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NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

MSC INTERNAL NOTE NO. 70-FM-78

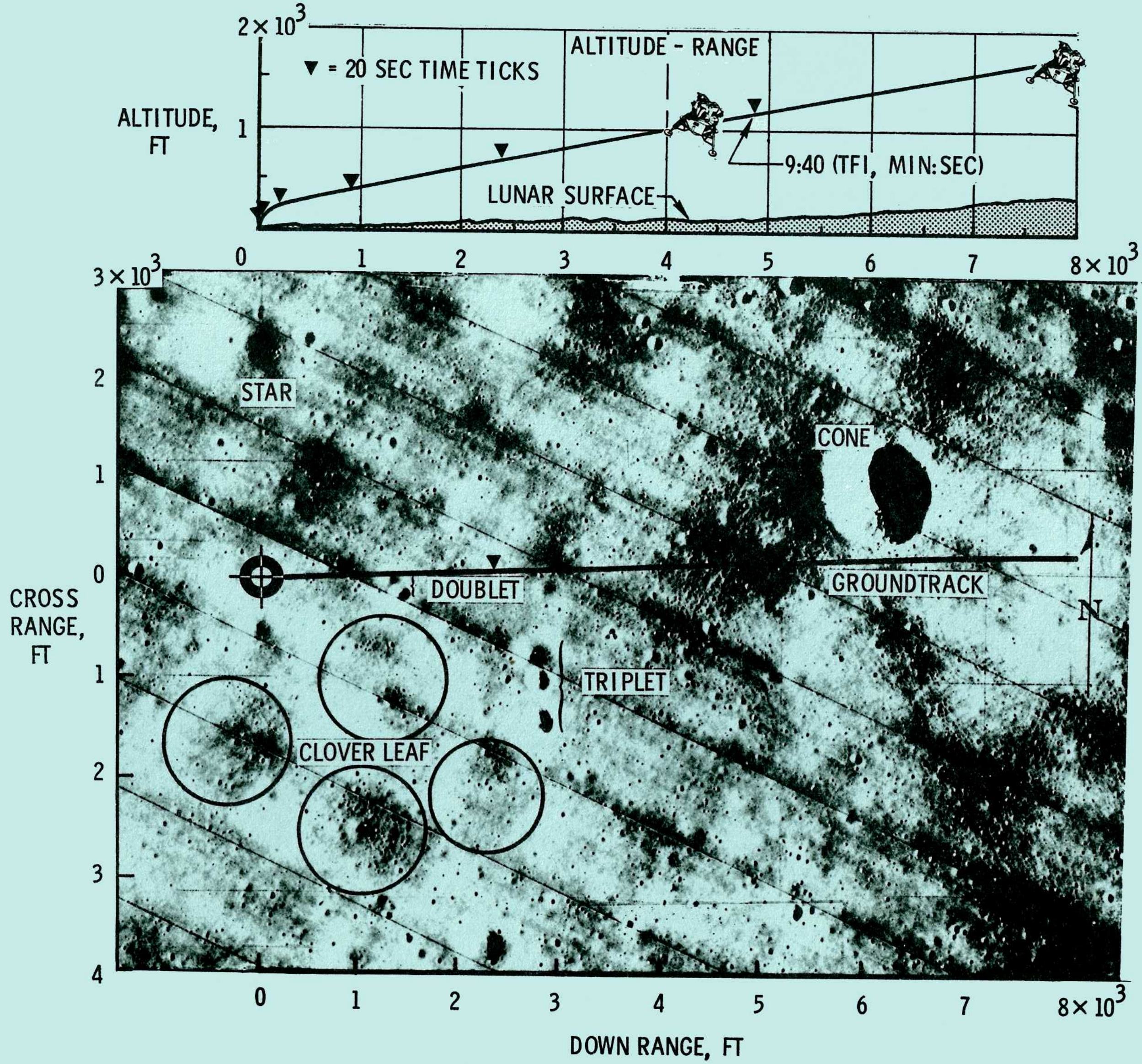
April 8, 1970

APOLLO 13
LUNAR TRAJECTORY NOTES

MISSION PLANNING AND ANALYSIS DIVISION

MANNED SPACECRAFT CENTER
HOUSTON, TEXAS

FRA MAURO LANDING



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PROJECT APOLLO

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Approved: ______________________
John P. Mayer, Chief
Mission Planning and Analysis Division

FOREWORD

The purpose of this document is to provide some general information about the design of lunar trajectories and some specific information about the Apollo 13 mission.

This report, which was written by MPAD and TRW personnel, is composed of the following three sections.

I Environment

II Trajectory planning

III Apollo 13 mission summary

Some interesting background information and definitions are presented in section I. General information about lunar trajectory planning is presented in section II, and the Apollo 13 trajectory is explained in detail in section III.

The major changes in the Apollo 13 mission from the Apollo 12 lunar mission are as follows.

a. The landing site for Apollo 13 is Fra Mauro. This site lies in an elongated valley bordered by north-south trending ridges. These ridges are the Fra Mauro formation and are thought to be ejecta from the Imbrium Basin, 500 kilometers to the north. Although the area around the landing site is probably mantled by post-Imbrian volcanics, several large craters are thought to have penetrated this mantle and to have excavated Fra Mauro material, e.g., Cone and Sunrise Craters. The scientific objectives at this site are to sample material from both Fra Mauro and from the overlying mantle. It is expected that the Fra Mauro material will be older than the samples returned by Apollo 11 and 12. Analysis of the mantle material may yield a cleaner picture of the moon's period of active volcanism.

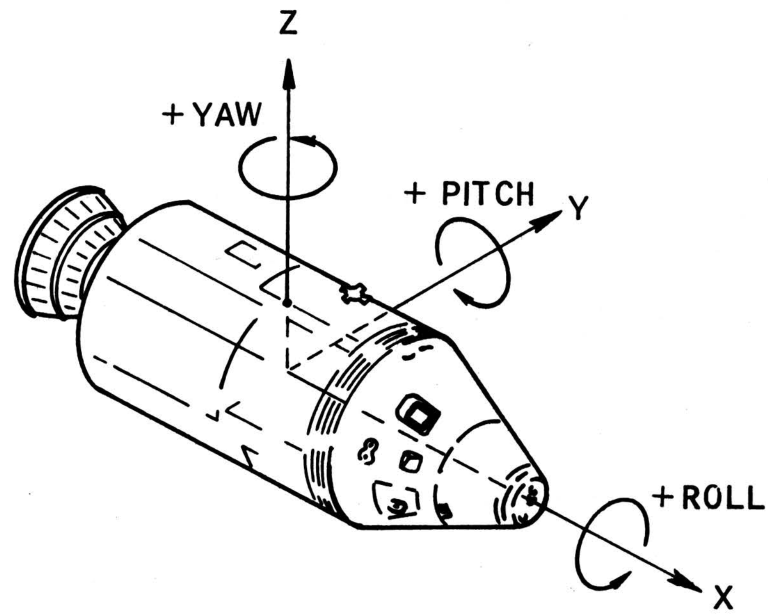
b. After the S-IVB performs the nominal evasive maneuver following transposition and docking, the discarded S-IVB stage will be targeted for lunar impact. The S-IVB impact of the moon will give another test to the Apollo 12 seismometer. As on Apollo 12, the discarded ascent stage from Apollo 13 will be targeted to impact the lunar surface near the vicinity of the Apollo 13 seismometer.

c. Two revolutions after the LOI burn places the CSM/LM on a 60- by 170-n. mi. elliptical lunar orbit, the SPS will perform the DOI burn. This burn, DOI, will place the CSM/LM in a 60- by 8-n. mi. elliptical orbit from which the LM will perform PDI on revolution 14. The CSM will circularize the orbit at the beginning of revolution 13. On Apollo 12, the DOI burn was performed by the LM descent engine.

SYMBOLS

ALHTC	Apollo lunar hand tool carrier
ALSCC	Apollo lunar surface close-up camera
ALSEP	Apollo lunar surface experiments package
AOS	acquisition of signal
ARIA	Apollo Range Instrumentation Aircraft
CDH	constant differential height maneuver
CDR	commander
CSI	concentric sequence initiation
CSM	command/service module
DAP	digital autopilot - a digital computer programed to control the attitude of the spacecraft
EI	entry interface
EMU	extravehicular mobility unit
ETB	equipment transfer bag
EVA	extravehicular activity
g.e.t.	ground elapsed time from lift-off
h_a	apolune altitude - the highest point in lunar orbit
h_p	perilune altitude - the lowest point in lunar orbit
LEC	lunar equipment conveyor
LGC	lunar module guidance computer
LM	lunar module
LMP	lunar module pilot
LOS	loss of signal
MC-1	first LM midcourse correction during ascent phase

MCC	midcourse correction during transearth or translunar phase
MESA	modularized equipment stowage assembly
MSFN	Manned Space Flight Network
PLSS	portable life support system
RR	rendezvous radar
RTG	radioisotope thermoelectric generator
SEQ	scientific equipment
SIDE	superthermal ion detector experiment
SRC	sample return container
SWC	Solar Wind Composition (experiment)
SXT	sextant
TPI	terminal phase initiation
TPF	terminal phase finalization
ΔV	change in velocity



Spacecraft body coordinate system.

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I. ENVIRONMENT

EARTH-MOON-SUN RELATIONSHIP

EARTH-MOON-SUN RELATIONSHIP

Before launching into discussions of trajectories to and from the moon, it is necessary to review some basic geometric relationships between the earth, moon, and sun (fig. 1). The moon rotates 360° about the earth in a near circular orbit every 27.32 days. The distance from the earth to the moon varies between 222 000 miles at the nearest point and 253 000 miles at the furthest point, distances equivalent to 28 to 32 earth diameters.

The earth revolves about the sun every 365.25 days; leap year every fourth year compensates for the small difference. The earth's orbital plane about the sun, called the ecliptic plane, is inclined 23.5° to the earth's equator, the latter defined by the earth's rotation about its north pole.

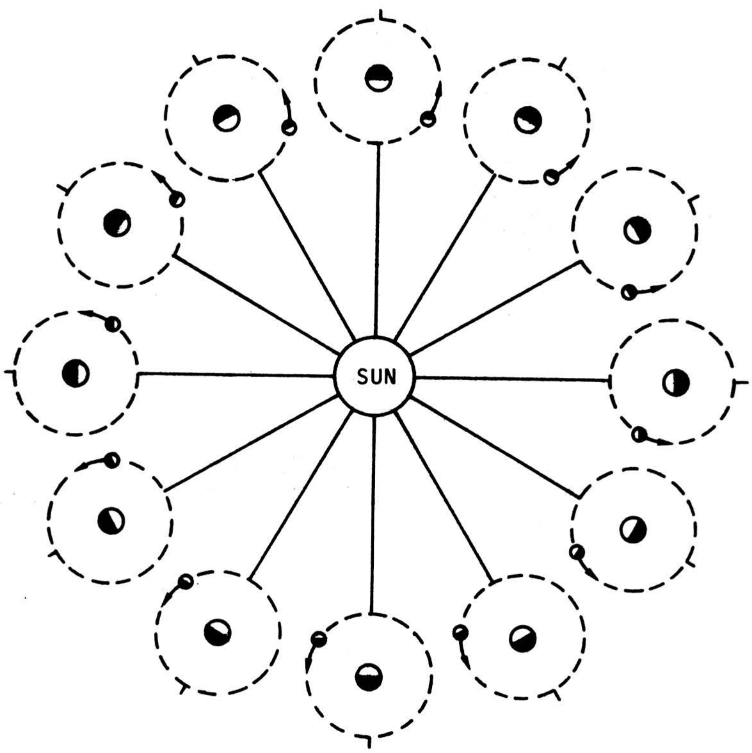


Figure 1.- Earth-moon-sun geometry.

The moon's orbital plane about the earth lies very close (within 5.1°) to the earth's orbital plane about the sun (the ecliptic plane). This relationship is illustrated in figure 2 for circa 1969; the moon's orbital plane is shown inclined 28.6° to the earth's equator. The path of the moon in this plane varies from 28.6° north of the equator to 28.6° south of the equator each revolution. The moon's angular distance measured from the earth's equator is called the moon's declination.

