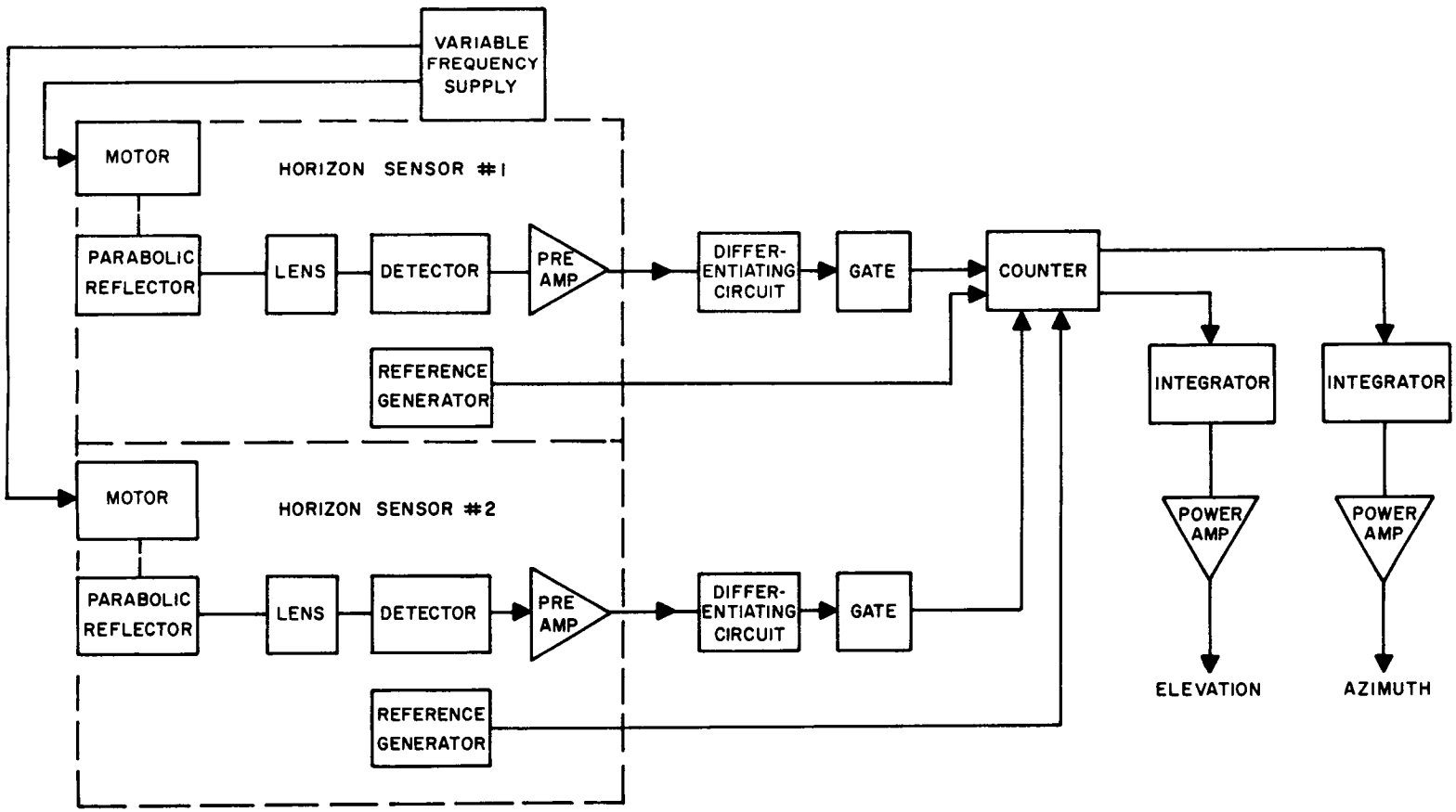
e. The materials used will be of matched temperature coefficients and of long term stability.

There will be two horizon scanners of the type mentioned above mounted so that their scanning planes are mutually perpendicular. The scanning planes will in turn be rotated either continuously or in steps about an axis parallel to the intersection of the two scanning planes. The axis mentioned above can be positioned in azimuth and in elevation by means of gimbaling. It is the error signals from the horizon scanners which control the movement of this axis. By scanning the object in two mutually perpendicular planes errors are obtained which can be resolved to drive the gimbaling and hence the intersection of the two scanning planes until this intersection is pointed toward the center of the object being scanned. The intersection of the scanning planes is rotated in steps. This permits scanning the object across several different diameter, and averages out the error due to variations in terrain.

Knowing the diameter of the object and the angle subtended at the sensor by the object, the range to the object is also determined. This may be determined by a simple trigonometric relationship.

A block diagram of the electronics is shown in Figure II-3-31. The variable frequency supply drives the motors which in turn drive the parabolic reflectors. The detector signals are fed to preamplifiers and thence to a differentiator and gate circuit. A counter is then gated when the lines of sight encounter and leave the edges of the object

[redacted]



[redacted]

Figure II-3-31. Electronics block diagram

being scanned. The reference generator signals also feed the counter and the counter output with D/A conversion becomes the error signal to drive the gimbaling of the horizon sensor configuration.

Since the scanning lenses rotate through an angle of 360 degrees it may be necessary to provide a shield between the horizon scanners and the vehicle to prevent false signals caused by heat from the vehicle reaching the detectors. It may also be possible to eliminate this problem by programming the output of the detectors so that signals are not accepted during a particular portion of the scan cycle.

The error due to noise using cooled detectors (space environment) is expected to be ±15 seconds of arc. Other errors are expected to be ±15 seconds of arc for an overall error of ±30 seconds of arc.

In addition to this instrument error there will be an error when scanning the earth due to variations in the horizon due to the tropopause. This variation is estimated to be ±6 miles and, at a range of 20,000 miles, causes an error of approximately one minute of arc. This error is a function of altitude and tends to decrease as mission accuracy requirements increase. This error may also be reduced by data processing bias because the state of the tropopause may be known a-priori.

Control Loop for the IR Horizon Scanners

The function of this control loop is to sample the output from the scanner's detectors, and by use of this information, to drive the D&E axes gimbals until the F axis is pointed directly to the center of the body being scanned. When the F axis is pointed toward the body's center, the D&E angle resolvers will read out the body's azimuth and elevation angles with respect to the stellar reference coordinates.

A generalized block diagram of the proposed control loop follows in Figure II-3-32. Following Figure II-3-31 is Figure II-3-32 which shows the geometry of the scanning motion with respect to the target.

With frequent reference to Figures II-3-32 and II-3-33, the operation of this loop may be described as follows.

The E&D optical systems will give an electrical output when they cross the target (See Figure II-3-33). For original aquisition of target it will be best to orient the

[redacted]

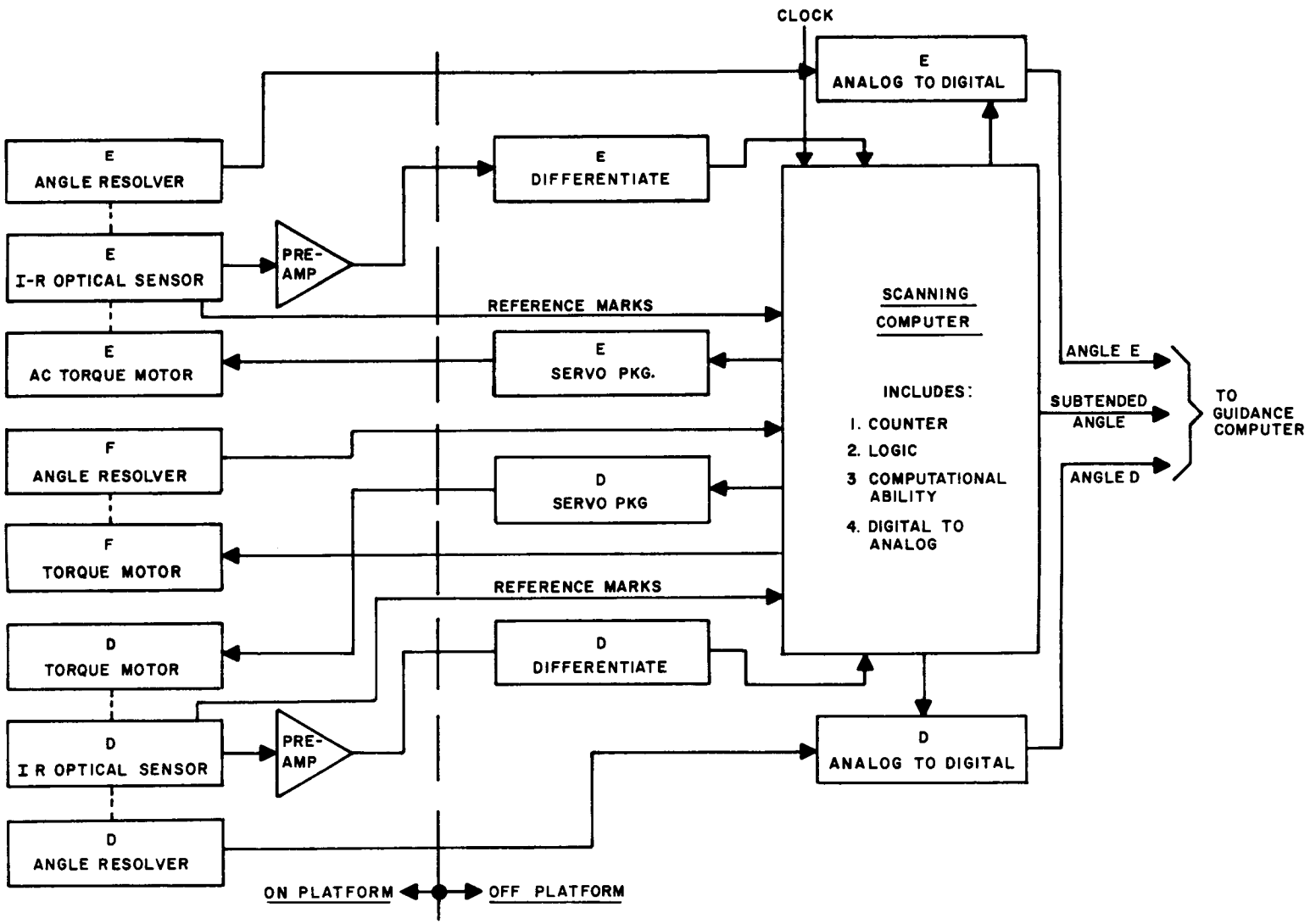


Figure II-3-32. Controls for I-R horizon scanners

[redacted]

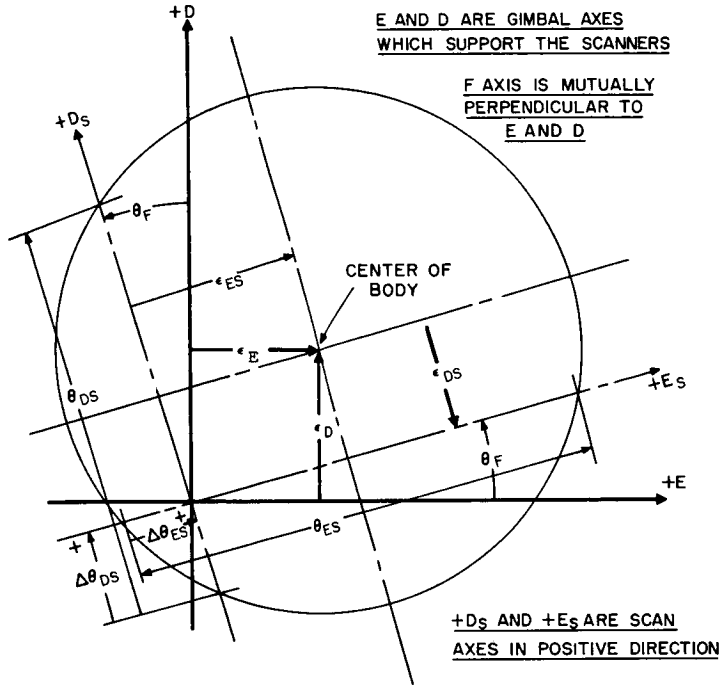


Figure II-3-33. Geometry of a body scan by horizon sensors

gimbals such that the target falls within the sweep path of the optical plane. For general navigational sightings during the APOLLO mission, the desired line of sight to the target will be known within the accuracy of the scanning field of view. However, if for some reason, no information concerning the position of target is available, a search mode can very conveniently be implemented by commanding the angle O_F to proceed in steps until the scanners acquire the target. As soon as a scanner sweeps across the target, the loop will automatically take hold and drive the gimbals so that azimuth and elevation to the target can be read from the D&E angle readouts.

To explain a nulling cycle, refer to Figure II-3-32. Assume that the sweep axes are oriented as shown in Figure II-3-32.

E_S and D_S indicate the positive scan directions across the target. E and D give the orientations of the gimbals axes to the target, and also show that E_S and D_S are related to E and D through the angle O_F, which is read from angle resolver F (5 min accuracy). Note that the F axis is mutually perpendicular to E & D at all times.

Consider now the scan motion of one optical beam. For clarity, refer to Figure II-3-33 and follow the scanning beam which moves on the E_S path. As this beam moves onto the body (lower left hand corner) it generates an electrical signal in the bolometer detector. This signal remains on the bolometer until the scan beam leaves at the right part of the body. In addition, a reference mark is generated as the beam crosses the F axis (point 0 in Figure II-3-33). If the bolometer output signal is differentiated, a pulse will be generated at the time the scan beam crosses the surfaces of the body. In addition, if the target body is partially illuminated (such as the Moon may well be) a pulse will also be generated when the scan beam crosses the line of differential illumination. Presently, it appears that special logic circuitry will be necessary to perform the pulse recognition function. For example, since the scanned target can never subtend more than 180 degrees, the initial scan of the body will determine the targets subtended angle by noting the pulses which indicate the maximum subtended angle that is less than 180 degrees. This measured value of subtended angle can then be used in conjunction with logic circuits to prevent the undesired pulses from falsely triggering the pulse counter.

The purpose of the counter (part of the scanning computer) is to measure the various angles by use of the pulse information which arrives from the optical system. Referring once again to Figure II-3-33, and following the E_S scan beam, the angles $\Delta\ \Theta_{ES}$ and Θ_{ES} will be determined by the counter. The counter will determine these angles by counting the clock pulses between the signal pulses which come from the optical system. These angles will be in digital binary form.

The other channel will also perform the same function, and similiarly will determine $\Delta\ \Theta_{DS}$ and Θ_{DS} from the D_S scan motion.

Since the purpose of this control loop is to center the scan pattern on the body, the desired error functions to feed back to the torque motors are as follows:

Along the E_S & D_S axes the error functions are

$$ES = 1/2\ \Theta_{ES} - \Delta\ \Theta\ ES \qquad \text{(Q-1)}$$

$$ES = 1/2\ \Theta_{DS} - \Delta\ \Theta_{DS} \qquad \text{(Q-2)}$$

The scanning computer will be required to compute these error functions by use of the pulse counter information giving Θ_{ES}, $\Delta\Theta_{ES}$, and $\Delta\Theta_{DS}$. The error functions (ϵ_{ES} and ϵ_{DS}) must then be converted from digital to analog form.

However, these error functions referenced to the E_s and D_s axes must be resolved into their respective components on the E and D axes before correction can be applied. This is necessary because the E and D axes are the control axes of the optical system.

Since the angle Θ_F is available, (F - Resolver), the required error functions can be generated by the following equations:

$$\epsilon_E = \epsilon_{ES} \operatorname{Cos} \Theta_F - \epsilon_{DS} \operatorname{Sin} \Theta_F \qquad \text{(Q-3)}$$

$$\epsilon_D = \epsilon_{ES} \operatorname{Sin} \Theta_F + \epsilon_{DS} \operatorname{Cos} \Theta_F \qquad \text{(Q-4)}$$

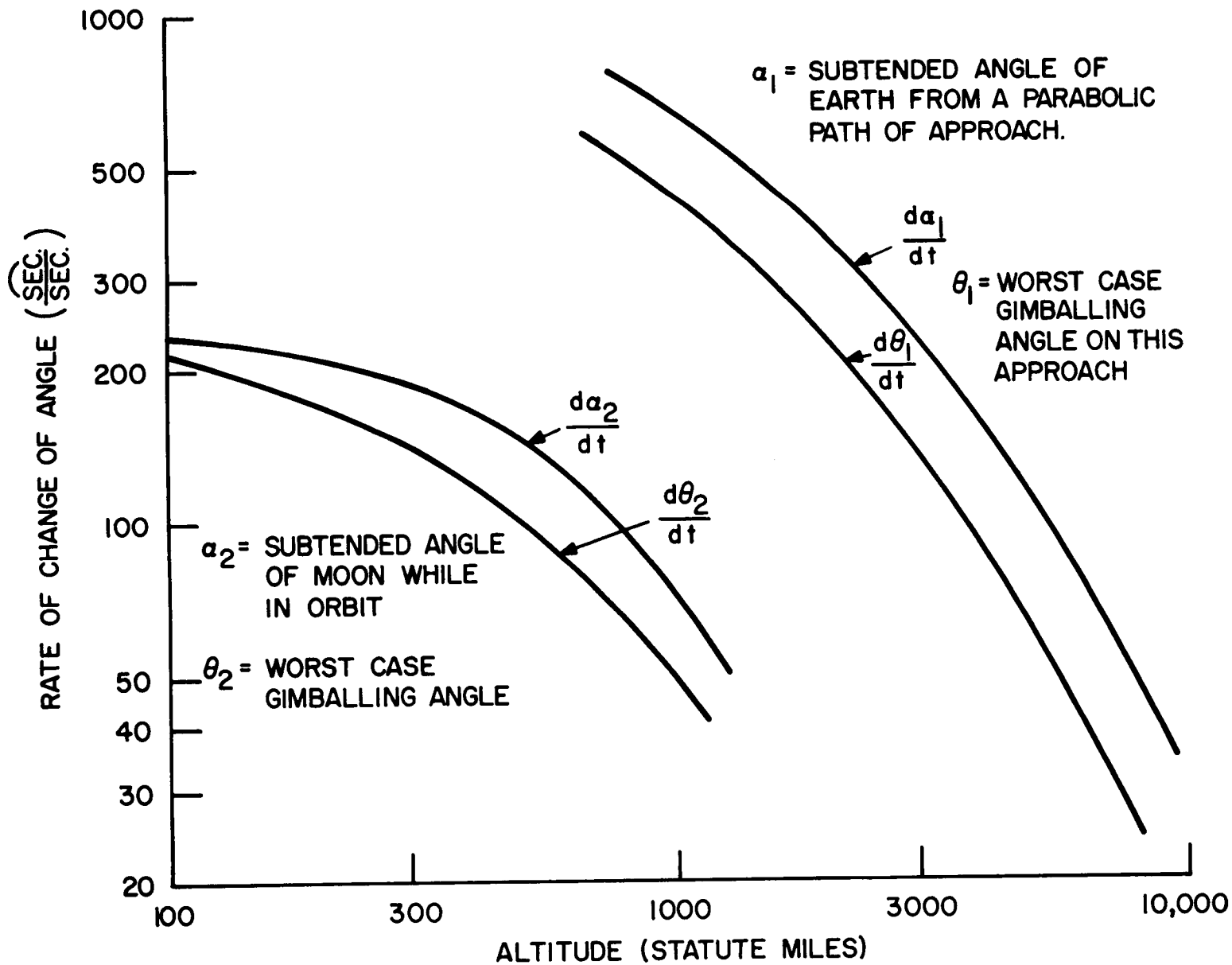
The error functions of Equations (Q-3) and (Q-4) are sent to the servo amplifiers which amplify and otherwise modify them such that they are in form to drive the torque motors on the D and E axes.

This complete loop is shown in block form in Figure II-3-32.

Another minor function of the scanning computer is the control of input and output data. Special logic blocks will be employed to send out the angles Θ_{DS} and Θ_{ES} after every nulling operation only. That is, the horizon scanners will acquire a target, then track and null to its center as previously described, and then read out the various angles to the guidance computer from the scanning computer and the D and E angle resolvers. Also, at this time a step change in the angle Θ_F will be made. The process will then be repeated. The purpose of changing the angle Θ_F is to smooth out the errors due to surface irregularities in the target. During the course of a navagational fix, the angle Θ_F will be rotated through 360 degrees or more in order to obtain good smoothing data.

Rates of change of tracking angles are not expected to present any problem. The maximum rate of change of any tracking angle will not exceed 4 min./sec. for either the earth approach (down to 4,000 miles) or while in the Moon orbit. Rates of change of tracking are plotted vs altitude on Figure II-3-34.

[redacted]



[redacted]

Figure II-3-34. Rate of change of angles vs altitude

3.6 SYSTEM SUMMARY

Figure II-3-35 illustrates the weight and power requirements for the Navigation, Guidance and Control subsystems.

The Inertial Platform and Electronics shown includes no redundancy. Peak power occurs during thrusting and more violent maneuvers, such as can be expected during re-entry.

Peak power is consummed by the celestial sextant during slewing or the acquisition phase. The 40 watt average power is consumed mostly by the extended body tracker scanning mechanism.

Peak power is consumed by the digital computer during periods of fast solution rates. It is expected that slower clock rates during midcourse will allow the average power to be reduced to 80 watts.

Attitude control power is not consumed during launch or re-entry.

Re-entry flight control power is consumed only during re-entry.

Power for manual equipment is required only during manual fix taking. It can be considered as sporadic with very low duty cycle.

A graph of peak and average power consumption vs mission phase is shown in Figure II-3-36.

CONFIDENTIAL

Subsystem or Component	Weight Lbs.	Peak Power Watts	Average Power Watts	Location	Remarks
Inertial Platform and Electronics	60	200	50	R/V	Includes Accelerometers and Star Trackers
Celestial Sextant and Electronics	45	120	40	Outside	Assembly of 2 Star Trackers and 1 Extended Body Tracker
Digital Computer					Delay Line Memory
	85	160	80	R/V	(Non-Redundant)
	250	460	230		(Redundant)
Attitude Control	230	150	150	M/M	Includes Propulsion - Fuel + Tank = 190#
Re-entry Flight Control	100	100	100	R/V	Roll Jet System Fuel Hot Gas Hydraulic Flap Actuator
Manual Navigation Equipment	60	100	100	M/M	Camera on Stabilized Platform
Navigation Aids	25	30	30	R/V	Radar Beacon for Radio Backup
Spare Parts	50				

Figure III-3-35. Navigation Guidance and Control Subsystem Summary

CONFIDENTIAL

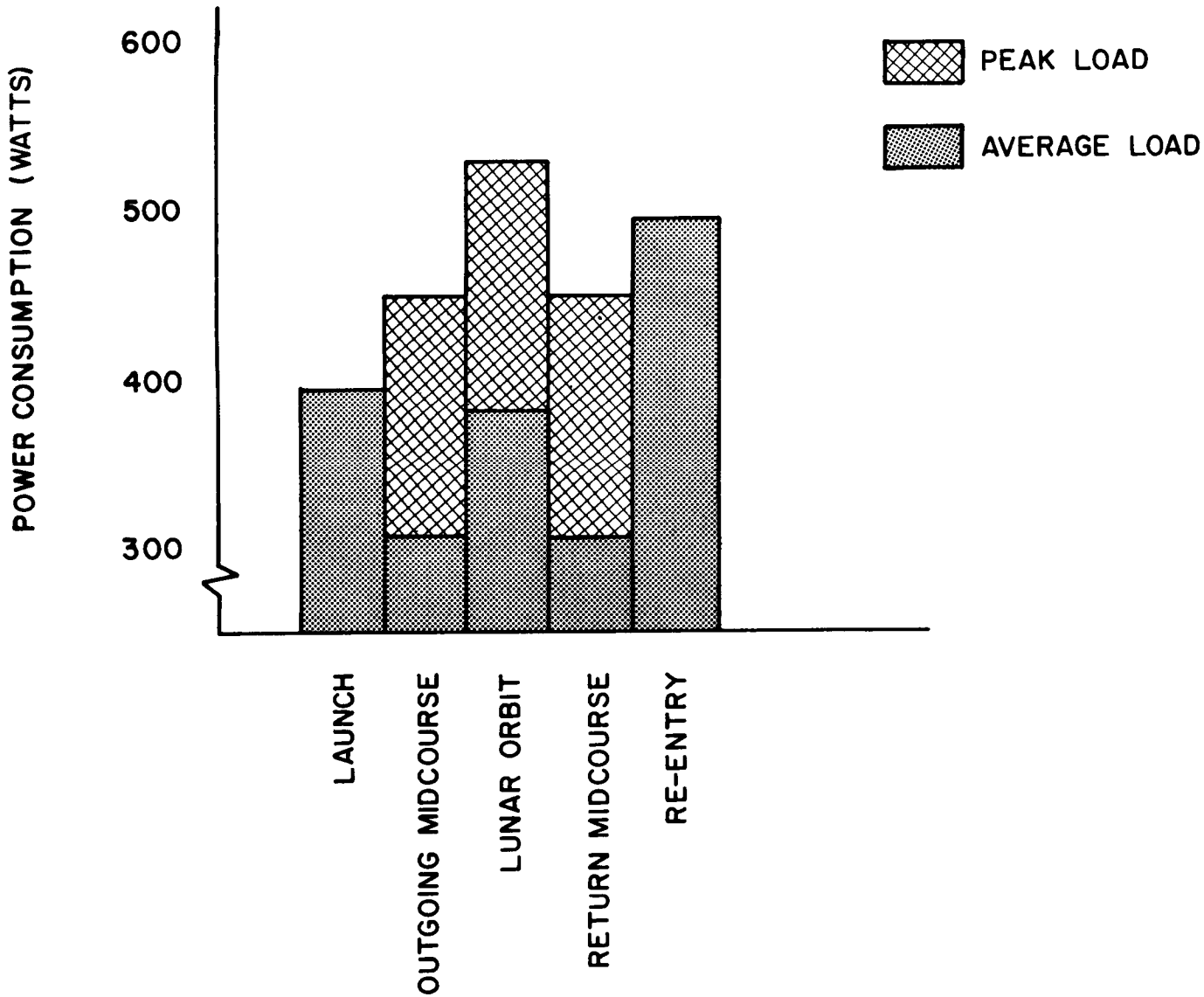


Figure II-3-36. Average Power Consumption vs. mission phase

4.0 Crew Functions

4.1 INTRODUCTION

The following two guidelines influence the basic approach to the design of the navigation, guidance and control subsystem for the APOLLO vehicle:

1. The spacecraft is designed for manned operations only.
2. Primary command is on-board the vehicle.

When basing the design of the navigation, guidance and control subsystem on these ground rules, it is of the utmost importance to determine how the crew can be used to the greatest advantage in simplifying the resulting system and in increasing the reliability of its operation. As a result of these considerations and the two basic guidelines, the following statements can be made which summarize the philosophy used in determining the crew's functions with respect to the navigation, guidance and control subsystem:

1. In general, the crew performs command functions rather than control functions. Thus the crew:
   a. monitors the operation of the navigation, guidance and control subsystem, by observing appropriate displays.
   b. makes appropriate decisions based on all available information.
   c. initiates the appropriate course of action but does not necessarily carry it through as a continuous control function.

However, the crew always has the capability of overriding the automatic equipment (except in the case of abort) by inserting the proper commands. In addition, the crew can manually take over many of the navigation, guidance and control subsystem functions which are normally performed automatically such as attitude control or flight control during re-entry.

2. The crew performs maintenance and repair operations with respect to navigation, guidance and control equipment.

3. The crew makes navigational observations using manual equipment.

4. The crew performs navigation and guidance computations based on information obtained from manual and automatic navigation equipment.

Thus the crew on the APOLLO mission performs in essentially the same manner as does the crew of a large modern airplane which is equipped with displays, automatic flight control equipment and communication links. The actual detailed crew tasks will evolve largely as a function of the navigation, guidance and control subsystem design.

4.2 CONSIDERATIONS AFFECTING NAVIGATION, GUIDANCE AND CONTROL SUBSYSTEM DESIGN

It is appropriate at this point to consider in more detail the basic philosophy used in determining crew functions with respect to the navigation, guidance and control subsystem and then consider specific tasks and decisions which involve the crew during the various phases of an APOLLO mission. The basic philosophy used in determining crew functions is summarized in the following statements:

1. The capabilities of human beings in the spaceship environment determine the extent to which the crew can participate in tasks relating to the operation of the navigation, guidance and control subsystem. The area of human performance capabilities is covered in considerable detail in Volume V entitled "Human Factors". The tasks which have been assigned to the crew, as well as the design of the displays and controls for the navigation, guidance and control subsystem, have been considered in the light of presently available information on human performance to assure that the assigned tasks are compatible with our present knowledge of human capabilities in the spacecraft environment. A detailed listing of crew tasks, including those relating to the navigation, guidance and control subsystem, is presented in the following section. The design of navigation, guidance and control subsystem

components and parameters, such as displays, attitude control acceleration levels, and manual controls, specifically considers human capabilities.

2. Crew participation in subsystem operation is included when greater subsystem simplicity and reliability results. Thus crew participation leads to a reduction of automatic sequencing and eliminates the need for ground command and control functions (except possibly for re-entry when ground control in emergency situations may be desirable). Therefore, the crew is used for making gross attitude changes, for the initial gross alignment of star-trackers, and for determining the mission profile.

3. The operation of the navigation, guidance and control subsystem is made automatic in many respects in order to relieve the crew of tedious and/or repetitive tasks such as taking numerous navigation fixes, performing tasks which must be done very accurately and at high speed such as thrust vector control during velocity corrections, and performing complex tasks such as complicated data processing. However, the system is designed so that in emergencies the crew can manually perform certain critical tasks which would normally be accomplished automatically, such as tracking and thrust vector control. The crew always has the capability of overriding the automatic equipment except in the case of abort.

4. The probability of achieving a successful mission can be increased by maintenance and repair procedures. Thus, navigation guidance and control subsystem components are designed so that critical parts can be replaced with spare parts carried along on the mission. Self checking tests and manual tests are devised, where possible, to check on component operation. Redundancy of equipment in critical areas opens additional opportunities for crew improvisation in the event of component failures. The crew can turn equipment off, when its operation is not required, in order to increase its useful life. The inertial platform, for example, can be turned off for long periods of time during the mission. A detailed consideration of maintanance, reliability and system design is presented in Volume V, Chapter II under "Man -- Machine Integration".

5. Emergency conditions can confront the crew with respect to the operation of the semi-automatic navigation, guidance, and control subsystem. This is true even though equipment is well designed and made redundant where required. Therefore, the navigation, guidance and control subsystem includes manually operated navigation equipment as a back-up for the semi-automatic navigation equipment. Actually, this provision is desirable even under normal circumstances since the crew can independently check on the normal semi-automatic mode of subsystem operation.

4.3 HUMAN FACTORS ASPECTS OF NAVIGATION, GUIDANCE AND CONTROL SUBSYSTEM DESIGN

Before considering some of the details with respect to the human factors aspects of the navigation, guidance and control subsystem design, it is desirable to first briefly consider the relationship of the crew with the various equipments involved in the navigation, guidance and control functions. This information is shown, in block diagram form in Figure II-4-1.

4.3.1 Displays and Manual Controls

The introduction of man into the operation of the navigation, guidance and control subsystem opens a large area of investigation with respect to manual controls and the presentation of information. The design of the controls and displays, as well as the selection of crew tasks, is influenced by the capabilities of man in the spaceship environment. This aspect of the problem is covered in Volume V, Chapter II. The material which follows provides details on the controls and displays relating to navigation guidance and control. This material is also presented in Volume V, Chapter II "Man -- Machine Integration".

Since all crew tasks influence the design of the controls and displays, a listing of crew activities as an approximate function of time, including navigation, guidance and control tasks, is given first. (See Table II-4-I.) This is followed by functional allocation charts for the navigation, guidance and control and the rocket power subsystems which essentially describe what monitoring and control tasks are required and how they are assigned during each mission phase, e.g. to the crew, the automatic systems or the ground.

[redacted]

This is followed by a detailed description of the controls and displays related to navigation, guidance and control wherein the nature of the suggested hardware, the scale factors, and interrelationship among controls and displays are described.