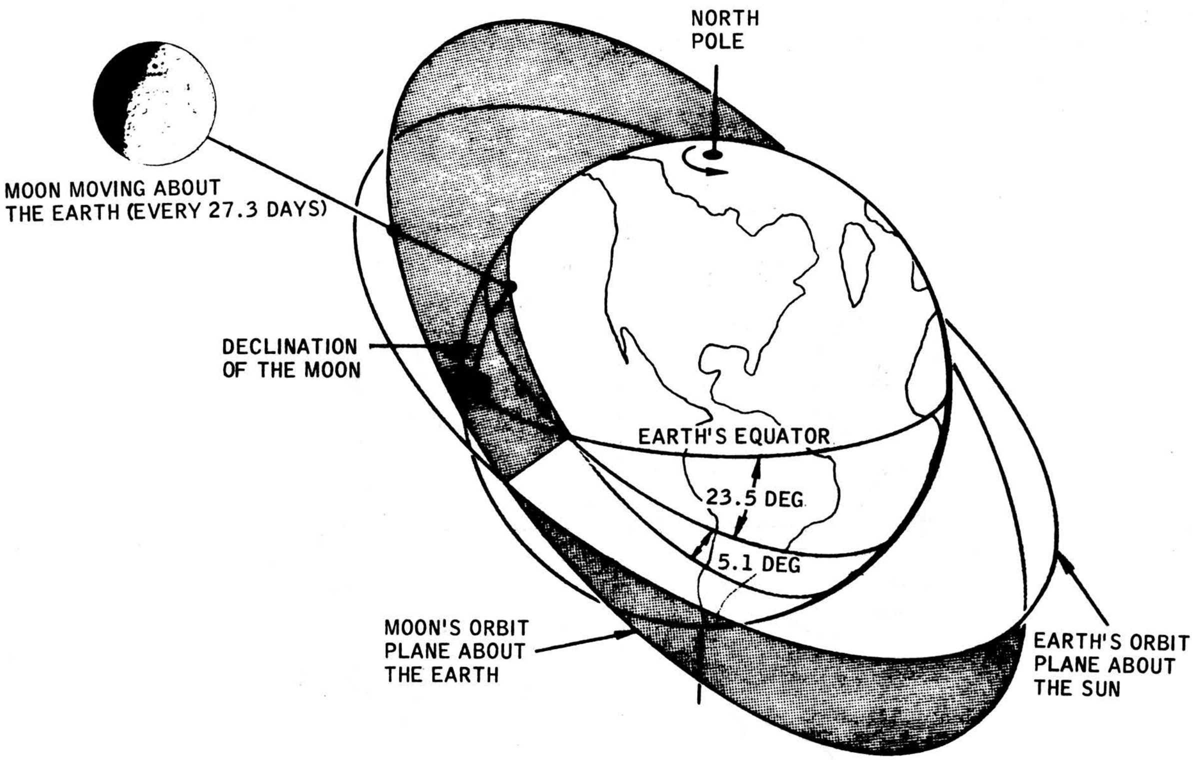


Figure 2.- Earth-moon-sun plane geometry - circa 1969.

The moon rotates about its north pole at the same rate that it moves about the earth. Therefore, the same face of the moon is always pointing at the earth, and the back side of the moon can never be seen from the earth. The visible surface of the moon is illustrated in figure 3.

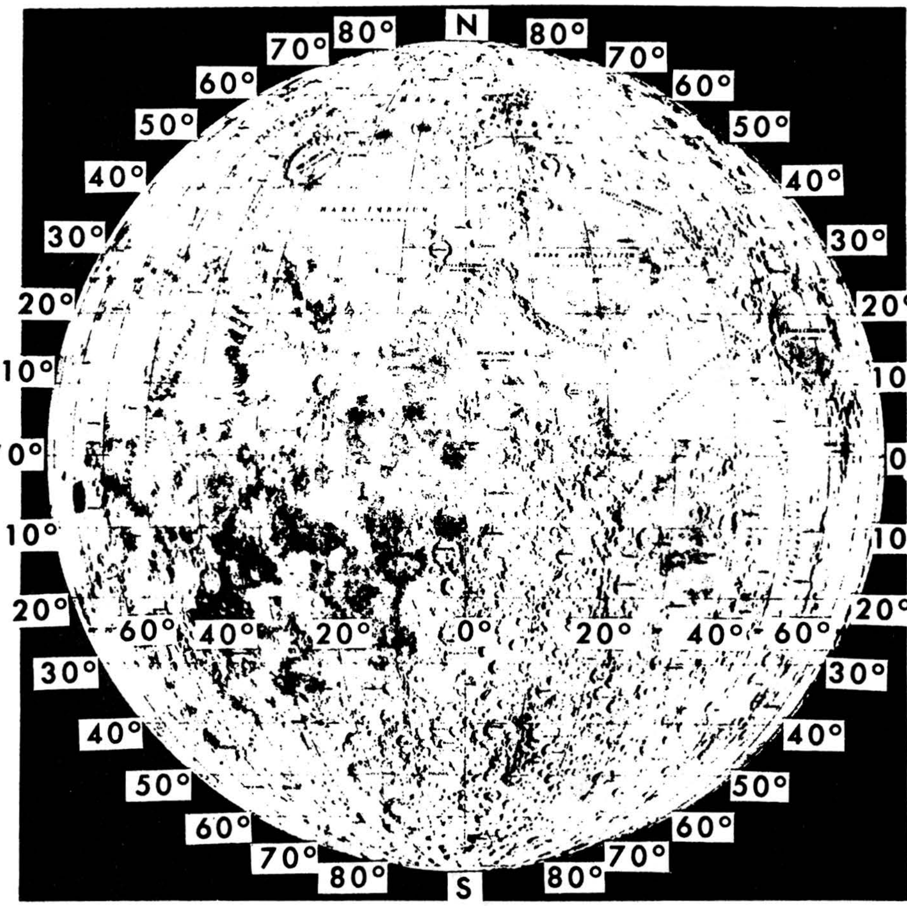


Figure 3.- The visible face of the moon.

While only 27.32 days are required for the moon to rotate through 360° about the earth, a slightly longer period is required for the sun lighting conditions to repeat themselves on the surface of the moon (fig. 4). The variation is caused by the earth's moving approximately 29° around the sun during this period. The 29° movement is significant because the lighting conditions at a particular position on the surface of the moon are repeated only once every 29.5 days, and the lighting conditions are important in establishing the time of the month that an Apollo mission is launched to the moon, as will be discussed later.

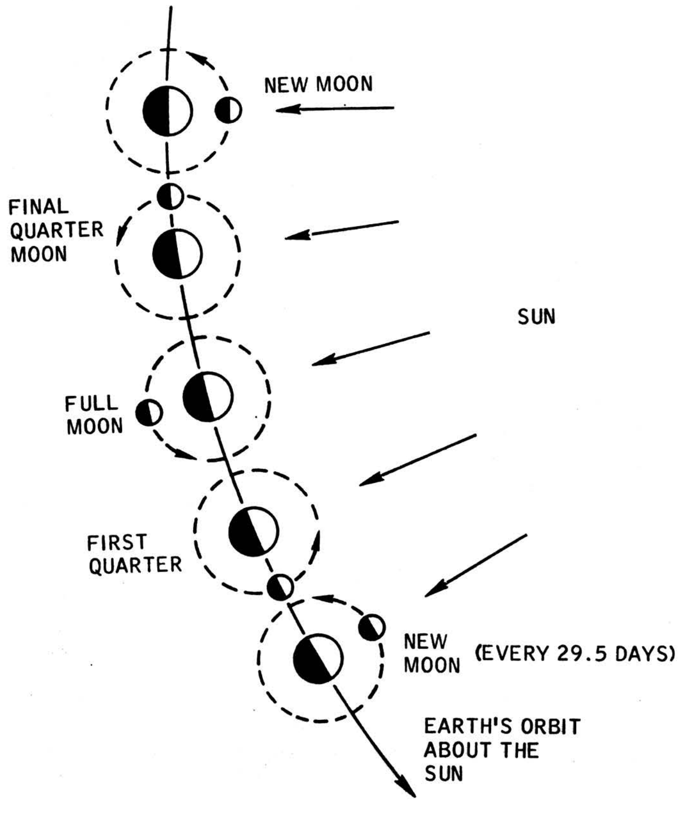


Figure 4.- The phases of the moon.

GRAVISPHERE AND SPHERE OF INFLUENCE

GRAVISPHERE AND SPHERE OF INFLUENCE

When the effect of the moon's gravity on a vehicle traveling between earth and moon is being considered, there are two important areas of interest: the gravisphere and the sphere of influence. The area closest to the moon is the sphere of equal gravitational attraction between the earth and moon. This area is called the lunar gravisphere. Many people think it is at this point that the vehicle stops decelerating with respect to the moon. This is not true; because of the motion of the moon in its orbit around the earth, the spacecraft has actually been speeding up with respect to the moon long before the gravisphere or the sphere of influence is encountered.

Of much greater importance to trajectory calculations than the gravisphere is a locus of points called the sphere of influence. The sphere of influence is defined as the locus of points where the ratio of the force with which the earth perturbs the selenocentric (moon centered) motion of a vehicle (PE) to the force of the moon's gravitational attraction (AM) is equal to the ratio of the force with which the moon perturbs the geocentric (earth centered) motion of a vehicle (PM) to the gravitational attraction of the earth (AE). Stated mathematically

$$\frac{PM}{AE} = \frac{PE}{AM}$$

where PM and PE represent the perturbing force of the moon and earth respectively and AM and AE represent the gravitational attraction of moon and earth, respectively.

At the point of crossing the sphere of influence, the vehicle is said to change reference bodies. Thus, it is at the sphere of influence that the vehicle changes from earth reference to moon reference on a translunar trajectory and from moon reference to earth reference on a transearth trajectory (fig. 1).

The perturbing effect of the earth on a selenocentric trajectory is a maximum at the boundary of the sphere of influence and becomes less as the vehicle gets closer to the moon.

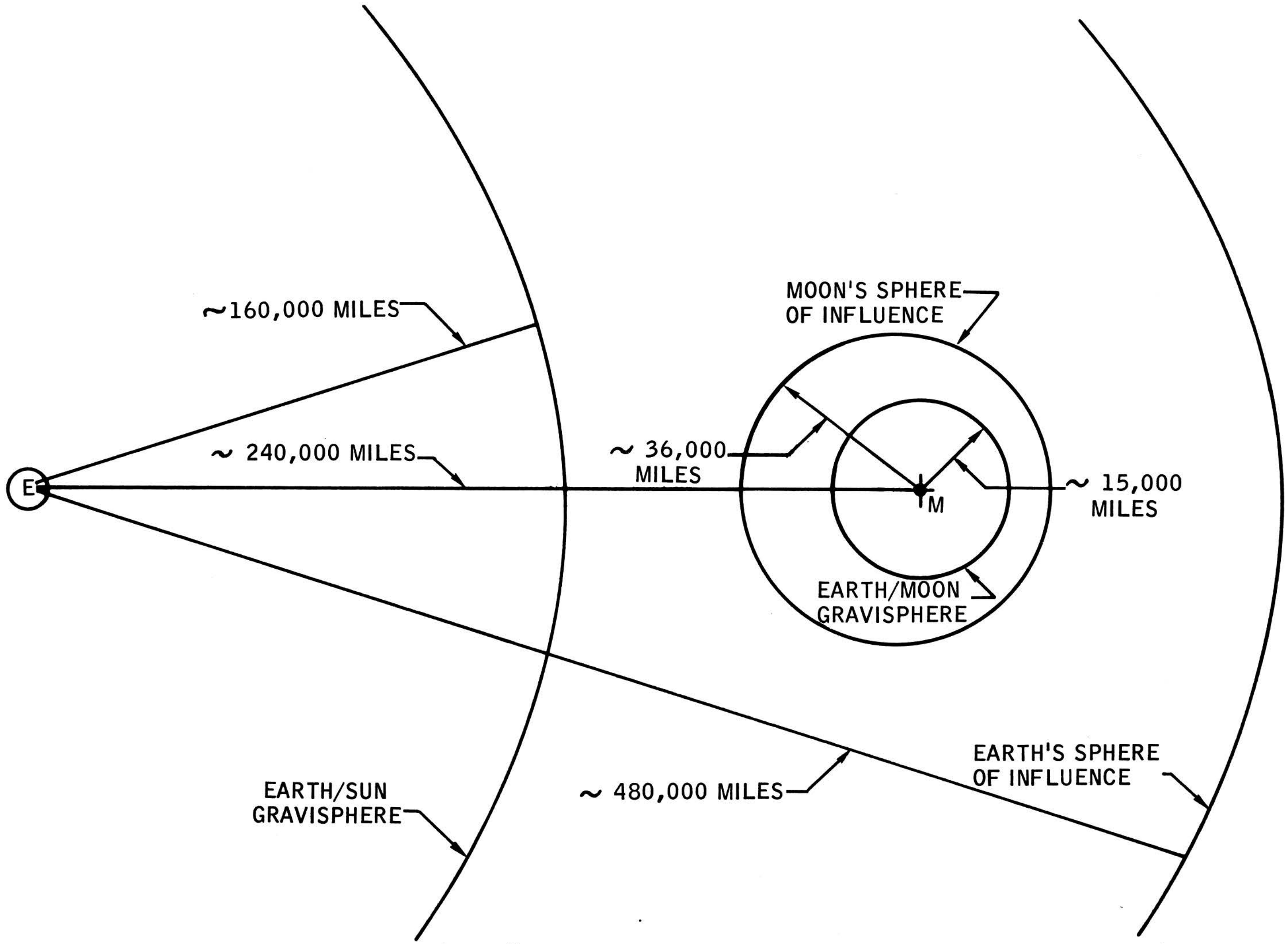


Figure 1.- Gravitational influences on lunar trajectories.

INERTIAL AND RELATIVE VELOCITIES

INERTIAL AND RELATIVE VELOCITIES

The two terms most frequently mentioned in discussions of the speed of the spacecraft are inertial and relative velocities. (Velocity has direction as well as magnitude, but this distinction is frequently ignored in conversation.) Sometimes the difference between the two velocities can seem quite confusing. Inertial velocity is the rate at which the vehicle's position is changing in a frame of reference which is assumed to be fixed. For instance, the term geocentric inertial velocity suggests that the velocity (rate of change of position) is determined with respect to the earth's center (geocentric) and that the earth's center is motionless; that is, only the change of the position of the vehicle with respect to the center of the earth is considered.

The term relative velocity refers to the velocity of the vehicle relative to a body which is also in motion with respect to the reference body. Thus, if the vehicle is on the out-going leg of a lunar trajectory, the relative velocity would be the velocity of the spacecraft relative to the moon, which is the difference in the velocity of the vehicle with respect to the earth and the velocity of the moon with respect to the earth. However, if the moon is the reference body (selenocentric) and the moon is considered to be motionless or fixed in space (inertial), then the rate of change of the position of the spacecraft with respect to the moon is called selenocentric inertial velocity.

II. TRAJECTORY PLANNING

LAUNCH AND TRANSLUNAR INJECTION

LAUNCH AND TRANSLUNAR INJECTION

Trajectories to the moon are almost always flown in two steps, launch into earth orbit and then injection into the translunar trajectory. Apollo missions are launched from the Kennedy Space Center in Florida. The acceptable direction of launch (called the launch azimuth) is restricted between 72° and 108° east of north (fig. 1). The restriction is based on crew safety requirements.

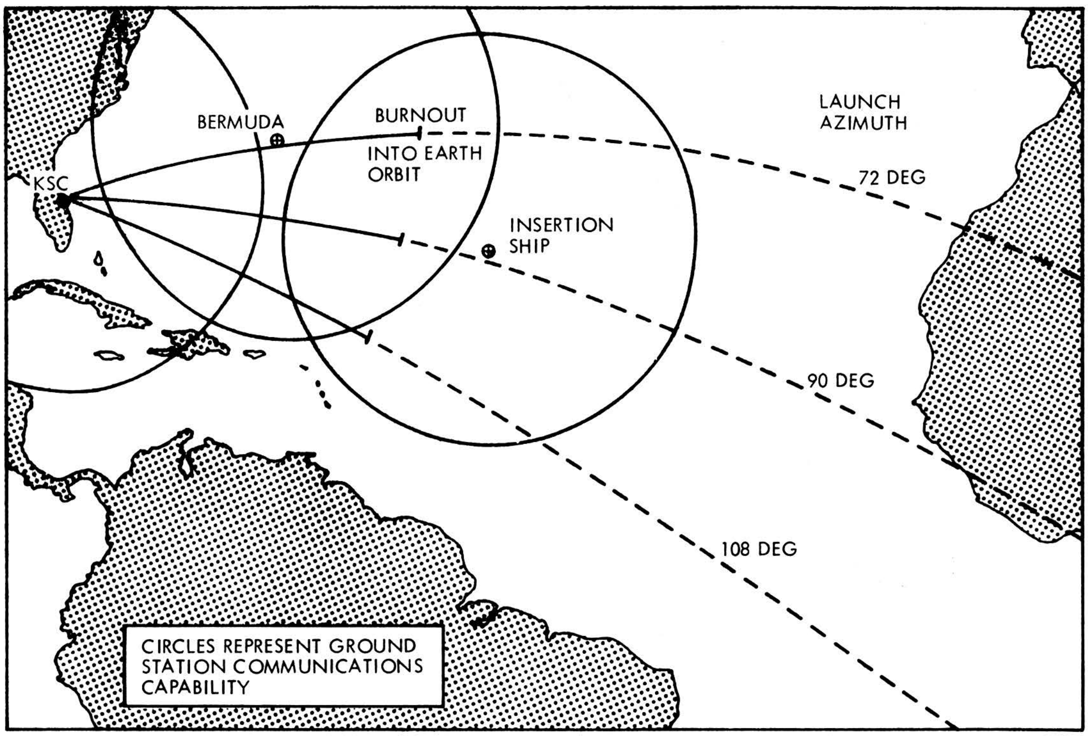


Figure 1. Launch azimuth from Kennedy Space Center.

The buildup in speed and altitude during the launch into earth orbit phase is illustrated in figure 2. Note that the inertial speed at lift-off is approximately 913 miles per hour (1340 feet per second), the speed of Cape Kennedy caused by the earth's spin about the north pole. Both the S-IC first stage and the S-II second stage are completely expended, and part of the S-IVB final stage is expended in achieving earth orbit. The change in slope of the speed curve at staging is caused by the change in thrust acceleration of the different rocket engines.

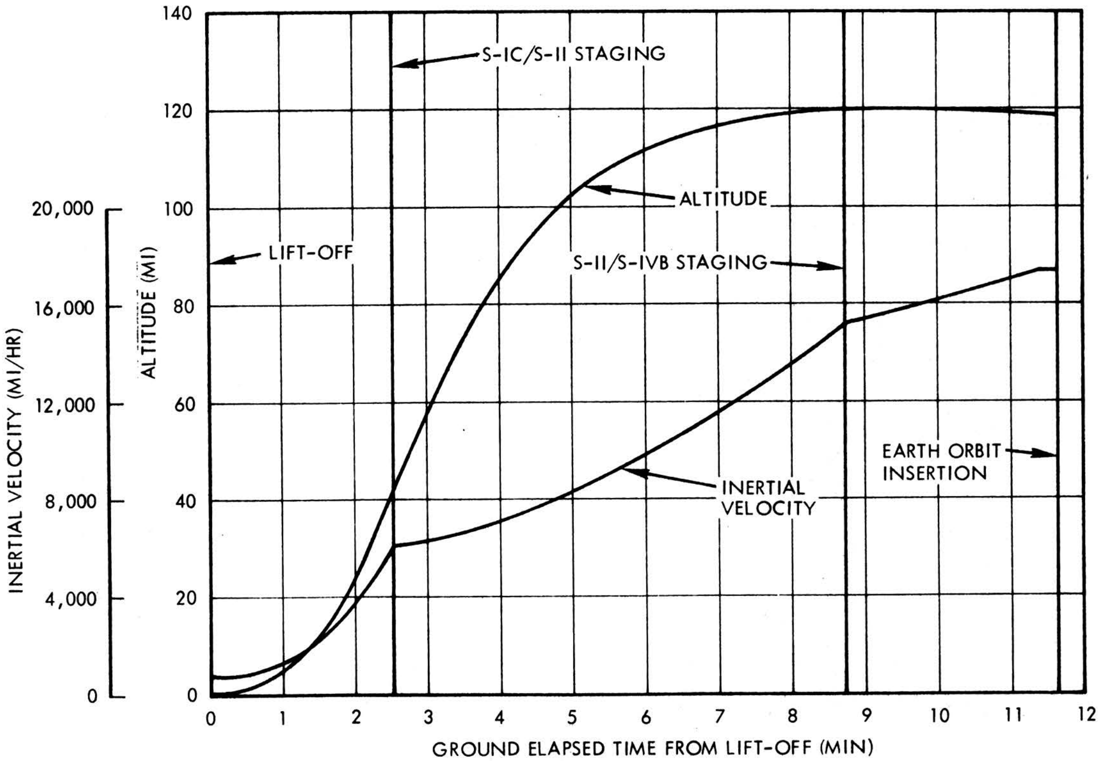


Figure 2. Launch into Earth orbit.

Burnout into the 115-mile (100-nautical mile) altitude circular orbit normally flown in the Apollo missions occurs with an orbital speed of 17 440 miles per hour (25 580 feet per second). Achieving earth orbit has been compared to throwing a stone off a mountain (fig. 3). The harder (or faster) the stone is thrown, the farther it will travel before it hits the earth. If the stone is thrown fast enough (orbital speed), it will fall around the earth. If the stone were thrown a little harder (superorbital speed), it would rise away from the earth, fall back towards it, and then rise again as illustrated in the figure. Finally, if the stone were thrown fast enough, it would rise so high that it would completely leave the influence of earth's gravity and escape into space. The Apollo trajectories to the moon travel at near escape speeds.

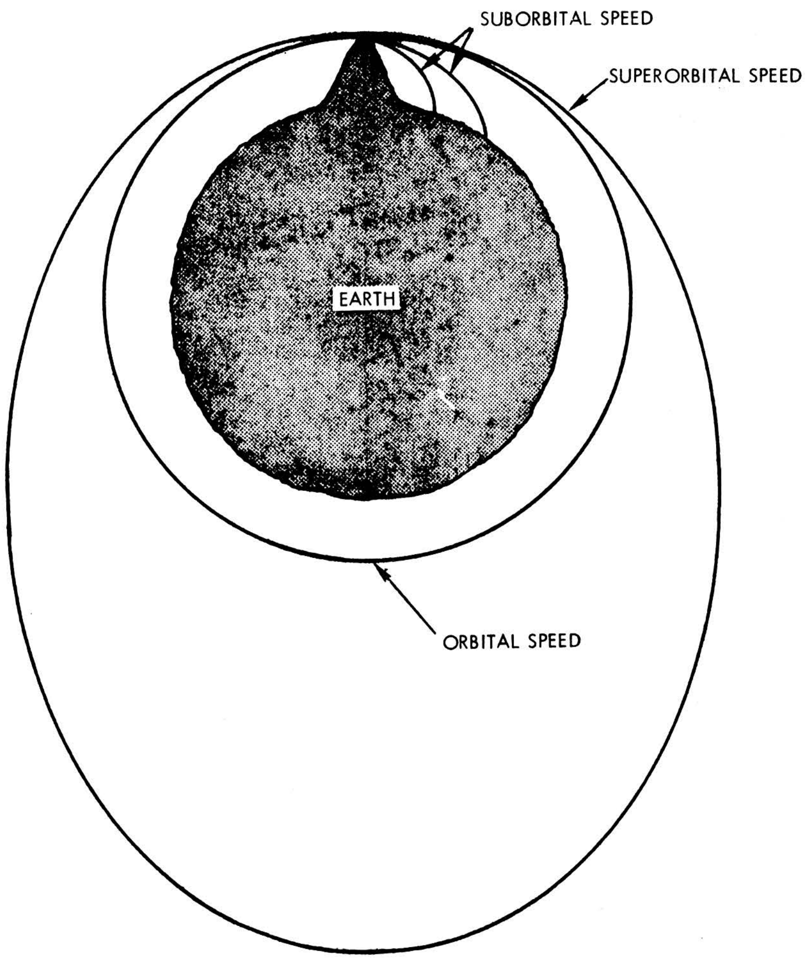


Figure 3.- Throwing a stone off a mountain.

The mission plan calls for the Apollo spacecraft (Saturn launch vehicle) to remain in earth orbit for systems checkout for a period of one to three revolutions about the earth prior to translunar injection (TLI). The S-IVB stage is reignited when the correct position relative to the moon is reached, and the spacecraft speed is increased almost to escape velocity. The speed buildup is illustrated in figure 4. The apogee altitude also shown in figure 4, is not the actual altitude but is the highest altitude that would be achieved in the orbit if the rocket engine stopped firing at that time. The curve illustrates the rapid buildup of apogee altitude toward the very end of the TLI rocket burn. Escape from earth's gravity would be characterized by an apogee altitude of infinity. Most Apollo missions will have speeds of 24 300 miles per hour (35 600 feet per second) and apogee altitudes of approximately 320 000 miles (the moon's greatest distance from the earth is only 253 000 miles). It would take approximately 9 days for the spacecraft to travel from TLI burnout to the apogee altitude of the orbit if it were not for its destined encounter with the moon.

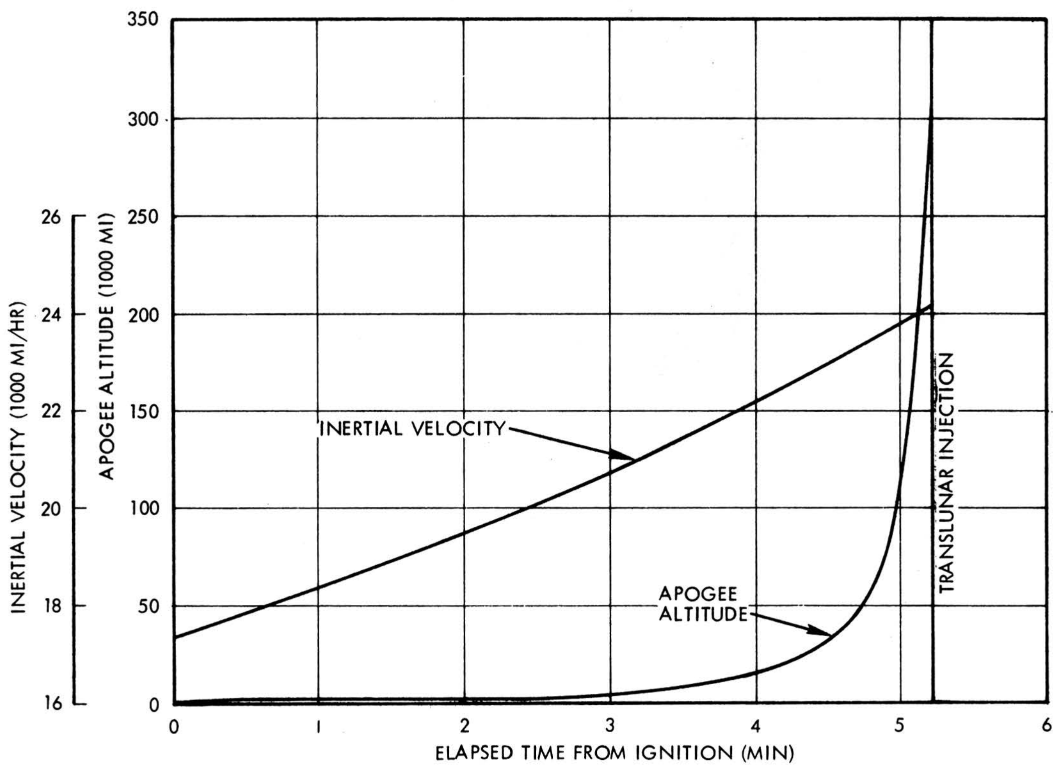


Figure 4. Translunar injection.