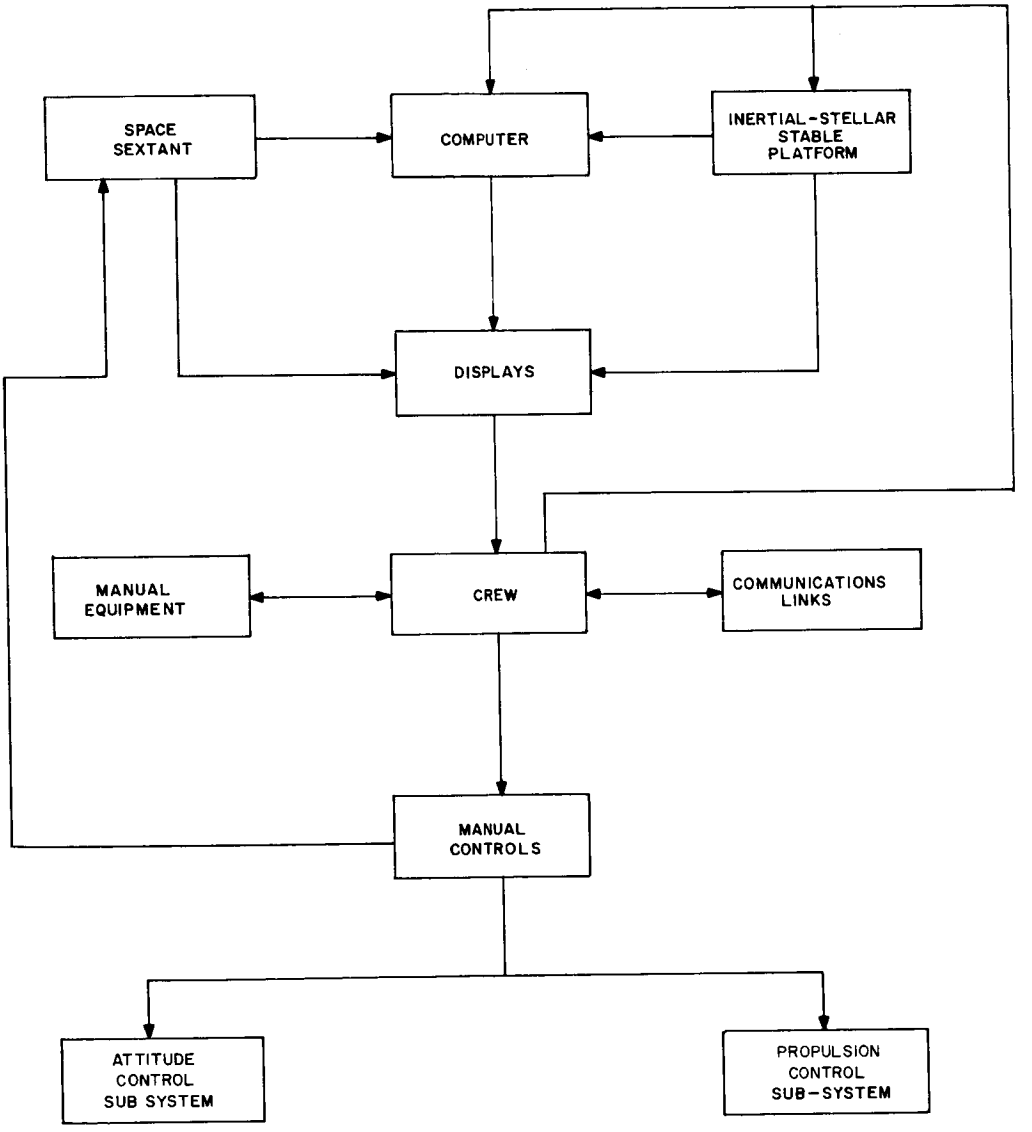
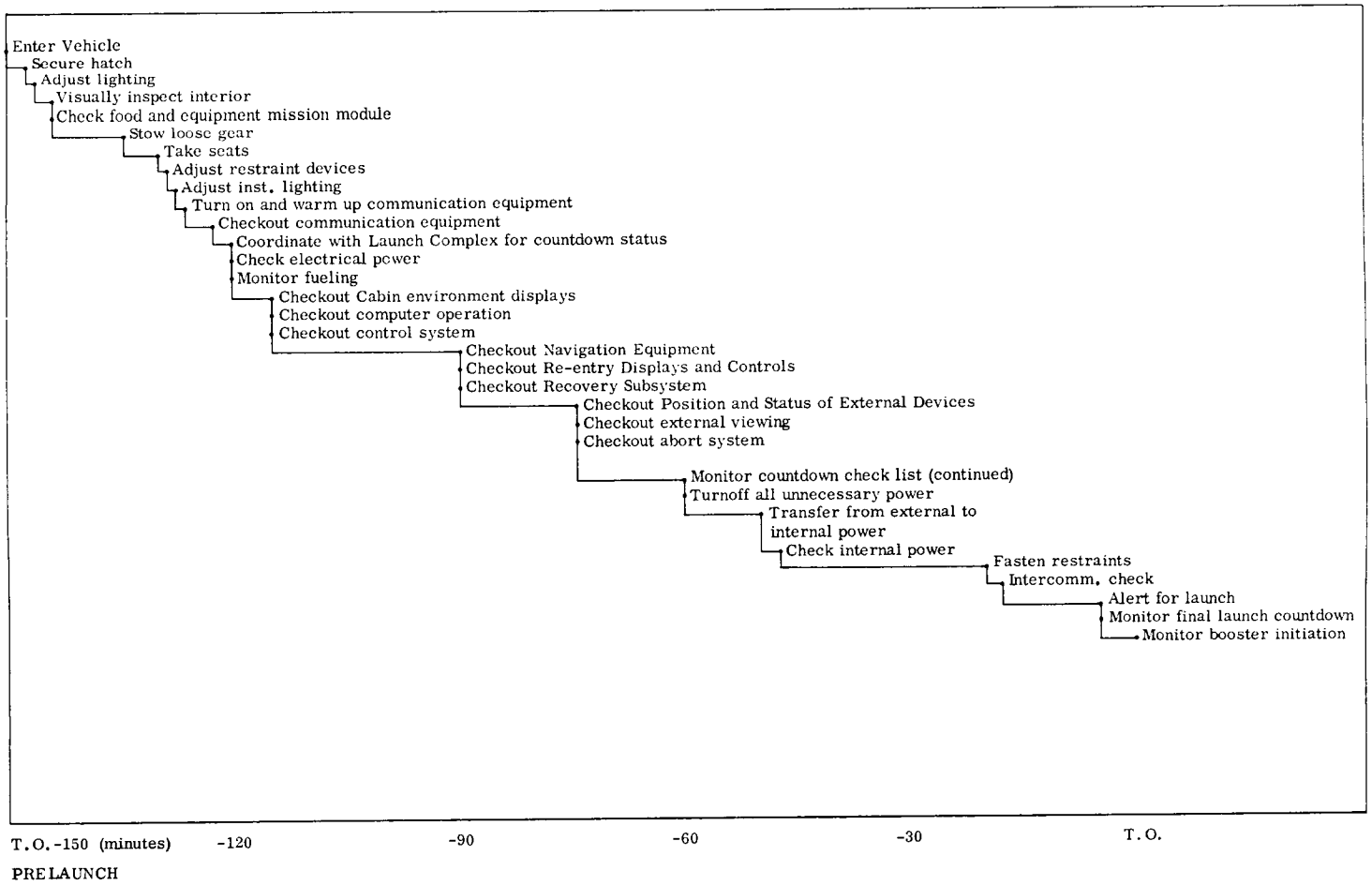


Figure II-4-1. Relationship of crew with navigation, guidance and control equipment

[redacted]

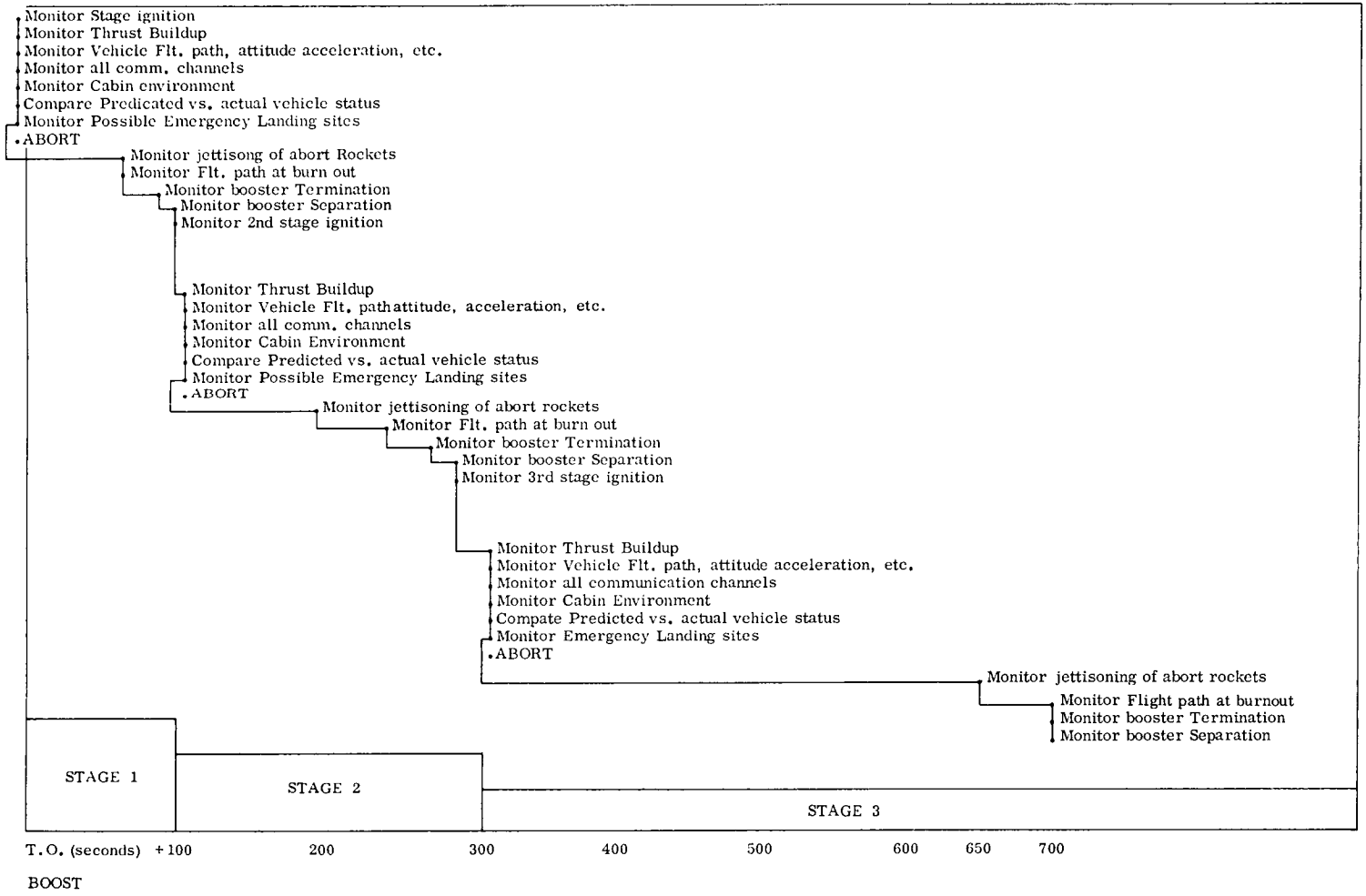
[redacted]

TABLE II-4-I. UPDATED TIME LINE ANALYSIS



[redacted]

TABLE II-4-I (Continued)



[redacted]

[redacted]

TABLE II-4-I (Continued)

Orient Vehicle attitude
Monitor Vehicle Flt. path
Extend antenna and peak tune
Extend solar collector
Inflate solar collector
Loosen restraints
Inspect Vehicle interior

Check communication subsystem
- UHF
- VHF
- TV
- Telemetry
- intercom

Check Rocket power subsystem
- Rocket engine (actual vs. Programmed)
  1. pressure
  2. condition
  3. gimbal position
- Check Fuel and Oxidizer System (actual vs Programmed)
  1. Flow rate
  2. Total consumed
  3. Total remaining
  4. Time to Pressurize
  5. Fuel - oxidizer pressure

Check secondary power
1. Monitor input from solar modules
2. Capacity-fuel cell batteries
   - hydrogen pressure
   - oxygen pressure
3. Output of fuel cell batteries-solar modules
   - voltage
   - current
4. Circuit breakers

Check environmental subsystem
1. Lighting
2. Air conditioning
   - gas supply (primary) flow rate & quantity
   - gas supply (emergency) flow rate & quantity
   - Temperature (atmospheric and wall temperature)
3. Cabin pressure
   - monitor - regulate total pressure
   - p_{O_2}
   - p_{H_2}
   - p_{CO_2}
   - Leak detection
4. Toxic gas
   - Sensors
   - Removal system
5. Fire detection & control system
6. Radiation and ion balance

T.O. 13+00 (hours) 14+00 15+00 16+00 17+00

MIDCOURSE (60 hrs. ± 5)

[redacted]

[redacted]

TABLE II-4-I (Continued)

Check out and align astro-tracker
- Orient vehicle for astro-tracker fix
  - Smooth, correct and plot astro-tracker fix
    - Prepare for manual check of position
      - Make celestial observations
        - Plot celestial observations
          - Determine need for correction
          - Monitor predicted vs. actual position
          - Determine time for correction
          - Crew alerted for course - change, in seats, and restrained
            - Prepare for velocity correction (check list)
              1. Select VELOCITY CORRECTION on sequence display
              2. Check power distributed panel observing buss load displays
              3. Select COMPUTER or MANUAL input thrust control system if MANUAL
                 a. Select DATA LINK on computer readout or obtain parameters by voice communications
                 b. Set magnitude
                 c. Set azimuth angle
                 d. Set elevation angle
                 e. Set time to fire
                 f. Select ROCKET FIRE on time-to-go mode selector
              4. Check thrust level on engine desired
                 a. Select "THRUST CHAMBER" mode on commanders side stick
                 b. Observe chamber position on display
                 c. Manipulate stick to obtain normal pominal position
              5. Select THRUST CONTROL on command needle mode selector
              6. Select THRUST CONTROL on flight control mode selector
              7. Observe command needles, attitude display, vehicle rate display
              8. Pressurize desired fuel and oxidizer tanks
              9. Open fuel valves from pressurized tanks
              10. Select "LUNAR AIM POINT" on computer readout
              11. When time-to-go approaches zero, turn on ignition system
              12. At 5 seconds to go, actuate normally open safety switch on commander's left armrest
              - As rocket fires, observe:
                a. Thrust
                b. Total velocity on computer readout
                c. Command needles on attitude indicator
                d. Body axis rate display
              - Release safety switch, if automatic cut off fails, or if for any reason premature cut off is desired
              - Extend solar collector
              - Select SOLAR CELL or command needles
              - Select SOLAR CELL on flight control system
              - Check vehicle re-orientation
              - Check solar cell out put on signal strength meter
              - Check charge rate of individual fuel cell batteries
              - Select antenna on commanders side stick
              - Observe antenna position on signal strength display
              - Rotate antenna about one axis to achieve maximum signal strength
              - Adjust gain of Signal Strength display as desired
              - Rotate antenna about other axis to achieve maximum signal strength
              - Repeat about first axis
              - Inform ground station of all course changes
              - Release restraints
              - Resume normal operating mode - Eat, personal hygiene, maintenance as required, etc.
              - TIME UNDETERMINED

USE PROCEDURES OUTLINED ABOVE FOR ALL MIDCOURSE CHANGES EXCEPT FOR FINAL MIDCOURSE CONNECTION PRIOR TO LUNAR ORBIT.

T.O. + 1900 (hours) T.O. + 1800 19+00 20+00 21+00 22+00 23+00

MIDCOURSE (Continued)

[redacted]

[redacted]

TABLE II-4-I (Continued)

- Orient vehicle for astro-tracker fix
- Smooth, correct and plot astro-tracker fix
- Prepare for manual check of position (sextant)
- Make celestial observations
- Plot celestial observations
- Monitor predicted vs. actual
- Check position velocity
- Checkout functioning of all lunar sensing and experimental equipment
- Calculate moon orbit parameters
- Select LUNAR ORBIT on sequence display
- Select LUNAR ORBIT on computer readout
- Check power distribution panel
- Check secondary power system
- Evaluate fuel requirements to achieve desired orbit
- Crew in seats and restrained
- Initiate velocity change to achieve orbit (follow check list described previously)
- Check program for lunar circular orbit

T.O. 57+30 58+00 59+00 60+00-5 hrs.

FINAL MIDCOURSE CORRECTION PRIOR TO LUNAR ORBIT

[redacted]

[redacted]

4.3.1.1 FUNCTIONAL ALLOCATION AND CONTROL-DISPLAY PARAMETER CHARTS

This sub-section contains Functional Allocation Charts (Tables II-4-II through II-4-III) and Control Display Parameter Charts (Tables II-4-IV and II-4-V).

The Functional Allocation charts present, in summary form for each subsystem, all the vehicle functions which relate to the crew, and describe for each mission phase (1) how the function is allocated to the vehicle system, e.g., M = man, G = ground, S = system, and (2) the nature of crew participation in this function, e.g. (M) monitor or C = control.

For each subsystem the C-D Parameter Charts define the implementation of these functions in terms of hardware. Presented on these charts is the nature of the hardware in terms of types of instrument and control, scales required, accuracy of reading, power, and weight figures, and the relationship of hardware items to one another.

4.3.1.2 Individual Panel Layout

This section of the report is designed to show and describe the specific controls and displays which are provided the crew in order for them to perform their respective functions. This is accomplished by describing, for each subsystem, the panel and the displays located on it.

4.3.1.2.1 ROCKET POWER PANEL

The lower right corner of the commander's panel contains the thrust and propellant quantity displays, Figure II-4-2. Individual pressurization and cutoff valves are provided for each compartment of the propellant tanks. Separate oxidizer and fuel pressurizing system mode selectors provide individual control and/or manual override of either system. The tanks would normally be left unpressurized to minimize boil-off.

Prior to rocket firing the automatic pressurizing mode would be selected and the tank would come up to, and remain at operating pressure. If the pressure fails to rise or overshoots the normal value, the tank pressure may be controlled by actuating the pressurization switch from "off" to "manual". The cutoff valves beneath the quantity

[redacted]

TABLE II-4-II — FUNCTIONAL ALLOCATION CHART — NAVIGATION AND GUIDANCE

SUBSYSTEM 1 ▷ Navigation & Guidance	MISSION PHASE									
FUNCTIONAL ANALYSIS — SYMBOLS: (C) = CONTROL, G = GROUND, (M) = MONITOR, M = MAN, S = SYSTEM	LAUNCH		SECONDARY MISSION EARTH ORBIT		MID-COURSE TO MOON EARTH		TRANSFER TO MOON EARTH		RE-ENTRY	
	(C)	(M)	(C)	(M)	(C)	(M)	(C)	(M)	(C)	(M)
A. Position & Velocity Parameters										
1. Orbital Parameters (Earth-Moon Reference)										
a. Input Computational Source Selection	(M)	(M)	M	M			M	M		
1. Computer-platform	M	M	M	M			M	M		
2. Data Link	M	M	M	M			M	M		
b. Apogee	SM	M	SM	M			SM	M		
c. Perigee	SM	M	SM	M			SM	M		
d. Inclination Angle	S	M	S	M			S	M		
e. Period	SM	M	SM	M			SM	M		
f. Ascending Node	S	M	S	M			S	M		
g. Central Angle of Vehicle	S	M	S	M			S	M		
h. Central Angle of Perigee	SM	M	SM	M			SM	M		
2. Distance-Velocity Parameters (To Earth - To Moon)										
a. Input Computational Source Selection	M	M	M	M	M	M	M	M	M	M
1. Periscope - Time			M	M	M	M	M	M		
2. Horizon Seekers - Time			M	M	M	M	M	M		
3. Altimeter (Radar) - Time	M	M	M	M	M	M	M	M	M	M
3. Predicted Data - At Some Future Time										
- At Occurence of Some Future Event										
a. Orbital Parameters	S	M	S	M			S	M		
b. Distance-Velocity Parameters (Earth Reference)										
1. Min.-Max. Range as Function of Vehicle									S	M
capabilities, initial conditions, "G",										
temperature and skip limits										

[redacted]

TABLE II-4-II — FUNCTIONAL ALLOCATION CHART — NAVIGATION AND GUIDANCE (Cont)

SUBSYSTEM 1 Navigation & Guidance	MISSION PHASE									
FUNCTIONAL ANALYSIS — SYMBOLS (C) = CONTROL, G = GROUND, (M) = MONITOR, M = MAN, S = SYSTEM	LAUNCH		SECONDARY MISSION EARTH ORBIT		MID-COURSE TO MOON EARTH		TRANSFER TO MOON EARTH		RE-ENTRY	
	(C)	(M)	(C)	(M)	(C)	(M)	(C)	(M)	(C)	(M)
2. Landing Location of Vehicle on Earth									S	M
3. Location of Alternate Landing Sites on Earth									S	M
c. Predicted Miss Distance - Velocity Error	S	M			S	M	S	M		
Parameters (From Desired Position and										
Velocity For Initiating Moon Orbit and										
Re-entry) (Earth-Moon Reference)										
1. Input Computational Source Selection	M	M	M	M	M	M	M	M		
a. Computer and Astrotrackers	M	M	M	M	M	M	M	M		
b. Computer and Horizon Seekers	M	M	M	M	M	M	M	M		
c. Theodolite	M	M	M	M	M	M	M	M		
1. Computer Assist	M	M	M	M	M	M	M	M		
2. No Computer Assist	M	M	M	M	M	M	M	M		
d. Data Link	M	M	M	M	M	M	M	M		
2. Predicted Miss Distance and Allowable Limits										
a. X Axis	SM	M			SM	M	SM	M		
b. Y Axis	SM	M			SM	M	SM	M		
c. Z Axis	SM	M			SM	M	SM	M		
d. Re-entry Angle	SM	M	SM	M	SM	M	SM	M		
3. Predicted ΔV Error and Allowable Limits	SM	M			SM	M	SM	M		

TABLE II-4-II — FUNCTIONAL ALLOCATION CHART — NAVIGATION AND GUIDANCE (Cont)

SUBSYSTEM 1 ▷ Navigation & Guidance	MISSION PHASE									
FUNCTIONAL ANALYSIS (SYMBOLS: (C) = CONTROL, G = GROUND, (M) = MONITOR, M = MAN, S = SYSTEM)	LAUNCH		SECONDARY MISSION EARTH ORBIT		MID-COURSE TO MOON EARTH		TRANSFER TO MOON EARTH		RE-ENTRY	
	(C)	(M)	(C)	(M)	(C)	(M)	(C)	(M)	(C)	(M)
B. Comparison (Checks) of Position & Velocity Parameters										
1. Type of Data to Check, i.e., Orbital Parameters,	M	M	M	M	M	M	M	M		
Distance-Velocity Parameters, Predicted Miss										
Distance-Velocity Parameters										
2. Source of Data For Check, i.e., Astrotrackers	M	M	M	M	M	M	M	M		
Horizon Seekers, Theodolite, Data Link										
Computations, Platform										
3. Corrections (Updating Computer)										
a. Manual or Automatic Input	M	M	M	M	M	M	M	M		
C. Time To:										
1. Initiate "Safest" abort sequency	S	M								
2. Initiate Secondary Mission	S	M								
3. Initiate Moon Orbit							S	M		
4. Initiate Escape From Moon Orbit							S	M		
5. Initiate Mid-course Corrections					M	M				
6. Initiate Re-entry									S	M
7. Release Drogue Chute									SM	M
8. Release Reefed Main Chutes No. 1 and No. 2									SM	M
9. Unreef Main Chutes No. 1 and No. 2									SM	M
10. Release Heat Shield									SM	M
11. Expand Impact Bags									SM	M

TABLE II-4-II — FUNCTIONAL ALLOCATION CHART — NAVIGATION AND GUIDANCE (Cont)

SUBSYSTEM I ▷ Navigation & Guidance	MISSION PHASE									
FUNCTIONAL ANALYSIS — SYMBOLS: (C) = CONTROL, G = GROUND, (M) = MONITOR, M = MAN, S = SYSTEM	LAUNCH		SECONDARY MISSION EARTH ORBIT		MID-COURSE TO MOON EARTH		TRANSFER TO MOON EARTH		RE-ENTRY	
	(C)	(M)	(C)	(M)	(C)	(M)	(C)	(M)	(C)	(M)
D. Thrust Corrections										
1. Manual or Automatic Insert of Required ΔV					M	M	M	M		
2. Program Selection										
a. Min. Time					M	M	M	M		
b. Min. Fuel					M	M	M	M		
3. Required ΔV										
a. Time to Initiate					M	M	M	M		
b. Magnitude					SM	M	SM	M		
c. Direction					SM	M	SM	M		
d. Thrust Level					M	M	M	M		
E. Attitude Control										
1. 3 Axis Attitude										
a. Present	S	M	SM	M	SM	M	SM	M	SM	M
b. Command	S	M	SM	M	SM	M	SM	M	SM	M
2. 3-Axis Attitude Rates	S	M	SM	M	SM	M	SM	M	SM	M
3. Attitude Reference Source										
a. Platform			M	M	M	M	M	M	M	M
b. Computer			M	M	M	M	M	M	M	M
c. Periscope			M	M	M	M	M	M	M	M
d. Horizon Seekers			M	M	M	M	M	M	M	M

[redacted]

[redacted]

TABLE II-4-II — FUNCTIONAL ALLOCATION CHART — NAVIGATION AND GUIDANCE (Cont)

SUBSYSTEM I ▷ Navigation & Guidance	MISSION PHASE									
FUNCTIONAL ANALYSIS — SYMBOLS: (C) = CONTROL, G = GROUND, (M) = MONITOR, M = MAN, S = SYSTEM	LAUNCH		SECONDARY MISSION EARTH ORBIT		MID-COURSE TO MOON EARTH		TRANSFER TO MOON EARTH		RE-ENTRY	
	(C)	(M)	(C)	(M)	(C)	(M)	(C)	(M)	(C)	(M)
F. Aerodynamic Control										
1. Present "G" and Limits									SM	M
2. Total Heat Absorbed and Limits									SM	M
3. Present Skin Temperature and Limits									SM	M
4. Skip Limits									SM	M
5. Altitude Rate									SM	M
a. Acceleration									SM	M
6. Obstacle Avoidance and Terrain Clearance									M	M
G. Celestial and Lunar Observations										
1. Periscope			M	M	M	M	M	M		
a. Focus			M	M	M	M	M	M		
b. Lens Select			M	M	M	M	M	M		
c. Filters			M	M	M	M	M	M		
2. TV Cameras			M	M	M	M	M	M		
3. Scientific Observations of Moon's Surface										
a. Spectrometer							M	GM		
b. Camera										
c. 1. Area Mapping							MS	GM		
2. Steroscopic Photo Mapping							MS	GM		

[redacted]

[redacted]

TABLE II-4-III — FUNCTIONAL ALLOCATION CHART — ROCKET POWER

SUBSYSTEM 2 Rocket Power	MISSION PHASE									
FUNCTIONAL ANALYSIS SYMBOLS (C) = CONTROL, G = GROUND, (M) = MONITOR, M = MAN, S = SYSTEM	LAUNCH		SECONDARY MISSION EARTH ORBIT		MID-COURSE TO MOON EARTH		TRANSFER TO MOON EARTH		RE-ENTRY	
	(C)	(M)	(C)	(M)	(C)	(M)	(C)	(M)	(C)	(M)
A. Time to Fire	G	MG			SM	M	SM	M		
B. Disconnect Unbilical Cord	G	G								
C. Rocket Engine Condition										
1. Saturn - Stage No. 1 - 8 Engines	S	G								
Stage No. 2 - 4 Engines										
Stage No. 3 - 4 Engines										
2. Mission Engines 4 Main Engines										
2 Low Thrust Engines										
a. Pressure					SM	M	SM	M		
b. Condition										
1. TV Observations					M	M	M	M		
c. Gimballed Position					SM	M	SM	M		
D. Rocket Engine Select										
1. Desired Thrust Level					M	M	M	M		
a. Combination of Main and Low Thrust Engines(1-6)					M	M	M	M		
E. Fuel and Oxidizer System (Mission Engines)										
(Fuel Tank Compartments for Mid-course - 2)										
(Oxidizer Tank Compartments for Mid-course - 4)										
1. Total Consumed					SM	M	SM	M		
2. Total Remaining					SM	M	SM	M		
3. Select Tanks To Supply Individual Engines					M	M	M	M		
4. Select Tanks to Empty and to Fill					M	M	M	M		

[redacted]

TABLE II-4-III — FUNCTIONAL ALLOCATION CHART — ROCKET POWER (Cont)

SUBSYSTEM 2 Rocket Power	MISSION PHASE									
FUNCTIONAL ANALYSIS (SYMBOLS: (C) = CONTROL, G = GROUND, (M) = MONITOR, M = MAN, S = SYSTEM)	LAUNCH		SECONDARY MISSION EARTH ORBIT		MID-COURSE TO MOON EARTH		TRANSFER TO MOON EARTH		RE-ENTRY	
	(C)	(M)	(C)	(M)	(C)	(M)	(C)	(M)	(C)	(M)
F. Pressure System (Mission Engines)										
1. Time to Pressurize					M	M	M	M		
2. Fuel and Oxidizer Pressure					SM	M	SM	M		
G. Staging Sequence										
1. Abort										
a. Engine Shutdown (Boost Only)	SMG									
b. Jettison of Escape Capsule	SMG									
c. Abort Rockets - Thrust										
1. Magnitude	Preset	G								
2. Direction	Preset	G								
3. Thrust Level	Preset	G								
2. Secondary Mission - Lunar Mission	S	MG								
H. Jettison of Abort Rockets									SM	M
I. Jettison of Mission Engines									SM	M
J. Jettison of Mission Module									SM	M

[redacted]

TABLE II-4-IV — CONTROL-DISPLAY PARAMETERS CHART — NAVIGATION AND GUIDANCE

PHASES.....
1. LAUNCH
2. TRANSFER ORBIT
3. LUNAR ORBIT
4. RE-ENTRY

CONTROL - DISPLAY PARAMETERS

SUBSYSTEM 1
PAGE 1 OF 2

NAVIGATION AND GUIDANCE

ITEM NO.	PARAMETERS		INSIDE COCOON	REQUIRED DURING PHASES 1	2	3	4	CONTROL MODES	FUNCTIONALLY RELATED TO. SUB. SYS.	ITEM	DISPLAY RANGE	MOVING ELEMENT TAPE	POINTER	COUNTER	PAGE HEIGHT INCHES	WIDTH INCHES	DEPTH INCHES	WEIGHT POUNDS	POWER WATTS DC	AC	
1.	BODY AXIS RATES - YAW, PITCH, ROLL			X	X	X	X		1	2-4,38	0 ± 10°/SEC				3	3	3	.9		9	MOVING SPHERE DISPLAY WITH COMMAND NEEDLES
2.	VEHICLE ATTITUDE AZIMUTH			X	X	X	X		1	1,38	0-360°			INCLUDED IN ITEM 4							
3.	ELEVATION			X	X	X	X		1	1,38	0 ± 90°										
4.	ROLL			X	X	X	X		1	1,38	0 ± 180°				5	5	10	8	-	10	
5.	ALTITUDE RATE						X		1	6-11,38	0 ± 3500/FT/SEC				SINGLE DISPLAY						LIMITS ARE A FUNCTION OF FLIGHT PATH ANGLE AND DYNAMIC PRESSURE.
6.	ALTITUDE RATE LIMITS						X		1	5-11,38											
7.	ALTITUDE			X			X		1	5-11,38	0 - 400,000 FT										
8.	LONGITUDINAL "G"						X		1	5-11,38	0 - 15 G										
9.	SKIN TEMPERATURE						X		1	5-11,38	0% - 100%			INCLUDED IN ITEM 14							
10.	HEAT SINK TEMPERATURE						X		1	5-11,38	0% - 100%										
11.	HORIZONTAL VELOCITY						X		1	5-10,38	0 - 36,000 FPS										
12.	GREENWICH MEAN TIME			X	X	X	X		1	15	0 - 24 HR										
13.	ELAPSED TIME			X	X	X	X		1	15	0 - 14 DAYS										
14.	TIME TO GO				X	X	X		1	15,47	0 - 100 MIN				9	9	4	8	-	20	
15.	COMPUTER READOUT: LAUNCH MONITOR			X					1	26,27					4	6	10	20	-	40	READOUT IS ON A CATHODE RAY TUBE. MODE SELECT IS ON COMMANDER'S CONSOLE INSIDE COCOON. ACTUAL, OR INTERROGATE VALUES MAY BE OBTAINED WITH ANY OF THE MODES LISTED. DATA MAY ALSO BE INSERTED BY MEANS OF A KEYBOARD INCLUDED IN ITEM 26.
16.	EARTH ORBIT			X			X														
17.	SPACE COORDINATES				X	X															
18.	LUNAR AIM POINT				X																
19.	LUNAR ORBIT					X															
20.	EARTH AIM POINT				X																
21.	RE-ENTRY			X			X														
22.	SENSORS				X	X															
23.	DATA LINK				X	X	X														
24.	ATTITUDE				X	X	X														
25.	COMPUTE				X	X	X														
26.	COMPUTER READOUT MODE SELECTOR		X	X	X	X	X	ITEMS 15-25							3	6	2	2	-	-	
27.	COMPUTER READOUT BRIGHT-FOCUS-CONTRACT			X	X	X	X	0 - 10	1	15											
28.	ASTROTRACKER MODE (3 REQUIRED)				X	X		POINT LOCK ON	1	22				INCLUDED IN ITEM 32							
29.	HORIZON SEEKER MODE (3 REQUIRED)				X	X	X	POINT LOCK ON	1	22											
30	RE-ENTRY DISPLAY SCALE CHANGE						X		1	21											2000, 1000, 500, 100, 50 MILES/INCH

TABLE II-4-IV — CONTROL-DISPLAY PARAMETERS CHART — NAVIGATION AND GUIDANCE (Continued)

PHASES.....
1. LAUNCH
2 TRANSFER ORBIT
3 LUNAR ORBIT
4. RE-ENTRY

CONTROL - DISPLAY PARAMETERS

SUBSYSTEM 1
PAGE 2 OF 2

NAVIGATION AND GUIDANCE

ITEM NO.	PARAMETERS	INSIDE COCOON	REQUIRED DURING PHASES 1	2	3	4	CONTROL MODES	FUNCTIONALLY RELATED TO. SUB. SYS.	ITEM	DISPLAY RANGE	MOVING ELEMENT TAPE	POINTER	COUNTER	PAGE HEIGHT INCHES	WIDTH INCHES	DEPTH INCHES	WEIGHT POUNDS	POWER WATTS DC	AC	
31	PLATFORM STELLAR REFERENCE MODE SELECT			X	X		SYNCH LOCK ON	1	24		INCLUDED			IN	TEM	32				
32	COMMAND NEEDLE MODE SELECT		X	X	X	X	SEE ITEM 48	1	2,3,4					6	9	3	4	-	-	
33	PERISCOPE:	X	X	X	X	X								10	10	-	50	-	-	
34	ALTITUDE RETICLE SET	X	X	X	X	X	MOON EARTH	1	33	0 - 1000 MI										
35	LENS SELECTOR	X	X	X	X	X	WIDE ANGLE NORMAL	1	33											
36	FILTER SELECTOR	X	X	X	X	X	1,2,3,4	1	33											VIEWING-ORIENTATION-COMPUTER-ROCKETS
37	MODE SELECTOR	X	X	X	X	X		1	33											3 AXIS CONTROLLER
38	SIDE STICK FLIGHT CONTROLS	X		X	X	X		1	1-4											
39	AERODYNAMIC CONTROLS					X		1	1-11											
40	SOLAR COLLECTOR			X	X															
41	ANTENNA			X	X															
42	ROCKET CHAMBERS			X	X															
43	ASTROTRACKERS			X	X															
44	HORIZON SCANNERS			X	X	X														
45	LEFT ARMREST CONTROLS ABORT	X	X				OFF ON													
46	SECONDARY MISSION	X	X				OFF ON													
47	SAFETY SWITCH	X		X	X	X	OFF ON													
48	FLIGHT CONTROL MODE SELECT	X	X	X	X	X					INCLUDED			IN	TEM	33				ITEMS 32 AND 48 INCLUDE "LAUNCH", "SOLAR COLLECTOR", "THRUST CONTROL", "RE-ENTRY AIRBORNE COMPUTER", "RE-ENTRY GROUND COMMAND" AND ITEM 48 INCLUDES "PILOT ASSIST", IN ADDITION.
49																				
50																				

CONFIDENTIAL

CONFIDENTIAL

TABLE II-4-V — CONTROL-DISPLAY PARAMETERS CHART — ROCKET POWER

PHASES.....
1. LAUNCH
2. TRANSFER ORBIT
3. LUNAR ORBIT
4. RE-ENTRY

CONTROL - DISPLAY PARAMETERS

SUBSYSTEM 2

ROCKET POWER

ITEM NO.	PARAMETERS		INSIDE COCOON	REQUIRED DURING PHASES 1	2	3	4	CONTROL MODES	FUNCTIONALLY RELATED TO. SUB. SYS.	ITEM	DISPLAY RANGE	MOVING ELEMENT TAPE	POINTER	COUNTER	PAGE HEIGHT INCHES	WIDTH INCHES	DEPTH INCHES	WEIGHT POUNDS	POWER WATTS DC	AC	
1.	THRUST	(FOUR REQUIRED)			X	X			2	15	0 - 6000 LB										
2.	FUEL AND OXIDIZER TOTAL QUANTITY				X	X			2	3,5	0 - 7500 FT/SEC										ASSUMES NORMAL SPECIFIC IMPULSE.
3.	OXIDIZER QUANTITY	(FOUR REQUIRED)		X	X	X		ON - OFF	2	4	0 - 1600 LB										
4.	OXIDIZER PRESSURE	(FOUR REQUIRED)		X	X	X		AUTO,OFF-ON	2	3	0 - 110 PSI										
5.	FUEL QUANTITY	(TWO REQUIRED)		X	X	X		ON - OFF	2	6	0 - 400 LB		ELECTROLUMINESCENT								
6.	FUEL PRESSURE	(TWO REQUIRED)		X	X	X		AUTO,OFF-ON	2	5	0 - 110 PSI										
7.	FUEL QUANTITY	(ATTITUDE CONTROL SYSTEM)		X	X	X			2	8	0 - 40 LB		ITEMS 1 - 11 ARE TOTALED								
8.	FUEL PRESSURE	(ATTITUDE CONTROL SYSTEM)		X	X	X			2	7	0 - 110 PSI		UNDER ITEM 11								
9.	OXIDIZER QUANTITY	(ATTITUDE CONTROL SYSTEM)		X	X	X			2	10	0 - 260 LB										
10.	OXIDIZER PRESSURE	(ATTITUDE CONTROL SYSTEM)		X	X	X		AUTO,OFF-ON	2	9	0 - 110 PSI										
11.	HELIUM PRESSURE	(ATTITUDE CONTROL SYSTEM)		X	X	X			2	8,10	0 - 3000 PSI				6	10	3	6		40	
12.	THRUST CHAMBER POSITION	YAW			X	X					± 5°										MAY BE CONTROLLED MANUALLY THRU COM-
13.	THRUST CHAMBER POSITION	PITCH			X	X					± 5°										MANDER'S SIDE STICK & MODE SELECTOR
14.	LOW THRUST ROCKETS	(TWO REQUIRED)		X		X	X	OFF ARM													
15.	MAIN ROCKET THRUST SELECT	(FOUR REQUIRED)		X	X	X	X	OFF, 2000 LB 6000 LB	2	1											
16.	MAIN ROCKET EMERGENCY PRESSURIZE			X		X	X	OFF - ON	2	2-6											
17.	MAIN ROCKET IGNITION			X	X	X	X	OFF - ON	2	1											
18.	THRUST CONTROL MODE SELECT			X		X	X	COMP. OFF MANUAL	2	15			ITEMS 14 - 22 ARE TOTALED								
19.	THRUST CONTROL MANUAL INPUTS:	AZIMUTH		X		X	X	0 - 360°	3				UNDER ITEM 22								THESE ALLOW GROUND SUPPLIED
20.		ELEVATION		X		X	X	0 ± 90°	3												COMMANDS TO BE EXECUTED
21.		MAGNITUDE		X		X	X	0 - 3000 FT/SEC	3												WITHOUT THE AID OF THE ON BOARD
22.		TIME		X		X	X	0 - 60 MIN	3									5	-	-	COMPUTER.
23.	OVERRIDE ROCKET CUTOFF			X		X	X	ON - OFF	2	1											ON LEFT HAND GRIP.
24.																					
25.																					
26.																					
27.																					
28.																					
29.																					