FREE-RETURN TRAJECTORIES

FREE-RETURN TRAJECTORIES

Very early in the development of the Apollo lunar landing program, it was decided for the paramount reason of crew safety that the coast to the moon should be on a free-return type trajectory. The name free return trajectory to some extent is a misnomer. A true free return trajectory would be a trajectory which, after translunar injection, would provide for circumnavigation of the moon and a safe return to an acceptable entry corridor at the earth with no expenditure of propellants. Such a trajectory is virtually impossible to achieve because the translunar injection would have to be accurate to about 0.05 mph. In Project Apollo, free-return trajectories refer to those that are free return within the propulsive capabilities of the reaction control system in case of a failure of the main spacecraft engine after translunar injection. Therefore, a free-return trajectory will usually require midcourse corrections of up to 100 fps for a safe circumnavigation of the moon and return to the earth entry corridor.

The translunar injection maneuver places the spacecraft on an initial trajectory which would extend much further than the average 240 000-mile distance to the moon and which would take weeks to complete if the lunar gravitational field did not affect the motion of the spacecraft. This trajectory is aimed to intercept the moon's orbit not where the moon is located at launch or at translunar injection, but where it will be located when the spacecraft has reached the moon, about three days after lift-off. Because translunar injection occurs at the low part of the trajectory (at the parking orbit altitude of 115 miles), the burn will take place on the side of the earth facing away from the moon (fig. 1).

If the timing and orientation are selected properly so that the spacecraft and the moon arrive at the intercept point together, the gravitational pull of the moon will influence and distort the trajectory away from the initial elliptical shape. If the spacecraft passes close to the moon (so that the gravitational pull is strong) and is traveling slowly (so that the pull acts over a long time), the spacecraft can be pulled into the surface or whipped around the moon and swung into an orbit which does not return very closely to earth again (the slow trajectory of fig. 1). If the spacecraft is further away from the moon and travels faster, the moon will not affect the path enough, and an orbit like the fast trajectory of figure 1 will result. However, if the speed and distance are arranged exactly right, the spacecraft will be swung around the moon at just the right direction and speed so that it will coast back to earth again. The trajectories that satisfy these special conditions are called free return trajectories.

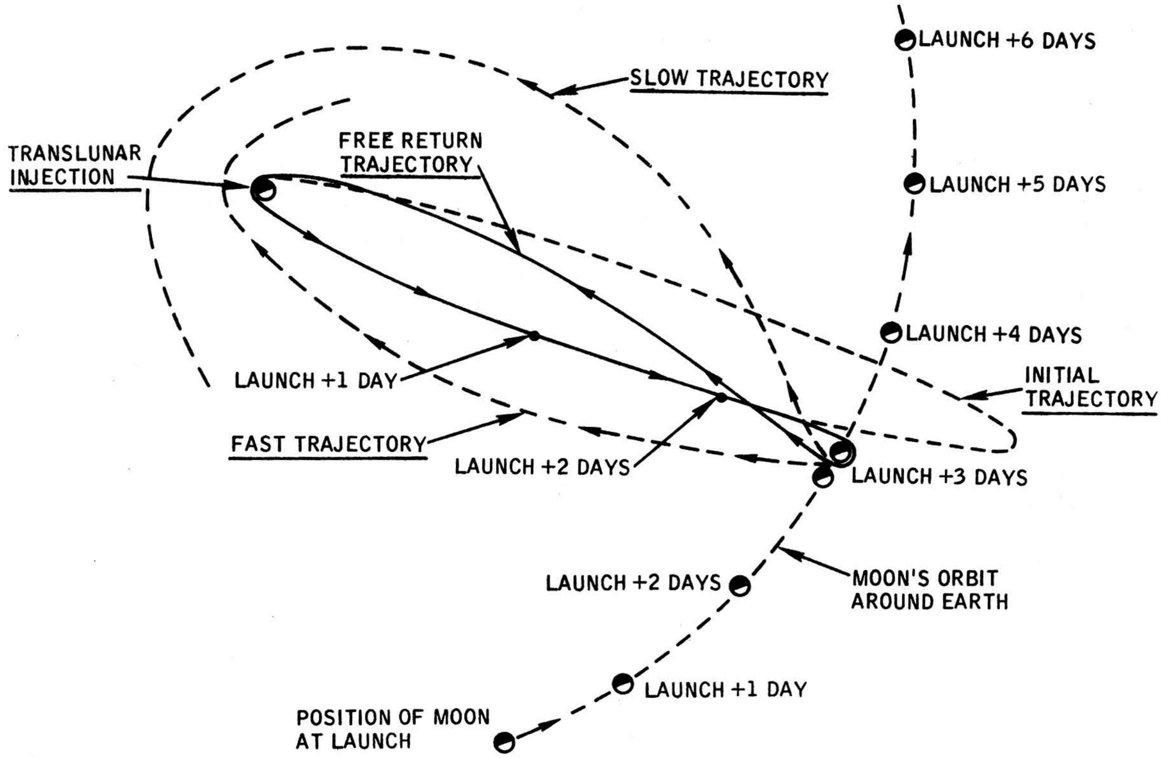


Figure 1.- Geometry of a free return trajectory.

In figure 1, the initial trajectory of the spacecraft is shown placed well ahead of the moon at time of launch. Nominally, the velocity of the vehicle at translunar injection will be free-return which means that, as the vehicle passes behind the moon, the lunar gravitation will deflect the trajectory back toward the earth into a safe reentry corridor.

If the velocity at translunar injection is slightly below that required for a free-return trajectory, the moon will deflect the trajectory too much and, without a midcourse correction, the vehicle would return to a perigee altitude well above the atmosphere and thus would not reenter. This is illustrated by the slow trajectory in figure 1.

If the velocity at translunar injection is greater than that required for a free-return trajectory, the spacecraft would not be deflected enough by the moon and would approach the earth from the opposite side, similar to the fast trajectory shown in figure 1. The precision at translunar injection required to accomplish the return to earth within the 20-mile limit required for safe capture is staggering. Because this precision far exceeds what can be expected from the launch vehicle, a number of small midcourse maneuvers are planned to correct the errors.

If the velocity at translunar injection were very much slower than that required for a free-return trajectory, the spacecraft would pass around the trailing edge of the moon as opposed to the leading edge as illustrated by the third stage (S-IVB) trajectory in figure 2. Inertially, the S-IVB stage is slower than the CSM and, therefore, arrives at the moon later; thus, the moon has had time to get ahead of the approach path. Instead of being deflected back toward the earth as is the spacecraft trajectory, the trajectory of the S-IVB is deflected away from the earth in the direction of the moon's motion. This deflection will accelerate the S-IVB to an escape trajectory with respect to the earth and will place it in a solar orbit. The process of slowing the S-IVB after spacecraft separation is accomplished by venting the residual propellant out of the fuel tanks and is called S-IVB blow-down.

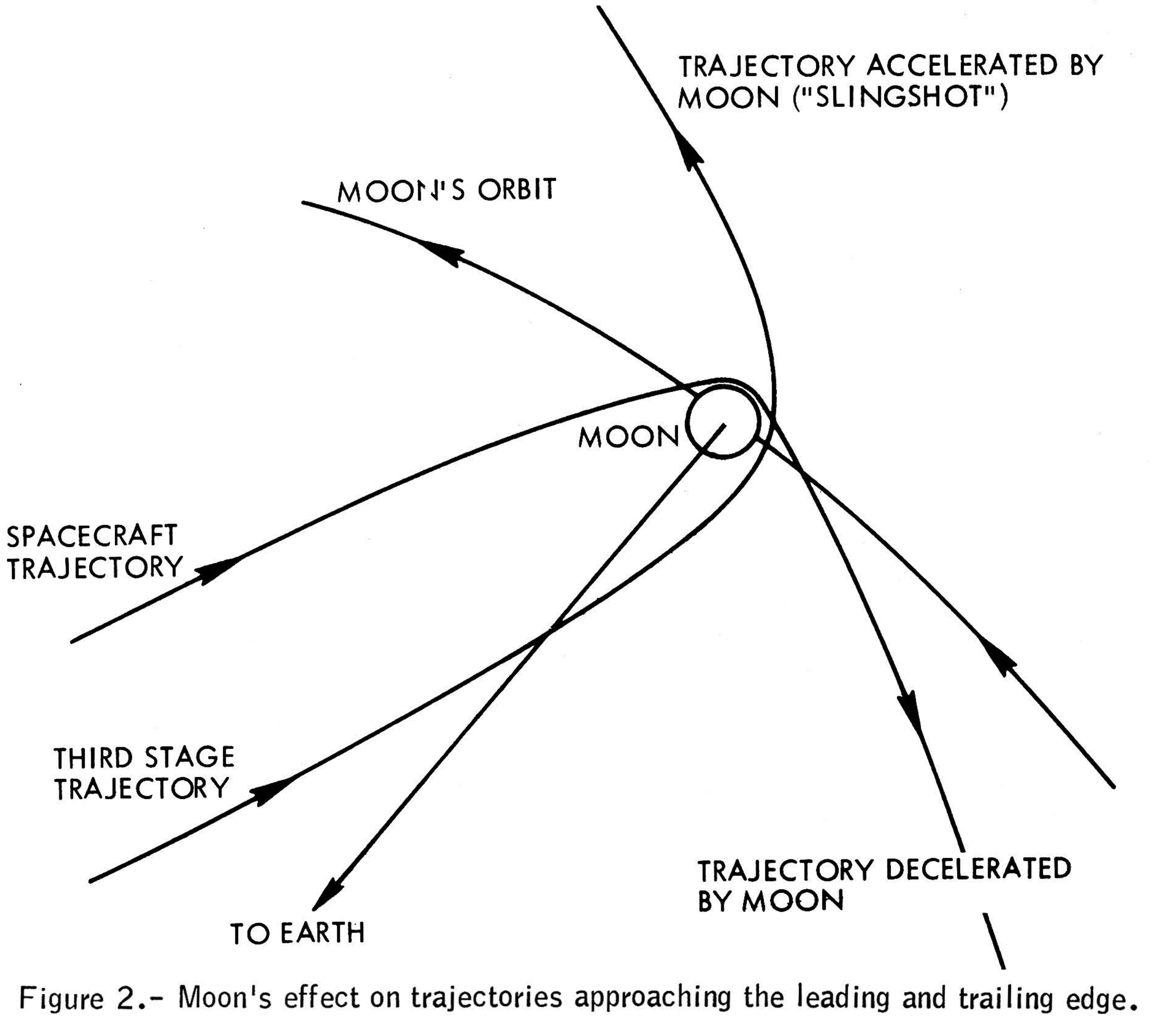


Figure 2.- Moon's effect on trajectories approaching the leading and trailing edge.

As the Apollo spacecraft falls toward the moon on the free-return trajectory, it is traveling about 5690 miles per hour (8347 feet per second) at the point of closest approach to the moon (perilune). To land on the surface, it is necessary to reduce this speed to zero to avoid arriving with spectacular effects. Because there is no atmosphere on the moon which could be used to slow the spacecraft down, the only means of braking is the spacecraft's rocket engine. Because more fuel is always required if more weight is to be slowed down, it is important to try to minimize the weight to be decelerated. For the Apollo mission, the lunar rendezvous scheme has been selected as the most efficient technique. In this method, the entire spacecraft is first partially slowed to a speed of about 4070 miles per hour (5972 feet per second). This speed is slow enough that the gravitational field of the moon will capture the spacecraft and place it in a slightly elliptic orbit about the moon. Two revolutions later, the vehicle is slowed to 3643 miles per hour to circularize the orbit. Then a portion of the entire spacecraft (the lunar module or LM) is detached, and its velocity is slowed to zero for the lunar landing, while the remainder of the spacecraft continues to orbit the moon. When it is time to leave the surface, an even smaller portion (the ascent stage of the LM) ascends back into orbit. In this way, only relatively small masses must be decelerated from and accelerated to the 3643-mile per hour speed required for the lunar orbit. This technique saves many tons of fuel which would otherwise be required if the entire spacecraft arrived at the moon and descended directly to the surface.

A lunar orbit altitude of 69 miles has been selected for Apollo. Therefore, as the spacecraft falls toward the moon on its free-return trajectory, when it reaches an altitude of 69 miles, the braking (or lunar orbit insertion) burn is performed to place the spacecraft into lunar orbit. The trajectory could be designed as shown in figure 3 so that the burn would be visible from the earth. However, if the rocket engine should fail to ignite, a crash onto the surface would result. For obvious reasons of crew safety, the free-return trajectory is designed so that its lowest altitude is about 69 miles, and the burn is performed when it reaches this point. Because the point of closest approach to the moon is on the back side, performance of the burn at the lowest altitude forces the lunar orbit insertion burn to be made while the spacecraft is hidden from the earth.

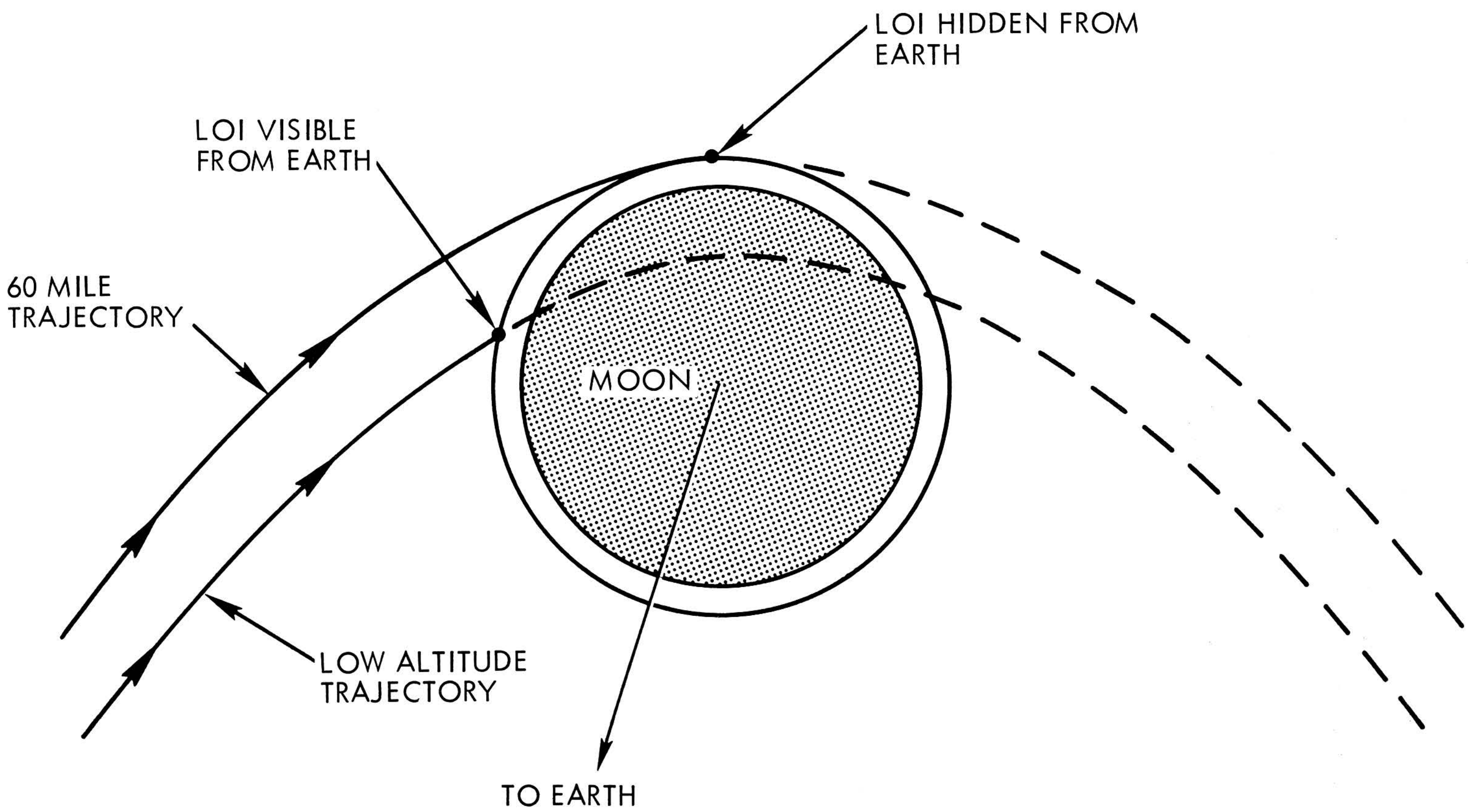


Figure 3.- Position of the lunar orbit insertion (LOI) maneuver.

In future lunar missions, NASA will use hybrid trajectories. A hybrid trajectory is a combination of a free return (within RCS capabilities) trajectory and a non-free-return trajectory to the moon which permits more efficient use of fuel. However, even these hybrid trajectories can be thought of as free-return trajectories because they are designed so that the non-free-return part of the trajectory can be converted to a safe earth return by use of the lunar module descent propulsion system (approximately 2000-feet per second capability).

LAUNCH WINDOWS

LAUNCH WINDOWS

There are a number of considerations which determine the unique time period called the launch window from which the lunar mission is flown. These considerations are as follows.

1. Daylight launch from Kennedy Space Center

2. Launch azimuth (direction) from Kennedy Space Center restricted from 72° to 108° from north

3. Translunar injection to occur over the Pacific Ocean (as opposed to the Atlantic Ocean)

4. Low sun elevation at the lunar landing site

5. Goldstone, California, radar coverage of the lunar landing phase

6. Daylight earth landing in the prime recovery area

The time of lunar landing is almost uniquely determined by the location of the lunar landing site and by the acceptable sun elevation angle range. Low sun elevation angles from 5° to 13° are required to create visible shadows of the craters to aid the crew in viewing the lunar terrain (fig. 1).

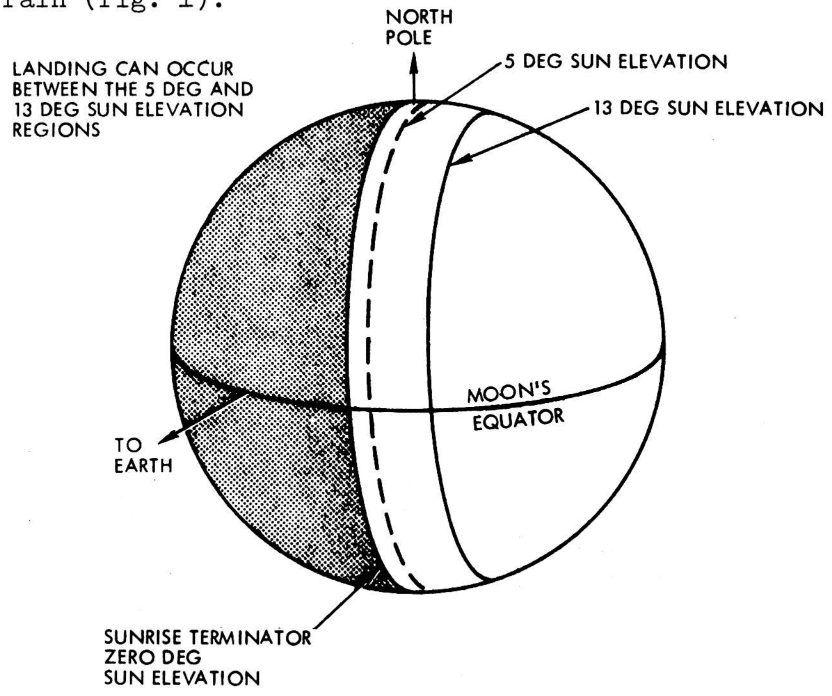


Figure 1. Sun elevation angle for lunar landing.

This lighting is equivalent to an early morning sunrise at the landing site. Because lunar sunlight incidence changes 0.5° per hour (as opposed to 15° per hour on the earth), the above elevation angle restriction established a 16-hour period which recurs approximately every 29.5 days, when the landing should be attempted. A backwards calculation of the total amount of time required for launch from earth, for flight to the moon, and for preparation for landing will establish the earth day of launch.

Because there are several candidate landing sites, there are several days of launch opportunity available. The most easterly landing site, where the proper lighting conditions will occur first, is assigned to the first opportunity. If, for some reason, the mission is not flown at this opportunity, a site further west will be selected for a later day. With the five candidate lunar landing sites illustrated on the picture of the moon presented earlier, an 8-day period of launch opportunity is established. An opportunity does not occur every day, but within the 8-day period, five launch opportunities will occur, one for each candidate landing site.

When the landing site and appropriate day of launch have been selected, it is necessary to determine the time of launch. The two major considerations involved are the acceptable launch azimuth (direction) range from the Kennedy Space Center and the location of the moon at spacecraft arrival. The geometry of the launch window as seen by an observer in space is presented in figure 2. The north pole, the equator, and the Kennedy Space Center latitude are included in the drawing. Because the earth rotates about the north pole one revolution each day (15° per hour), the launch site would rotate in the minor circle (labled KSC latitude) at the same rate. Also illustrated on the figure is the orbital plane of the moon about the earth, the expected location of the moon at spacecraft arrival, and the lunar antipode (the opposite direction of the moon). As viewed from inertial space, the moon moves relatively slowly about the earth (one revolution every 27.32 days or approximately 0.5° per hour). It is necessary for the spacecraft to be launched into an orbital plane that contains the position of the moon and its antipode at spacecraft arrival. Because the direction of launch is restricted[a] from 72° to 108° east of north, launch can occur only when the direction of launch is within the required range to intercept the moon. The 72° launch azimuth is always the first opportunity; and as the launch site rotates to the east, the launch direction moves from northeast to east and to southeast until the 108° launch azimuth restriction is encountered. The 36° band of launch azimuth allows approximately a 4-hour, 30-minute period of launch opportunity. This period is called the daily launch

[a]This restriction is made for crew safety reasons.

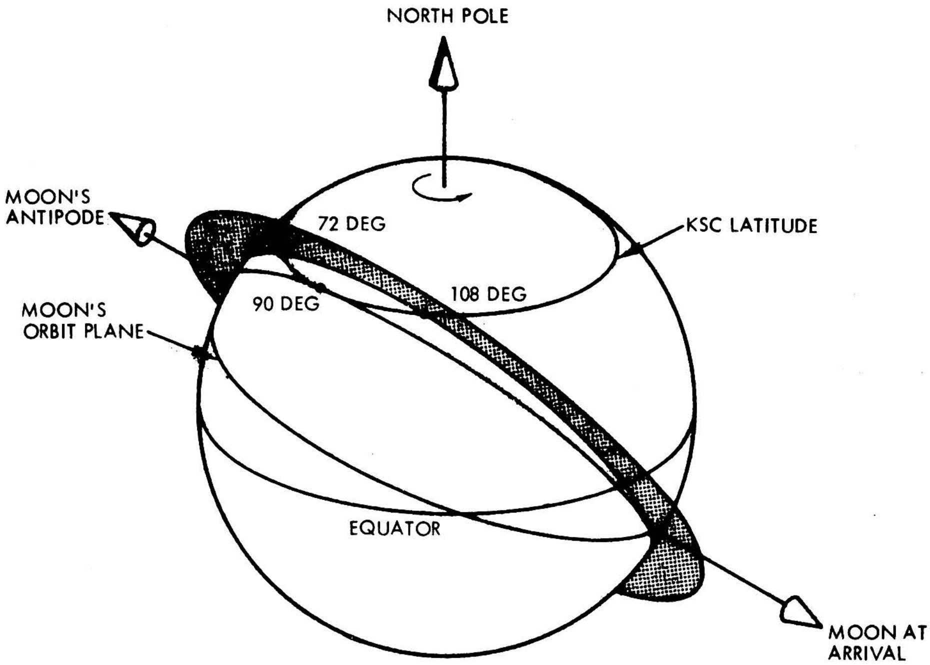


Figure 2. Daily launch window.

window. The daily launch window can be further restricted by the requirement to provide Goldstone, California, radar coverage of the lunar landing phase. This restriction occurs because there is only a period of approximately 12 hours a day when Goldstone is able to see the moon.

Two daily launch windows occur each day; one results in a translunar injection burn out of earth orbit in the vicinity of the Pacific Ocean, and the other one results in a burn in the vicinity of the Atlantic Ocean. These daily launch windows are referred to as the Pacific injection opportunity and the Atlantic injection opportunity. The Pacific injection opportunity is preferred because it normally requires the highly desirable daytime launches while the Atlantic injection opportunity normally requires nighttime launches. The Pacific injection also provides a less hectic earth orbit timeline and improved post-TLI MSFN coverage.

The last consideration, that of a daylight earth landing, may eliminate the launch opportunities to the far eastern lunar landing sites. This elimination is primarily an effect of the required sun lighting conditions at the lunar landing sites.