CONFIDENTIAL

CONFIDENTIAL

[redacted]

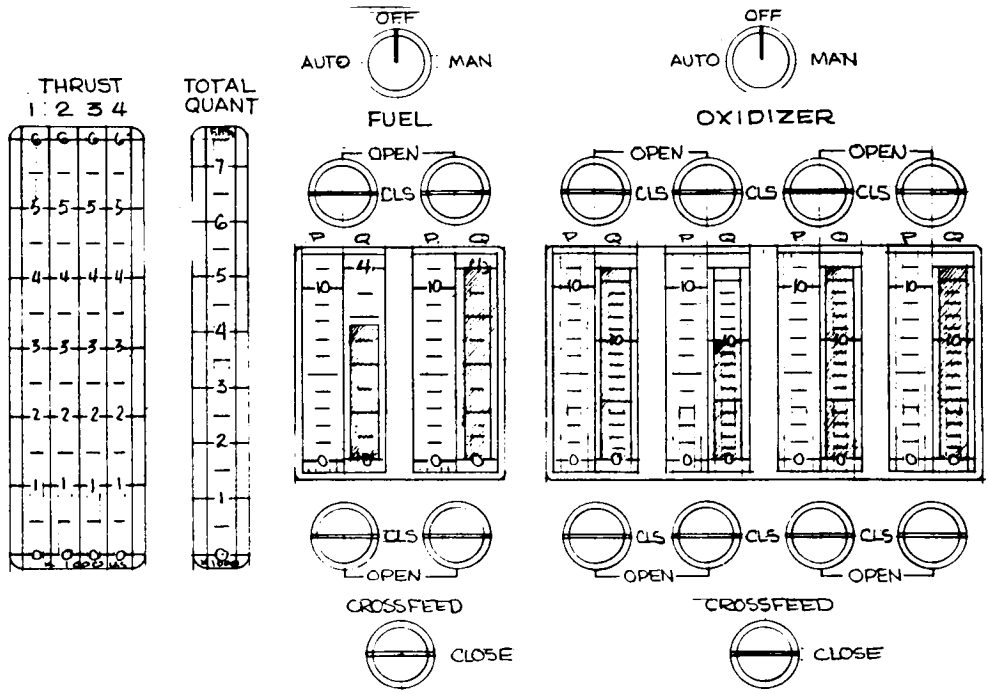


Figure II-4-2. Rocket power panel

and pressure displays allow the crew to use propellant so as to maintain nearly equal quantities in all tanks, thus minimizing the danger of a leaking tank. Of course, rocket engines 1 and 2 will normally operate only from oxidizer tanks 1 and 2 and fuel tank 1. To allow complete flexibility of the system, two crossfeed valves are provided which will allow any engine to be run from any combination of tanks. As an aid to fuel energy management, a total quantity display is provided for both fuel and oxidizer. The scale on this display is common to both variables and is calibrated in terms of feet-per-second of velocity change capability. This assumes a constant specific impulse and further requires that vehicle mass be known. It appears that the principal change in vehicle mass will be due to expenditure of rocket propellants; hence, the mass could reasonably be made a function of rocket propellant remaining, thus making this scale calibration possible. The display of rocket power capability in terms of ft/sec is important since these are the units in which the computer will read out required rocket performance. The thrust display shows individual indications of both actual and commanded thrust for each of the main engines.

[redacted]

The remaining controls for the rocket system are on the console within the commander's cocoon. The control system shown is based upon the assumption that sufficiently accurate thrust application is not within the capability of the crew without providing a display system of such complexity as to require the same sensing components which are needed for an automatic thrust control system. On these grounds it is felt that the task can be done better with the man serving as an alternate input source to the thrust control system to provide backup for the computer. This presumes that the thrust control system can be built to operate independently of the computer and with sufficient reliability to assure mission success. This is not an unreasonable assumption since the system requires little computation and is composed principally of an inertial measurement system and a servo system. The controls on the console allow the commander to choose either manual input or computer input to the thrust control system. In the case of manual inputs, provisions are made for input of azimuth angle, elevation angle, magnitude, and time. If computer input is selected, the values of these parameters will be shown on the computer readout display (described elsewhere). Rocket operation is initiated by: pressurizing and opening valves on desired tanks, selecting thrust level (2000 lb. or 6000 lb.) on desired engine, arming the desired low-thrust engine, orienting the thrust chamber of the selected rocket through the commander's side stick and mode selector (chamber position is shown on a display to the left of the attitude indicator), actuating ignition system, and when the preset firing time approaches, actuating the normally open arming switch on the commander's left grip. Operation may be monitored by observing velocity buildup on the computer readout and by comparing commanded and actual thrust on the thrust display. If automatic cutoff does not occur or if attitude is not maintained — or for any other reason — the rocket may be cutoff by releasing the normally-open arming switch on the left grip. For operation with the cocoon closed, an emergency pressurizing control is provided on the console which will pressurize and open feed lines on all tanks. There would be some added propellant lost with this procedure, but it allows rocket operation to be accomplished without opening the cocoon.

Additional displays of attitude control system propellant quantities and pressure, along with pressurizing controls, are located to the left of the computer readout display. Required circuit breakers for the rocket engine systems are located overhead on the power distribution panel.

4.3.1.2.2 VIEWER

Centrally located on the commander's console is an optical viewer with several modes of operation, Figure II-4-3. In the Viewing mode, the optics provide a choice of wide angle or normal lenses and several filter colors and densities. Integral with the external view are reticle lines which may be adjusted so as to just frame either the earth or the Moon when viewed through the wide angle lens. By choosing the correct reference (earth-Moon) an approximation of altitude may be read off a counter attached to the recticle position system. The wide angle lens also allows rough attitude information to be derived from the viewer by noting the position of the earth or Moon image in the field of view. In the orientation mode, the same external view is presented with the addition of a superimposed predetermined matrix of stars. By orienting the vehicle manually so as to superimpose the view of the actual stars with the projected matrix, the attitude of the vehicle can be re-established in the event the platform loses its reference. A "synchronize" — "normal" mode switch is provided to allow the platform to be synchronized to the proper position. At the time the star images coincide with the matrix, the platform would be switched to normal.

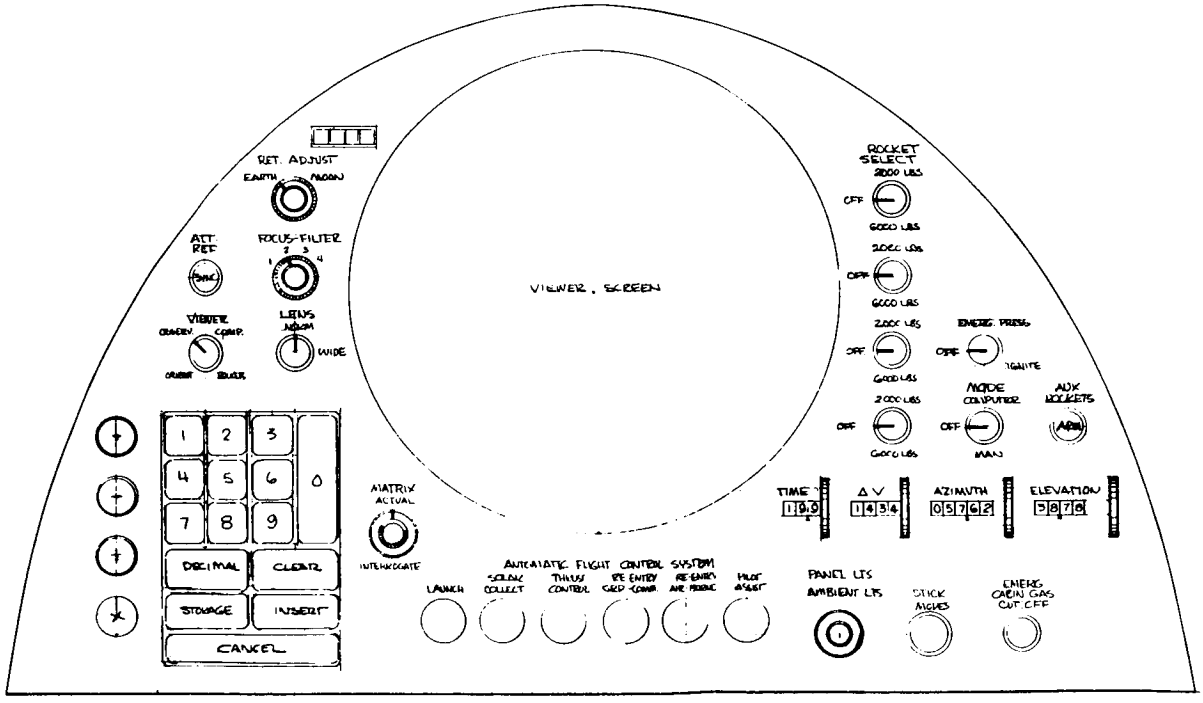


Figure II-4-3. Viewer

A third mode, "computer", allows a redundant composite readout cathode ray tube to be displayed through the viewer.

In the "re-entry" mode the viewer would have that angle of view and that magnification which would allow the operator to see the portion of the earth toward which he is heading in the late stages of re-entry. He would use this information first in aerodynamic control and later in manipulating the parachute to avoid local obstacles and correct for lateral velocities.

To allow maximum use to be made of the computer, a flexible data insertion and readout capability has been provided, utilizing a key board, a mode selector, and a cathode ray tube. The mode selector allows any one of several predetermined display matrices to be displayed on the cathode ray tube, Figure II-4-4.

A typical matrix is shown below:

LUNAR AIM POINT

MISS DISTANCE

X _____ 1
Y _____ 2
Z _____ 3
Time _____ 4

VELOCITY INCREMENT

Magnitude _____ 5
Azimuth _____ 6
Elevation _____ 7
Time _____ 8

VELOCITY, TOTAL

Magnitude _____ 9
Azimuth _____ 10
Elevation _____ 11
Time _____ 12

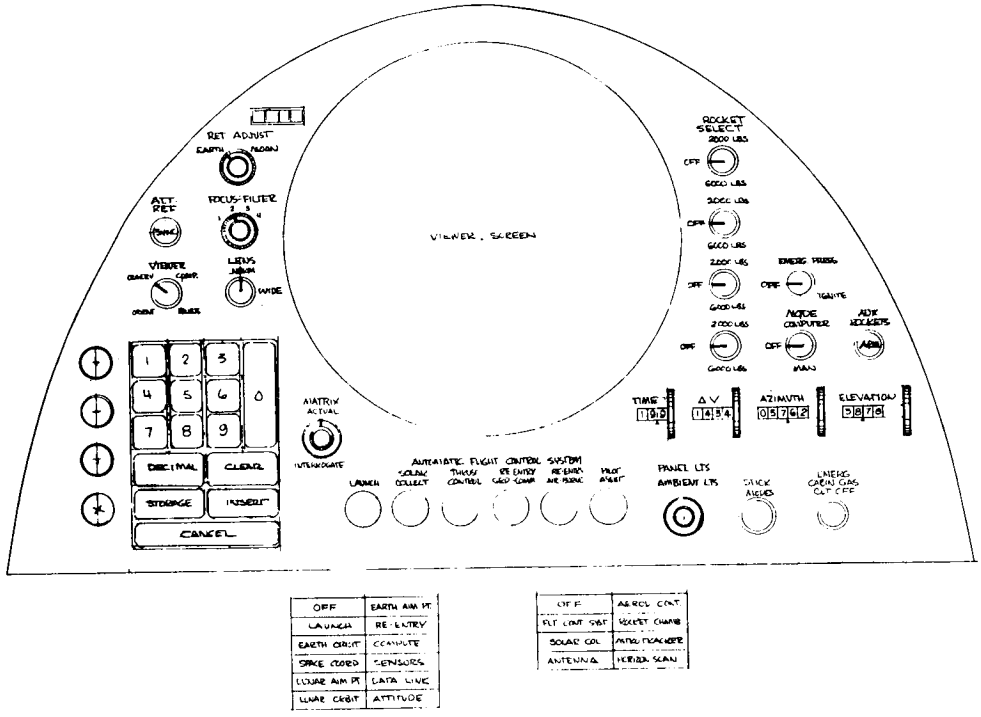


Figure II-4-4. Computer readout

Each of these parameters would appear in a numbered space on the display. The mode selector also provides for a selection of actual, or interrogated, values of those parameters. If "actual" is selected, the readout of appropriate values will occur immediately. If "interrogate" is selected, no parameters will be read out until sufficient initial conditions have been inserted to define the problem. Inserted data will be differentiated from readout data by means of color. Data insertion is accomplished by first punching the number of the desired parameter on the keyboard and depressing the address button. This directs further inputs to the proper address within the display matrix selected. The desired value of the selected parameter is punched in and appears on the readout for checking. If the correct number is displayed it is then inserted into the problem by means of the insert button. When sufficient inputs have been provided to allow a computation to be made, such will automatically occur. Provisions for decimals, clearing previous mode selections, and cancelling inserted data are available. This type of display and insertion system allows a great deal of information transfer without burdening the system with an extremely large, difficult-to-interpret mode selector.

Further study will be required to completely define the number and exact configuration of display matrices required to allow optimum interchange of information. The following is a list of possible modes with a brief statement of the type of information each might display.

LAUNCH	- Data to monitor launch and determine abort procedures.
EARTH ORBIT	- Standard orbital parameters plus velocity increment parameters for evaluating the effect of thrust application.
SPACE COORDINATES	- Position and velocity in space coordinates plus velocity increment parameters.
LUNAR AIM POINT	- Miss distances, velocity increment parameters, and total velocity.
LUNAR ORBIT	- Same as earth orbit, but based on a Moon-centered coordinate system.
EARTH AIM POINT	- Similar to lunar aim point, plus re-entry parameters.
RE-ENTRY	- Plot of attainable landing area with relative positions of desired and predicted landing sites.
SENSORS	- Provision to monitor inputs from astrotrackers, horizon scanners, etc.
DATA LINK	- Ground command data
ATTITUDE	- Inertial measurement unit outputs.
COMPUTER	- A blank matrix for insertion and manipulation of numbers.

4.3.1.2.3 SIDE STICK CONTROLLER

A three axis controller is mounted on the right armrest inside the commander's cocoon. A mode selector for the stick is located on the commander's console. Modes are provided to allow movement of the antenna and the rocket chambers. When one of

these modes is selected, the position of the device being controlled is shown on a time-shared display to the left of the attitude indicator. Astrotracker and horizon scanner pointing capability is also provided. In either of these modes the position of the sensor being pointed must be monitored on the computer readout display. The desired astrotracker or horizon scanner is selected by selecting the "point" mode by means of the mode switcher located to the left of the computer readout cathode ray tube. When a signal sufficient for lock-on is obtained, it will light the "lock-on mode". At this time the lock-on may be selected for that sensor. A similar mode selector is provided for the platform stellar reference. If maneuvering limits are exceeded and the reference looses its stellar monitor, the monitor will automatically go into the "synch" mode. At that time the vehicle must be re-oriented to a predetermined attitude and the reference monitor locked on again. Orientation of the vehicle may be done several ways. The side stick may be used directly to control the aerodynamic controls during atmospheric flight. During space flight the "flight control mode" may be selected which then allows the attitude to be controlled through the reaction jet control system. In this mode, control action will command angular rates about each of the vehicle axes.

Other automatic flight control modes are available and may be selected by means of switches on the commander's console. In order to anticipate the control action which will occur when a mode is selected, it may first be displayed on the command needles. This also allows the operation of the flight control system to be monitored after engagement, by observing the command needles which should remain centered.

4.3.1.2.4 LEFT ARMREST CONTROLS

The commander's left armrest, Figure II-4-5, contains three controls, an abort switch, a secondary mission switch, and a safety switch. The first two are self-explanatory and the latter is provided as a safety interlock for all irreversible switching operations. For example, before the antenna, solar collector, or mission module can be jettisoned, this switch must be actuated as well as the mode switch for the item being jettisoned. This guards against accidental operation of an important switch while entering or leaving the seat. The switch is also interlocked with the rocket engines and must be held closed any time the rockets are fired.

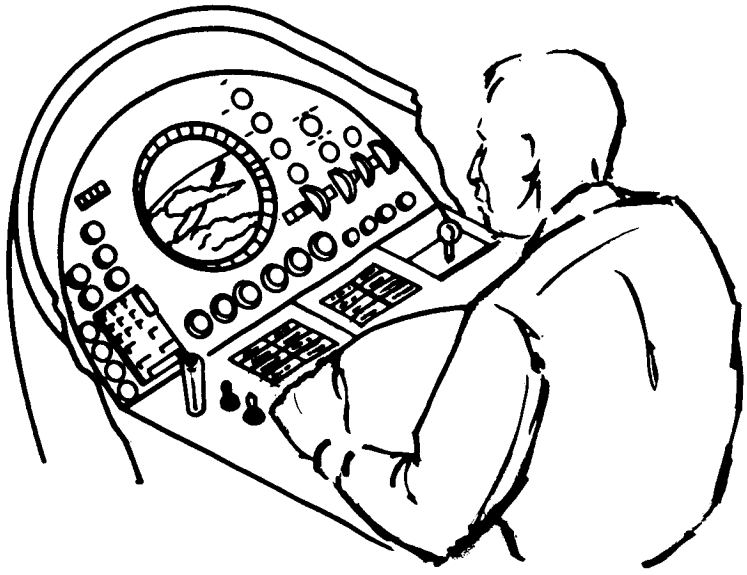


Figure II-4-5. Armrest control

4.3.1.2.5 CHECKLISTS

The information contained on the following check lists are tasks required by the operator in performing some of his control functions within the system. It should be noted that the tasks are associated with abort, midcourse navigation, lunar orbit, earth orbit, and attitude control during trajectory corrections and are viewed as discrete rather than of a continuous nature. The principal function where this is not the case is attitude control during periods of relative light control inactivity, i.e., when no changes in the trajectory are being staged. This situation may be contrasted to the continuous control function of the operator in the manual mode of re-entry flight control.

When control tasks of the pilot are discrete, they may be adequately described in terms of check lists. In effect, these check lists provide a description of the contemplated control tasks required of the operator.

Some preliminary checklists are as follows;

4.3.1.2.5.1 Velocity Correction

1. Select "VELOCITY CORR" on sequence display
2. Check power distribution panel observing buss load displays
3. Select "COMPUTER" or "MANUAL" input to thrust control system. If "MANUAL" —
   a. Select "DATA LINK" on computer readout or obtain parameters by voice communication
   b. Set magnitude
   c. Set azimuth angle
   d. Set elevation angle
   e. Set time to fire
   f. Select "ROCKET FIRE" on time-to-go mode selector
4. Check thrust level on engine desired
   a. Select "Thrust Chamber" mode on commander's side stick
   b. Observe chamber position on display
   c. Manipulate stick to obtain nominal position
5. Alert crew to take normal duty stations
6. Select "Thrust Control" on command needle mode selector
7. Select "Thrust Control" on flight control mode selector
8. Observe command needles, attitude display, vehicle rate display
9. Pressurize desired fuel and oxidizer tanks
10. Open feed valves from pressurized tanks
11. Select "Lunar Aim Point" on computer readout

[redacted]

12. When time-to-go approaches zero, turn on ignition system

13. At 15 seconds to go, actuate normally open safety switch on commander's left armrest.

14. As rocket fires, observe:

    a. Thrust

    b. Total velocity on computer readout

    c. Command Needles on attitude indicator

    d. Body Axis Rate Display

15. Release safety switch if automatic cutoff fails or if for any reason premature cutoff is desired.

16. Re-orient Solar Collector and Antenna

[redacted]

4.3.1.2.5.2 Re-Entry

1. Select "Re-entry" on sequence display and computer readout
2. Check power distribution panel
3. Jettison solar collector
4. Jettison antenna
5. Crossfeed excess oxygen from rocket system to environment
6. Seal hatch to mission module
7. Purge mission module
8. Check hatch for leaks
9. Turn on re-entry vehicle attitude control system
10. Jettison mission and propulsion module
11. Re-align vehicle with velocity vector
12. Select desired mode on command needles
13. Select desired mode on flight control
14. Observe re-entry displays and monitor system generation
15. Change mode of operation or fly manually if required
16. Verify parachute deployment through periscope
17. Actuate mechanical backup controls if parachute fails to deploy or impact bag does not inflate
18. Observe periscope to avoid obstacles
19. Verify operation of rescue aids.

4.3.1.2.5.3 Lunar Orbit

1. Select "Lunar Orbit" on sequence display
2. Select "Lunar Orbit" on computer readout
3. Check power distribution panel
4. Check secondary power system
5. Evaluate fuel requirements to achieve desired orbit
6. If feasible, initiate velocity change to achieve orbit
7. Observe Moon through periscope
8. Verify altitude readings with periscope
9. Monitor antenna signal strength to maintain communications
10. Turn off telemetry when behind Moon
11. Turn on tape recorder when behind Moon
12. Check operation of fuel cell batteries when transitioning between dark and light portion of orbit
13. Select "Earth Aim Point" on computer readout
14. Determine desired lunar orbit departure time
15. Initiate velocity change to intercept earth aim point

4.3.1.3 Re-Entry Controls and Displays

4.3.1.3.1 CONTROL TASKS

There are several operating modes for re-entry flight control and navigation. These are: a completely manual mode (damping system operative), a completely automatic mode based upon commands from the flight data computer or ground-based computer for some nominal trajectory, and pilot control mode based upon commands of the flight data computer or ground-based computer for some nominal trajectory.

In the complete automatic mode, the pilot must decide which source of command information he wants to fly and then switch to it. In addition, he is able to monitor the situation using the displays provided him for the complete manual mode. To make his monitoring task of the "how goes it" situation much easier, the flight director will display the commanded values of vehicle attitude.

In the pilot control mode based upon commanded signals, the pilot again must select the command source. He then tracks the director needles in flying the nominal trajectory for navigating to the pre-selected target (desired x, y, z position). Definition of the quickening function to be displayed on the director needles requires further study.

Of primary interest, is the complete manual mode of operation. The task objective of the operator in this mode is to navigate the vehicle safely to some pre-selected destination from some set of initial conditions without explicit reference to some precomputed nominal trajectory. If it is assumed that the vehicle is within the re-entry corridor (has a safe re-entry angle) and that the stability augmentation system adequately damps all short period oscillations, the operator's control task may be described as follows:

1. Maintain a trim pitch attitude angle (recovery vehicle attitude control system) and a zero roll and yaw angle when the vehicle is re-entering at escape velocity and dynamic pressure is gradually increasing from zero. The predesignated target should be within the precomputed minimum-maximum down range and lateral range capability of the vehicle at this time.

2. Maintain this flight condition until zero altitude rate is obtained. When zero altitude rate is arrived at, initiate a 180 degree roll so that negative lift may be employed in flying a constant altitude trajectory.

3. Maintain a constant altitude trajectory by controlling negative lift (variable pitch attitude) until sub-orbital velocity is obtained. It is also possible at this time to make some lateral and down range corrections so as to keep the predesignated target within the center of the precomputed range capability of the vehicle. This is accomplished by varying pitch attitude (lift) for down range corrections. The same technique is used for lateral range corrections in that only lift is used. However, the vehicle is rolled over to some desired roll angle where the lift vector is lateral to the relative wind.

4. When sub-orbital velocity is arrived at, initiate a 180 degree roll angle so that positive lift may now be employed in flying an equilibrium glide trajectory. This equilibrium glide trajectory may be approximated by maintaining a constant sinking rate or altitude rate. Adjustments should also be made to minimize target miss distance. It should be noted, however, that most of the range capability of the vehicle has been expended in the higher velocity regions of the re-entry trajectory.

5. By the time 50 percent of orbital velocity is obtained, the "G" and heating problems are no longer present. In addition, there is little remaining range capability and consequently, the vehicle should be zeroed in on its target at this point. At the approximate altitude, the drogue chute and reefed main parachutes should be released.

If the initial conditions are ideal, i.e., correct re-entry angle and target directly within center of the minimum and maximum lateral and down range capability of the vehicle, the pilot should be able to control and navigate his vehicle to the pre-selected target by following the above procedures. If the initial conditions are not ideal and the pilot must maneuver the vehicle close to the upper or lower re-entry corridor boundaries, the task will be considerably more difficult. The important consideration in defining the pilot's task under these circumstances is that he does not have continuous aerodynamic control (the control surfaces are ineffective during various portions of re-entry). In addition, the closer to the boundary limits the vehicle operates, the less there is of effective aerodynamic control with respect to time. Consequently, the pilot must be able to anticipate the result of his control actions.

When operating at the upper limit, i.e. shallowest re-entry angle, because of guidance error or attempting to fly maximum down range, the problem is one of remaining in the atmosphere, e.g., preventing skipout. The approach taken here is that this can be best accomplished by employing negative lift rather than by employing drag. The danger, of course, is that negative lift may not be employed by the pilot soon enough or long enough. For this reason, some type of safety factor should be built into the displayed computations of predicted maximum-minimum range and skip limits.

On the other hand, when operating at the lower limit, the problem is not to exceed the "G" or heating limits. In this situation, positive lift should be employed. However, it must be employed with care because a certain magnitude of "G" and heat absorbtion is necessary and desirable. Although these desirable amounts have not been defined for display purposes, it would conceivably be advantageous to incorporate desired "G" or temperature rate in the display of these parameters. This has not been done because of a possible confusion between command signals originating from flight director needles and these parameters when flying re-entry in the nominal trajectory command mode.

In developing displays for re-entry, information which reflects three different aspects of the operator's control task during re-entry were conceptualized. These task categories are: 1 long range planning, 2 immediate situation and 3 immediate vehicle orientation. A fourth category was established which provided redundancy with respect to the critical parameters having well defined limits, i.e., skin temperature, total heat absorbed, and "G". Within this general framework, an analysis of the control and feedback loops required for operator control during re-entry was possible.

The block diagram below, Figure II-4-6, illustrates these information categories, starting from the general planning parameters to the more specific attitude control parameters. In effect, the diagram defines the feedback parameters and the control modes available to the pilot during re-entry.

It should be noted that angle of attack is not a displayed parameter. It has been omitted because of the close parallel between its value and that of pitch attitude during most of the re-entry trajectory. Furthermore, as the selected vehicle configuration (D-2) uses parachutes for landing, there was no requirement for its display.

Another omission not to be overlooked is the lack of re-entry angle information. This display is considered to be part of the midcourse guidance requirements and thus, is not discussed in terms of re-entry. The midcourse guidance accuracies must be adequate in producing safe initial re-entry conditions that place the target destination within the range capability of the vehicle. In the event that these initial conditions are not safe, only fate can prevail. However, if the target destination is not within range, re-entry can still be accomplished although landing will not be at the desired location.

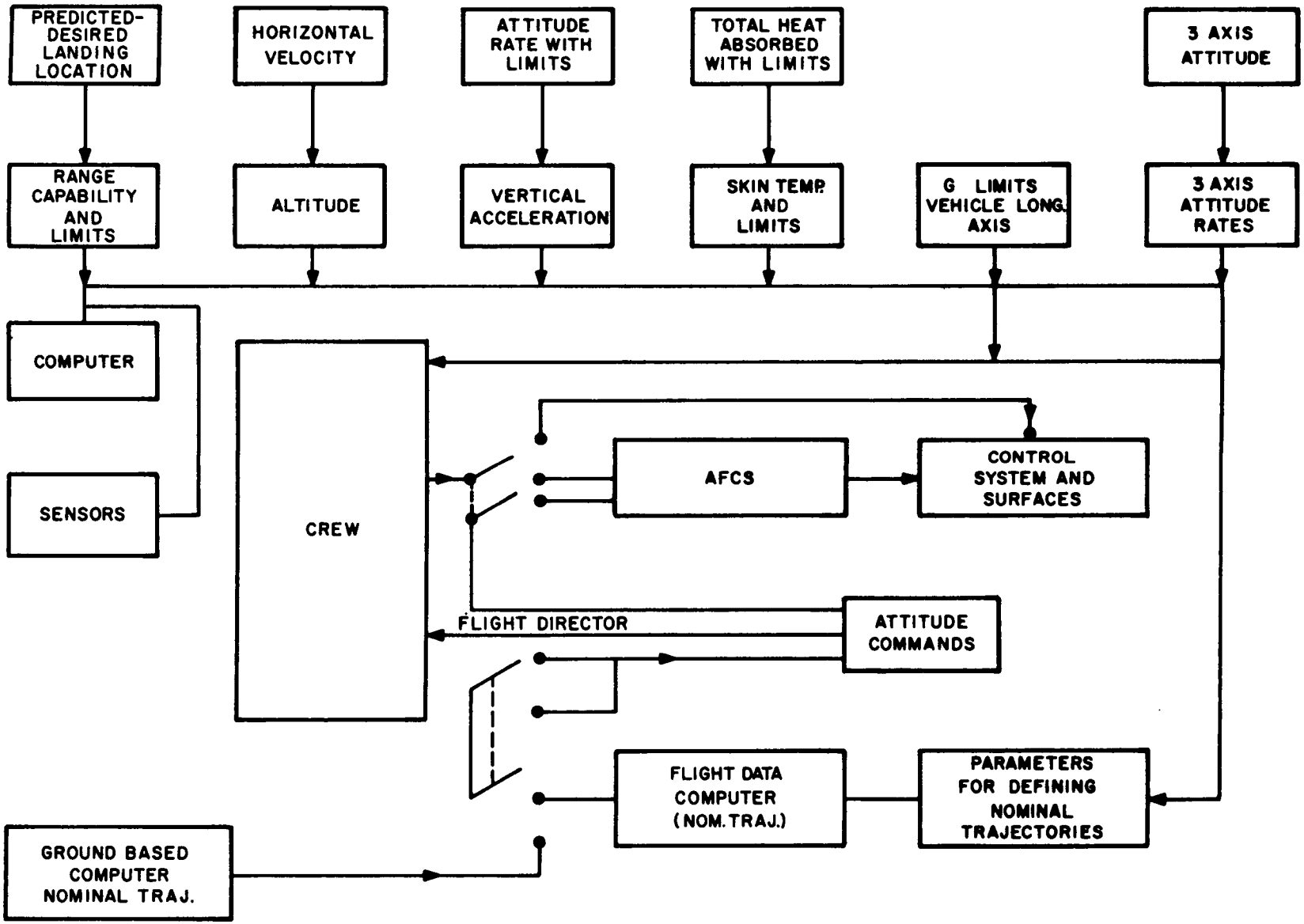


Figure II-4-6. Block diagram describing the feedback parameters and control modes available to the pilot during re-entry

With respect to landing preparation, chute releases will be automatic. However, control redundancy is built into the system by providing the crew with manual backup for parachute release. These controls are not located on the panel but are assigned to the space vehicle engineer's station.

4.3.1.3.2 RE-ENTRY PANEL AND DISPLAYS

Figure II-4-7 shows the panel location of the displays used for re-entry. The energy management, attitude indicator, and 3-axis attitude rate displays are located directly in front of the pilot-commander while the remaining re-entry displays (altitude rate, altitude, horizontal velocity, "G", total heat absorbed, and skin temperature) are located immediately adjacent and to the right.

Because of the importance of this display, the possibility of time-sharing them by locating them in a crew-shared viewing area has been examined. However, because of the effects of deceleration (limited peripheral vision) this notion was discarded in favor of locating the displays directly in front of the pilot commander where visual distortion due to deceleration is at a minimum.

The energy management display, altitude rate display, and the attitude indicator, are individually described and discussed on the following pages.

4.3.1.3.3 ENERGY MANAGEMENT DISPLAY

This display presents the desired destination and the predicted destination of the re-entering vehicle on a variable range scale, Figure II-4-8. The distance error between predicted and desired destination can be manually computed from the variable scale. By using a variable scale, reading accuracy may be increased as the range capability and the velocity of the vehicle are reduced. It should be noted, however, that the lateral range scale is fixed. This is because of the limited lateral range capability of the vehicle.

The displayed maximum-minimum "range bars" not only permit direct reading of the vehicle's range capability at any one time but also allow for the possibility of navigating the vehicle within the center-of-range capability. The upper range bar is driven as a function of predicted skip limits plus some safety factor and vehicle capability. The lower range bar, on the other hand, is driven by a function of predicted limits with respect to "G", total heat absorbed, and temperature plus some safety factor and vehicle capability.