The spacecraft earth landing always occurs near the lunar antipode as defined at the time of lunar departure (transearth injection). Sun lighting conditions are illustrated in figure 3 for a far eastern lunar landing site (≃ 40° east longitude) and for a near central lunar landing site (≃ 15° east longitude). The earth landing near the lunar antipode is also indicated in this figure. With a far eastern lunar landing site, earth landing occurs in the early morning before sunrise (this was the case with the Apollo 8 mission); while for a near central site, earth landing occurs in the morning. Finally, for lunar landing sites to the west, afternoon earth landings can be expected.

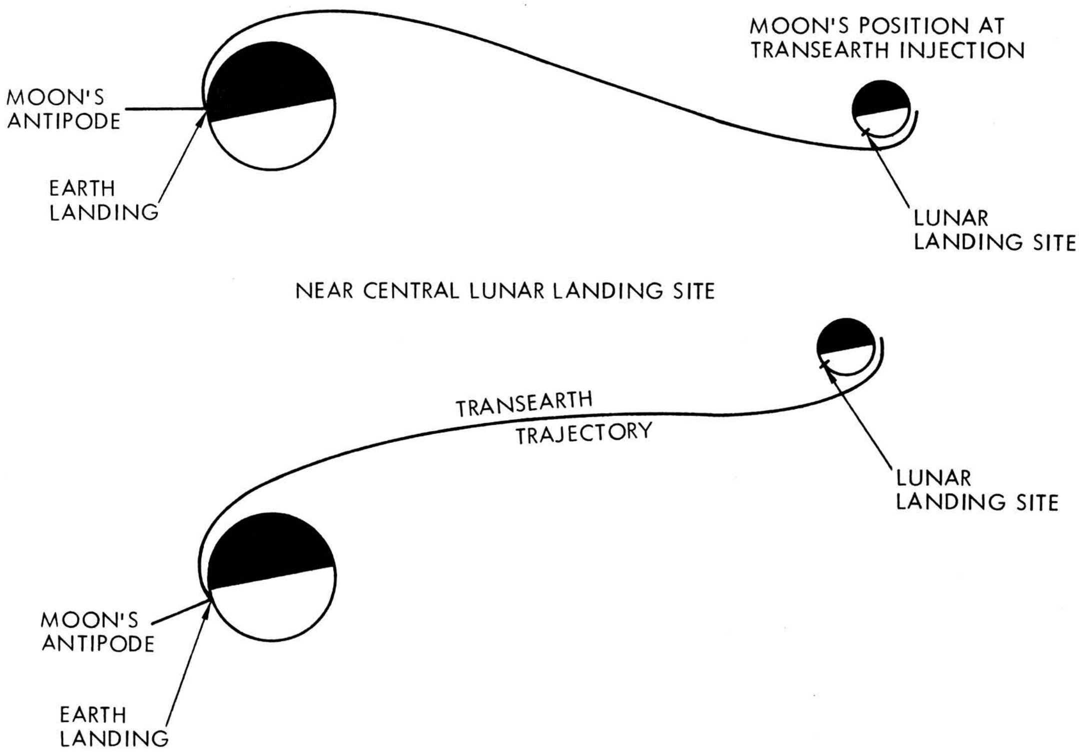


Figure 3.- Daylight at earth landing.

NAVIGATION FOR LUNAR LANDING MISSION

NAVIGATION FOR LUNAR LANDING MISSIONS

Navigation of a spacecraft is a process which involves the taking of measurements that are suitable to define the vehicle's trajectory, and the sending and receiving of information between ground control and the vehicle. The complete navigation network for the Apollo 11 mission includes several measuring devices onboard each of the vehicles and on the ground.

The command/service module (CSM) has an optical tracking system that consists of a 1 power telescope and a 28 power sextant. The sextant can be used to measure the angle between the line of sight to a star and the line of sight to a horizon or to a landmark. Both the sextant and telescope can be used to measure the line of sight to a landmark or to the lunar module (LM). The CSM is also equipped with a VHF ranging device to measure the distance to the LM. The CSM also has a transponder to send and receive communications, data, and tracking signals to and from the earth based tracking stations of the Manned Space Flight Network (MSFN). This transponder can be connected to a high gain antenna array or to any one of four omnidirectional antennas. The high gain antenna can be turned so that it can be used for a certain range of spacecraft attitudes. The omnidirectional antennas cover a much broader range of spacecraft attitudes, but their use results in a weaker signal.

The LM is equipped with a telescope, a landing radar, and a rendezvous radar. The LM also has antennas and a transponder for communication with the MSFN tracking stations and with the CSM. The landing radar measures the velocity of the LM relative to the lunar surface as it approaches the landing site and also the slant range from the vehicle to the lunar surface. (When the landing radar is pointed straight down, slant range coincides with altitude.) The rendezvous radar tracks the CSM while the LM is ascending from the lunar surface. It measures range and range rate (closing rate) between the two vehicles as well as two angles which define the line of sight from the LM to the CSM. These data enable the LM computer to determine its orbit and to compute the maneuvers that will place the LM on a rendezvous course.

These various navigation systems are often used simultaneously. For example, during the LM ascent, all three major navigation systems (MSFN, LM rendezvous radar, CSM optics and VHF ranging device) are in operation. Each system determines an independent solution for each maneuver. These solutions are compared with each other and with premission predictions to insure that the correct maneuver is performed.

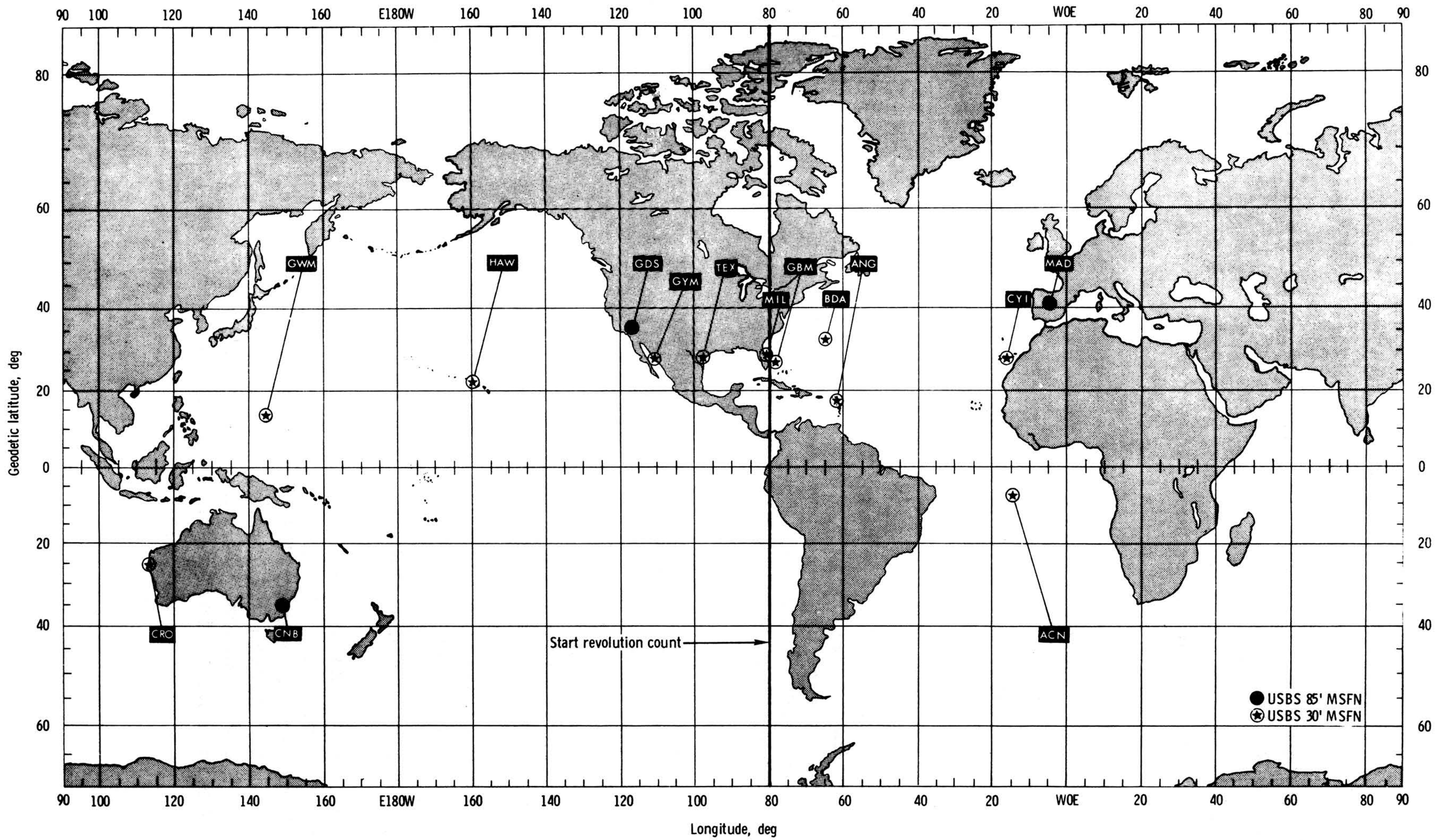


Figure 1.- Manned Space Flight Network (MSFN).

The measurements to determine the trajectory differ from time to time depending on the phase of the mission. During the time the vehicle is on its way to the moon or returning to earth, the primary measurements are obtained by the Manned Space Flight Network (MSFN), which consists of tracking stations located around the earth. The approximate locations of these tracking stations are shown on the world map presented in figure 1. These stations are separated in longitude such that at least three stations can view the vehicle simultaneously except when the vehicle is very near the earth or is behind the moon.

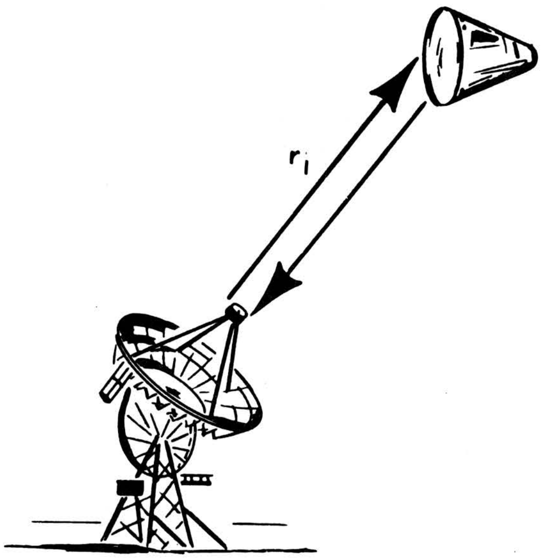
Primarily, the tracking stations measure the speed of the vehicle as it approaches or recedes from the station. These velocity data are known as two-way or three-way Doppler depending on which station receives the signal. Only one station sends the signal to the spacecraft, but any station within view of the spacecraft can receive the signal. The data at the station that both sends and receives the signal are called two-way Doppler, and the data at the stations that only receive the signal are called three-way Doppler. In figure 2, these two data types are depicted by showing the path of the signal for the two tracking modes.

The tracking data are sent to MSC where they are resolved into position and velocity of the spacecraft by use of a computerized orbit determination program. The position and velocity, which define the spacecraft trajectory, are then returned to the spacecraft by way of a tracking station to complete the cycle. The spacecraft can then maneuver into a new trajectory if such is required. This cycle of data processing continues throughout the flight.

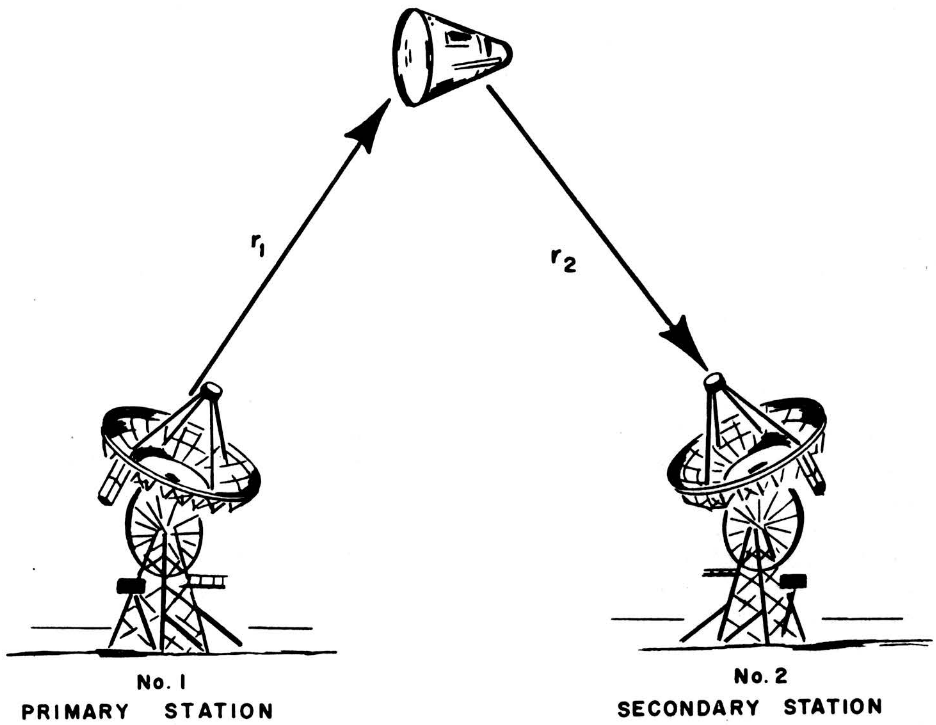
Navigation plans develop from premission studies performed with computer programs that model navigation data and orbit determination techniques. These premission studies are designed to evaluate navigation capabilities for a planned mission and to discover problem areas if they exist.

Translunar navigation is required primarily for the support of midcourse correction maneuvers and a deboost into lunar parking orbit. The primary target parameter for the midcourse corrections is the radius of closest approach to the moon or perilune radius. This radius can be determined to within an accuracy of 5 miles for each of the midcourse corrections and can be determined to within 2 miles by the time of the deboost maneuver. The lunar deboost maneuver occurs approximately at the time the vehicle reaches the perilune radius and is known as lunar orbit insertion.

The vehicle configuration during translunar flight is the command/service module (CSM) with the lunar module (LM) attached. Navigation and communications are performed using the CSM equipment. If the CSM equipment fails in the translunar phase, the mission will be aborted, and



(a) Two-way Doppler



(b) Three-way Doppler

Figure 2.- Two-way and three-way Doppler.

the LM electronic equipment will be powered up and used for navigation. A second backup to the primary navigation system is the onboard optics which consist of a scanning telescope and a sextant.

The sextant is used to measure angles between a star and the lunar horizon or between a star and the earth horizon. The angle measurements are then used in the onboard computer to determine the trajectory. The probability of having to resort to the onboard optics for translunar navigation is almost nonexistent.

In lunar orbit, the major events that require navigation support are the descent orbit insertion maneuver, the circularization maneuver, the descent and landing of the LM, the rendezvous of the LM with the CSM, and the transearth injection maneuver to start the spacecraft on its return to earth. The navigation techniques for the circularization maneuver and the transearth injection are the same as those used on the translunar coast.

Lunar navigation prior to and during descent and landing of the lunar module poses a particular challenge because of the Apollo mission requirement to land in a precisely specified location. For this it is necessary to have a satisfactory model of the lunar gravitational field to be able to predict the magnitude and direction of the gravitational pull on the spacecraft. Much effort has been expended to infer such a model from tracking data gathered from Lunar Orbiters, Apollo 8, 10, 11, and 12 spacecraft during their passes in front of the moon. From these data, it is clear that the gravity force is often influenced by an uneven distribution of the moon's mass about its center. This mass distribution can be thought of as a homogeneous moon, somewhat distorted from a spherical shape, or as a spherical moon with local mass concentrations (mascons) embedded in its surface; but in fact it is probably caused by a combination of these effects.

The orbital phases thus determined by MSFN navigation based on the best moon model available is further refined by the crew's tracking of prominent lunar landmarks near the landing site with an onboard sextant. In this way, the relative position of orbit and landing site are determined; and, if necessary, orbit corrections are incorporated to insure that reaching the landing site during powered descent is within the propulsive capabilities of the lunar module.

During the rendezvous phase, the rendezvous radar onboard the lunar module is used to provide tracking data to the computer onboard. The navigation of spacecraft in lunar orbit is a relatively new experience, and the techniques for processing the tracking data are frequently updated based on new experiences. The accuracy of navigation in lunar orbit varies from time to time as new techniques are developed.

Transearth navigation consists primarily of trajectory determination for support of midcourse corrections and to define the entry interface for reentry guidance.

The primary target parameter for transearth midcourse corrections is the angle at which the spacecraft intersects the earth atmosphere defined at 400 000 feet above the earth surface. This target angle is known as the reentry flight-path angle and can be determined with MSFN tracking data to an accuracy of 0.5° for all but the last midcourse correction. The last midcourse correction is planned to occur at approximately 3 hours prior to entry into the earth's atmosphere, and the entry flight-path angle can be determined to within 0.01° of the actual angle.

If communications between ground and vehicle are lost during the transearth phase, the backup navigation system may be required. Unlike the translunar phase, the LM communications system is not available during the transearth phase because the LM was left in lunar orbit. Therefore, the onboard sextant is used to obtain angle data between a star and the earth horizon or between a star and the moon horizon. These angle data are used in the onboard orbit determination computer program to determine the trajectory for midcourse corrections.

Considerable effort is expended each mission to prepare crew charts which contain the stars available for transearth navigation. That is, the crew charts include the stars that are bright enough to be seen through the sextant, are close enough to the earth or moon to satisfy the maximum trunnion angle capability of the sextant, and are far enough away from the sun to prevent sun shafting through the sextant. The basic rules for onboard navigation are also provided in these charts so that for any conceivable transearth return the crew will have a transearth navigation capability.

The onboard optics data are not as accurate for orbit determination as are the MSFN tracking data. However, they are accurate enough to provide navigation to an acceptable reentry. The flight-path angle can be determined to within 0.5° by the time of the last midcourse correction.

The accuracies of orbit determination are sufficient for acceptable navigation for all phases of the mission, even when the backup navigation system is employed.

DESCENT, LANDING, AND ASCENT

DESCENT, LANDING, AND ASCENT

Lunar Module Descent and Landing

In preparation for the lunar module (LM) descent to the lunar surface, the LM (fig. 1) is mechanically undocked from the command/service module (CSM) 4.5 hours before landing. This event takes place while both vehicles are in a 59-n. mi. by 8.2-n. mi. orbit.

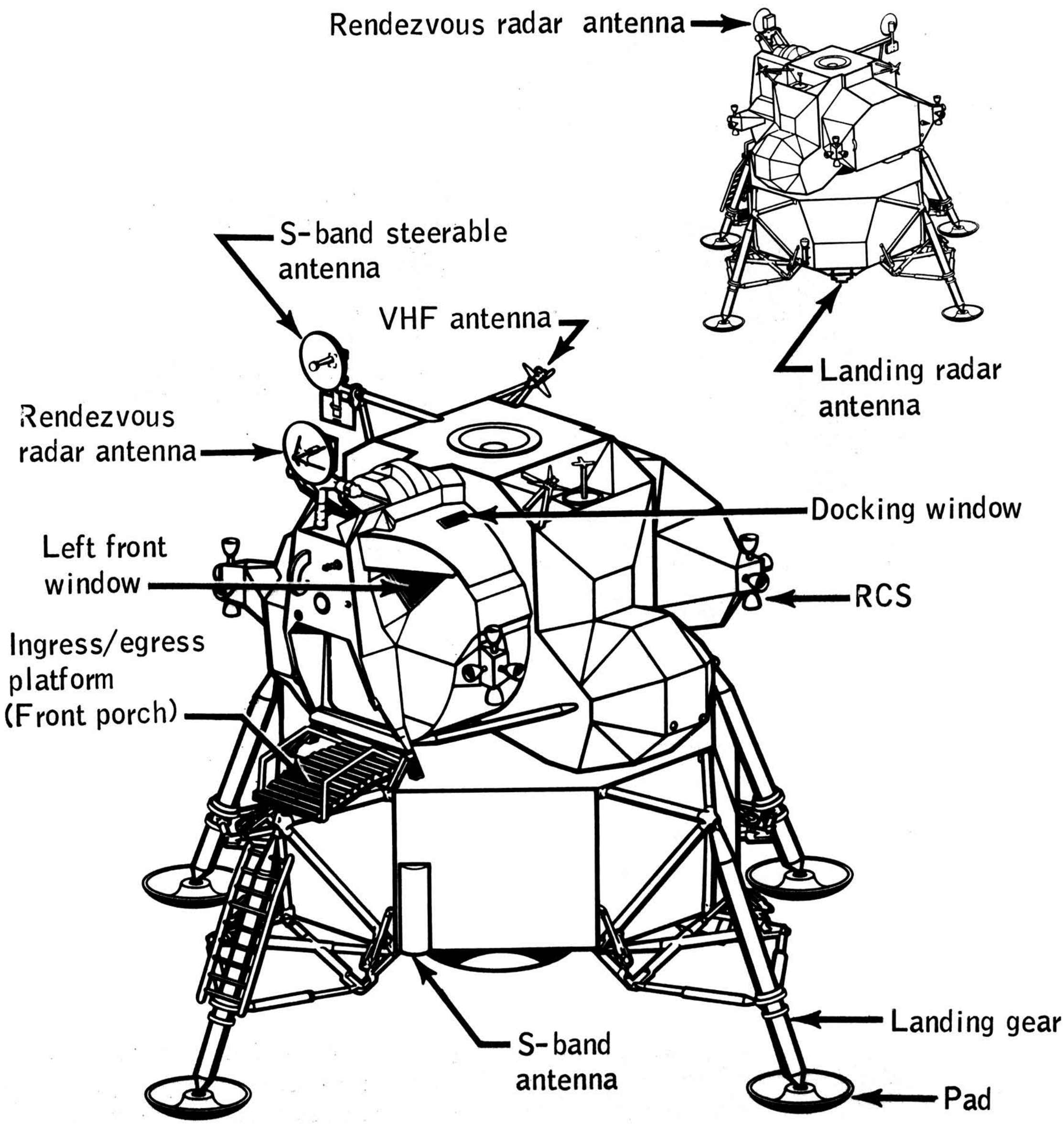


Figure 1.- Lunar module.

Immediately after undocking, the CSM performs a manual maneuver to provide an increase in CSM/LM separation distance.

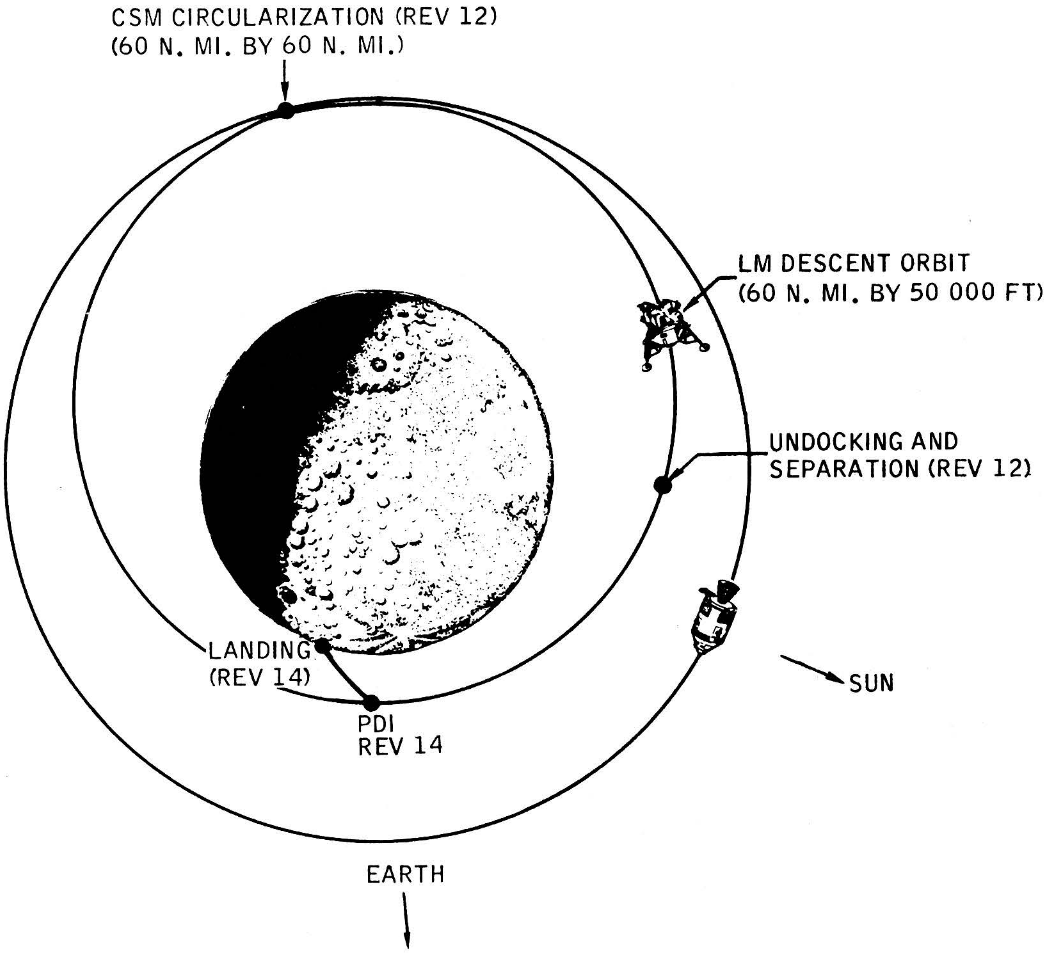


Figure 2.- LM descent.

Approximately 79 minutes after undocking, at the apolune of the LM descent orbit, the CSM performs a circularization maneuver (fig. 2). This maneuver places the CSM in a parking orbit which facilitates rendezvous phasing and landmark tracking.