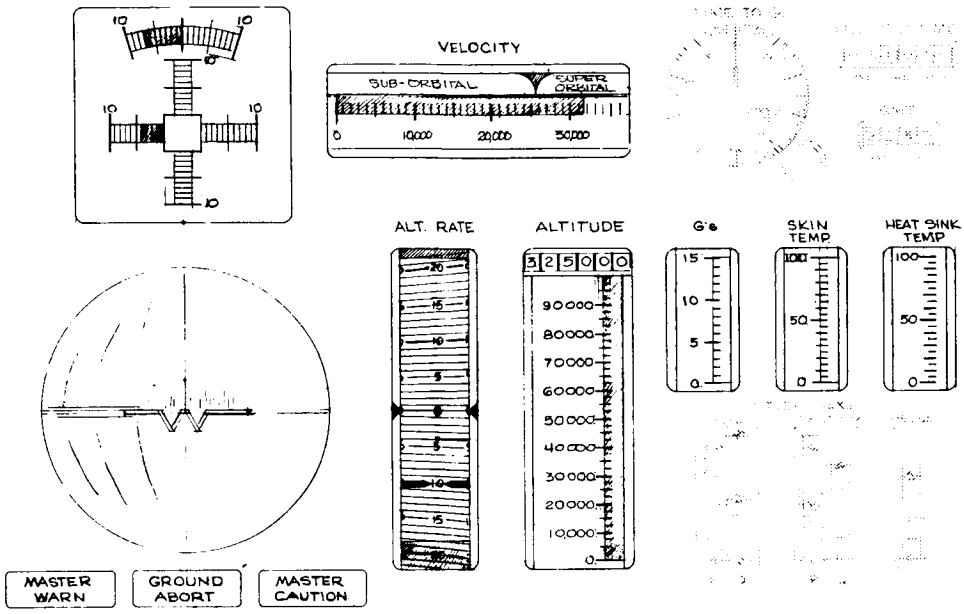


Figure II-4-7. Re-entry displays

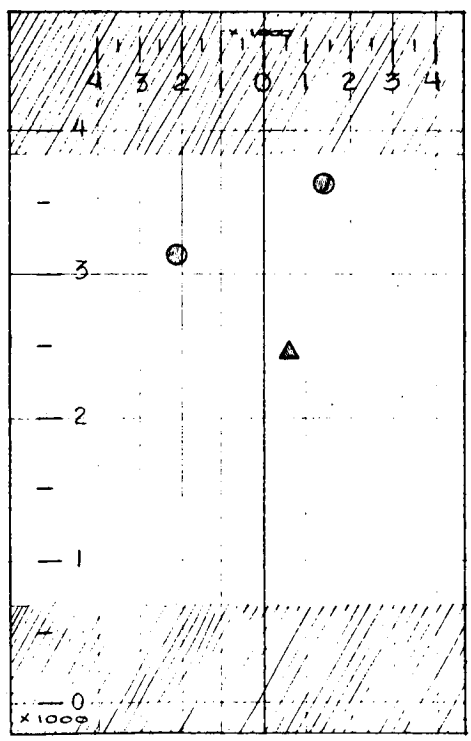


Figure II-4-8. Energy management display

In effect, this display permits the planning of a trajectory which stays within the projected corridor boundaries and range capability of the vehicle. At the same time position error is obtained with various degrees of accuracy.

4.3.1.3.4 ALTITUDE RATE DISPLAYS

The altitude rate scale, Figure II-4-9, is designed to give present situation information in relation to the re-entry boundary limits. In addition, "quickened" information is provided the pilot to help him avoid these limits.

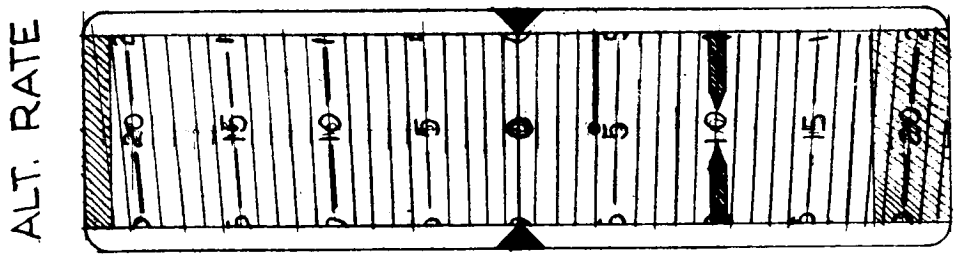


Figure II-4-9. Altitude rate display

This display is the principal input to the pilot indicating when it is time to roll 180 degrees and to apply negative lift e.g., when altitude rate changes to zero during the initial re-entry run. Moreover, it will provide adequate feedback for the tasks which require the holding of a constant altitude during the super-orbital phase of re-entry and a constant altitude rate during the suborbital phase. The acceleration dot will facilitate these tasks.

The altitude rate limits are conceived to be actual rather than predicted limits. Predicted limits in the form of range limitations are incorporated in the Energy Management Display. Thus, the pilot is always aware, in terms of altitude rate, of where he is in relation to his upper and lower re-entry corridor limits. The "quickened" display permits anticipation of direction or deviations from a constant altitude or constant altitude rate flight condition.

The information is presented on a variable scale (moving horizontal scale) in order to provide varying degrees of sensitivity during different phases of re-entry. This scale is driven as a function of horizontal velocity so that the scale interval is expanded during the portions of re-entry where horizontal velocity is large. The scale interval is contracted, although the visible scale range is increased, during the portions of re-entry where horizontal velocity is relatively small and the vertical component is large.

4.3.1.3.5 ATTITUDE INDICATOR AND RATE DISPLAYS

Because the attitude and attitude rate displays are the principal vehicle orientation displays, they serve a general function in all mission phases. Of particular importance are the attitude rate displays for controlling vehicle attitude during the midcourse guidance and orbital phases of the mission, Figure II-4-10. They greatly aid pilot performance when the task is to null various attitude rates or to hold a particular vehicle attitude.

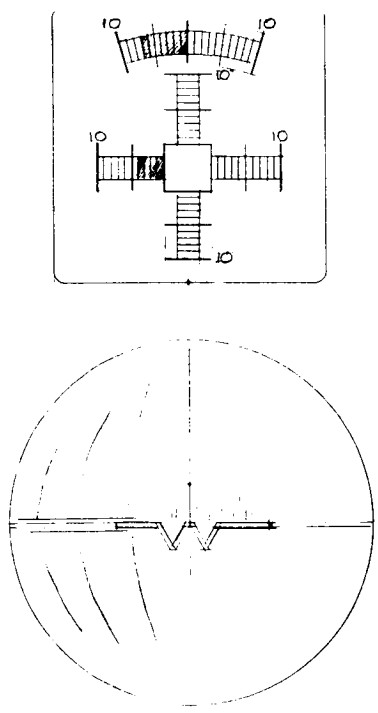


Figure II-4-10. Attitude indicator and rate displays

The 3-axis attitude indicator, of course, provides the pilot with information of the present vehicle orientation during all the mission phases. Commanded attitude information required during launch and thrust correction maneuvers are provided by the director needles on the display.

During re-entry the 3-axis attitude rate displays provide feedback to the pilot with respect to the build-up in dynamic pressure as it affects vehicle response. When dynamic pressure is small, the aerodynamic response of the vehicle is sluggish. However, as dynamic pressure increases, vehicle response becomes more dynamic. This display will indicate these different response rates and, consequently, aid the pilot in controlling the variable lift trajectory of the vehicle.

The 3-axis moving horizon attitude indicator was selected over other attitude displays because it permits the display of interpretable attitude information in all axes. For example, during re-entry the pilot must be able to control pitch attitude in flying a variable negative lift trajectory when rolled 180 degrees. This upside down position is a problem in control-display design because in this attitude the usual flight control-display relationships are reversed. Rather than design two separate display systems, one for 0 degree roll angle and the other for 180 degree roll angle in attempting to provide control display compatibility, it was considered advisable to show this situation through an attitude display. It is believed that the 3-axis moving horizon attitude indicator adequately provides the operator with this information, i.e., pitch attitude and heading orientation throughout all roll angles.

It should be noted that the director needles on the attitude indicator will permit the pilot to monitor the command attitude signals originating from either the flight data computer or ground computer when re-entering in the automatic mode. A variation of this mode permits the pilot to control the vehicle while tracking these commanded attitude inputs. However, this latter mode of operation is seen as the least desirable of all the re-entry modes because servomechanisms are more capable in performing this task than the pilot.

[redacted]

4.3.1.3.6 ALTITUDE AND HORIZONTAL VELOCITY DISPLAYS

The moving pointer horizontal velocity scale, Figure II-4-11 is primarily used to indicate when the vehicle passes from super-orbital velocity to suborbital velocity, so that the pilot may anticipate when to re-initiate the 180 degree roll angle and commence an equilibrium glide trajectory. In addition, he is able to obtain some indication of his gradually decreasing range capability, and indications as to when problems such as heating, high G levels, and skipping out of the atmosphere are definitely over.

The moving pointer, digital readout altitude indicator provides both gross and refined altitude information. This information is considered as planning data for the z-axis while the energy management display is considered as a planning display for the x- and y-axis of an earth reference system. It will also provide the pilot with an indication of remaining range capability of the vehicle.

Itemized on the scale are altitudes for drouge chute release, release of reefed main parachutes, and un-reefing of main parachutes.

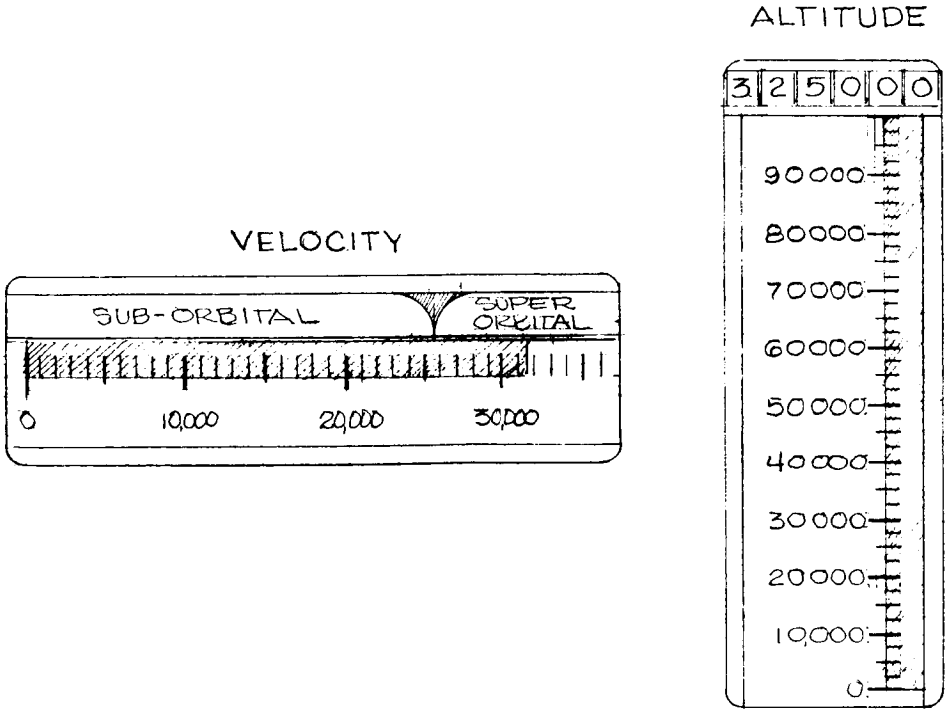


Figure II-4-11. Altitude and horizontal velocity displays

[redacted]

4.3.1.3.7 G, TOTAL HEAT ABSORBED AND SKIN TEMPERATURE DISPLAYS

The G, total heat absorbed and skin temperature displays, Figure II-4-7 are used to call out the exact values for these constraint parameters and to show, for the individual parameters, the relationship between limits and present condition. These displays are essentially backup displays to the predicted range limits on the Energy Management Display and the computed limits on the Altitude Rate Display. However, these are considered necessary information requirements in the attempt to provide the pilot with complete information about his present status. In this context, skip limits would be another required display. However, this parameter, as yet, is not as clearly defined as those above and, hence, it is only indicated among the computed items.

4.3.1.3.8 GLIDE RE-ENTRY LANDING (FOR THE R-3 RE-ENTRY VEHICLE)

In order to permit maneuvers below orbital velocity, glide vehicles capable of re-entry and landing at a pre-selected landing site would require additional displays besides those which have been described.

To provide an indication of the lift-drag ratio, the stall limits and the proper flare out of the vehicle, angle of attack information is required. Angle of attack being indicative of the lift-drag ratio, is used to define the maximum and minimum range capability of the vehicle. This permits the pilot to home in on his landing site if it is within these maximum and minimum range limits by estimating the future trajectory from the existing angle of attack. In the case of stall and flare out, angle of attack provides feedback for their proper control.

Another important requirement is for greater reading accuracy of gross altitude and velocity. With respect to the altitude and horizontal velocity displays which have been described, this would involve the expansion of the scale interval size on the displays of altitude below 100,000 ft and of velocity below 8,000 ft/sec.

A third requirement is a turn and bank indicator (lateral acceleration) to be used in conjunction with the attitude indicator. This will permit the pilot to make coordinated turns during adjustments prior to landing and at the lower speeds.

[redacted]

Not to be overlooked, of course, are the stringent requirements for flight path control during landing. What this means, in terms of display requirements for a re-entry vehicle configuration, has not been studied in detail. The attitude indicator which has been described has director needles and thus, is capable of being incorporated into an ILS system. Performance would be more adequate when a separate glide slope and localizer display was utilized. Another consideration, and perhaps the most adequate, is to provide for a GCA system and utilize this as the prime landing mode for a glide re-entry vehicle.

[redacted]

4.3.2 Semi-Automatic Navigation, Guidance and Control Subsystem Components

The major semi-automatic navigation, guidance and control subsystem components are described in detail in Section 3. The Inertial-Stellar Stable Platform is described in Section 3.4, the Celestial Sextant in Section 3.5, and the Computer in Section 3.3. The descriptions of the components include a discussion of design considerations which take advantage of crew capabilities for maintaining, testing and repairing the equipment.

4.3.3 Manual Navigational Equipment

4.3.3.1 PHOTOGRAPHIC NAVIGATIONAL EQUIPMENT

Some aspects of the implementation of photographic techniques as a navigational aid have been studied. Conceivably a stellar field with an image of the earth or Moon may be photographed. The linear dimension of the extended body as well as its position in the stellar background may be analyzed to yield inertial position of the space vehicle at the time the exposure was made.

The length of exposure of the stellar field is related to the illumination of the image of the object and sensitivity of the photo sensitive emulsion. The illumination of a star image, or point source of illumination, is a function of the square of the effective diameter and the size of the image on the focal plane (0.001-in. is a typical value). The illumination of a subtended body image such as the earth or Moon, is related to the square of the ratio of lens focal length to effective aperture and directly proportional to the brilliance of the object. Scale factor of the equipment is related to focal length. Resolution is related to focal length, aperature, type of emulsion and processing facility. Consequently, trade-offs in the selection of these parameters must be examined to optimize, or even render feasible, the photographic configuration for a navigational camera.

Preceding the camera design, the basic pros and cons of the utilization of a camera, rather than visual interpretation of the image directly by the human eye alone, must be considered.

In the question of sensitivity, the unaided human eye can detect illumination in the order of 10^{-12} Lumens/meter^2, (about the brilliance of a 6th magnitude star). The

integration quality of photosensitive emulsions allow them to detect less intense illumination by summing its effect over time. The addition of lenses to either detector has the effect of increasing their sensitivity by their light gathering or condensing characteristics. Time is, therefore, the price to be paid for the intergration characteristic of the photosensitive emulsion to assimilate point data. If extremely dim objects are to be the objects of a navigational camera, base motion isolation from the vehicle must be provided for lengthy exposures. On the other hand, for more brilliant objects, a camera can store more data in a short period of time (e.g. typical exposure to photograph the moon is 1/5 sec, f/32, on Kodak Panatomic X film) than a man can interpret in the same time. If the short time assimilation of great quantities of data is a requisite for a navigation scheme, photography far outstrips man's capability alone. But if the determiniation of angular separation between two objects is the navigational objective, the retention of the occurrence as a photographic image seems to be an undue complication of the required instrumentation.

Resolution is a consideration for either photographic or manual interpretation of data. The storage capability of photographic systems is related to its optical qualities and the accutance of the photosensitive medium. Since lens quality is common to both methods it can be ignored when evaluating their comparative advantages. The Kodak Panatomic X material previously cited can resolve 100 lines per millimeter at 30 to 1 contrast ratio when processed in an optimum fashion. The resolution of the unaided normal eye is specified differently and an acceptable figure is 1 arc minute. It is apparent that if the photograph is to be analysed by the unaided normal eye, it can only display the detail that was imaged upon it, degraded by the limiting accutance of the photographic process itself. It seems that, given sufficient time and illumination as previously discussed, the unaided eye may perform better by analyzing the real image itself. If a magnifying glass or projection techniques are to be employed to enhance the visual analysis of the photographic image, similar optical aids may be employed in the visual analysis of the real image. It is evident then that the eye's perception is not increased by the photographic process itself. It is noted however, that mechanical or electronic reduction of the photographic information (such as map matching) can conceivably surpass visual interpretation of the data.

In conclusion, then, if the assimilation of great quantities of imaged data, of if extremely long or extremely short viewing times, or if mechanical or electronic scanning techniques are dictated by a navigational concept, a photographic medium shows promise. However, if the navigational concept involves the simple analysis of the diameter of extended celestial bodies (e.g. earth, Moon, sun) and the angular separation between them and stars, it seems the human eye offers greater promise.

4.3.3.2 NAVIGATIONAL PERISCOPE

A navigational periscope offers some promise as a manual backup navigational aid. Conceivably the instrument is visualized as possessing design features similar to two precision alignment theodolites. The instrument is intended to be mounted in the mission module.

Two celestial bodies (earth, Moon, star, sun) are viewed simultaneously and their images are superimposed in the same field of view for presentation to the operator. Each object objective is allowed to articulate independently for acquiring its respective object. The angular headings of the respective bodies are determined by the orientations of the respective objectives with respect to the vehicle. The angles may be detected electro-mechanically, by precise instrumentation such as phasolvers or read optically from precision divided circles mounted directly to each objective's azimuth and elevation axis. The azimuth and elevation heading of each object is displayed in the image field of view to the operator. Simultaneously the vehicle heading in inertial coordinates, as determined by the space or celestial sextant or the stable platform, is displayed in the same field of view along with sideral time.

Once the images are superimposed and oriented with respect to a recticle by adjustment by the operator, the object and vehicle heading angles, as well as the time readouts, are "frozen" for readout by the operator. Electronic tracking aids are visualized to be required to aid the operator in centering the target. The information is then fed to the Guidance Computer as elements of a vehicle fix in the appropriate coordinates. Fix elements observed in this manner can conceivably be accurate to 5 arc seconds.

Such instrumentation is presented as conceptually showing some promise. Further study is required to justify the feasibility of building hardware to meet this purpose.

4.4 Manual Back-up Subsystem Operation

The discussion of the manual back-up subsystem operation is divided into the following three categories:

1. Navigation
2. Guidance
3. Control

4.4.1 Navigation

A brief discussion of the proposed manual optical equipment is given in the preceeding section. The purpose of the equipment is to provide the crew with a manual source of navigation fix information which serves as a back-up for corresponding information which is normally obtained from the celestial sextant. If photographic or visual observation techniques for providing data are utilized, it appears possible to obtain a position accuracy of about 5 seconds of arc by taking three fixes at one-half hour intervals. Photographic data can be manually reduced by means of reference to a grid which appears on the photograph or by means of an engine dividing machine. These data can be introduced into the computer by means of the keyboard.

In the case of computer failure, the crew can make use of charts which solve the simple two-body equations. This simple model is a good approximation of the actual physical system for ranges of up to 85,000 nmi with respect to the earth and 13,000 nmi with respect to the Moon. Two-body and patched conic models are presently being investigated as possible aids in simplifying the navigation problem for manual calculations. This aspect of the problem is discussed in Appendices T-R3 and T-T.

4.4.2 Guidance

The two major aspects of manual guidance techniques are as follows:

1. Manual calculations
2. Manual control of vehicle attitude and thrust vector.

4.4.2.1 MANUAL CALCULATIONS

Manual calculations of the required velocity correction must include the determination of the time at which the correction should be made, and the magnitude and the direction of the correction. Tables, combined with simple calculations performed by the crew, should be sufficiently accurate to insure a safe return to earth even though it may not be possible to reach a specific target point.

4.4.2.2 MANUAL CONTROL

The crew must be capable of manually controlling the following three subsystems:

1. Space vehicle attitude control subsystem.
2. Re-entry vehicle attitude control subsystem.
3. Propulsion subsystem.

The attitude control subsystem is instrumental in determining the direction of the thrust vector and will be discussed in the next section.

Control over the propulsion system is required so that the crew can command the initiation and termination of thrust in order to introduce the required change in the magnitude of the velocity correction. Displays provide information on the progress of the velocity changes. This information is obtained from integrating accelerometer outputs which are properly resolved with respect to the body and inertial coordinate systems. A detailed description of the manual velocity correction process is included in the section on Displays and Manual Controls. It should be noted that manual velocity corrections can, of course, be based on information from several sources, including corrections based on ground tracking and computation obtained by means of the voice communications link.

The thrust level utilized during the correction is decided upon by the crew since one, two or four of the large gimbaled engines can be used or the two fixed low thrust

ullage engines can also be used. Many factors must be considered in making this decision. The important factors are as follows:

1. Resulting thrust level
2. Crew reaction time for terminating thrust
3. Thrust cut-off uncertainties
4. Crew capability for manually controlling attitude in the presence of disturbance torques resulting from thrust misalignments
5. Operational status of engines and associated equipment
6. Safety considerations associated with the effects of errors on the mission

The effects of errors introduces by manual thrust vector control and manual calculations will be investigated during the course of the simulation studies now in progress.

4.4.3 Attitude Control

The crew has the perogative of taking over the control of the vehicle's attitude by means of manual controls and displays which are described in detail in Section 4.3.1 on "Displays and Manual Controls".

The requirements and the resulting design parameters for the attitude control subsystems for the mission module and the re-entry vehicle are described in Section 3.0 on System Design and in Appendix T-N. The resulting designs are based on available information with regard to operational requirements. However, from an examination of the results obtained, it seems reasonable to believe that the angular acceleration control levels and the angular velocities can be satisfactorily handled by the crew when using the manual mode of operation.

Attitude control outside of the atmosphere is obtained by a mass expulsion system. The maximum angular acceleration designed into this type of control system is 0.55 deg/sec^2 which is well within the capabilities of man. This is based on attitude control simulation studies carried out by Lear in which angular acceleration levels up to 2.5 deg/sec^2 were easily controlled manually. The maximum angular velocity expected is 16-deg/sec which is less than 3 rev/min. Tests by NASA at the Lewis

Laboratory show that manual control with rates of up to 30 rev/min. about each of three mutually perpendicular axes are within man's capability.

Stability augmented manual control will probably be necessary because of re-entry vehicle dynamics. With this implementation, manual control throughout re-entry should pose no serious problems.

During velocity corrections, the attitude of the vehicle is not only determined by the required direction of the thrust vector, but also by the selection which the crew has made with regard to the engine(s) to be used for obtaining thrust. This aspect of attitude control has been briefly discussed in Section 4.4.2.2 with respect to velocity corrections.

Simulation studies are planned which will utilize the actual manual controls and displays for demonstrating the compatibility of the attitude control subsystem designs with the corresponding vehicle configurations and man's capabilities.

4.4.4 Conclusion

It is concluded that manual navigation, guidance and control back-up equipment and procedures are adequate to insure a safe return to the earth. The capability of returning to a specific location is questionable because of the simplifications introduced in the mathematical model in order to permit manual navigation calculations and the errors introduced by manual navigation fixes. Further study of this problem is required when computation methods and specific hardware capabilities are known.

4.5 MAJOR CREW DECISIONS

One of the most significant advantages resulting from the inclusion of man in the spacecraft system is that many hardware implementation problems are simplified because of man's ability to make decisions during the course of the mission. The major decision areas are as follows:

1. Information sources
2. Mission profile
3. Velocity corrections
4. Override of automatic system operation

4.5.1 Information Sources

The crew, by means of appropriate instrument panel displays, manual navigation equipment and communications equipment, obtains information upon which to base decisions relating to navigation, guidance and control from the following sources:

a. On-board semi-automatic navigation, guidance and control subsystem.

b. Ground tracking and computation centers.

c. Manual navigation fixes and computations performed by the crew.

d. Ground computation centers which process either manual or semi-automatic on-board position fix information.

e. Navigation aids such as earth satellites (Transit) and possibly beacons located on the Moon.

f. Predicted navigation and guidance information which is based on previous measurements.

There may be cases when information from various sources conflict. The crew must decide, under the existing circumstances, which information source is providing the most valid data.

4.5.2 Mission Profile

The mission profile for a given flight is greatly influenced by circumstances which cannot be anticipated prior to lift-off, and it is in this area that crew decisions play a vital part. The crew, by means of appropriate instrument panel displays, manual navigation equipment and communications equipment obtains information upon which to base decisions. Assuming that accurate navigation, guidance and control information is available from at least one of several possible sources, the crew must use this information, in combination with much additional information relating to the other subsystems, as the basis upon which to make decisions with respect to the mission profile. The following is a list of basic missions among which the crew must choose. It is obvious that the navigation, guidance and control subsystem is initimately involved in all mission profile decisions.

A. Abort or emergency return.

1. Return to earth in minimum time with emphasis on safe re-entry only.

2. Return to a specific chosen location on earth in minimum time.

B. Earth orbit.

1. Circular orbit

2. Highly elliptical orbit

C. Circumlunar mission

D. Orbiting around the Moon.

1. Orbit the Moon for a pre-planned number of orbits, with the orbital elements and the sequence of operations completely pre-planned.

2. The number of orbits is determined by the crew while in orbit, and

a. the crew makes decisions relating to the shape of the final orbit and the actual sequence of operations used in achieving the final orbit, or

b. the crew follows a pre-planned sequence of operations to achieve an orbit with elements specified prior to the time of injection.

E. Other possible missions

1. Rendezvous

2. Docking and mating

3. Lunar landings, take-offs, and selection of landing site

4.5.3 Velocity Corrections

When the crew is satisfied with the validity of the navigation information being used to determine the need for velocity corrections, the decision of whether or not to make a velocity correction must be made. Once again there may be more than one source for obtaining velocity correction information. As an illustration of what can be involved in making this decision, consider the case in which the semi-automatic equipment is

operating properly. The crew has information from displays as to whether the following conditions are fulfilled:

A. The magnitude of the velocity correction is greater than K_1 times the minimum possible correction, and is K_2 times the 3σ uncertainty in making the correction.

B. The target miss distance being corrected is greater than K_3 times the 3σ uncertainty of the position estimate at the target.

C. The target miss distance is greater than K_4 miles.

The values of K_1, K_2, K_3 and K_4 are determined from simulation runs. The various resulting situations are listed below along with the appropriate action required for each case.

Situation	Action
A and B true, C false	No action required.
B and C true, A false	Choose a later correction time and recompute the correction.
A and C true, B false	Take more data to increase the accuracy of the estimate.
A, B and C true	Make correction at the proper time.
A true, B and C false	No action required.
B true, A and C false	No action required.
C true, A and B false	Take more data to increase the accuracy of the estimate.
A, B and C false	No action required.

In addition to these strictly navigation, guidance and control considerations, the crew's decision must also include such factors as the amount of propulsion available, the status of the other subsystems and the effect on the overall mission.

An additional crew decision which involves velocity corrections is that of determining the thrust level to be used. This was discussed in Section 4.4.2.2 on manual guidance control. The decision as to whether to use one, two or four of the main gimballed engines or to use the fixed low-thrust ullage engines is influenced by manual and automatic attitude control accuracy in the presence of misalignment distrubances, crew reaction times, and the operational status of the various engines.

4.5.4 Override Automatic Subsystem Operation

The crew can be faced with the decision of whether or not to override the automatic mode of subsystem operation. This situation occurs when information from more than one source is conflicting or when an equipment malfunction is detected. Automatic functions include taking navigation fixes, and attitude control and thrust vector control during velocity corrections. Safety considerations are intimately involved in this decision, especially when making corrections to obtain low perilune altitudes when in lunar orbit and when making corrections to achieve permissible re-entry flight path angles.

4.6 FUTURE WORK

Much valuable information with respect to the crew's role in the navigation, guidance and control area of the APOLLO mission will result from simulation studies.

Simulation studies described in Section 3.—Digital Computer Simulation of the APOLLO Mission, will provide opportunities for determining the effects of manual control on typical APOLLO missions. Thus the effects of manual navigation fixes, manual calculations using tables and simplified cis-lunar trajectory calculation models such as patched conic and two-body approximations, and errors in manual attitude control during velocity corrections can be determined.

Other important areas for future simulation studies include manual atttitude control for the removal of initial separation rates, confirmation of the design parameters of the space vehicle and re-entry vehicle attitude control subsystems with respect to manual attitude control, and complete manual control during re-entry, rendezvous, docking and mating, and lunar landing. Such investigations would use the proposed display and manual control equipment in both static and dynamic (centrifuge) test environments.

5.0 Error Analysis

5.1 GENERAL

An error analysis of most phases of the APOLLO mission is given in this section. The mission considered establishes an orbit around the Moon and subsequent return to earth. The phases of mission considered are given in the following list:

Insertion

Outgoing Midcourse

Transition from Midcourse to Terminal

Terminal Phase Retro-thrust

Lunar Orbit Establishment

Terminal Escape Thrust

Transition to Return Midcourse

Return Midcourse

Transition from Midcourse to Re-entry

Re-entry

Landing

As will be shown in later sections, the overall guidance concept from launch to landing maintains the APOLLO spacecraft on the reference trajectory and on time schedule. Deviations are permitted to the extent that mission objectives are not compromised. These objectives are to establish an orbit around the Moon with a 1000 to 2000 n mi apolune and a 50 to 100 n mi perilune. The spacecraft must then be returned to earth and the command module landed within a 10 n mi (RMS) accuracy. The command system to achieve these objectives will be on board.

For purposes of study, a reference trajectory with a 53 hour time-of-flight to lunar distance of closest approach is chosen for the outgoing leg. The return leg takes 99 hours to vacuum perigee. Longer time of flight, of 72 hours, is analyzed in less detail.

The overall problem may be considered in three major portions. These are from insertion to injection into lunar orbit, in lunar orbit, and ejection from lunar orbit to earth landing.

Major system considerations are influenced by the accuracy with which the guidance problem may be solved in these phases.

These considerations are listed below for each of the above portions of the mission.

5.1.1 Insertion to Lunar Orbit Injection

1. The accuracy with which the outgoing leg midcourse measurements and last velocity corrections are made determine the initial condition position and velocity accuracies prior to injecting into orbits around the Moon. The accuracy with which the initial conditions are established can be traded-off with possible subsequent lunar orbit corrections. Corrections for deviations in this phase which effect the total flight schedule must be considered.

2. Instrument accuracies and data processing techniques establish vehicle position and velocity accuracies. These in turn establish the number, frequency of occurrence, and magnitude of the midcourse velocity corrections required to achieve desired aim/point accuracy.

3. Insertion position and velocity errors determine the initial deviations from the aim/point in the position and velocity. In addition, if these errors are large, insertion errors influence to a large extent the magnitude of the first midcourse correction. The instrumentation accuracy and guidance concepts to achieve insertion initial conditions must be balanced against the requirements for midcourse instrument accuracies.

4. Uncertainties in the physical constants of the equations of motion must also be considered to assess the increased fuel requirements (or velocity) to correct these uncertainties. These considerations apply for the return leg as well.

5. To monitor effectively the boost phase and to utilize the information derived for subsequent guidance dictates the accuracy of the on-board system. The trade-off is then whether the on-board guidance system should do the boost guidance as well.

5.1.2 Lunar Orbit

1. The on-board measurement system determines the accuracy with which position, velocity and orbital parameters may be determined. If necessary, the velocity correction accuracy will also influence how accurately these elements may be maintained.

2. Initial condition errors introduced by the outgoing last velocity correction may be reduced by subsequent in-orbit velocity corrections. Thus a compromise must be made as to how to allocate accuracy and velocity corrections between these two phases.

3. In addition, in orbit velocity corrections may be made to establish desirable initial conditions for the return leg phase.

5.1.3 Lunar Orbit Ejection to Earth Landing

1. The self-contained re-entry navigation and guidance accuracy must allow an accurate return to a pre-selected landing sight within the allowable CEP from re-entry (400,000 ft) to landing.

2. The midcourse position and velocity determination must establish position and velocity velocity at the re-entry point so that the landing accuracy can be achieved. Thus items 1 and 2 interact. These accuracy requirements establish instrumentation accuracies and methods of processing data.

3. The return leg midcourse velocity corrections establish the re-entry corridor variations or velocity and position deviations from the reference re-entry conditions. These deviations influence the atmospheric re-entry vehicle maneuver requirements and as a result the L/D requirements for eliminating re-entry point inaccuracies.

4. The magnitude, number and time of application of return leg midcourse velocity corrections are established by position and velocity measurement accuracy and the

accuracy with which the velocity correction may be made. A statistical estimate of the velocity required yields an estimate for the fuel requirements for the return leg mission.

5. The accuracy with which ejection final conditions may be established from lunar orbit determines the initial errors for the return leg. The measurement system accuracies and guidance concept used to establish the lunar escape velocity increment are influenced by the required accuracy for this phase.

In essence then, the accuracy of the on-board inertial, optical, radio and back-up ground based radio tracking equipment and maneuvering requirements of the re-entry vehicle are determined from the enumerated considerations.

Compatibility of the overall mission is demonstrated when all the conflicting requirements are compromised to achieve one simple basic system design. Additional study is required to fulfill this objective.

Emphasis will be first placed on the outgoing leg of the mission from boost to lunar aimpoint acquisition.

Emphasis has been placed on incorporating into the analysis what is presently considered to be feasible equipment capability.

5.2 SUMMARY OF RESULTS

5.2.1 Insertion

At insertion the position error will be 1 n mi and the velocity error will be 15 ft/sec. These errors are for each coordinate axis. The errors are quoted for a high quality inertial measurement unit. As will be mentioned the midcourse velocity requirements for the entire mission are determined primarily by the insertion errors. These errors may be reduced by updating the inertial platform with a stellar fix. The velocity error could then be reduced to 10 ft/sec or less.

5.2.2 Outgoing Midcourse

For the outgoing phase two velocity measurement errors were considered: one ft/sec and ten ft/sec. Also two guidance concepts: Case I - maintaining reference trajectory conditions in position, velocity, and time at the lunar aimpoint; and Case II - maintaining only position and time at the aimpoint. These concepts were studied by analytical methods assuming linear analysis is applicable. At this time no thorough evaluation has been completed. The results may be considered as preliminary estimates.

The study indicates that for Case I the total velocity requirements are 400 ft/sec and for Case II 150 ft/sec. These values can be reduced (or increased) proportionately to corresponding variations in insertion errors.

It was found that for an aimpoint located at approximately 20,000 miles from the Moon's center position uncertainties at this point would be from 2 to 20 miles.

5.2.3 Transition Phase

The cases studied aimed for an aimpoint remote from the distance of closest approach at the Moon. From this point (20,000 miles) the distance-of-closest approach could vary from 10 - 100 miles for velocity measurement uncertainties of one to 10 ft/sec.

5.2.4 Lunar Orbit

Several retro-thrust phases will be used for establishing the lunar orbit. Perilune can be established from 2 to 12 miles depending on measurement accuracy. Redundant data processing techniques have not been studied as yet for the lunar orbit phase but indications are that the 2 mile value can be met.

5.2.5 Terminal Escape

The velocity uncertainty used for this phase was 10 ft/sec and 2 to 5 miles position uncertainty. Since the return leg velocity corrections are less than outgoing requirements, terminal escape accuracy is not too critical when all velocity corrections for the mission are summed in an RMS manner. (i.e. 20 to 30 ft/sec errors can be tolerated.)

5.2.6 Return Midcourse

For 1 ft/sec velocity measurement accuracy and 10 ft/sec disorbit error the total return midcourse corrections will be 40 ft/sec.

5.2.7 Re-Entry

The down range error at initial re-entry will be ±50 to ±70 miles depending on measurement accuracy. The cross range error will be ±2 to ±15 miles.

5.2.8 Landing

For intial condition errors at the last aimpoint of 3 miles in position and 7 ft/sec in velocity, the impact down range error will be 2.5 miles and the cross range error will be 2.2 miles. The altitude error will be 4.1 miles.

5.2.9 Total Mission Midcourse Velocity Requirements

The total velocity requirements will be approximately 400 ft/sec for Case I and 150 ft/sec for Case II. These values in turn are primarily dependent on insertion errors.

5.3 INSERTION ERRORS

The first phase of the analysis deals with the insertion phase. This phase will include the boost phase, coast phase - if used, and final injection velocity increment to reach a burnout velocity sufficient to accomplish the intended mission.

An analysis of the insertion phase yields an estimate of expected burnout errors. These errors are required to establish the initial condition errors for the ensuing lunar trajectory and as a result miss components at the selected aim point.

This estimate yields expected accuracies for the boost guidance inertial measurement unit located in the Saturn C-2 booster. As a consequence this estimate also establishes the quality of the APOLLO spacecraft inertial measurement unit if it is desired that this unit perform the boost guidance function.

The booster escape trajectory profiles used for this error analysis are those given in the NASA Project APOLLO Working Paper No. 1002 "Saturn C-1 and C-2 Booster

Systems". Cases I, II, III and IV for the Saturn C-2 eight engine escape trajectories were considered.

The escape profiles were used with a digital computer program to compute the burnout errors listed in Table II-5-I.

The coordinate axes are shown in Figure II-5-1.

Table II-5-II lists the instrument errors. These values are typical of the performance expected by existing inertial components.

The results of Table II-5-I will be used to give an indication of expected Saturn C-2 burnout errors. For future analyses velocity errors of 15 ft/sec (three sigma) or 5 ft/sec (RMS) and position errors of 6000 feet (three sigma) will be used for each coordinate axes.

As will be seen burnout errors can influence to a great extent the midcourse fuel requirements. The following discussion indicates how improvements in accuracy can be achieved.

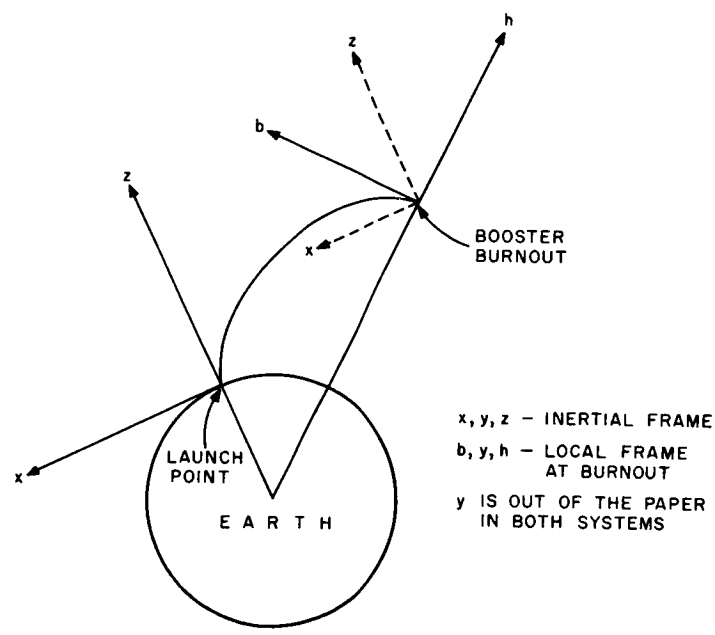


Figure II-5-1. Launch coordinator

TABLE II-5-I. SUMMARY OF BOOSTER BURNOUT ERRORS

Escape Trajectory	Total RMS Errors (1σ) Position (Inertial) δx (ft)	δy (ft)	δz (ft)	Velocity (Inertial) $\delta \dot{x}$ (ft/sec)	$\delta \dot{y}$ (ft/sec)	$\delta \dot{z}$ (ft/sec)	Position (Local) δb (ft)	δh (ft)	Velocity (Local) $\delta \dot{b}$ (ft/sec)	$\delta \dot{h}$ (ft/sec)
Case I	1059	1520	2326	2.031	4.99	6.56	2063	1508	3.94	5.63
Case II	1089	1529	2328	2.000	4.995	6.611	2091	1495	4.045	5.597
Case III	1120	1539	2326	1.989	5.002	6.637	2116	1480	4.118	5.573
Case IV	1150	1548	2328	1.979	5.007	6.675	2141	1470	4.215	5.542

Trajectory Information

Escape Trajectory	Burnout Surface Range (Degrees)	Burnout Path Angle (Degrees)	Burnout Altitude (n mi)	Burnout Velocity (ft/sec)
Case I	36.63°	88.4°	62.64	36,350.2
Case II	36.35°	87.5°	96.66	36,176.3
Case III	36.08°	87.0°	125.82	36,028.7
Case IV	35.79°	86.2°	157.68	35,861.3

The spacecraft will be equipped with stellar monitoring equipment. It may be possible to acquire the stars as soon as the vehicle is in light atmosphere (approximately 190 seconds after launch). The inertial platform could then be realigned to an accuracy of 5 seconds and maintained stable to this accuracy for the remainder of the flight. In this way, the major portion of inertial instrument error and nearly all of initial alignment error could be eliminated. The errors which would remain are the accelerometer errors, the error in velocity and position which was accumulated up to the time the stars were acquired and the five second misalignment error which will exist during the remainder of the flight.

TABLE II-5-II. INSTRUMENT ERRORS (THREE SIGMA)

Initial Misalignment	
x_o axis	30 seconds of arc
y_o axis	90 seconds of arc
z_o axis	30 seconds of arc
Constraint Drift	10 meru* (for all gyros)
Unbalance Drift	10 meru/g
Anisoelastic Drift	3 meru/g^2
Vibration	1.5 meru/g^2
Accelerometer Bias	0.03 cm/sec^2
Accelerometer Scale Factor	4×10^{-5}
*1 meru = 0.015 degrees/hr	
x_o, y_o, z_o - Fixed at launch	

To evaluate this approach, the previously described inertial system and a more advanced system were investigated and these cut-off velocity errors determined with and without the stellar correction scheme.

Without stellar correction, the velocity error is as before:

5 ft/sec (RMS)

With stellar correction introduced at approximately 190 seconds, the velocity error was reduced to:

3.5 ft/sec (RMS)

The second system was one of higher accuracy represented by a better quality inertial components. The velocity error without correction was:

4.5 ft/sec (RMS)

and with stellar correction, was reduced to:

1.7 ft/sec (RMS)

The specification of this advanced system appears in Table II-5-III. To calculate these results the acceleration profile of Case I was used.

TABLE II-5-III. ADVANCED INERTIAL SYSTEM

Gyros		
Constant Drift	3	Meru
Acc. Sensitive	3	Meru/g
$(Acc)^2$ Sensitive	1	Meru/g
Accelerometers		
Scale Factor	25	Parts/Million
Bias	.05	cm/sec^2
Initial Misalignment		
All Axes	10	sec

If a long coast phase is introduced before insertion (the coast phase may be as long as 25 min) the accuracies given will be degraded by several ft/sec. Again stellar fixes during this phase could improve accuracy.

The foregoing discussion yields estimates of boost guidance accuracies for high quality systems. We can now turn our attention to the midcourse problem to see what effect insertion errors produce.

5.4 OUTGOING MIDCOURSE

5.4.1 The Guidance Concept and Aimpoints

The outgoing midcourse phase is shown in Figure II-5-2. At insertion the spacecraft will be on a ballistic flight to the Moon. Insertion errors will perturb the nominal flight path. These errors have been treated in the preceding sections.

In addition variations in thrust and launch time delays at lift off will cause the launch guidance system to control two of the three parameters of position, velocity and time at the expense of a variation in the third. Thus if the boost guidance requirement is to hold time of arrival at the aimpoint constant and position displacement vector zero, the velocity at arrival will differ from standard conditions. If significant variations occur midcourse corrections must be made to eliminate these errors.

A sensitive parameter for the APOLLO mission is the distance-of-closest approach to the Moon. Another sensitive parameter is the total time of flight of the mission since the spacecraft must arrive back at a given earth landing site. It is also important to minimize large plane change requirements. To solve the total guidance problem effectively requires further study. A concept may be studies, however, which provides useful answers. The guidance concept for which an error analysis is made attempts to maintain the APOLLO spacecraft on the reference trajectory and on time schedule. This is the most difficult problem in terms of fuel requirements. There are two aimpoint position vectors to select for missions which go into orbit around the Moon. One is located an hour or so away from nominal distance-of-closest approach to the Moon. This point is shown on Figure II-52. The spacecraft will be forced to arrive at this point in a fixed time after insertion. The spacecraft will arrive with a velocity error from standard conditions.

[redacted]

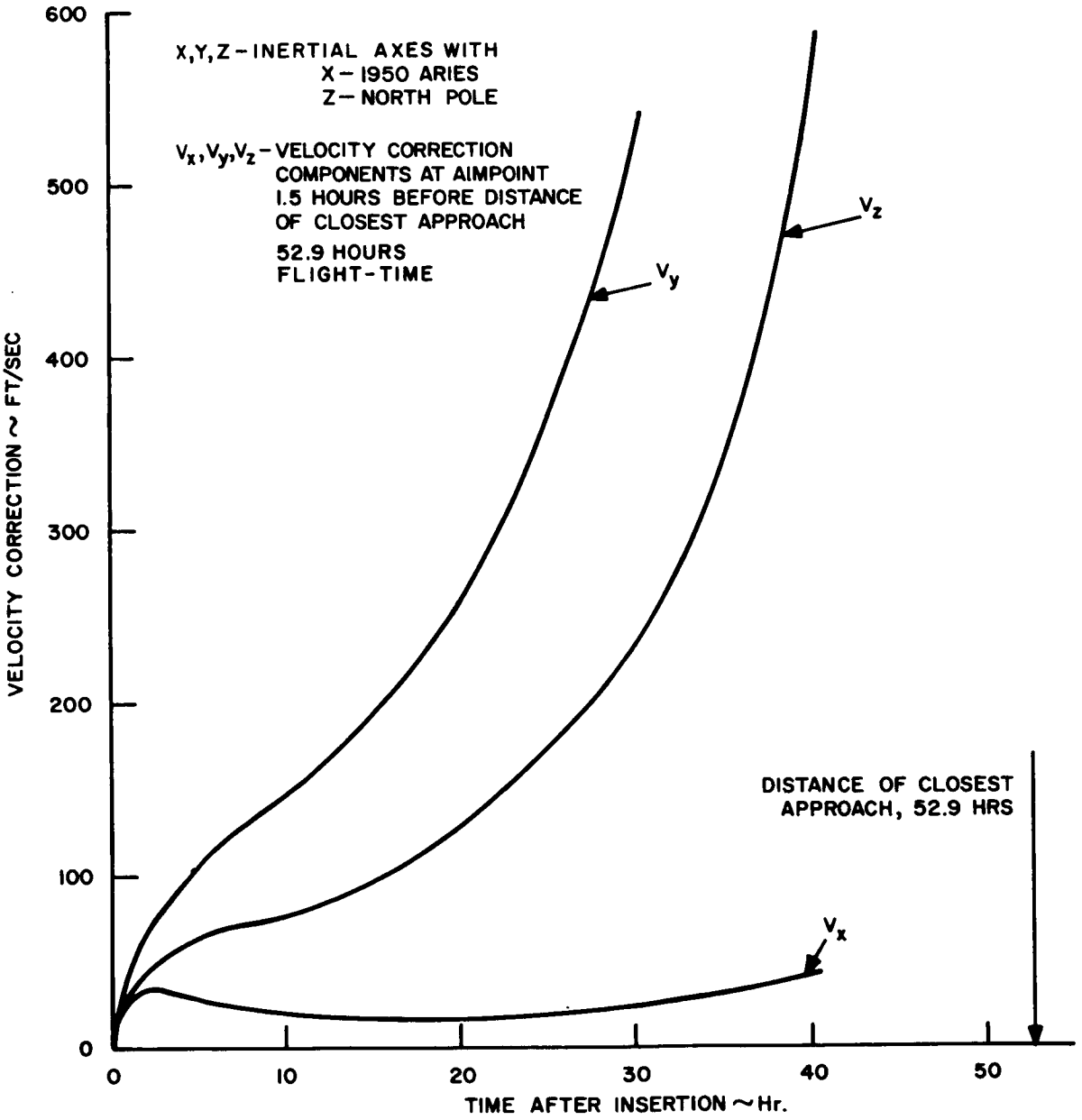


Figure II-5-2. First midcourse velocity correction required to null 15 ft/sec insertion errors

[redacted]

At the aimpoint this error can be eliminated by returning the spacecraft to standard conditions.

Standard conditions will be selected to yield a safe return trajectory. Once the aimpoint is acquired an hour or so will elapse before the decision must be made to orbit the moon or continue on to earth return. For a circumlunar flight the aimpoint could be more naturally the distance-of-closest approach. The other aimpoint is located about an hour from earth re-entry. This aimpoint will be discussed in the return midcourse study.

It should also be mentioned that no major attempt has been made to determine optimum parameters. An optimizing study should be made when overall system parameters are more firmly established.

5.4.2 Method of Analysis

In the analysis of this midcourse problem it is assumed that deviations in the midcourse trajectories, perturbed as a result of insertion errors, are linearly related to the reference trajectory. Preliminary computer runs indicate that linearity conditions will prevail in the regions of interest for 15 FPS perturbations in insertion conditions. The linearity problem is treated in Appendix T-H.

A linear statistical error analysis may be conducted to determine effect of errors. This analysis is reported in Appendix T-X and will be referred to later.

5.4.3 Measurement Errors

The on-board optical measurement equipment will introduce errors in the measurements. For this analysis spherical distribution of errors is considered. As indicated in Appendix T-L the actual distribution may be quite assymetric. The results indicate that at the end of a three hour smoothing interval the variances of velocity errors are*

$$\sigma_x^2 = 3\ (\text{ft/sec})^2$$
$$\sigma_y^2 = 30\ (\text{ft/sec})^2$$
$$\sigma_z^2 = 8\ (\text{ft/sec})^2$$

*In this analysis X, Y, Z refer to inertial coordinate axes with X pointing to 1950 Aries.

Assuming spherical distributions eases computational labor and allows preliminary error analysis to be performed independent of data processing techniques and availability of data. However, conclusive results can only be obtained when the actual operation of the system is studied.

Smoothing times to achieve desired levels of measurement accuracy must be realistically accounted for, however. These times may be determined by the use of analytical expressions derived for polynomial smoothing as discussed in Appendix G.

5.4.4 Velocity Correction Errors

As will be shown midcourse velocity corrections are not expected to exceed 100 ft/sec. The 6000 pound thrust engines can be shut down with a three sigma uncertainity of 100 pound seconds. For the empty 7000 pound APOLLO spacecraft the velocity uncertainity will be

$$\Delta V = \frac{100 \times 32}{7000} = 0.46 \text{ ft/sec}$$

The thrust vector can be controlled to better than one-half degree by the gimballed engines. This angular error would contribute approximately 0.9 ft/sec.

For the analysis presented a one ft/sec error (three sigma) is considered for velocity corrections.

Two three dimensional outgoing trajectories were investigated: a 53 hour flight time to the lunar distance-of-closest approach and a 71 hour flight time. The two will be compared to see if one or the other has an advantage in achieving better fuel savings.

A sample of the sensitivity coefficients for these trajectories is given in Appendix T-I. These coefficients will be used to obtain the results in the following studies.

5.4.5 Initial Aimpoint Errors

Table II-5-IV shows the sensitivity coefficients given in Appendix T-I for aimpoint perturbations at 53 hour flight time, for perturbations in insertion conditions. (The position coefficients $\frac{\Delta X_A}{\Delta X_I}$ are not given in Appendix T-I and are only included here.)

Using insertion errors of 15 ft/sec and 1 n mi (three sigma) and the results of Table II-5-IV gives the aimpoint errors in Table II-5-V. These errors are calculated at a constant time of flight of 53 hours, and in an inertial frame centered at the earth.

The large values of aimpoint errors of Table II-5-IV indicate that midcourse corrections will be required to at least meet requirements of passing within 1000 to 2000 miles of the Moon.

TABLE II-5-IV. MATRIX ELEMENTS, C (t_0), RELATING LUNAR POSITION AND VELOCITY VARIATIONS TO INSERTION ERRORS

	X_I	Y_I	Z_I	$\dot{X}_I$	$\dot{Y}_I$	$\dot{Z}_I$
X_A	-335	421	56	73	-24	-48
Y_A	-479	734	161	143	-57	-85
Z_A	-260	341	103	-68	-25	-44
$\dot{X}_A$	-470	566	86	91.5	-5.1	+89
$\dot{Y}_A$	387	-535	-123	-48	+13.3	-55.0
$\dot{Z}_A$	175	-200	-106	-13.1	+33.4	-4.1

X_I, Y_I, Z_I - Position Errors at Insertion, n mi

X_A, Y_A, Z_A - Position Errors Lunar Aimpoint, n mi

$\dot{X}_A$, $\dot{Y}_A$, $\dot{Z}_A$ - Velocity Errors at Aimpoint, ft/sec

$\dot{X}_I$, $\dot{Y}_I$, $\dot{Z}_I$ - Velocity Errors at Insertion, ft/sec

Time of Flight to Aimpoint = 52.9 hr

TABLE II-5-V. LUNAR AIMPOINT ERRORS

X_A	1460 n mi
Y_A	2400 n mi
Z_A	1200 n mi
$\dot{X}_A$	1350 ft/sec
$\dot{Y}_A$	1070 ft/sec
$\dot{Z}_A$	530 ft/sec

Using Equation 6 of Appendix T-X and the appropriate sensitivity coefficients the midcourse correction required to compensate for 15 ft/sec (3 σ) insertion errors can be determined. The results of these computations are shown on Figure II-5-2. The aimpoint for these calculations is located at 51.4 hours from insertion or 1.5 hours before reaching distance of closest approach. Figure II-5-2 can be used to assess the penalty in increased velocity (and consequently fuel) if the first correction is delayed in time.

The on-board data processing techniques being considered will be able to process large amount of data and smooth this information to get very accurate estimates of position and velocity. For example (see Figure III-100) in 4 hours of smoothing data, taken at a rate of one measurement every 60 seconds, velocity may be determined to within 1 ft/sec and to within 10 ft/sec in less than an hour.

Figures II-5-3, II-5-4, II-5-5 are curves of sensitivity coefficients relating aimpoint position displacement for perturbed reference velocity. As may be seen the coefficients drop rapidly in the first two hours and then level off.

Aimpoint perturbations as a result of position errors are second order effects when compared to velocity errors. The effect of position errors has not been included for the midcourse correction analysis except for initial insertion errors and re-entry corridor studies.

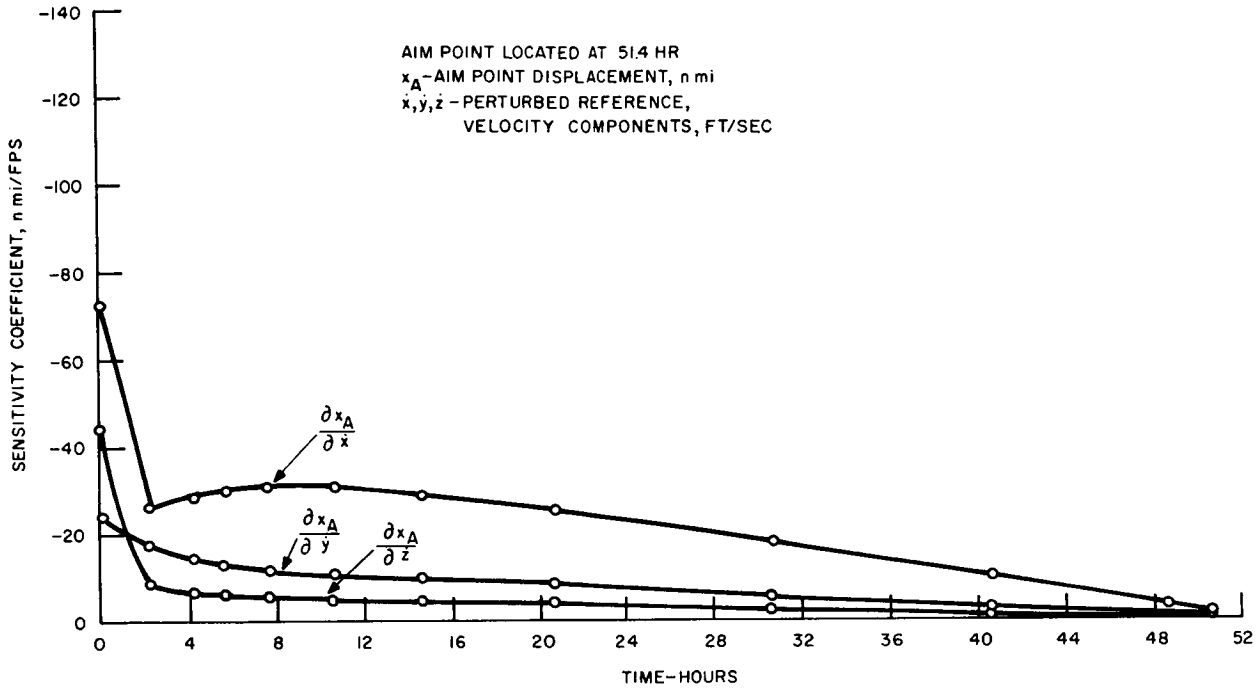


Figure II-5-3. Elements of $C_2(t)$, sensitivity coefficients for 52.9 hr nominal out going flight

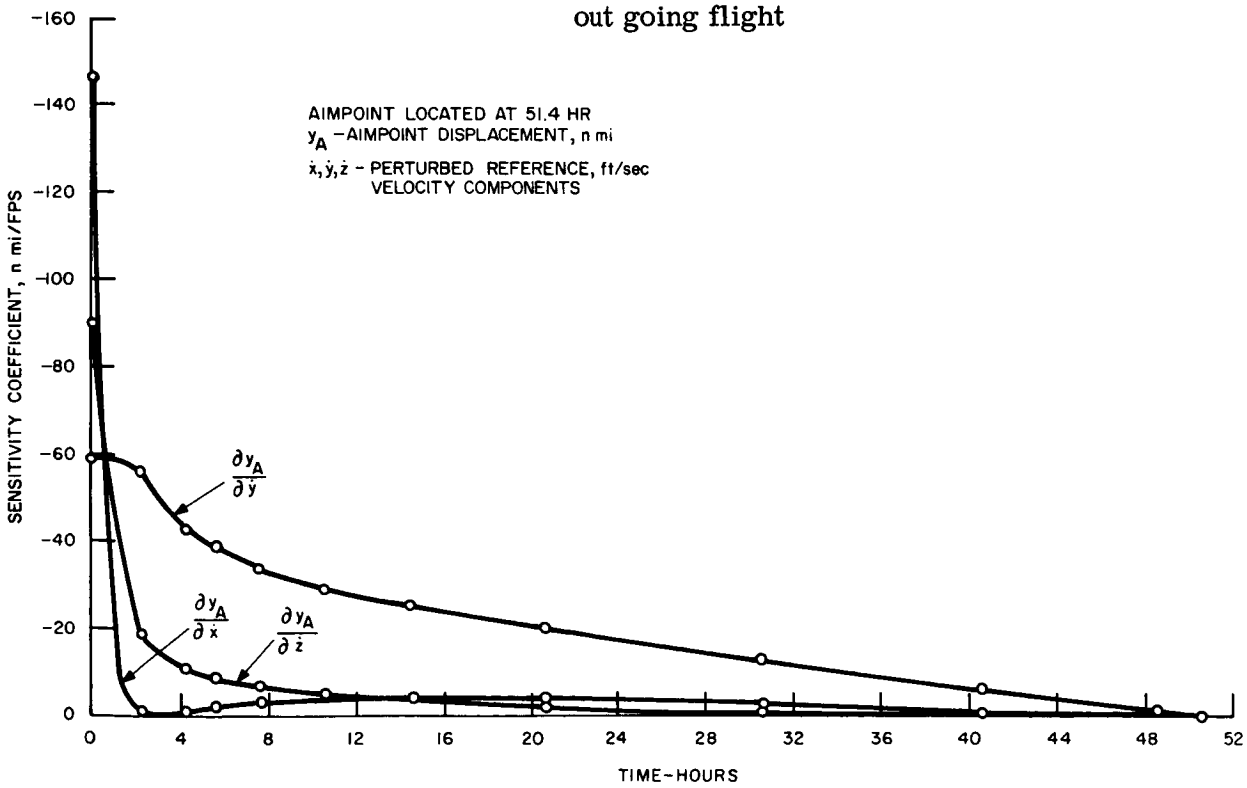


Figure II-5-4. Elements of $C_2(t)$ sensitivity coefficients for 52.9 hr nominal out going flight

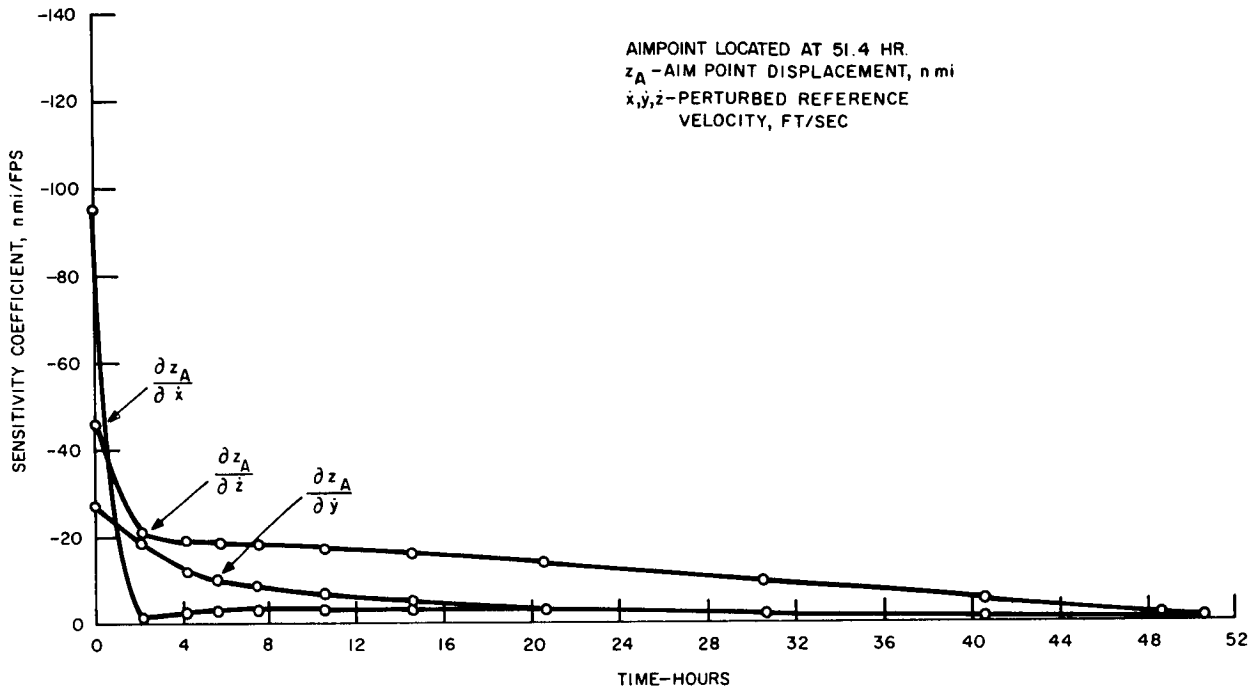


Figure II-5-5. Elements of $C_2(t)$ sensitivity coefficients for 52.9 hr nominal out going flight

To indicate representative results we can consider that measurements can be taken to an accuracy of one foot per second and a correction made at 4 hours. Since the velocity correction can only be made with an accuracy of 1 ft/sec further data processing will not be too effective.

At 4 hours, from Figure II-5-2 the three sigma values of correction components are:

$$V_x = 30 \text{ ft/sec}$$

$$V_y = 90 \text{ ft/sec}$$

$$V_z = 60 \text{ ft/sec}$$

And using Figures II-5-3, II-5-4, II-5-5 the aimpoint miss distance with a 1 ft/sec (3σ) error is

$$x_A = 33.5 \text{ NM}$$

$$y_A = 43 \text{ NM}$$

$$z_A = 22.5 \text{ NM}$$

This result is obtained from Equation 9 of Appendix T-X with the measurement and correction velocity moment matrix

$$M_{ev\,ev_c} = \begin{vmatrix} 1 & 0 & 0 \\ 0 & 1 & 0 \\ 0 & 0 & 1 \end{vmatrix}$$

and the elements of $C_2(t_1)$ obtained from Figures II-5-3, II-5-4, II-5-5.

After this first correction if the residual miss distance is too great additional corrections can be made.

The next correction velocity is determined with the use of Figures II-5-6, II-5-7, and II-5-8. These coefficients are the inverses of the ones shown in Figures II-5-3, II-5-4 and II-5-5 and are used to complete the velocity correction to null miss distance components. The miss distance to be corrected is given by the previous residual errors.

For the problem at hand, if all that is required is to come within 1000-2000 miles of the Moon the task is easily accomplished.

If we proceed to make corrections to obtain better accuracy at the aimpoint the following results would be obtained.

At t = 40 hours a correction of

$$V_x = 4 \text{ ft/sec}$$
$$V_y = 7 \text{ ft/sec}$$
$$V_z = 0$$

would be made.

The aimpoint error now would be

$$x = 10 \text{ nmi}$$
$$y = 6 \text{ nmi}$$
$$z = 6 \text{ nmi}$$

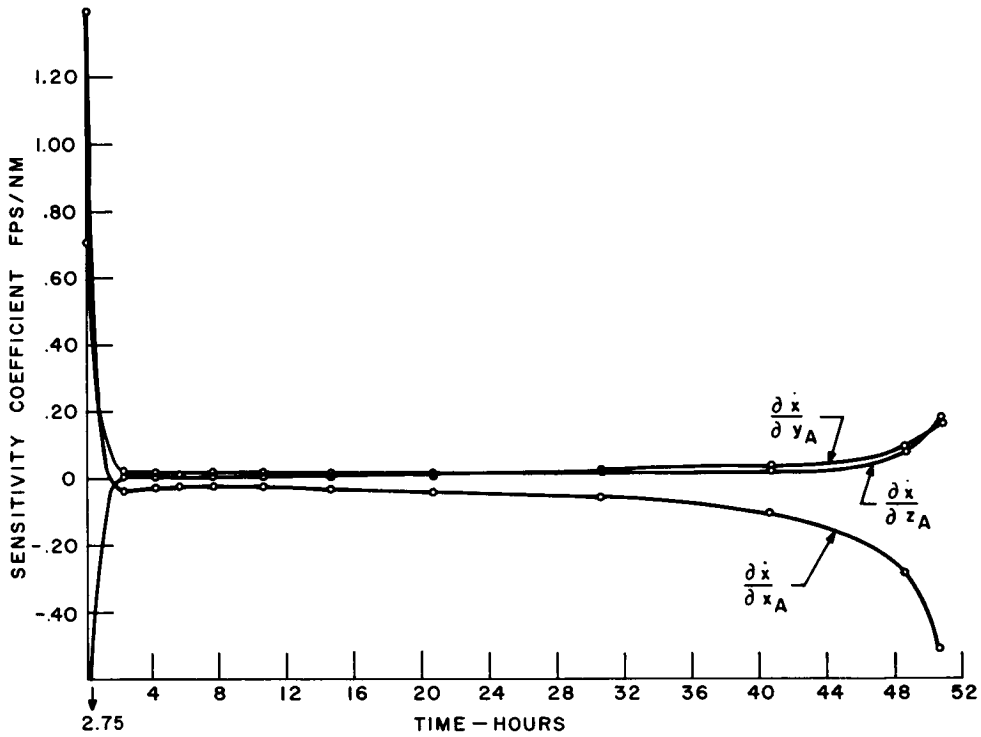


Figure II-5-6. Elements of the $A_2(t)$ matrix

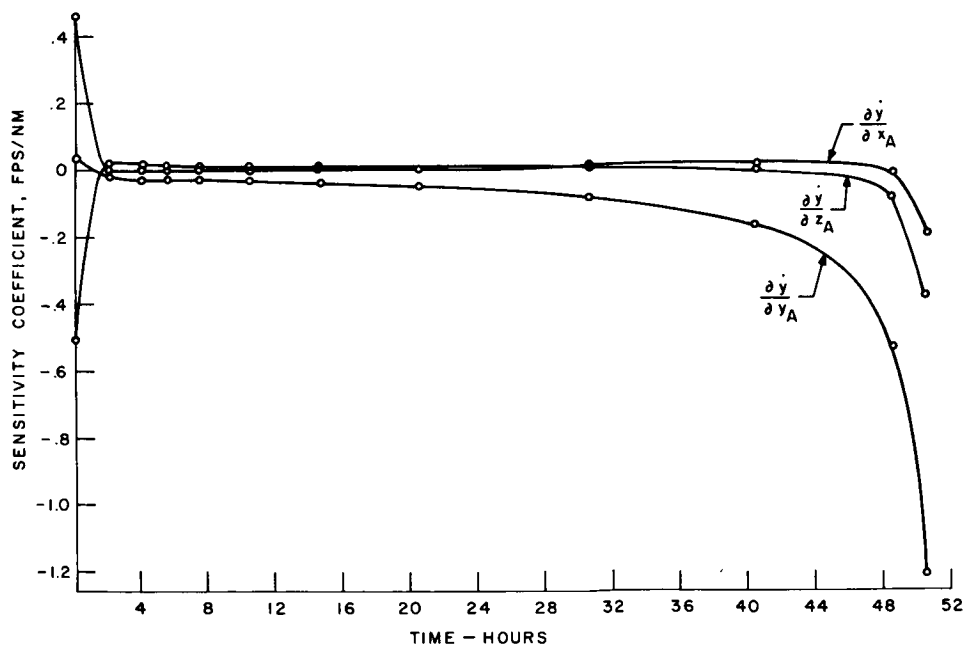


Figure II-5-7. Elements of the $A_2(t)$ matrix

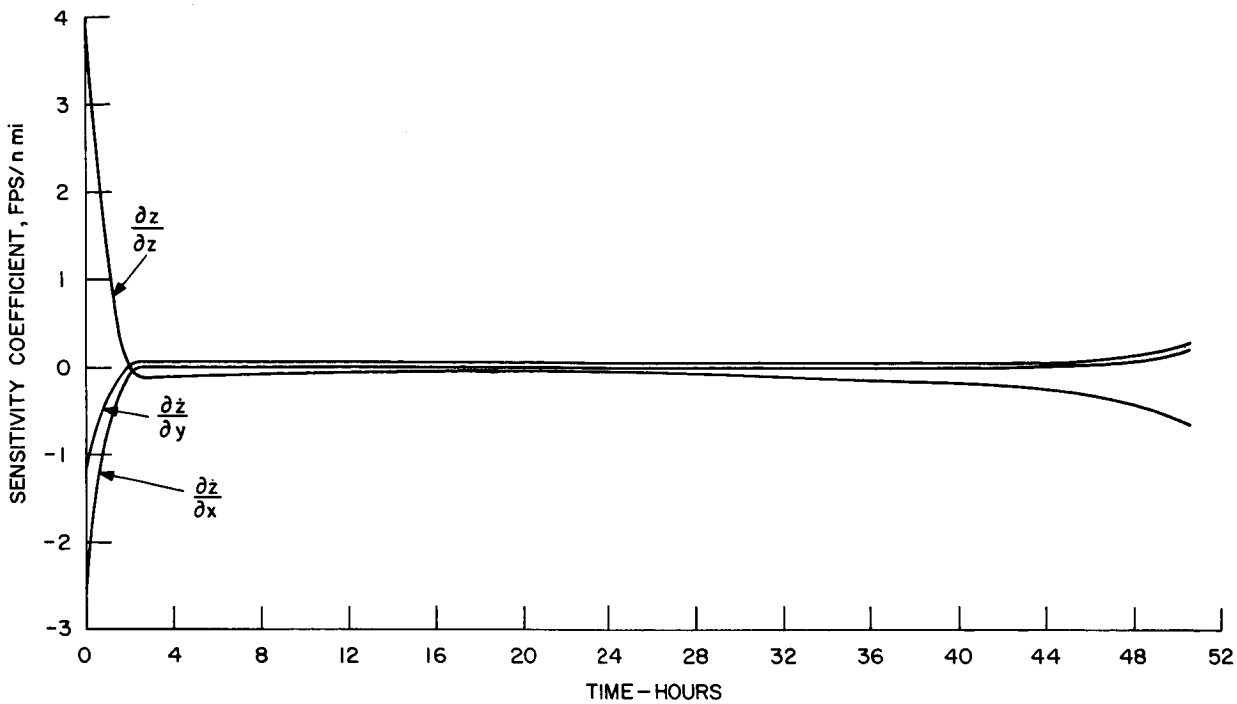


Figure II-5-8. Elements of the $A_2(t)$ matrix

and a last correction at 48 hours would yield:

$$V_x = 2.8 \text{ ft/sec}$$

$$V_y = 3 \text{ ft/sec}$$

$$V_z = 3 \text{ ft/sec}$$

with a resulting aimpoint error of

$$x = 4 \text{ nmi}$$

$$y = 2 \text{ nmi}$$

$$z = 1 \text{ nmi}$$

Table II-5-VI summarizes these results. The Root Sum Square (RSS) of the total maneuver is 113 ft/sec.

TABLE II-5-VI — OUTGOING MIDCOURSE CORRECTIONS

Time of Correction	Velocity Correction			Miss Distance		
	V_x	V_y	V_z	X_A	Y_A	Z_A
(HRS)	ft/sec	ft/sec	ft/sec	(NM)	(NM)	(NM)
4	30	90	60	33.5	43	22.5
40	4	7	--	10	6	6
48	2.8	3	3	4	2	1
TOTAL (3σ)	30	90.5	60			
RSS		113				

The results of Table II-5-VI can be summed in an RMS manner to get total velocity requirements. It is seen that only the first correction is of significance.

At 20,000 miles from the Moon the sensitivity coefficients relating perilune (at 2000 nmi) of the hyperbolic approach orbit are

$$\frac{\Delta r_p}{\Delta \gamma} = 187 \text{ nmi/degree}$$

$$\frac{\Delta r_p}{\Delta v} = 2.12 \text{ nmi/ft/sec}$$

where r_p = perilune

v = velocity magnitude

γ = flight path angle (measured from local horizontal)

The resulting perilune variation can be summed in a RMS manner to get

$$\Delta r_p = \sqrt{\left(\frac{\Delta r_p}{\Delta} \quad \Delta\right)^2 + \left(\frac{\Delta r_p}{\Delta v} \quad \Delta v\right)^2}$$

$$\text{For} \quad v = 2500 \text{ ft/sec}$$
$$\gamma = -69 \text{ degrees}$$
$$\Delta\gamma = \frac{\Delta v}{v \cos\gamma} \times 57.3 \text{ degrees}$$

The perilune variation would be (for $\Delta v = 1$ ft/sec)

$$\Delta r_p = \sqrt{\left[\frac{187 \times 1 \times 57.3}{870}\right]^2 + \left[2.12\right]^2} = 12.4 \text{ NM}$$

5.4.6 Effect of Measurement Error

If 10 ft/sec measurement errors are allowed the midcourse correction sequence would appear as given in Table II-5-VII.

TABLE II-5-VII — OUTGOING MIDCOURSE CORRECTIONS

Time of Correction	Correction			Miss Distance		
	V_x	V_y	V_z	X_A	Y_A	Z_A
(HR)	ft/sec	ft/sec	ft/sec	(NM)	(NM)	(NM)
4	30	90	60			
34	19	34	26			
41	26	24	29			
50	40	60	40	20	10	10
TOTAL (3)	60	117	82			
RSS		154				

The example cited is typical of what can be expected for the return leg if a measurement of velocity to 1 ft/sec is possible. Further study is necessary to confirm these estimates.

Table II-5-X yields the expected position errors at the aimpoint (10,000 miles) in inertial Cartesian coordinates. We can convert these errors to spherical coordinates at re-entry to obtain the re-entry corridor and expected down range and cross range errors at re-entry. For the assumed one ft/sec velocity error only the aimpoint position errors are important.

From Appendix T-W (Figure III-131) we obtain

$$\frac{\Delta\ rp}{\Delta\ X} = \ .2$$

$$\frac{\Delta\ rp}{\Delta\ Y} = +\ 0.65 \qquad (1)$$

$$\frac{\Delta\ rp}{\Delta\ Z} = +\ 0.85$$

$$\text{and } \Delta\ rp = \sqrt{\left(\frac{\Delta\ rp}{\Delta\ X}\ \Delta\ x\right)^2 + \left(\frac{\Delta rp}{\Delta Y}\ \Delta Y\right)^2 + \frac{\Delta\ rp}{\Delta\ Z}\ \Delta Z^{\ 2}} \qquad (2)$$

From Equation 20 of Appendix T-W, we obtain for the cross plane error, ΔCR, at re-entry

$$\Delta\ CR = 2.4 \text{ nmi}$$

This result is derived from

$$\Theta = 111 \text{ degrees}$$

$$v = 21{,}000 \text{ ft/sec}$$

$$\gamma = 52 \text{ degrees}$$

$$r = 10{,}000 \text{ nmi}$$

In this case the perilune variation would be ± 124 nmi. For a nominal distance of closest approach of 1000-2000 miles, the perilune error is still tolerable. To reduce this error would require moving the aimpoint closer to perilune.

From Table II-5-V and II-5-VI it is seen that although aimpoint accuracy has decreased, the total velocity requirements have not changed appreciably. However, if insertion errors are redueed the velocity requirements are reduced proportionately until measurement errors become predominant.

5.4.7 Outgoing Velocity of Arrival

Insertion errors cause aimpoint miss distances and velocity of arrival errors resulting in non-standard lunar orbital elements. Either miss distance or arrival velocity errors may be nulled, but not both simultaneously. The velocity of arrival errors are correlated with miss distance errors. (Appendix T-X). If the velocity errors are eliminated by a velocity correction at a later time, such as at the aimpoint, the RMS magnitude of the correction must be added directly to the RMS value of the midcourse correction. Excessive arrival velocity errors may change the lunar orbital elements from nominal for purposes of satisfying return leg requirements and establishing a particular lunar orbit.

In the first analysis of this problem it is desired to establish the spacecraft to near nominal conditions. Thus both position vector and velocity vector at arrival will be controlled. It is found that a velocity correction of (3σ values)

$$\begin{aligned} V_X &= 150 \text{ ft/sec} \\ V_Y &= 90 \text{ ft/sec} \\ V_Z &= 250 \text{ ft/sec} \end{aligned} \qquad (1)$$

will be required at the aimpoint to restore standard conditions. Equation 13 of Appendix T-I were used to derive this result.

These values should be added to the total results of Table II-5-VI to get for the entire outgoing maneuver (3σ values)

$$\begin{aligned} V_X &= 180 \text{ ft/sec} \\ V_Y &= 180 \text{ ft/sec} \\ V_Z &= 310 \text{ ft/sec} \end{aligned} \qquad (2)$$

Again these values are dependent on insertion errors. If the errors were reduced (or increased) from 15 ft/sec the results would change proportionately.

For the mission in which a lunar orbit will be established a sizable (2000 to 3000 ft/sec) velocity increment will be subtracted from the hyperbolic approach velocity. Arrival velocity errors may be eliminated simultaneously if the outgoing aimpoint is located at the retro-thrust point. Out-of-plane velocity components (essentially $\dot{V}_Z$ in this analysis. (See Chapter I for nominal lunar orbit elements) may be eliminated with little penalty.

When total guidance requirements for all phases of the lunar mission are satisfactorily compromised the restriction on returning to a standard position and velocity at a given time may be relaxed. Total velocity requirements for the outgoing leg would then approach the values given in Table II-5-VI.

5.4.8 Low Energy Outgoing Trajectory

A lower energy trajectory than the 52.9 hour flight time is analyzed to indicate if velocity corrections are influenced appreciably by the choice of trajectories. The time of flight of this trajectory is 71 hours from insertion to within 1000 nmi of the Moon. Figure II-5-9 shows the midcourse correction required to correct for 15 ft/sec insertion errors.

Table II-5-VIII shows a comparison of the two trajectories.

TABLE II-5-VIII — COMPARISON OF 53 HOUR AND 71 HOUR FLIGHT TIMES

	Magnitude of Correction (RSS)	
Time of Correction (hr)	53 Hr Flight (ft/sec)	71 Hr Flight (ft/sec)
5	130	112
10	168	170
20	280	260

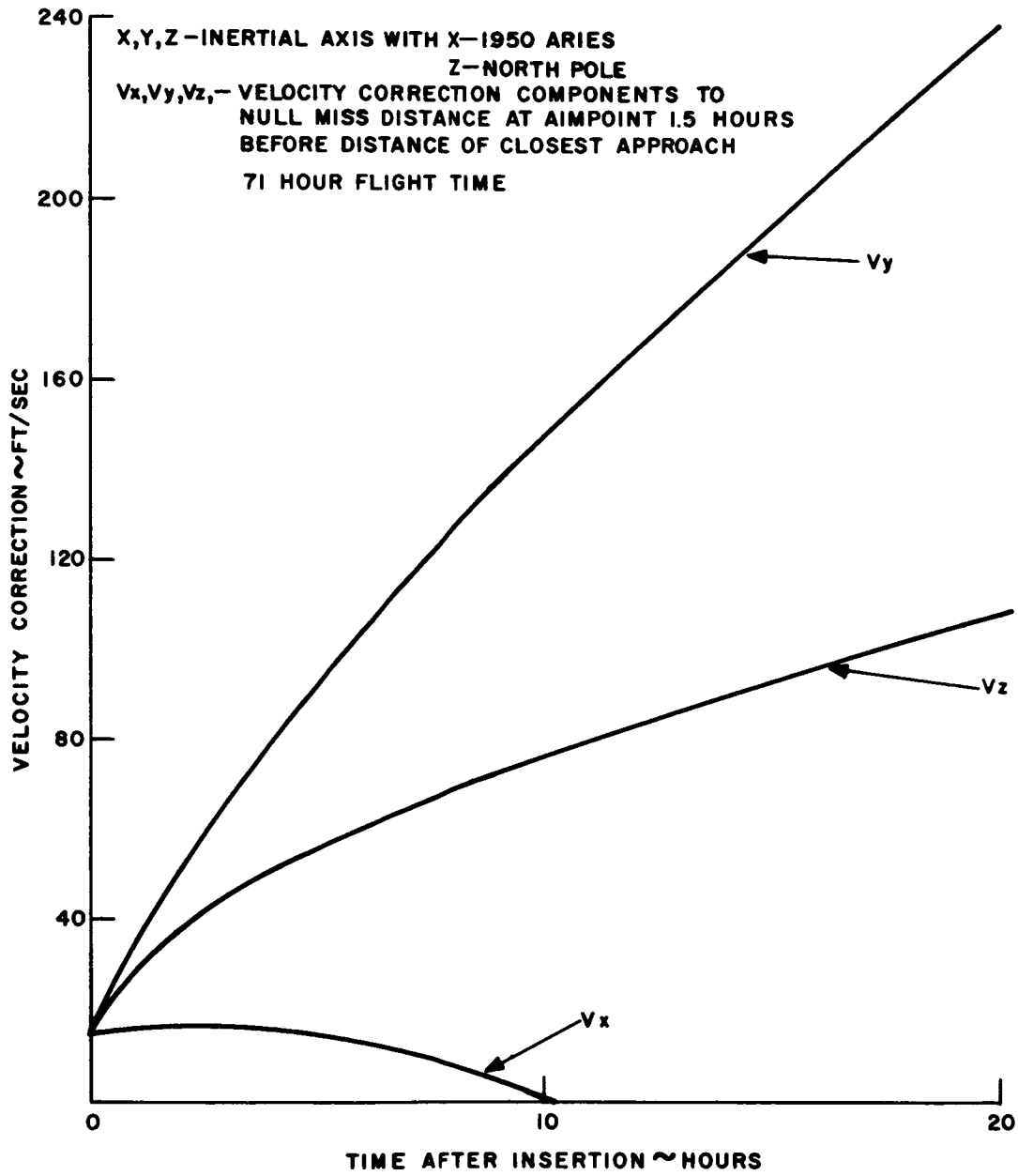


Figure II-5-9. First mid-course velocity correction required to null 15ft/sec insertion errors

The results of this comparison indicates that there is little to choose between trajectories on a basis of magnitude of the first velocity correction.

5.5 TRANSITION FROM OUTGOING LEG TO TERMINAL PHASE

The transition from the outgoing leg to the terminal phase may be considered to occur where the last midcourse velocity correction is made prior to applying a retro-thrust to orbit the Moon for the nominal mission. The transition is demonstrated by reference to Figure VII-5-10. At r_2 a midcourse correction is made which places the vehicle nearly on the nominal trajectory. In this case r_2 would constitute the nominal aimpoint and velocity-of-arrival errors would be eliminated at this point. The vehicle would then arrive at r_1 with residual velocity and position errors from nominal conditions as a result of errors existing at r_2. The distance r_2 must be established such that the desired time delay before initiating the retro-thrust C_2 at r_1 is compatible with allowable arrival errors at r_1.

From the results of previous investigation for outgoing leg midcourse corrections the errors propogated to point r_1 would be of the order of 12 to 120 miles in distance of closest approach. (For 1 to 10 ft/sec measurement errors). Any velocity errors may be eliminated simultaneously with the retro-thrust starting at r_1 and terminating at r_0 and with subsequent orbital corrections.

For the "modified ellipse" trajectory used, r_2 would be about 20,000 miles for a two hour time delay before reaching the nominal distance-of-closest approach at 53 hours.

Since first a circular orbit then a highly elliptic orbit will be established, errors may be compensated for easily in this phase when large velocity corrections are made.

5.6 INJECTION INTO ORBIT AROUND THE MOON

An investigation of injection sensitivities was made in order to determine the effects of measurement inaccuracies. Since there are no critical requirements on a 1000 nmi circular orbit and the 50 nmi altitude perilune can be approached in steps which can be monitored, the results indicate that lunar orbit injection places no highly critical requirements on hardware unless a highly precise (± 1 nmi perilune altitude tolerance) orbit must be achieved. Sensitivities for the case of first transferring to a circular

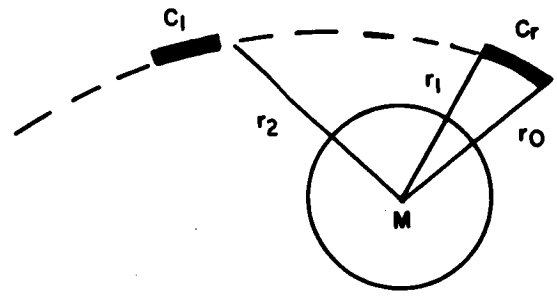


Figure II-5-10. Transistion/from out going leg to terminal phase

orbit about the Moon at an altitude of 1000 nmi above the surface of the Moon are summarized below:

(a) $\frac{\partial h'}{\partial h_i} = 3$ nmi/nmi

where

h_i = Injection altitude

h = Altitude 180 degrees from point of injection

(b) $\frac{\partial h'}{\partial \epsilon} = 25$ nmi/degree

where

ϵ = Angular error of incremental velocity vector with respect to the local horizontal in the plane of the orbit.

(c) $\frac{\partial h'}{\partial v_h} = 2$ nmi/ft/sec

where

v_h = horizontal velocity at injection point

(d) $\frac{\partial \beta}{\partial \alpha} = 0.76$ degree/degree

where

α = angular error of incremental velocity vector with respect to the vehicle velocity vector in local horizontal plane

β = Angle of orbital plane with respect to nominal plane determined by vehicle velocity vector.

After the circular orbit has been achieved by the application of a velocity decrement of about 2345 ft/sec, additional incremental velocity corrections, which will total about 699 ft/sec, can be made at the selected apolune position in order to bring the vehicle safely to a minimum altitude of 50 nmi. The sensitivity coefficients also change.

For Example,

$$\frac{\partial h'}{\partial v_h} = 0.93 \text{ nmi/ft/sec}$$

$$\frac{\partial h'}{\partial h_i} = 1.25 \text{ nmi/nmi}$$

when transferring directly from the circular orbit to the specified elliptical orbit.

In order to achieve a perilune altitude of 50 nmi with an RMS tolerance of 1 nmi, the error in velocity at the injection point must be less than 0.75 ft/sec and the altitude error must be less than 0.56 nmi for the case where velocity and position errors contribute equally to the total error. The anticipated errors at the time of closest

approach (10 ft/sec in velocity and 5 nmi in altitude) would result in an RMS error of 11.2 nmi at perilune if a direct transfer from the hyperbolic approach trajectory to the elliptical orbit were made.

Using data processing techniques discussed in APPENDIX T-L, an improvement in measurement accuracy is expected (i.e., velocity measured to 1 ft/sec and position to several miles). The perilune error could thus be reduced to 1 to 2 nmi (RMS) errors.

5.7 TRANSITION FROM TERMINAL PHASE TO RETURN LEG

The transition from the terminal phase to the return midcourse leg may be considered to occur at the escape thrust phase termination. Range and cross-range position should be known to 5 nmi or better for altitudes of 2000 nmi or less. It is estimated that velocity can be measured to 10 ft/sec or better, using the data processing methods described elsewhere. (See APPENDIX T-G and T-L).

An escape velocity increment of the order of 3000 ft/sec must be measured. With a stellar monitored reference accurate to 0.01 degrees (3σ) or better and accelerometers with scale factor errors of 1×10^{-4} the velocity increment can be measured to better than 2 ft/sec. Thrust tail-off uncertainties are expected to be ± 0.5 ft/sec or less for the 6000 pound engines.

The foregoing results are applicable to the retro-thrust accuracy for establishing the lunar orbit for the terminal retro-thrust phase on the outgoing leg.

5.8 RETURN MIDCOURSE ERROR ANALYSIS

5.8.1 General

The guidance objective for the midcourse return leg is to return the spacecraft to a nominal aimpoint with sufficient accuracy to allow a successful re-entry. A brief description of the events of interest will be aided by reference to the two dimensional figure shown in Figure II-5-11. At point 1 the spacecraft is injected into the nominal return path. Velocity and position errors will exist with respect to nominal conditions. These errors, in general, will not allow meeting practical re-entry corridor

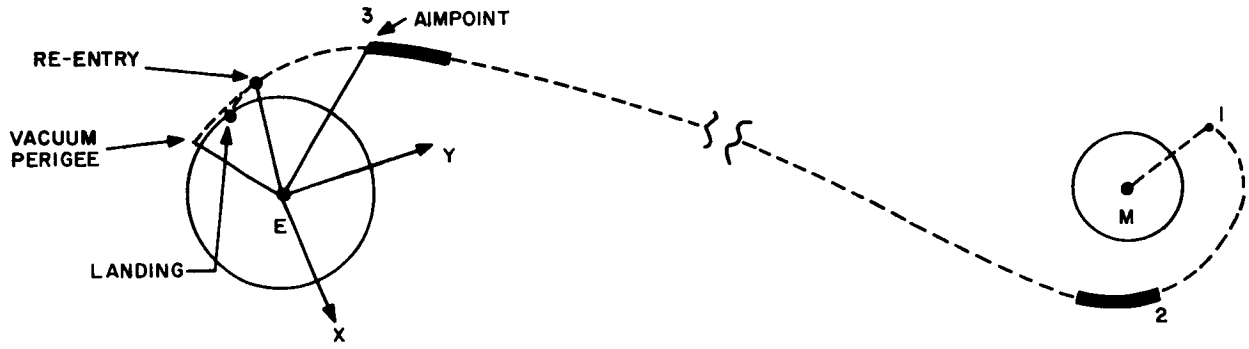


Figure II-5-11. Return midcourse phase

requirements. The allowable variation in vacuum perigee is 30 nmi ± 1.2 degrees re-entry flight path angle at an altitude of 400,000 feet. Midcourse corrections will be required.

The first correction, started at point 2, reduces the effect of injection errors. The miss distance components at the aimpoint are first corrected. These components are specified, as for the outgoing leg, in an inertial coordinate system indicated by X and Y axes. The three dimensional sensitivity coefficients to be discussed later are given in the X, Y, Z inertial coordinate frame with X along 1950 Aries.

The aimpoint,point 3, is selected to be one-half hour away from the re-entry point. This time is used to prepare for re-entry. The spacecraft, within small error, arrives at the aimpoint in position and time correspondence with reference conditions by virtue of a previous velocity correction. At the aimpoint a velocity correction is made to eliminate the arrival velocity error if necessary. The spacecraft is now on schedule. The mission module will be separated at this point and prior to separation the last celestial fix is made. This last position and velocity fix must be sufficiently accurate to be able to achieve a landing within a 10 nmi CEP by an onboard

command system. This requirement, as well as the minimum allowable time before re-entry, (in this 30-60 minutes), establishes the accuracy with which the last fix must be made.

In addition the velocity and position accuracy of the aimpoint after the last velocity correction must be sufficiently accurate as to not impose excessive atmospheric maneuvering requirements on the re-entry vehicle. The maneuvering requirements that will be imposed are those necessary to correct errors introduced by the inaccuracy in midcourse guidance in adhering to reference conditions. The degree to which reference conditions have been established, in the absence of guidance errors, also play a prominant role in determining vehicle maneuver requirements.

With this description of the problem we may now turn our attention to seeking a solution. In general non-optimum but representative solutions are discussed.

5.8.2 Return Leg Aimpoint Measurement Accuracy Requirements

It will be shown in Section 5.10 that the allowable measurement errors at 10,000 nmi can be 4 nmi and 6 ft/sec (RMS) which allow more than adequate accuracy to land within 10 nmi of the landing site.

The aimpoint is located at approximately 10,000 nmi from the earth. For the nominal 99.7 hour trajectory (flight-time to vacuum perigee) the aimpoint is 99.2 hours after insertion on the return leg trajectory.

5.8.3 Sensitivity Coefficients

The flight time of the sample return trajectory studied is 99.7 hours. Samples of the sensitivity coefficients relating aimpoint position (C_2 matrix) and velocity errors (C_4 matrix) to velocity errors along the reference trajectory are given in Appendix T-I.

Again these coefficients are determined by perturbing the reference trajectory by small increments (10 ft/sec) and observing the perturbations in aimpoint conditions at the reference time-of-flight.

By inverting the C_2 matrix we obtain the A_2 matrix which relates the required velocity correction along the reference trajectory to correct for an aimpoint position error.

5.8.4 Aimpoint Perturbations as a Result of Insertion Errors

We first determine the aimpoint perturbations for the insertion errors given in Section F. These errors will be of the order of 2 to 10 ft/sec. A linear analysis is used similar to that described in the outgoing leg discussion. The miss distance components are computed using equations given in Appendix T-X.

Table II-5-IX gives the three sigma estimate of the miss components in the X Y Z inertial frame at the aimpoint for 10 ft/sec (3σ) insertion errors. See Figure II-5-12.

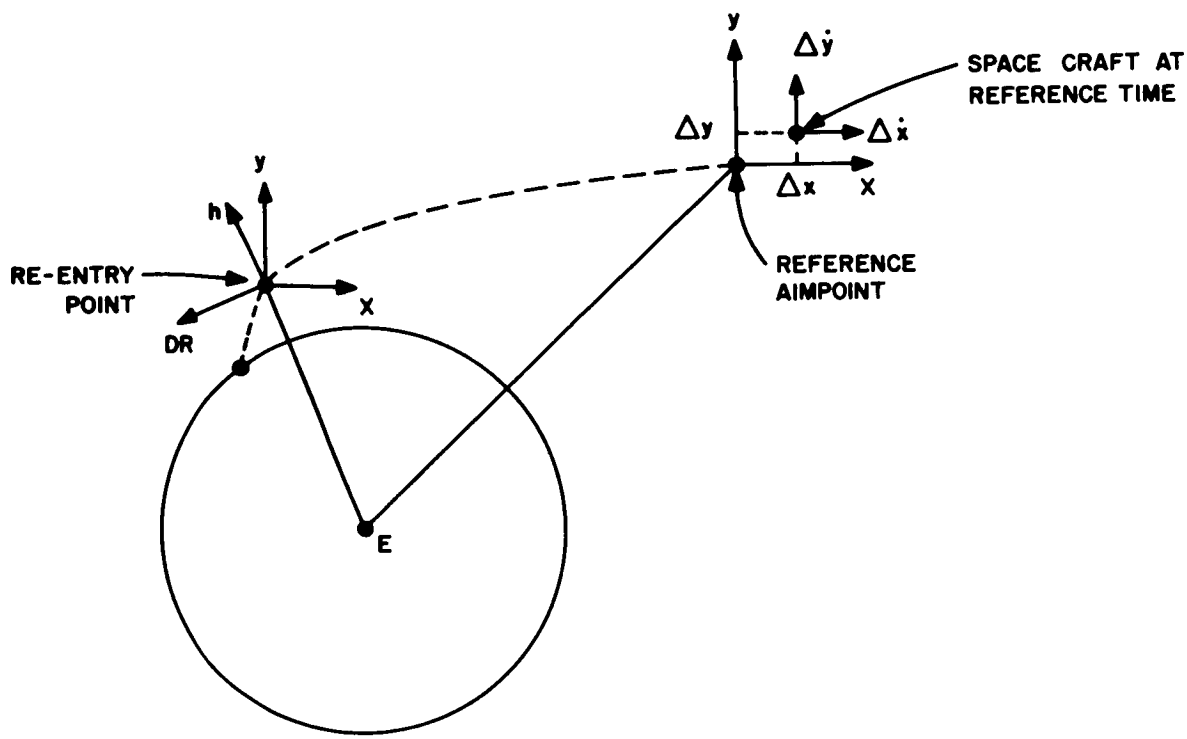


Figure II-5-12. Transition from midcourse to re-entry

TABLE II-5-IX — MISS COMPONENTS FOR LUNAR INSERTION VELOCITY ERROR

$X = 1750$ nmi

$Y = 680$ nmi

$Z = 1000$ nmi

$\dot{X} = 850$ ft/sec

$\dot{Y} = 345$ ft/sec

$\dot{Z} = 250$ ft/sec

These errors result in a re-entry vacuum perigee variation of 1000 nmi.

Figure II-5-13 shows the required midcourse velocity correction to null the effect of a 10 ft/sec insertion error. Shown in Figure II-5-14 are the expected arrival velocity errors at the aimpoint.

As indicated in the analysis of the outgoing midcourse phase smoothed measurement data can yield accurate velocity data after a few hours of data taking.

A representative return leg correction sequence may now be described.

Using the sensitivity coefficients in Appendix T-I a sequence of corrections can be demonstrated for a 1 ft/sec measurement error as indicated in Table II-5-X.

TABLE II-5-X — RETURN LEG CORRECTION SEQUENCE

Correction Time	Correction Velocity			Aimpoint Position Error		
(hr)	V_X (ft/sec)	V_Y (ft/sec)	V_Z (ft/sec)	X (nmi)	Y (nmi)	Z (nmi)
0	-	-	-	1750	680	1000
10	35	8	4	179	29	18
75	8	3	3	25	9	9
89	3	2	2	2	4	4
TOTAL (3σ)	36	8.8	5.4			
RSS	37					

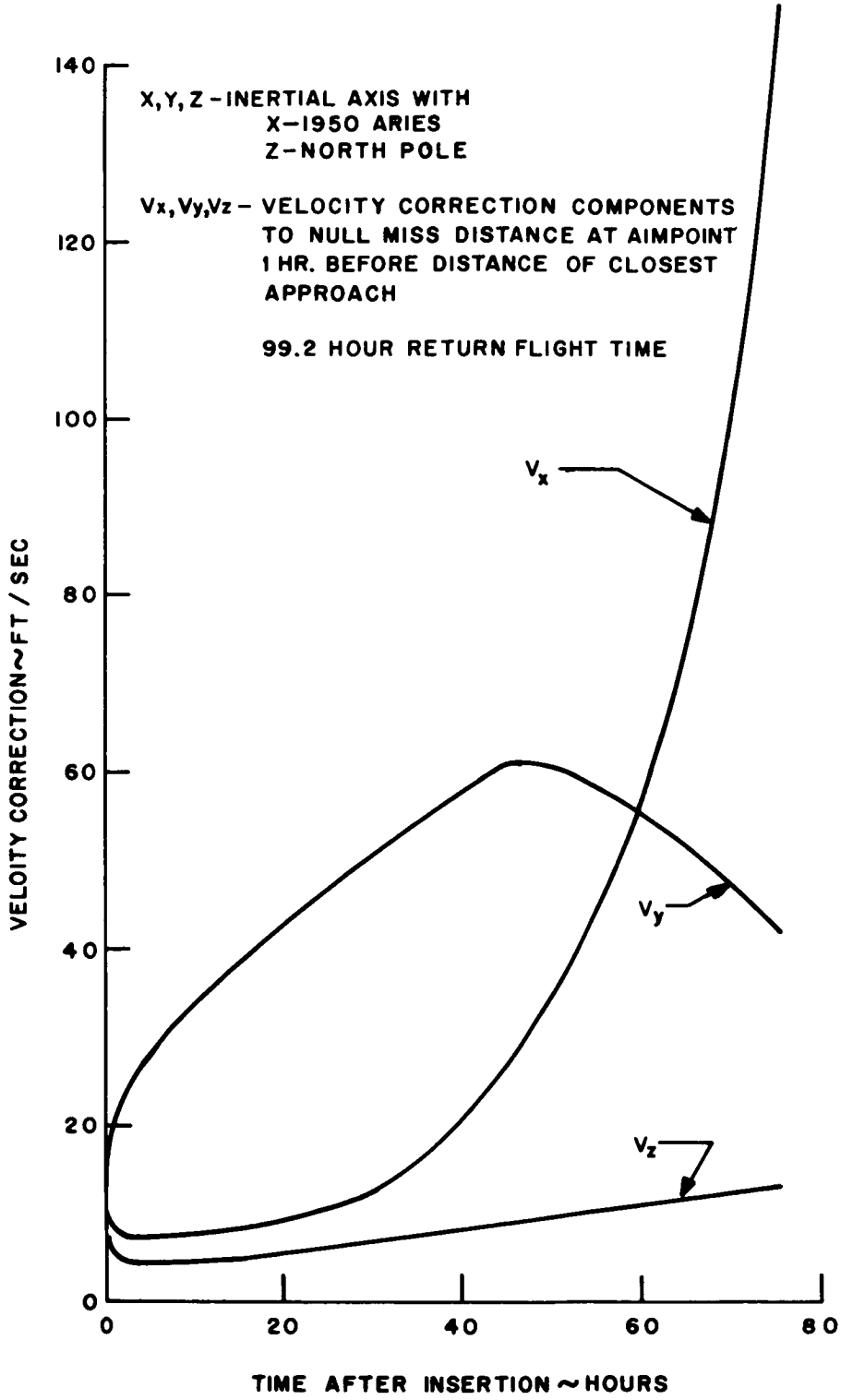


Figure II-5-13. First midcourse velocity correction required to null 10ft/sec insertion errors for 99 hour return trajectory

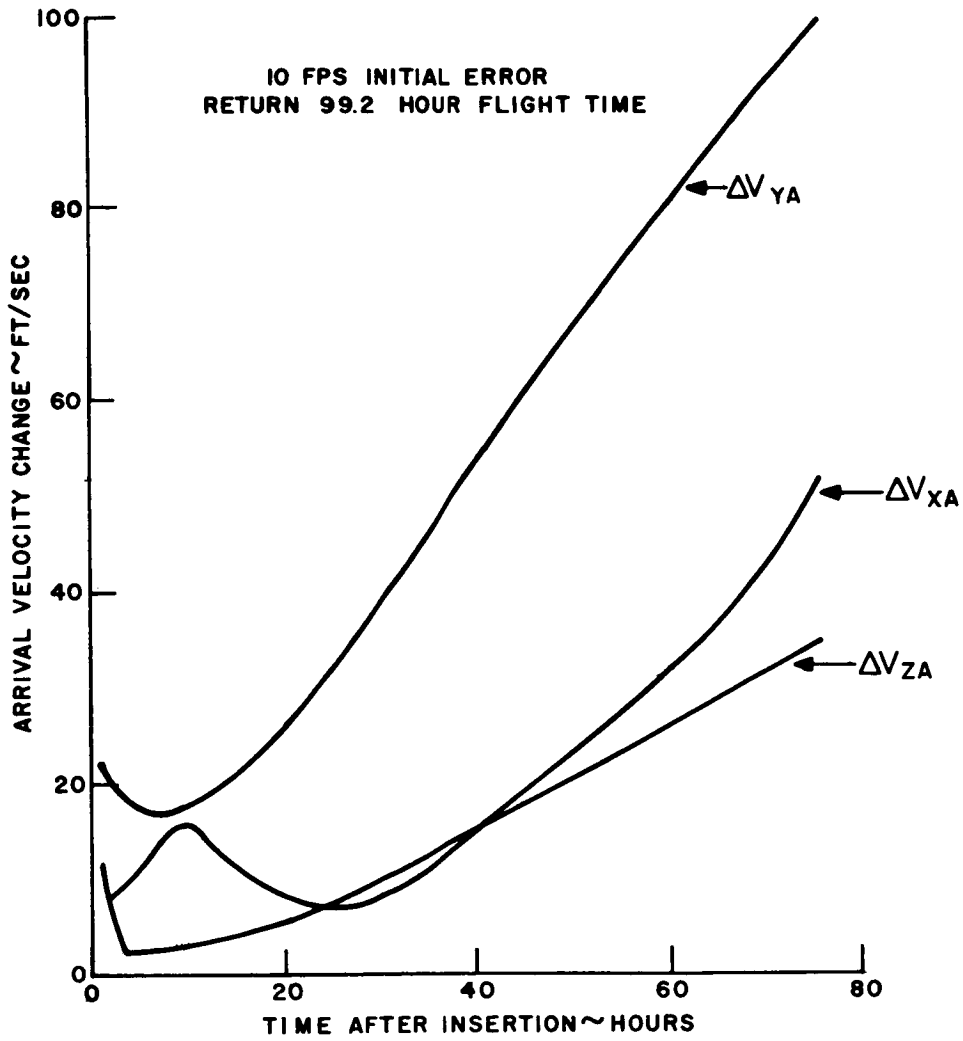


Figure II-5-14. Changes in arrival velocity after the first mid-course correction

In addition a time-of-flight variation to re-entry will occur which is approximately

$$\Delta t = 4.5 \text{ seconds}$$

for a sensitivity of 4.5 sec/ft per second error.

In this time the earth will rotate (and thus the landing site for Polar return) by

$$AW = W \Delta t = 0.019 \text{ degrees}$$

where

$$W = \text{earth rotation} = 15 \text{ degrees/hr}$$

or

$$\Delta CR = .019 \times 60 = 1.2 \text{ nmi (for a Polar return)}$$

From equation 2 we obtain a root Sum Square estimate of the re-entry corridor of 4.8 nmi.

The corridor will vary between ± 4.8 nmi.

The allowable corridor is 30 nmi (See Chapter I, Section 1.7).

This re-entry corridor variation corresponds to a re-entry flight path angle variation of ± 0.38 degrees (See Chapter I, Section 17) or Appendix T-W, Equation 17 & 18).

The down range, DR, location of the re-entry point at 400,000 feet will change from the nominal point by ± 0.76 degrees or ± 46 nmi. (See Appendix T-W, Equation 19.)

Thus

$$\Delta DR = \pm 46 \text{ NM} = \text{Down Range error}$$

These results are summarized in Table II-5-XI.

TABLE II-5-XI – RE-ENTRY ERRORS

Source of Error	Vacuum Perigee (nmi)	Re-Entry Down Range Error (nmi)	Re-Entry Cross Range Error (nmi)	Time-of-Flight Error (sec)
Initial Position	4.8	46	1.2	
Initial Velocity	0.64	small	small	4.5
RSS*	4.83	46	1.2	

5.8.5 Effect of Measurement Error

If 10 ft/sec measurement errors existed the last two corrections (at 75 and 89 hours) and aimpoint errors would be 10 times as large.

The arrival velocity error will be near the value as determined after the first correction. From Figure II-5-14 at 10 hours the arrival error is

$$V_{XA} = 16 \text{ ft/sec}$$

$$V_{YA} = 18 \text{ ft/sec}$$

$$V_{ZA} = 4 \text{ ft/sec}$$

This error can be corrected for at arrival at the aimpoint at 99.2 hours. The velocities would add directly to results of Table II-5-X.

For this case the vacuum perigee error would be ± 48 nmi and allowable re-entry corridor limits (30 nmi) would be exceeded. Thus if a 10 ft/sec measurement error exists we recompute the time of midcourse corrections to yield allowable corridor

* All errors have been summed by taking the square root of the sums of squares of all individual error components. This result does not yield the three sigma error ellipsoid.

[redacted]

limits. The last correction prior to reaching the aimpoint (in this case, at 10,000 nmi from the earth, 99.2 hours after leaving the lunar orbit) must be such that re-entry corridor requirements are preserved.

If a velocity correction is made at 97 hours from leaving the lunar orbit the sensitivity coefficients at t = 97 hours and the errors at the aimpoint will be

t = 97 hours are

$$\frac{\Delta XA}{\Delta \dot{X}} = 0.6 \text{ nmi/ft/sec}; \quad \frac{\Delta YA}{\Delta \dot{Y}} = 0.4 \text{ nmi/ft/sec}; \quad \frac{\Delta ZA}{\Delta \dot{Z}_A} = 0.4 \text{ nmi/ft/sec}$$

$$\Delta X = 6 \text{ nmi}$$
$$\Delta Y = 4 \text{ nmi}$$
$$\Delta Z = 4 \text{ nmi}$$

for a 10 ft/sec error.

These position errors would yield vacuum perigee variation similar to preceding results. However, the large measurement velocity error would result in an additional vaccum perigee variation of approximately 5 nmi (See Figure III-132 of Appendix T-W for 10 ft/sec velocity error).

This error combined, in an RMS manner, with the errors resulting from position errors yields a combined vacuum perigee variation of

$$\Delta r_p = \pm 7 \text{ nmi}$$

The time-of-flight variation will be approximately 45 seconds. For a Polar return the landing site would move 11.0 nmi. This error would be the maximum expected in the cross-range direction.

The vacuum perigee variation would result in a ±0.6 degree re-entry flight path angle variation.

[redacted]

This error in turn would result in a down range displacement of the re-entry point by ± 1.2 degrees or ± 72 nmi.

5.9 TRANSITION FROM MIDCOURSE TO RE-ENTRY

Figure II-5-12 shows the transition phase from midcourse to re-entry.

When the velocity of arrival error correction is made the measurement errors at the aimpoint (10,000 miles from earth center) will be:

1 to 10 ft/sec in velocity and
2 to 4 nmi in position (See Appendix T-G for discussion of measurement errors)

The re-entry position error will be (from Table II-5-XI) 46 nmi for a 1 ft/sec measurement and 72 nmi for 10 ft/sec errors for down range errors and 3 to 10 nmi for cross range errors.

The first set of errors will establish the initial condition errors for re-entry navigation.

The last set of errors establish the re-entry maneuver requirements resulting from midcourse measurement inaccuracies.

As may be seen the maneuver requirements can be small for measurement errors from 1 to 10 ft/sec.

5.10 RE-ENTRY ERROR ANALYSIS

After the last midcourse correction is made at 10,000 miles no more position measurements will be taken. Reference to the stars may still be obtained, however, and can be used to correct for platform drift. The re-entry vehicle will arrive at re-entry altitude in one-half-hour.

In Reference 7 it is shown that for re-entry vehicles of low $L/_D$, the re-entry flight time is always small compared with Schuler Frequency so that the effect of Schuler Tuning can be ignored in the error analysis. When this approximation is made, the

error analysis can be performed by integrating the erroneous accelerations that the inertial equipment measures by misorientation, gyro drift, accelerometer scale factor, etc., as explained in Reference 8. In such an analysis, the aero vector* is a quantity of considerable significance. For the nominal trajectory of Figure II-5-15, the components of the aero vector, in terms of landing point horizontal and vertical are:

At end of glide:

Horizontal	8.3 million feet
Down	.12 million feet

At Impact:

Horizontal	17 million feet
Up	1.5 million feet

Two sets of values are given because it is only the glide portion which must be performed inertially. The inertial system may be corrected from external measurements during vertical flight.

The error sources are divided into two categories:

(1) Inertial instrument errors occurring during re-entry, and

(2) Errors resulting from initial condition errors left over from midcourse guidance.

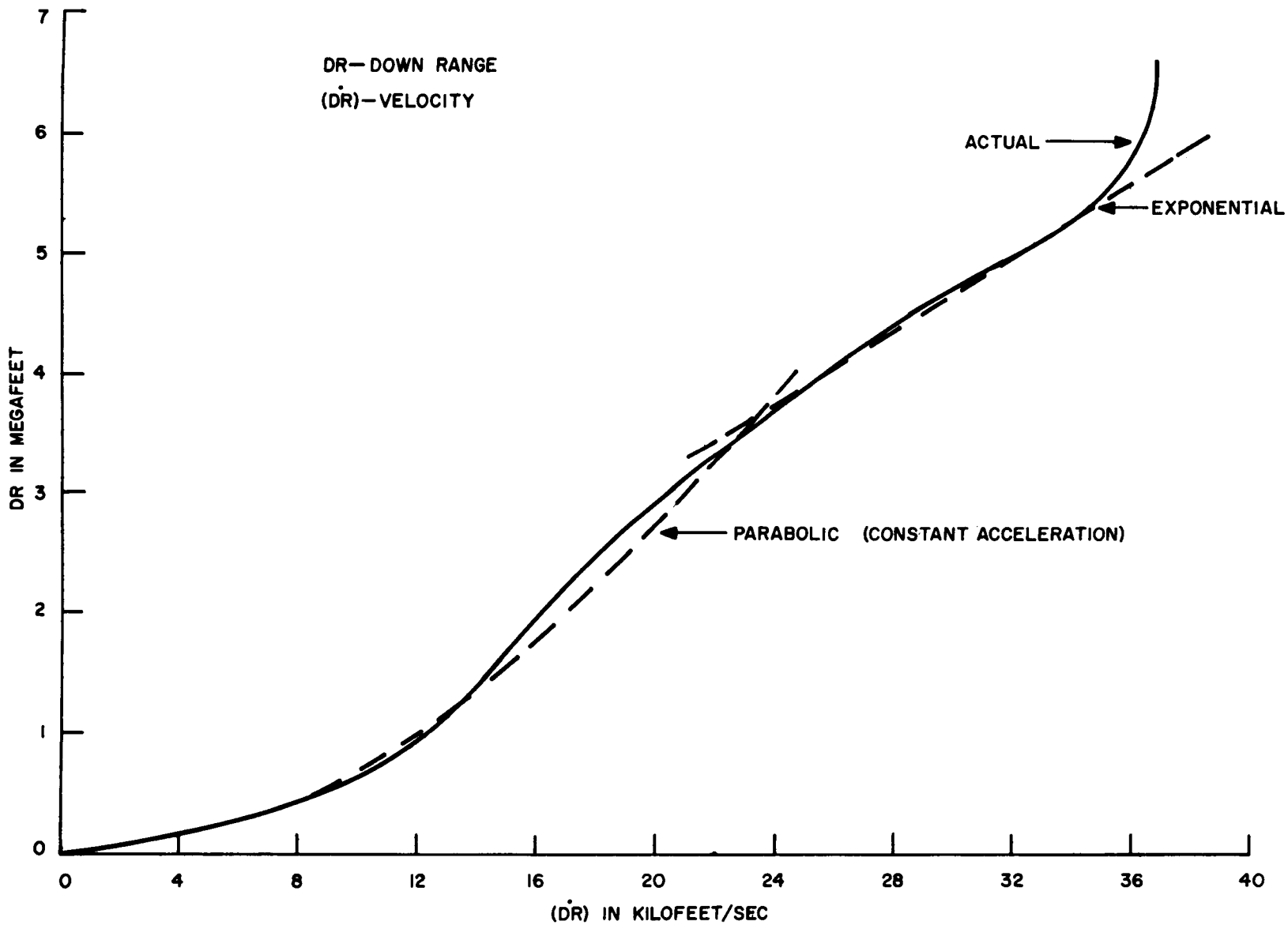
This division is convenient because the errors in the first category are uncorrelated; whereas, the errors in the second category are coupled and require a different method of handling.

Table II-5-XI and II-5-XII give error summaries for Category (1) errors at end of glide and impact. Tables II-5-III and II-5-IV give similar summaries for Category (2) errors. It is evident that Category (2) errors predominate.

---

* Double integral of acceleration measured by the accelerometers.

[redacted]



[redacted]

Figure II-5-15. (DR) vs (DR) plot of re-entry trajectory

TABLE II-5-XI

INERTIAL EQUIPMENT ERRORS AT END OF GLIDE

	Error in nmi*		
Source	DR	CR	h
Gyros			
Constant (15 meru)	.01	.01	
		.19	.19
Unbalance 1A (15 meru/g)		.03	
		.03	
Unbalance SRA (15 meru/g)	--		.03
Anisoelastic (2 meru/g^2)	--	--	.03
		.03	
Initial alignment (10 sec)	--	--	
		.02	.07
Accelerometer Bias (.1 cn/sec^2)	.05	.05	.05
Accelerometer Scale Factor (75 ppm)	.10		--
RSS Error	.11	.20	.21

* DR — Down Range Error
CR — Cross Range Error
h — Altitude error

TABLE II-5-XII

INERTIAL EQUIPMENT ERRORS AT IMPACT

Source	Error in nmi* DR	CR	h
Gyros			
Constant (15 meru)	.15	.15	
		.48	.48
Unbalance 1A (15 meru/g)		.38	
		.04	
Unbalance SRA (15 meru/g)	.04		.44
Anisoelastic (2 meru/g^2)	.02	.02	
		.06	.06
Initial Alignment (10 sec)	.01	.01	
		.14	.14
Accelerometer Bias (.1 cm/sec^2)	.12	.12	.12
Accelerometer Scale Factor (75 ppm)	.21		.02
RSS Error	.29	.66	.68

* DR — Down Range Error
CR — Cross Range Error
h — Altitude Error

TABLE II-5-XIII — ERRORS RESULTING FROM INITIAL CONDITIONS — AT END OF GLIDE

Error Source	Error (nmi)		
	Down Range	Cross Range	Altitude
Initial Position	1.08	1.73	4.02
Misalignment*	0.01	0.23	0.69
Initial Velocity Error	1.39		
Resultant Error	1.76	1.96	4.08

TABLE II5-XIV — ERRORS RESULTING FROM INITIAL CONDITIONS AT IMPACT

Error Source	Error (nmi)		
	Down Range	Cross Range	Altitude
Initial Position	1.08	1.73	4.02
Misalignment*	0.08	0.45	
Initial Velocity	2.19		0.88
	2.48	2.18	4.11

*Misalignment is 1.08 minutes in pitch and 1.73 minutes in roll. These errors occur as a result of initial position errors and are correlated with these errors.

The initial measurement position errors at 10,000 miles considered for these results is a spherically distributed 3 nmi (3σ) error and 6.5 ft/sec velocity error (3σ).

At re-entry at 400,000 feet the errors resulting from these errors would be: cross range error 1.08 miles; down range error 1.73 miles; and the altitude error 4.02 miles.

From 10,000 miles to re-entry stellar fixes may be taken to keep the platform aligned. The results reported here are based on up-dating platform alignment.

If no fixes were taken platform drift would be 0.1 degrees from the last fix taken at the last correction point at 10,000 miles. (0.2 degrees/hour drift for 1/2 hour).

From Table II-5-XI, for a misalignment of 0.1 degree, instead of 10 seconds the cross range error would be increased to 1.44 nmi instead of 0.02 nmi.

The results of Tables II-5-XIII give the errors at the end of glide — approximately Mach 5 at 100,000 feet.

Tables II-5-XIV yield the errors at nominal impact time. The errors are seen to be of the order of several miles.

These errors are well within a 10 mile (RMS) error.

[redacted]

[redacted]

6.0 Required Modifications for Alternate Missions

Primary emphasis in this study has been directed toward the lunar orbiting mission. However, it is recognized that the ultimate APOLLO mission is a manned lunar landing, and that initial APOLLO flights will include other missions such as earth orbit and rendezvous with an earth satellite. Therefore, one of our objectives has been to design an N and C subsystem for the lunar orbiting mission in such a way that these other missions can be accomplished with a minimum of modification. In the following paragraphs a discussion is given of the modifications required for three alternative missions.

1. Earth Orbit
2. Rendezvous with Earth Satellite
3. Lunar Landing

6.1 EARTH ORBIT

An earth orbiting mission can be accomplished with essentially no modifications to the N and C subsystem described in previous sections. This system has the capability of measurement and correction of lunar orbits and there is no basic differences for earth orbit. Capability is provided for determining all orbit parameters, for calculating velocity changes to change or correct the orbit and for accomplishing these velocity changes. Such a mission would constitute a good test of a large part of the APOLLO G and C subsystem. Thus after injection into an earth orbit on-board measurements and computations could be made to determine the orbital parameters. The system could then be used to make a change in the orbital plane or some other orbit parameter. This could be done either with a manned or unmanned vehicle.

6.2 RENDEZVOUS WITH EARTH SATELLITE

The objective of this mission is to cause the APOLLO vehicle to mate with a satellite. The terminal guidance function performed by on-board equipment can be divided into two parts: 1. rendezvous-starting from ranges from the target on the order of 20-50 miles, apply corrective thrust to bring the rendezvousing vehicle to some close range on the order of 200-1000 ft with a low relatively velocity, on the order of 1 ft/sec,

and 2. mating – starting with the final conditions from rendezvous, control relative position and attitude so as to establish a rigid physical link between the vehicles.

Modifications required for this mission are primarily in the terminal sensors. As indicated below a radar is required for rendezvous whose characteristics are somewhat different than those of the radar altimeter which might be used in the lunar orbiting. Another sensor system will be required for the short ranges during the mating phase. In both cases advantage will be taken of the fact that this is a friendly rendezvous so that radar and IR beacons can be located on the satellite.

The computer functions in rendezvous are somewhat different than for the lunar orbiting mission but are well within the capability of the APOLLO computer. One further change that will probably be required is the addition of one or more transverse engines for making cross-range corrections.

N and C requirements for the rendezvous mission are described in more detail in the following paragraphs. These are based on one system concept which allows rapid rendezvous.

6.2.1 Rendezvous Terminal Guidance

6.2.1.1 SYSTEM DESCRIPTION

The terminal guidance concept is to reduce the angular rate of turning of the line-of-sight (LOS) (as determined by rate gyros on the radar gimbals) to zero by firing the lateral engines and to reduce the range rate to zero by several successively shorter firings of the longitudinal engine. The cross range corrections are initiated when the angular rate exceeds a preselected value. Determination of the angular rate is based upon smoothing the angular momentum of the vehicle with respect to the target in order to minimize the effect of measurement errors. Magnitude of corrections are based on a computation of velocity necessary to reduce the angular rate of the LOS to zero. Measurement of the corrections are made by using the platform mounted accelerometers. Range rate corrections are based upon a desired range rate versus function that allows sufficient time for radar data smoothing between engine firings.

The terminal guidance system utilizes the following subsystems:

Radar

Inertial Measurement Unit (IMU)

Propulsion

Terminal Guidance Computer

Attitude Control

The function of each of these equipments in the terminal guidance system is described briefly in the following paragraphs.

The primary purpose of the radar is to determine the line-of-sight to the target satellite and provide range and range rate information along this line-of-sight. This information is provided from radar lock-on to 40 ft range as detailed in the section on radar. With the radar locked on the line-of-sight, two gimbal angles on the antenna give information on the attitude of the FSV with respect to the line-of-sight. A pair of rate gyros on the antenna measure the two components of the angular rate of the line-of-sight to 200 ft. These gyros on the antenna are also used to isolate vehicle body motion disturbances on the line-of-sight rate measurement.

The IMU (Inertial Measuring Unit) provides attitude reference signals from separation to radar lock-on, including radar search phases. After radar lock-on the vehicle longitudinal axis is aligned to the line-of-sight using the radar gimbal angles. This then permits radar, engine and TV to act along the LOS which is required for proper system performance. The IMU continues to provide roll attitude information but the other two gimbals are locked, either mechanically or through the gimbal servos, so that the IMU integrating accelerometers can be used to measure velocity increments in body-fixed coordinates.

In addition to the main APOLLO propulsion engines four engines 90 degrees apart with thrust axes normal to the longitudinal axis are assumed. Four engines are used for transverse corrections in spite of the increased weight of the propulsion systems because this approach offers the greatest probability of successful operation.

The terminal guidance computer has the following functions:

1. Store information on required attitude during radar search and provide this information to the attitude control subsystem.
2. Provide required smoothing for range, range rate, and angular rate of line-of-sight signals.
3. Calculate magnitude of cross-range and range corrections to be made during rendezvous.
4. Provide signals to initiate corrections based on stored logic and measured data.

Operation of the terminal guidance system is as follows: Upon acquisition of the target by the radar, pitch and yaw attitude control is switched from the IMU to the gimbal angle signals from the radar antenna. The attitude control system thereafter operates to null these signals thereby keeping the vehicle roll axis along the line of sight. The outer roll gimbal angle of the IMU continues to provide a roll attitude signal. After sufficient time to allow for smoothing range, range rate and LOS angular rate data, the first cross-range correction is made if the angular rate of the LOS exceeds 1 milliradian/sec. Additional cross-range corrections are made whenever the LOS angular rate exceeds 1 milliradian/sec provided sufficient time has elapsed from the previous correction to allow adequate smoothing.

When the range decreases to about 25,000 feet the first longitudinal range rate correction is made. This and subsequent range corrections are calculated to cause the range rate to follow a prescribed program as a function of range. Additional cross-range corrections are made between range corrections and in particular, one cross-range correction is made immediately prior to each range correction to minimize

cross-velocity during range corrections. After the first range rate correction cross-range corrections are initiated by transverse velocity rather than LOS angular rate. The final corrections must be made at a range of a few hundred ft. The final range correction is calculated to reduce range rate along the LOS to zero.

The cross-range velocity can be computed by multiplying the angular rate of the line-of-sight by the range. However, since these quantities must be smoothed in order to get accurate results the calculated cross-velocity will lag the actual value because of smoothing delays. This effect is not too important at long ranges but will have a large effect at short ranges where the smoothing delay must be longer because of radar target generated noise and because the growth rate of the line-of-sight angular rate is high. An alternative approach makes use of the fact that the angular momentum of the vehicle about the target is essentially constant. Thus, this quantity can be computed from smoothed range and line-of-sight rate data without being affected by the smoothing delays. Cross-range velocity is then computed by dividing the angular momentum by the smoothed range.

Initiation of cross-range corrections can be controlled by computed transverse velocity discussed above. However, in order to avoid making a large number of small corrections, it is desirable to use a higher velocity threshold at long ranges than at short ranges. The threshold can be reduced to zero as a function of range by using a constant threshold for the angular rate of the line-of-sight to initiate cross-range corrections. At short ranges, a constant cross-velocity threshold will control initiation of transverse engines, to prevent a large number of engine operations.

It is questionable whether the above measurements can be used to directly control the cross-range corrections due to noise on the range and angular velocity of the LOS signals from the radar. Since these errors are likely to be much worse when engines are being fired, it seems undesirable to use these measurements to shut off the engines. An alternative approach is to allow sufficient time for smoothing between corrections, use the smoothed data to compute corrections and then shut the engine off using the integral of body fixed acceleration signals.

[redacted]

Since it has not been definitely established that lags in LOS angular rate information can be tolerated it is necessary to compute cross-velocity from the angular momentum, even though more computing equipment is required. The threshold value for cross velocity should be a function of range at long range and a constant value at short ranges.

6.2.1.2 CROSS-RANGE CORRECTION

To understand the requirements of the normal velocity control system it is necessary to examine the functional form of the angular rate of the line-of-sight while thrusting and while coasting (zero thrust period).

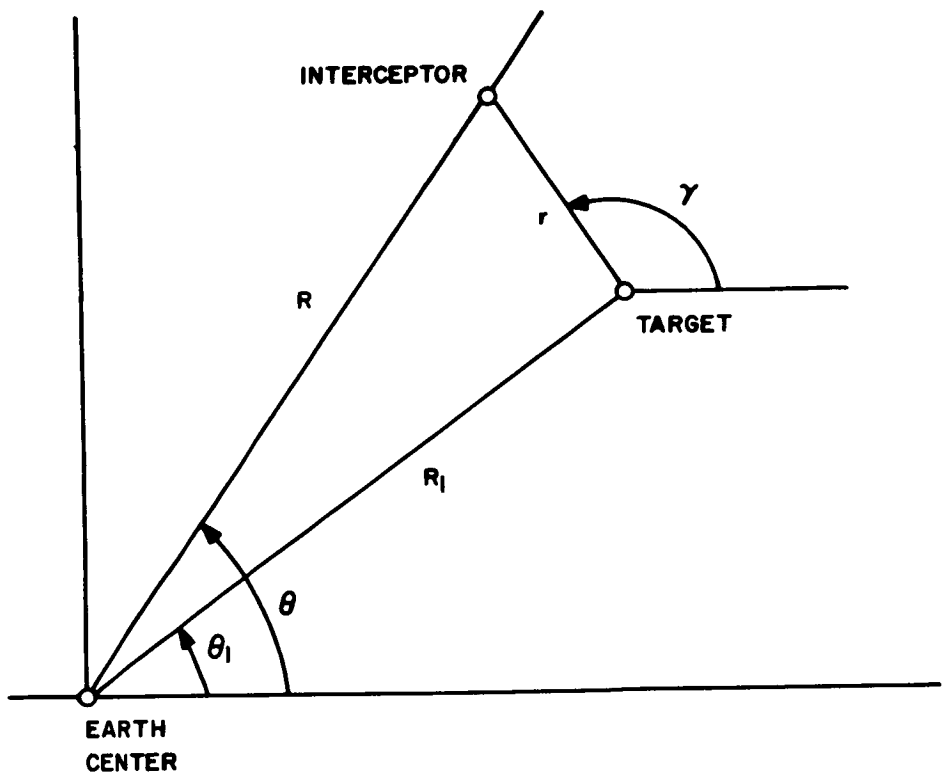


Figure II-6-1. Co-ordinate system

[redacted]

In the interest of conciseness and brevity, step by step derivations will be dispensed with and only the important equations will be given.

During zero thrusting periods, the behavior of the LOS angular rate is dependent only upon the conditions at the beginning of the period. From the conservation of angular momentum,

$$r^2 \dot{\gamma} = r^2 \dot{\sigma}_o \tag{S-1}$$

where the zero subscript indicates the conditions at the beginning of coast. During proper operation of the normal velocity control system, the range may be approximated by a linear function of time, since the interceptor is nearly on a collision course. This equation for range is:

$$r \cong r_o + \dot{r}\, t \tag{S-2}$$

Substitution of equation (S-2) into equation (S-1) results in the expression indicating the behavior of LOS angular rate during zero thrust period. It is:

$$\dot{\gamma} = \frac{\dot{\gamma}_o}{(1 - \frac{\dot{r}}{r_o}\, t)^2} \tag{S-3}$$

This equation is useful in determining the time between corrections for a constant threshold when the conditions resulting from the preceding thrust period are known.

The behavior of $\dot{\gamma}$ during thrusting periods is also of interest. Even with perfect measurement information, there will be a residual LOS angular rate, for the following reason. The normal velocity ($r\dot{\gamma}$) computed at the beginning of each thrusting period indicates the component of relative velocity perpendicular to the instantaneous LOS. Since the LOS is continually changing with time an instantaneous normal velocity correction would be required to completely reduce the measured LOS angular rate. Since this is not possible, a small component of angular rate remains. This residual LOS angular rate is considered next.

The simplified equations of motion for zero radial thrusting are:

$$\ddot{r} - \dot{r}\,\dot{\gamma}^2 = 0 \tag{S-4}$$

$$r\,\ddot{\gamma} + 2\dot{r}\,\dot{\gamma} = a_n \tag{S-5}$$

From equations (S-2) and (S-3), together with the assumption that $\ddot{r} \approx 0$ (and therefore $r\,\dot{\gamma}^2 = 0$), the residual LOS angular rate is:

$$\dot{\gamma}_2 = \frac{a_n\,\dot{r}\,\dot{\gamma}_1^{\,2}}{2\,(a_n - \dot{r}\,\dot{\gamma}_1)^2} \tag{S-6}$$

when

$$|a_n| >> |\,\dot{r}\,\dot{\gamma}_1\,|$$

$$\dot{\gamma}_2 = \frac{\dot{r}}{2\,a_n}\;\dot{\gamma}_1^{\,2} \tag{S-7}$$

where subscript 2 indicates LOS angular rate at end of thrusting period (therefore initial condition for coasting period) and $\dot{\gamma}_1$ is LOS angular rate at beginning of thrusting.

From equation (S-3) the time of coast t (zero thrust period) t_c, is expressed as:

$$t_c = \left(1 - \sqrt{\frac{\dot{\gamma}_o}{\dot{\gamma}}}\,\right)\frac{r}{\dot{r}} \tag{S-8}$$

and with the aid of equation (S-7)

$$t_c = \left[1 - \sqrt{\frac{\dot{r}}{2\,a_n}\;\dot{\gamma}_1}\,\right]\frac{r}{\dot{r}} \tag{S-9}$$

where r is that range at the beginning of the zero thrust period.

Also, time of thrusting period, t_n, is:

$$t_n = \frac{r_n \dot{\gamma}_1}{a_n} \tag{S-10}$$

where r_n is the range at beginning of thrust period.

6.2.1.3 RANGE CORRECTION

The function of the range control system is to provide longitudinal corrections in some manner so as to result in reduction of range and range data to zero simultaneously. An obvious approach to this problem is to control range rate as some function of range such that $\dot{r} = 0$ when $r = 0$. This approach using the function $\dot{r} = K\sqrt{r}$ has been studied by others assuming proportional thrust. It is possible to do essentially the same thing with a constant thrust engine by controlling its on and off periods by means of parabolic switching lines in the phase plane as shown in Figure II-6-2. Here the engine is turned on when the phase plane trajectory intersects the left switching line and off when it intersects the right one. The equation of the left switching line is chosen to match the thrusting characteristics of the engine assuming low thrust tolerance and high mass tolerance. The actual phase plane trajectory deviates from this switching line only because of system tolerance and variation of vehicle mass. The spacing between the switching lines is selected to result in constant drift times between corrections.

It is also possible to use straight switching lines as shown in Figure II-6-3. This results in a somewhat simpler control function for the guidance computer. Its main disadvantage is that it results in a large number of on-off cycles for the engine. Also at close ranges, the required on times become very short which are impractical due to propulsion time delays.

In practice, the operation of all of the above system approaches must be investigated further because of the problem of measurement noise errors. All of these schemes use measured data to shut down the engines and the data obtained during thrusting periods must be studied to determine if it is sufficiently accurate for this purpose when the FSV approaches close to the target and radar target noise increases.

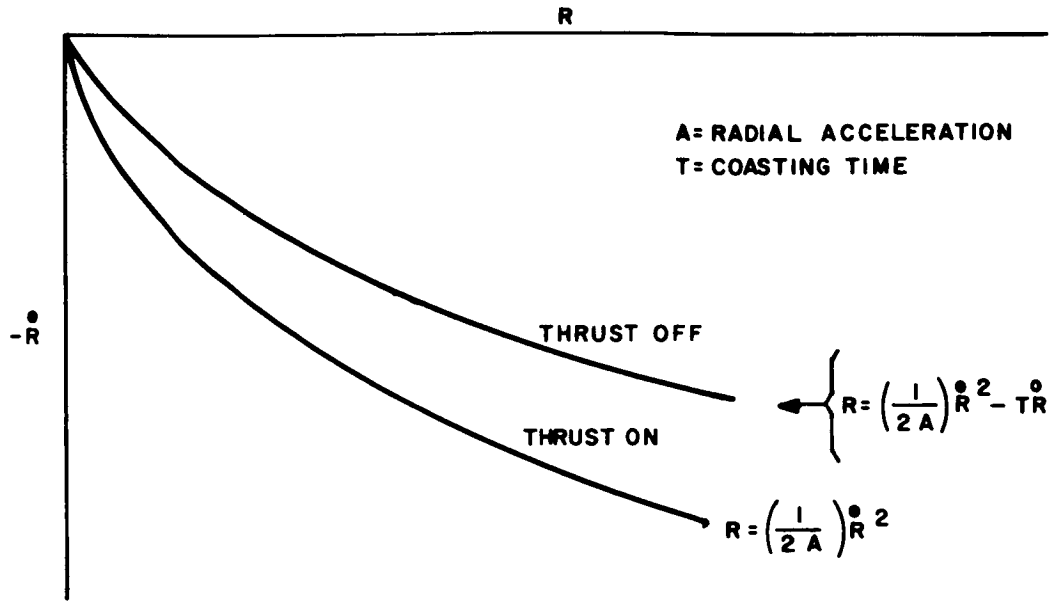


Figure II-6-2. Parabolic switching line-radial thrust program

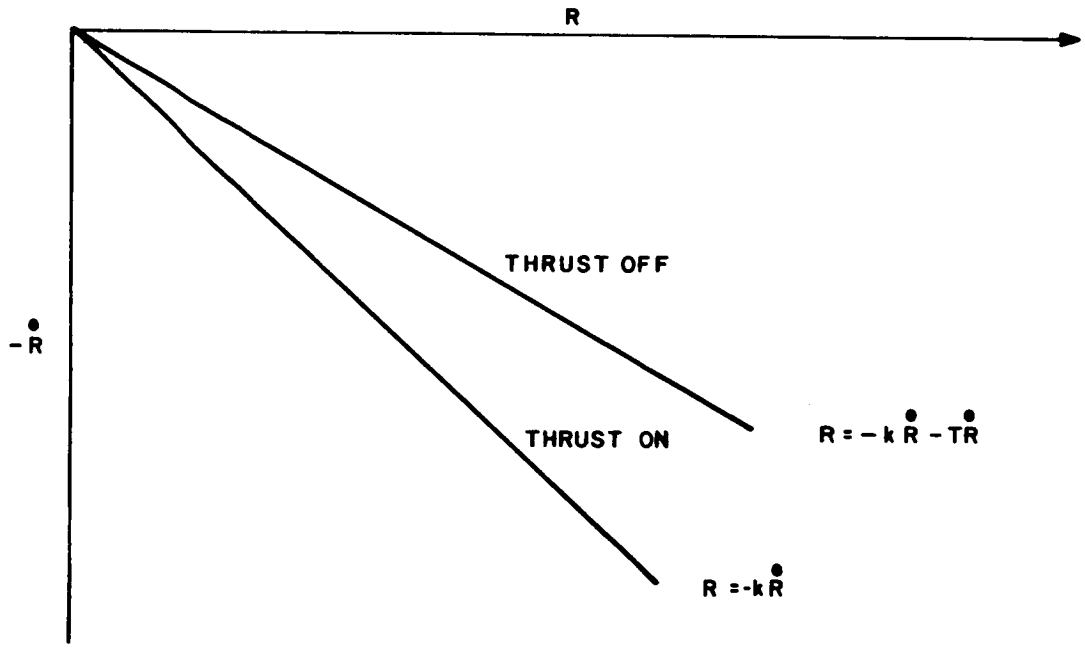


Figure II-6-3. Linear switching line-radial thrust program

Alternative approaches allow sufficient time between corrections for smoothing of measurement data. The smoothed data is used to calculate required range corrections which are then metered through the use of accelerometers and integrators. Two approaches are given. In one the range rate is made to follow a prescribed program as a function of range by means of several discrete corrections. The other approach makes use of straight switching lines similar to those in Figure II-6-3 except that the engine is shut off by the integral of acceleration instead of the lower switching line.

An alternate approach to all of the above schemes is to terminate thrust on accelerometer information. During coasting, smoothed data is used to calculate, first the range at which the correction begins, and then the range rate correction required. This range rate correction is metered against the output of an integrating accelerometer and thrust is terminated when desired correction is obtained.

Another method of range rate control, making use of either straight or parabolic switching lines, reduces the velocity to zero in several discrete steps. The advantage to this is that it permits coasting time between corrections for smoothing measurement data. The only errors remaining due to range control are those acquired during the final thrusting period.

The range control system chosen is based on parabolic switching lines where the velocity is reduced to zero in a number of discrete thrusting periods. Each thrusting period is terminated on integrated accelerometer information. This method yields the least errors and minimizes the number of engine operations, while making use of the measuring equipment (radar and accelerometers) under conservative conditions.

The range rate control can be analyzed with the equation of motion.

$$\ddot{r} = a_r + r\,\dot{\gamma}^2 \qquad \text{(S-11)}$$

For

$$r\,\dot{\gamma}^2 << a_r :$$

$$\ddot{r} = a_r \qquad \text{(S-12)}$$

Since $a_n = 0$, the value of $\dot{\gamma}$ at the beginning of longitudinal thrusting will be as derived in the Normal Velocity Control System section, i.e.

$$\dot{\gamma} = \frac{\dot{\gamma}_o}{(1 - \frac{\dot{r}}{r} t)^2} \tag{S-13}$$

Thus $\dot{\gamma}$, the residual LOS angular rate can be shown to be, even in the presence of noise, much less than 1.0 milliradian/sec (mr/sec); thus at a range of 25×10^3 feet, $r\dot{\gamma}^2$ (for $\dot{\gamma} = 0.5$ mr/sec) = .00625 fps^2. For $a_r = 8$ fps^2, the assumption that

$$a_r >> r\dot{\gamma}^2 \text{ is valid}$$

Successive integration of (16) yields

$$\dot{r} - \dot{r}_o = a_r t \tag{S-14}$$

and

$$r - r_o = \dot{r}_o t + \frac{1}{2} a_r t^2 \tag{S-15}$$

The limits of integration on t were from zero to t.

Solving for t from (S-14) and substituting into (S-15) yields the parabolic equation for range as a function range rate:

$$r - r_o = \frac{\dot{r}^2 - \dot{r}_o^2}{2 a_r} \tag{S-16}$$

where a_r is assumed constant during a given thrusting period.

This equation describes the relationship between r and $\dot{r}$ for constant acceleration. Therefore, it is desirable to match the actual vehicle trajectory to fit this parabola.

[redacted]

6.2.1.4 RADAR

A radar which can meet the requirements of this mission is a modification of the AN/DPN-34 radar produced by Westinghouse for the BOMARC missile. Components of this system have been developed on a production basis for the BOMARCA. General Electric Co.'s MSVD has had considerable experience with another version of this radar in the Mark IIIC Test Program conducted for AFBMD. Thus this is a well-designed thoroughly proven piece of equipment admirably suited for modification and quick delivery.

This radar system is designed to search a 40 degree x 40 degree scan sector and to acquire and track the target automatically.

During the acquisition and homing mode the output information provided to the FSV system comprises:

a. Antenna angular position

b. Antenna angular rates (line of sight rates)

c. Range

The transition from acquisition and homing radar to the mating phase can be made arbitrarily at a range of approximately 200 feet. This transition range is not critical since the minimum range in the acquisition mode is 100 feet. Because in the acquisition mode the radar provides additional information (line of sight rate), it is desirable to use this mode as long as possible.

[redacted]

During acquisition and tracking the radar provides the following outputs with the accuracies indicated:

Output	Measurement Range	Accuracy (3)
Range (High Power)	50 nmi to 1500 ft	± 1% or ± 300 ft, whichever is greater
Range (Low Power)	1500 ft to 100 ft	± 1% or ± 10 ft, whichever is greater
Range Rate	0 to 1000 ft/sec	± 1% or ± 0.3 ft per sec whichever is greater
Angle	± 45°	± 1% random error, ± 0.001 radian bias error
Angle Rate	± 0.1 radians/sec	± 5% random error, ± 0.001 radians/second bias error
Acquisition Range	59 nmi against $25m^2$ 22.7 nmi against $1m^2$	40° x 40° scan

All radar outputs are analog dc voltages.

The seeker antenna will consist of a parabolidal reflector 24 inches in diameter, having a focal length to diameter ratio of 0.4, and a vertex primary radiator.

The primary radiator which will be employed is a modified version of the "Hennessey Feed". The feed design is identical to that employed in the AN/DPN-34 and other Westinghouse X-band antennas resulting in equal half-power beamwidths in both the E- and H-planes, and high illumination efficiency. The impedance and radiation characteristics of this type of primary radiator have previously been optimized for a focal-length-to diameter ratio of 0.4 at X-band.

Lobing is accomplished by moving the primary radiator in a small circle about the axis of the paraboloidal reflector. The circular waveguide feeding the reflector is angularly offset. Since this radiator is circularly symmetrical and the rectangular waveguide

entering at the vertex of the reflector is fixed, the polarization of the radiated energy is unaffected by rotation of the antenna feed. A short circuited, half-wave length choke is employed between the stationary and rotating portions of the circular waveguide feed to maintain RF continuity.

The half-power beamwidth of the antenna will be approximately 3.6 degrees in both the E- and H-planes at a frequency of 9580 mc. The one-way gain will be 33.4 db at this frequency, and, based on previous experience, side lobe levels immediately adjacent to the main lobe will be at least -23 db (with respect to the peak of the main lobe). Side lobe levels beyond 30 degrees from the main lobe will be at least -28 db. Polarization will be horizontial linear; however, vertical linear polarization can be easily provided if necessary.

The feed will be offset sufficiently to shift the main lobe 1 degree from the "boresight" axis. This would provide a fundamental target error signal modulation of approximately 65 percent per degree.

6.2.1.5 COMPUTER

In the approach to the target, it will be required that the computer accept range, range-rate, line-of-sight inertial, pitch-rate and yaw rate and provide required smoothing. With the philosophy of transverse velocity corrections measured by integrating accelerometers, it will be required that the computer accept and operate upon these signals. The operations will be comparison and limiting. The comparison will be with the computed required correction magnitude, and the limiting will be achieved by control of the transverse engines.

Control of the longitudinal thrust engine will require engine ignition as a function of range and range rate. Control of the transverse engines will require comparison and limiting of transverse angular rates and the computation of required transverse velocity corrections as a function of the limit of angular rates and the range at the time of correction. Longitudinal corrections are to be made only after transverse thrusting in both channels, and no transverse thrusting will be permitted during longitudinal thrusting.

An estimate of computer requirements to accomplish rendezvous is as follows:

Computer Type	General Purpose
Clock Frequency	128 kc
Word Length	16 bits
Storage Capacity	
Instructions	512 words
Coefficients	128 words random access
Intermediate data	128 words random access
High speed loops	11 words random access
Number Code	Binary
Negative Number Representation	Two's Complement
Instruction Code	Single Address
Duration of Major Arithmetic Operations	
Add or Subtract	252 Microseconds
Multiply	2 milliseconds
Divide	2 milliseconds
High Speed Iteration Rate	1000 per second
Number of Inputs	32
Number of Outputs	32

It is seen that these requirements are well within the capability of the APOLLO computer described in paragraph 3.

6.2.2 Mating Control

6.2.2.1 GENERAL REQUIREMENTS

It is convenient first to distinguish between the operation of simply "docking" and that of "mating". In docking the relative velocity ultimately is reduced to zero, the relative position to a sufficiently small value maintained by some kind of substantial (but not necessarily rigid) physical link. No attitude control or only gross attitude control requirements are imposed on the target, and those on the chaser are principally the attitude requirements which accompany maneuvering. On the other hand, "mating" requires the establishment of a rigid physical link, generally in a rather precisely delineated regions, and normally requires very precise control of relative orientation and orientation rate during the final stages of approach to physical contract.

At the end of the coarse rendezvous maneuver the relative position of target and chaser are as shown in Figure II-6-4. They have a certain separation and relative velocity which are initial conditions from the point of view of the docking phase. To a greater degree than in rendezvous the orientation of each becomes important. Orientations are designated in the figure by heavy arrows (generally not coplanar with the relative position and velocity vectors), their conventional senses so chosen that the mating attitude is tail-to-tail.

The details of the mating guidance and control requirements depend sensitively on operational considerations, but the following broad functions normally are represented.

1. To bring to zero the relative separation of the mating surfaces on the target and chaser;
2. To bring the impact velocity of the chaser on the target between prescribed bounds V_{min} and V_{max}, where V_{min} is established as the minimum velocity needed to actuate the positive coupling mechanism and V_{max} is the mechanical design limit of the shock absorbing mechanism supporting the mating surfaces;
3. To bring the angular orientation of both target and chaser relative to the line of sight within a limit established by catcher or coupler entrance tolerance;

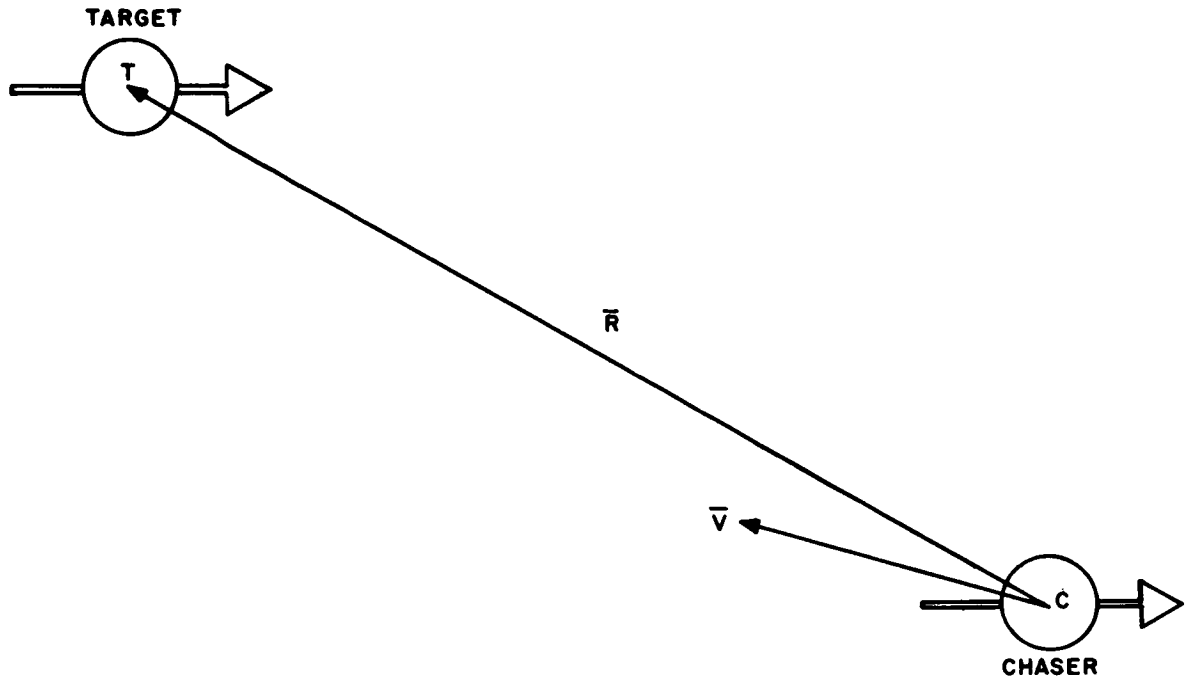


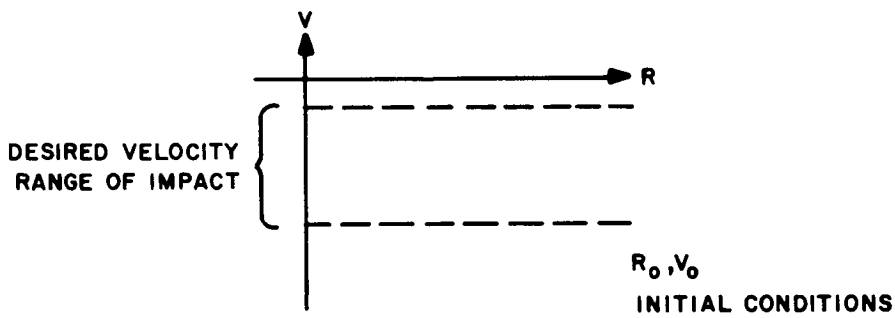
Figure II-6-4. The encounter - situation at beginning of docking phase

4. To bring the relative roll of the two vehicles about the line of sight (i.e. the angle between roll fiducial marks, one on each vehicle) within the roll tolerance limits of the catching or coupling mechanism;

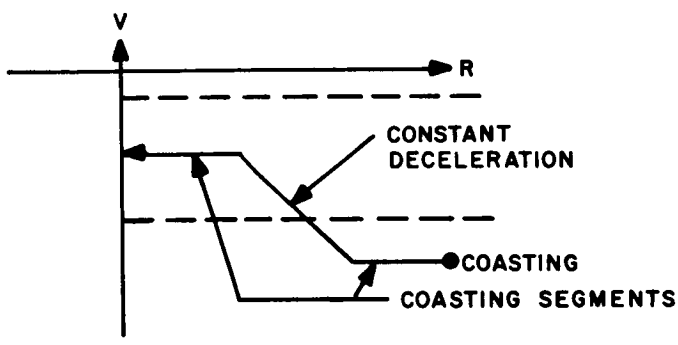
5. To hold the angular rates of both vehicles transverse to and about the line of sight to a sufficiently low value that lateral coupler loads are low and no interference with sensor operation arises (generally accomplished by keeping the attitude angles themselves low);

6. To hold as close to zero as possible the inertial angular velocity of the line of sight so that the mated configuration has only a small initial angular velocity in inertial space.

Of these functions, those involving attitude probably are most critical since the establishment of proper position and velocity is at least in part accomplished by the coarse rendezvous guidance system. It is convenient to visualize the remaining translation control operations in terms of a phase diagram of relative position and velocity as in Figure II-6-5. (Negative or closing relative velocities are of primary interest.) Figure II-6-5A shows the initial relative position and velocity established by the course rendezvous system and the terminal "box" (i.e., a segment on the negative V-axis) desired as a terminal state. Figure II-6-5B shows the subsequent phase trajectory: in general beginning with a horizontal coasting segment while the initial state is being measured by relative position and velocity sensors used for docking; next an essentially constant deceleration phase initiated on the basis of R and lasting for a time precalculated on the basis of V_0 (a parabolic phase trajectory); and finally another coasting phase within the tolerable final velocity band. It would appear that there should be no need for more than one terminal powered phase provided the precision with which the initial velocity can be established is high enough. If the error box about (R_o, V_o) were



A) INITIAL AND DESIRED TERMINAL CONDITIONS



B) PHASE TRAJECTORY FOR DOCKING

Figure II-6-5. Position - velocity phase diagram for docking

actually taller than the desired terminal velocity band, then the initiation of deceleration could be delayed until better velocity measurements were available. One other factor also must be weighed in deciding the number of terminal corrective thrusts, namely the degree to which imperfect attitude control (i. e. thrust which is not a pure couple) during this phase can affect translation.

In view of the functions listed above, the following guidance and control operations are required:

1. Measurement of relative position.
2. Measurement and control of relative velocity along line of sight.
3. Sensing and control of angular velocity of line of sight.
4. Sensing and control of relative orientation: roll, pitch, yaw (where roll is about line of sight).

The basic problem is one of measurement or sensing, which may be done in any of three basic ways: sightings from both target and chaser on the other; sightings of one from the other plus information transfer; sightings from both on other reference bodies such as the earth. The case of sightings on both from the earth is explicitly omitted from docking considerations.

It seems very desirable to consider two sensing regimes in the docking phase. The first is a coarse docking sensing mode which begins after the coarse rendezvous sensing mode. Perhaps it is a simple continuation of the same sensing, but now accompanied by a different control concept. The second is a fine mating sensing mode where emphasis is placed on precision sensing of vehicle attitudes. It can be expected that the physical devices used for sensing during these two modes will be quite different. It should be emphasized that the coarse phase may also occur in a mating maneuver, preceding the fine phase, for the purpose of getting within the acquisition limits of the fine system.

6.2.2.2 SENSING METHODS

The sensing that is required to implement the functional requirements for guidance and control may be based on any of a number of principles. One can conceive ways of performing it by methods of at least six classes:

- radar
- radio
- infrared
- visible optical
- magnetic
- inertial

and still others exist. Not all are equally attractive, of course, and further study is needed to made a specific selection.

The major problem with radar systems is that of the spatial distribution of the target, although this difficulty can be minimized by using a beacon. Radar is the primary sensor during the coarse rendezvous maneuver, providing in a coarse sense all of the information needed by the chaser in the docking phase provided its minimum operating range can be sufficiently reduced. However, it does not provide information about target orientation unless special provisions are made. (An obvious one is to put a second radar on the target and a beacon on the chaser.) It does not seem likely at this time that radar methods will be useful for precision mating, but it is proposed to study them further for coarse sensing. Radio methods, particularly some kind of radio direction finding, seem also to be an interesting possibility and will be investigated.

In the normal encounter shown in Figure II-6-4, the chaser orientation probably is not too far from the line of sight and its rendezvous engine is presented to the target. This suggests a coarse orientation of the target to the line of sight by infrared sensing of the chaser from the target, another possibility that will be examined.

The class of optical methods, realization of which can take a number of forms, seems especially promising for the fine sensing task. Range, first of all, can be measured over the range of interest by a simple optical range finder sighting upon a beacon light on the other vehicle. Its precision need not be great at ranges of the order of 100 m, since range is not critical to the postulated docking guidance concept. Even with a separation of as little as 10 m it may still be possible to accomplish all velocity and fine orientation control processes before impact. At this range one can measure not only range very precisely but also range rate, either by differentiation of range information or by mounting precision rate gyros on the movable and body-fixed optical axes of the ranging telescopes. The problem in this connection will be to investigate the practical range limits of these optical techniques.

Orientation also can be obtained optically. It would seem relatively straightforward to use an autocollimator technique with the body-fixed ranging telescope to determine orientation components transverse to the line of sight.

Roll measurement is a bit more difficult, but ways of doing it exist. Polarization is one possibility, with suitable attention to eliminating the 180 degrees ambiguity. Another is the use of multiple or distributed sources on the other vehicle with a means of pattern recognition. With multiple sources the problem may arise that the range sightings are confused, for the two ranging telescopes obviously must be careful to look at the same point source. Again there are solutions, for example in the form of colored sources and the use of suitable filters. Everything considered, it is felt that the optical methods are a fertile class for fine sensing and that they give wide scope to design ingenuity.

Magnetic methods have, in principle, some attractive features for performing several of the functions that can be handled optically. It is particularly appealing to attempt to measure range rate at relatively close range by the voltage induced in a coil in one vehicle by a calibrated field in the other. Again, in principle, one can imagine measuring relative orientation by the interactions between magnetic dipoles within the two vehicles. If this technique can be mechanized it would give a less accurate orientation than optical methods, but perhaps would still be satisfactory for fine sensing depending on the mechanical design of the mating mechanism. Major problems with magnetic methods

are related to the masking effects of the earth's field and that of the charged particle flows in space, the relatively low field strength that can be generated within power restrictions, and the fact that such field may interfere with other operational equipment aboard one of the vehicles.

Except by reference to a third body or the stars, the one self-contained means of measuring the antular velocity of the line of sight in intertial space is by a rate gyro. If the orientation of one of the bodies is controlled so that it always points at the other, two single-axis body mounted rate gyros measure that angular velocity and permit the calculation of the transverse velocity component needed to drive the angular velocity to zero. It is estimated that a bias error of 1.0 mr/sec (3) is reasonable, with an unsmoothed random component of the order of 5 percent of the measured angular rate. This would correspond to a transverse velocity error of 10 cm/sec at 100 m, reducing to 1 cm/sec at 10 m. This is negligible sensing error for the purposes of transverse velocity correction.

The methods discussed above are summarized in Table II-6-I. Still others are logically possible, including those depending on electric field interaction between devices on the two vehicles and the gravitational field disturbances produced at one body by the other. But these do not seem particularly interesting at this time.

TABLE II-6-I. POSSIBLE SENSING METHODS

Principle	Range	Range Rate	Ang. Velocity LOS	Roll	Pitch & Yaw
radar	C	C			C
radio					C
infrared					C
visible optical (close range)	F	F		F	F
magnetic (close range)		F		F	F
inertial (rate gyro)		F (with optical)	C, F		

Key: C = coarse, F = fine

6.2.2.3 OPERATING SEQUENCE

One can summarize the subsystem design facets by describing a typical operating sequence for the docking guidance and control. A final choice will depend upon the particular sensors used, but it will resemble the following.

1. At about 100 m, gross orientation of both vehicles to the line of sight. Begin position and velocity measurements and estimate time of approach velocity correction.
2. Between 50 and 100 m, longitudinal thrust for correction of approach velocity and lateral thrust for correction of the angular velocity of the line of sight. Lock on fine sensors for subsequent control. (From the view point of dynamic disturbances, it is preferable to do this after thrust, but it can be done before if higher quality sensing is needed for this maneuver.) Begin fine attitude control of both vehicles in pitch and yaw.
3. Between 30 and 50 m, begin coarse roll control. (Actually this may start whenever adequate sensing information becomes available.)
4. Between 10 and 20 m, fine attitude control brings pitch and yaw within steady state tolerances. Begin fine roll control. Monitor approach velocity and angular velocity of the line of sight to determine whether any subsequent correction is necessary.
5. At about 5 m, all velocity and attitude errors should be within their tolerance boxes. From this point, coast home while maintaining fine attitude control.
6. At 0^+ m, engage mating coupler. Transfer attitude control to master unit of combined vehicles.

6.2.2.4 PROPULSION

The fine docking maneuver requires attitude control of the vehicle about three axes and c.g. control in three directions. These control functions can be accomplished using the attitude control engines, with some system modifications. The location of the twelve attitude control engines is shown schematically in Figure II-6-6. It is seen that with

this arrangement we can control c.g. motion in the longitudinal direction by firing the appropriate pitch or yaw engines. However transverse motion can be controlled only in one direction, and this will simultaneously introduce a large pitch moment.

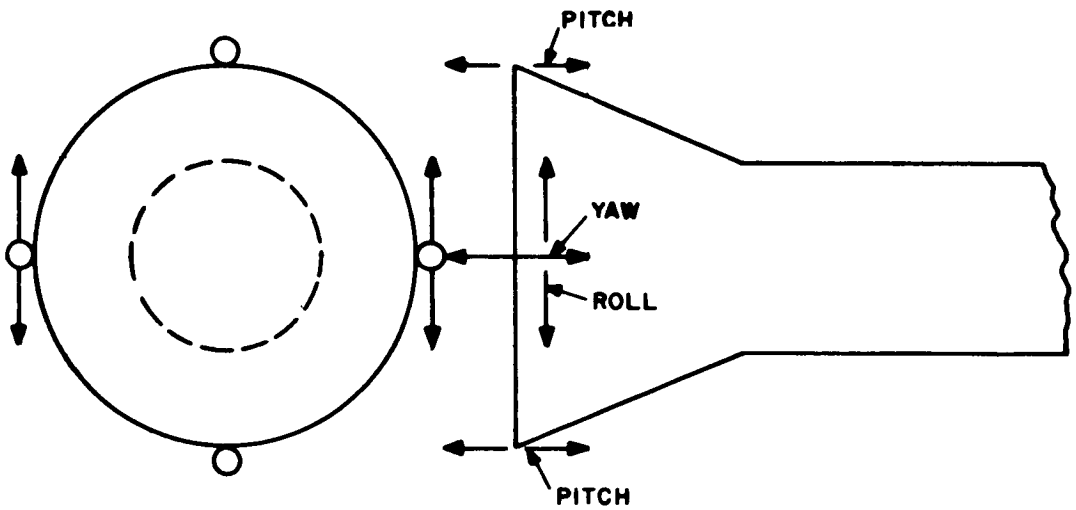


Figure II-6-6. Engine location for clocking maneuver

Thus for this mission it would be necessary to move the roll jets closer to the vehicle c.g. and add another pair 90 degrees around the periphery, as illustrated in Figure II-6-7. The jets are then used in various combinations to control the attitude and c.g. motions as indicated in Table II-6-II.

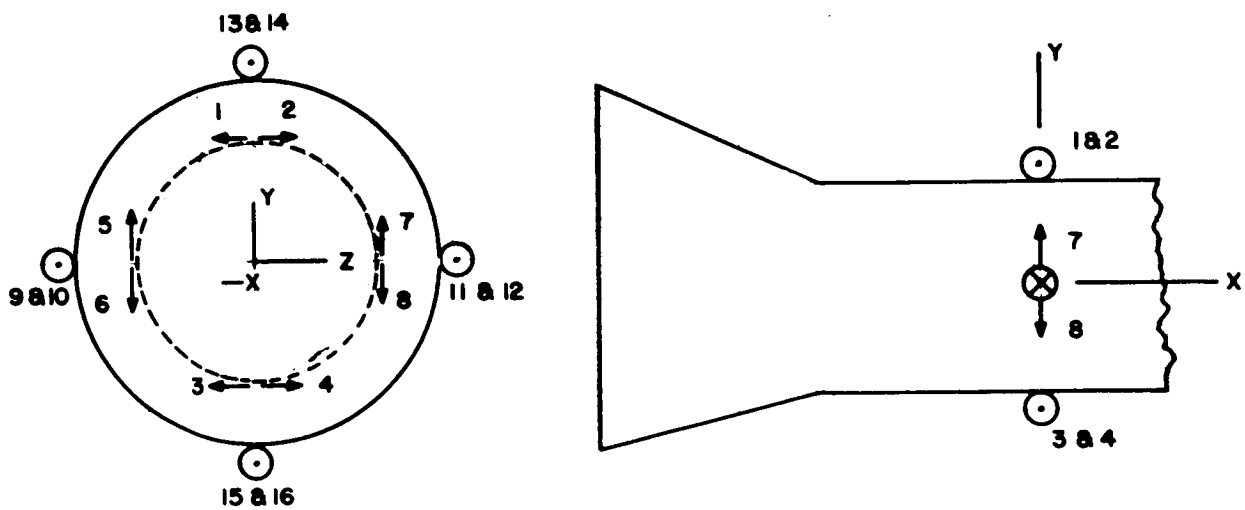


Figure II-6-7. Modified engine location for clocking maneuver

TABLE II-6-II. JET CONFIGURATIONS FOR VEHICLE CONTROL

+X	10, 12 or 14, 16 or 10, 12, 14, 16	XCW	2, 3 or 5, 8 or 2, 3, 5, 8
-X	9, 11 or 13, 15 or 9, 11, 13, 15	XCCW	1, 4 or 6, 7 or 1, 4, 6, 7
+Y	5, 7	YCW	9, 12
-Y	6, 8	YCCW	10, 11
+Z	2, 4	ZCW	13, 16
-Z	1, 3	ZCCW	14, 15

The present attitude control engines have thrusts of 6-8 lbs. Thus with two firing we have an acceleration of about .03 ft/sec^2. For c.g. control of the vehicle it would be necessary to raise these thrust levels by at least a factor of ten.

6.3 LUNAR LANDING

6.3.1 Introduction

The problem of guidance and control for lunar landing of the APOLLO vehicle has not been studied in great detail, primarily because it appears that this mission will not be feasible for some time after the circumlar mission. The purpose of this discussion is thus merely to define the G & C problem, describe a possible solution, and in particular to indicate how the system designed for the lunar orbiting mission can be used for landing and what changes and additions are required.

In making a lunar landing with the APOLLO vehicle, it is assumed that the vehicle would first be put into a lunar orbit as described in other sections of this report. In the same manner the parameters of the orbit would be accurately determined. At the same time the crew would observe the lunar terrain and select a desirable landing site in or near the plane of the lunar orbit. At the proper time a retrothrust is applied and the vehicle is controlled to make a soft landing at the selected site. While the vehicle is on the Moon its location is precisely determined by celestial observations using the celestial sextant and on-board computer. A nominal return trajectory is calculated along with required stored coefficients. The lunar launch guidance scheme will be essentially the same as that used for earth boost guidance.

Most of the studies of the lunar landing problem to date have considered only the problem of landing from a direct flight from the earth. In this case the vehicle approaches the Moon close to the vertical through the desired landing site. The G & C problem consists only of removing the small cross-range velocity and then controlling along the vertical in such a manner that altitude and relative velocity reach zero at the same time.

In considering the case of landing from lunar orbit, there are several reasons why it is desirable to control the vehicle in a manner such that the final landing phase is vertical. Thus the problem of landing the vehicle on its aft end with small cross velocity is somewhat simplified by using a vertical approach. Also since the control

function is simpler in this case, there is less probability of improper landing due to a control malfunction. A possible disadvantage is that such a procedure may not lead to optimum utilization of propellant. This requires further study.

6.3.2 Transfer to Vertical Descent

It seems like a reasonable assumption that an early APOLLO lunar landing flights landing at a precise point will not be required. Based on this assumption one possible approach to the landing problem is as follows. With the vehicle in orbit fixes are taken on the Moon to give a precise determination of the orbit. At the same time the crew will by visual observation, determine the desired landing site. The location of this site in lunar latitude and longitude can be obtained either from a lunar map, or by manual sightings from the vehicle. A velocity increment is then applied so as to simultaneously give the required orbit plane change and place the vehicle on a trajectory which passes through an aimpoint directly of the desired landing site.

The altitude of this aimpoint will be determined prior to launch. This altitude should be as low as possible but must be high enough so that sufficient time remains for final corrections even with the worst combination of measurement and correction errors and errors in knowledge of lunar gravity. It appears that this altitude will be 50,000 ft or less. Referring to Figure II-6-8 the first velocity increment ΔV_1 is applied at point 1. A second correction is applied at point 2 (actually applied prior to point 2 so correction ends at point 2) to null the cross component of velocity. The time and magnitude of ΔV_2 is also calculated prior to the start of the landing procedure. The calculation of ΔV_1 and ΔV_2 is made using a two-body trajectory program. A series of such trajectories is calculated for various points 1 and the actual correction sequence selected so as to minimize $(\Delta V_1 + \Delta V_2 + V_2)$ where V_2 is the residual vertical velocity after the application of ΔV_2. All the above operations can be performed without the addition of any different equipment.

6.3.3 Vertical Descent

After the application of ΔV_2 the vehicle will be falling essentially vertically. By aligning the inertial platform shortly before the application of ΔV_1, the direction of the local vertical can be determined from the platform gimbal angles. Control during the

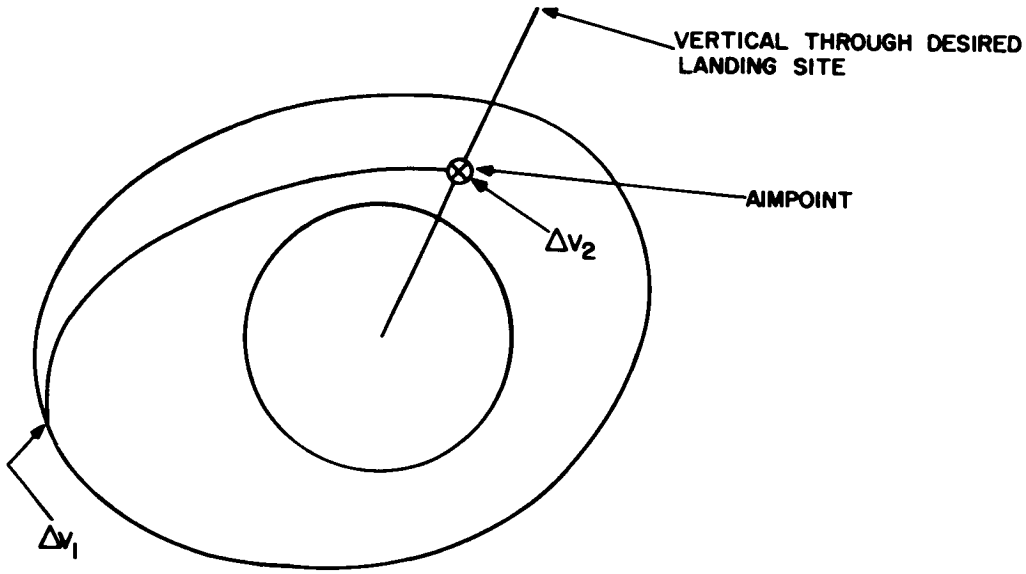


Figure II-6-8. Trajectory for transfer to vertical descent

vertical phase requires the addition of a doppler radar and a radar altimeter. If the altitude of the aimpoint is too high, two altimeters may be required, one for high altitudes and one for low. The doppler radar determines the velocity vector of the vehicle with respect to the lunar surface by measuring the doppler shift in each of three antenna beams fixed to the vehicle.

From this point on, control of the vehicle is very similar to that described for rendezvous in the previous section. The horizontal component of velocity is nulled using a combination of four engines with thrust lines normal to the vehicle axis. The vertical control channel uses the control law

$$\dot{h} = k \sqrt{h}$$

to result in essentially zero impact velocity. In this case the main engines are used at maximum thrust until some low altitude at which the variable thrust characteristics of the engines is utilized for better control of landing.

6.3.4 Return Insertion

The problem of launching from the Moon into a return trajectory is basically no different than launching from the earth except that equipment for aligning the platform and determining initial conditions must be carried along. The exact position of the lunar landing site can be determined with the navigation equipment and a nominal return trajectory can be computed in the guidance computer. Probably the most difficult problem is azimuth alignment of the platform, which can probably be accomplished using a theodolite, which will also be useful for manual midcourse fixes. Boost guidance from the Moon can then be accomplished in the same manner as described for earth boost guidance in section

6.3.5 Summary

The changes in G & C required for lunar landing are thus the addition of one or two altimeters and a doppler radar. If the APOLLO computer is not actually used to control the Saturn during boost guidance this control function would have to be added to the computer.

7.0 References

(1) General Electric Report R59ELC35 - Error Reduction By Digital Data Smoothing Techniques - By N. W. Harvey

(2) General Electric Report R60APS15 - Space Navigation - by H. B. Haake

(3) Random Processes in Automatic Control - by Lanning and Battin, McGraw Hill, 1956

(4) NASA Technical Report 32-38 - Analysis of Radio Command Mid-Course Guidance- by Noton, Cutting, Barnes

(5) NASA Seminar Proceedings - Tracking Programs and Orbit Determination - February 23-36, 1960

(6) Mathematical Methods of Statistics - by Cramer, Prinction Press - 1954

(7) "Glide Re-entry Vehicle Guidance Study" General Electric Ord. Dept. P. Cushman 3/29/61

(8) "Instrument Error Study of a Gyro Stabilized Inertial Platform General Electric Ord. Dept., L. B Harris et al R61POD7

(9) NASA Project APPOLO Working Paper No. 1002 "Saturn C-1 and C-2 Booster Systems". Confidential

(10) NASA Saturn Report

(11) "Interplanetary Mid-Course Guidance Study" GER 9136 Jan. 1959 Goodyear Aircraft Corp.

(12) "A Recoverable Interplanetary Space Probe" R-235 Instrumentation Laboratory, MIT July 1959

(13) "Error Coefficients in a Ballistic Circumnavigation of the Moon" R59EMH23, General Electric Co. April 1959

(14) "Lunar Trajectory Studies", A. Petty, I. Jurkevich, General Electric Co. AFCRL-TN-60-1132

(15) "A Dynamic Analysis and Preliminary Design of Guidance for Lunar Vehicles" Aeronutronic 4 Vols. C-590 (Confidential)

(16) "Trajectory Simulation Program for Polaris Missile" GE R58SD623 Sept. 1958, Immel, Martin, O'Neill

(17) Celestial Observation for Space Navigation, L. Larmore - Aero/Space Engineering, January 1959

(18) Modern Mathematics for the Engineer - Beckenbach - editor - McGraw Hill 1956.

(19) The Prediction of Ballistic Missile Trajectories from Radar Observations L.I. Shapiro-McGraw-Hill, 1957.

(20) Space Trajectories - Edited by Rand Corporation, Academic Press - 1960.

(21) Project APOLLO Data Book - Volume II (General Electric Midtern Report) 1961

(22) Space Flight, Vol. I, Environment & Celestial Mechanics. K.A. Ehriche, D. VanNostrand Co., Inc.

(23) Psychological Factors in the Manual Control of Attitude in the Mercury Vehicle R.B. Voas, A.R.S., Dec. 1960.

(24) Manned Re-Entry Vehicle Study - Contract AF33(616) - 6204, Flight Control Laboratory, WADD

(25) Final Stage Vehicle for the Satellite Inspector Program (SAINT) - MSVD Proposal in response to RFP04-647-61-105

(26) Early Rendezvous Demonstration Study - MSVD Proposal in response to RFQ-TY-64006.