The powered descent maneuver (fig. 3), which is initiated about 3 hours after the CSM circularization maneuver consists of three operational phases: braking, designed for descent engine performance efficiency; approach, designed for crew visibility; and landing, designed to provide for takeover of manual control to touchdown on the lunar surface. The transition from braking to approach phase is termed high gate (approximately 7500-ft altitude), and the transition from approach to landing phase is termed low gate (approximately 500-ft altitude).

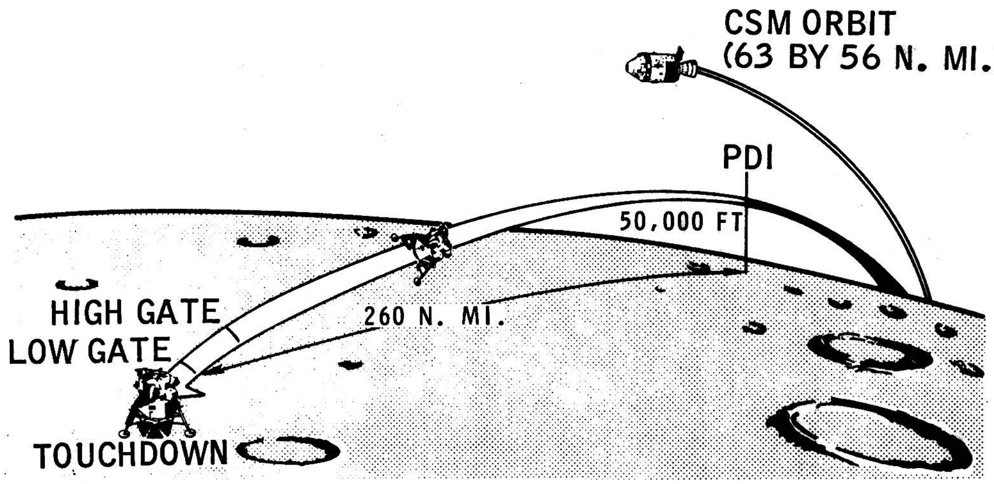
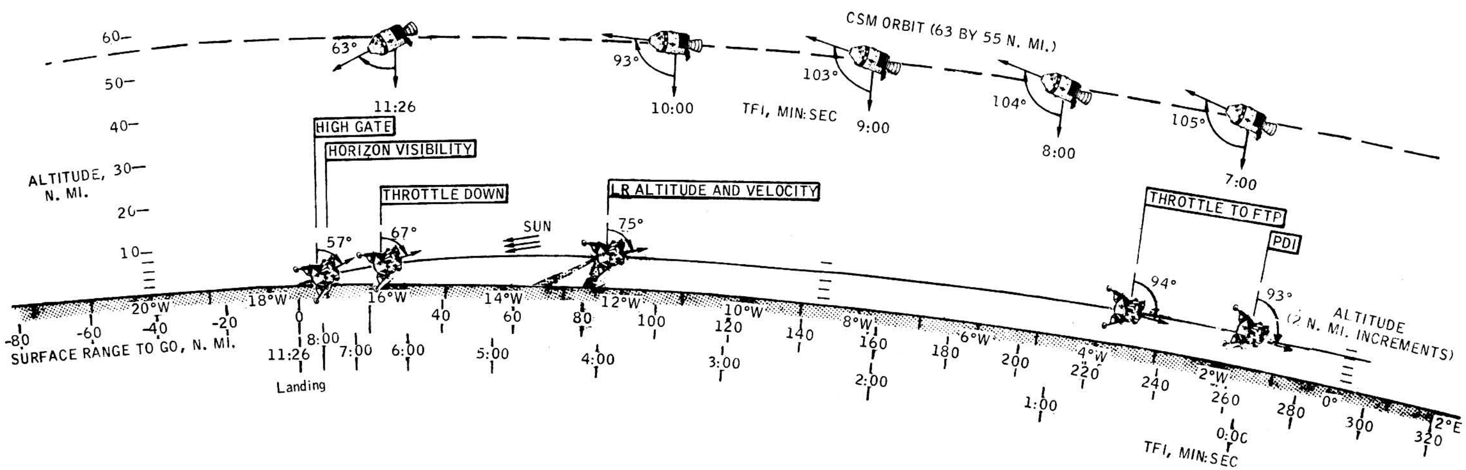


Figure 3.- Operational phases of powered descent.

The purpose of the braking phase is to efficiently reduce the orbital velocity of the LM. To accomplish that purpose, certain design constraints must be respected. The orientation or attitude of the vehicle must be essentially retrograde to allow the descent engine thrust vector to oppose the LM velocity vector. Maximum thrust is required from the engine throughout most of the phase; however, the engine is throttled during the latter part of the phase to allow guidance control for target achievement. The engine ignition time and range from the landing site as well as the high gate altitude are a few of the critical parameters that require analysis in the target selection and trajectory design of the braking phase. The final descent phase of the LM from powered descent initiation is shown in figure 4.



SUMMARY					
EVENT	TFI, MIN:SEC	V_i, FPS	$\dot{H}$, FPS	H, FT	ΔV, FPS
POWERED DESCENT INITIATION	0:00	5558	-3	51 488	0
THROTTLE TO MAXIMUM THRUST	0:26	5526	-3	51 410	33
LANDING RADAR ALTITUDE AND VELOCITY UPDATE	4:00	3309	-84	40 071	2289
THROTTLE RECOVERY	6:32	1422	-84	23 720	4291
HORIZON VISIBILITY	8:10	641	-184	11 411	5193
HIGH GATE	8:30	478	-182	7 464	5396
LOW GATE	10:10	57 (71)*	-13	498	6201
LANDING	11:25	-15 (0)*	-3	5	6620

* (HORIZONTAL VELOCITY RELATIVE TO SURFACE)

Figure 4.- Powered descent profile.

Two major events during the braking phase are acquisition of landing radar tracking of the lunar surface and throttling of the descent propulsion system (DPS). The first of these events normally occurs at an altitude of about 40 000 feet. Landing radar tracking is necessary to update the guidance system estimate of altitude and velocity relative to the lunar surface. The second event, DPS throttle recovery, normally occurs 2 minutes prior to high gate. Throttling is necessary to accurately achieve high gate target conditions.

The approach phase, which begins at high gate, permits visual monitoring by the pilot of the approach to the lunar surface. That is, the guidance system is supplied with targets which yield vehicle attitudes that permit crew visibility of the landing area through the forward LM windows throughout this phase. This requirement for crew visibility is the primary constraint on the trajectory design and target selection for the approach phase.

The landing phase begins at low gate and is designed to provide for continued visual assessment of the landing site and compatibility for pilot takeover from the automatic control. There is no change in guidance law or targets at this point because the approach phase targets are selected so that these additional constraints are satisfied. If the crewman should choose not to assume manual control, he can continue with the automatic guidance to touchdown. In this event, a vertical descent is initiated at an altitude of 100 feet. A velocity error nulling guidance is used to provide a 3-fps rate of descent throughout the vertical descent.

The powered descent maneuver traverses an arc of about 16° (260 n. mi.) across the lunar surface and is the most critical segment of the lunar landing mission.

After touchdown on the lunar surface, the crew will perform a complete systems check and prepare for a first-opportunity lift-off in the event any critical system shows a degradation beyond safe limits. If such an early lift-off is not required, the crew will initiate various surface experiments and collect lunar samples. After a stay of more than 1 day (~33.5 hr) on the lunar surface, the crew will make the necessary preparations for LM ascent.

Lunar Module Ascent

The lunar module ascent stage is shown in figure 5.

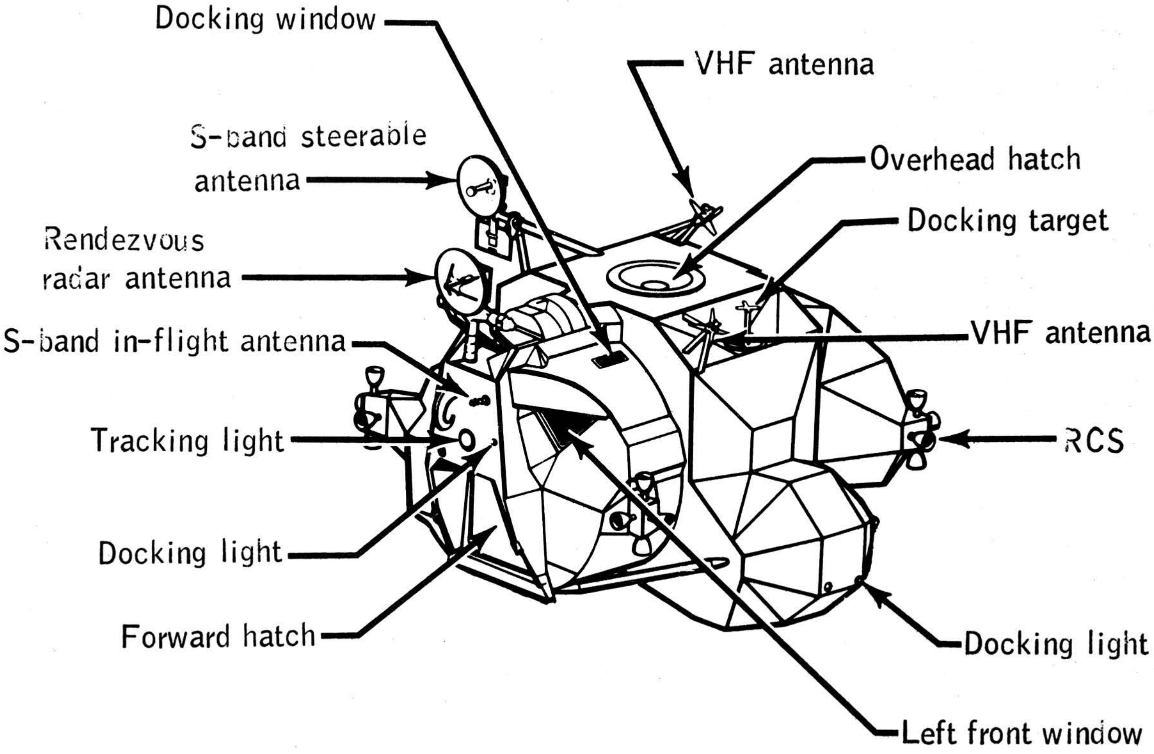


Figure 5.- Ascent stage of lunar module.

The precise time of LM lift-off will be determined primarily by the LM/CSM phasing requirements for rendezvous. The powered ascent (fig. 6) is divided into two operational phases: vertical rise and orbital insertion.

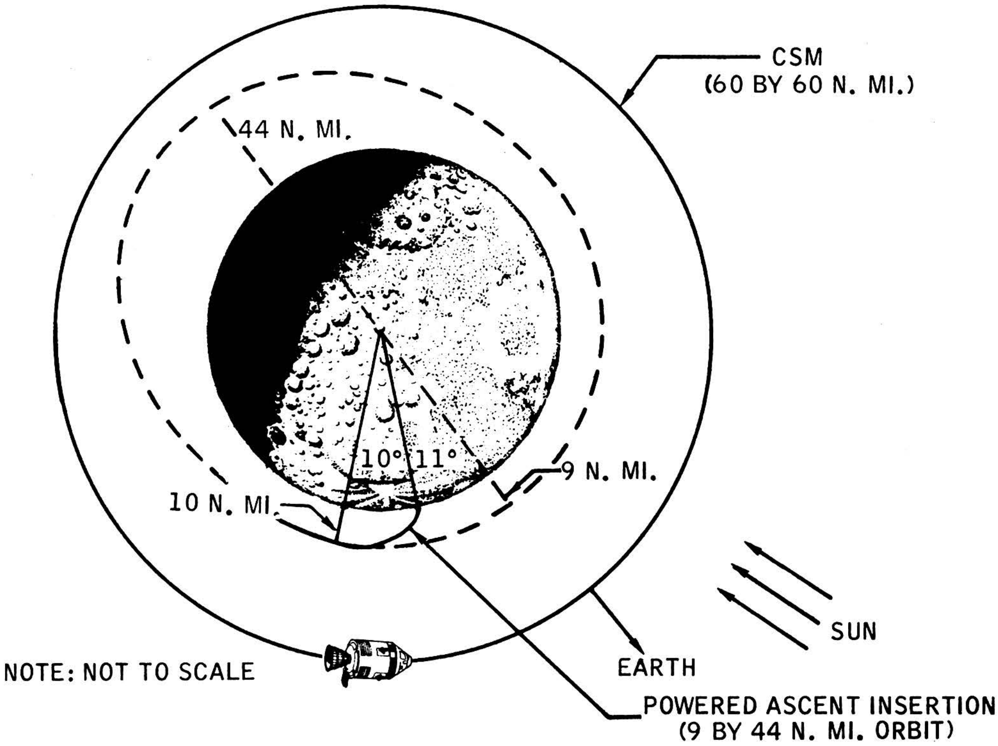


Figure 6.- LM ascent.

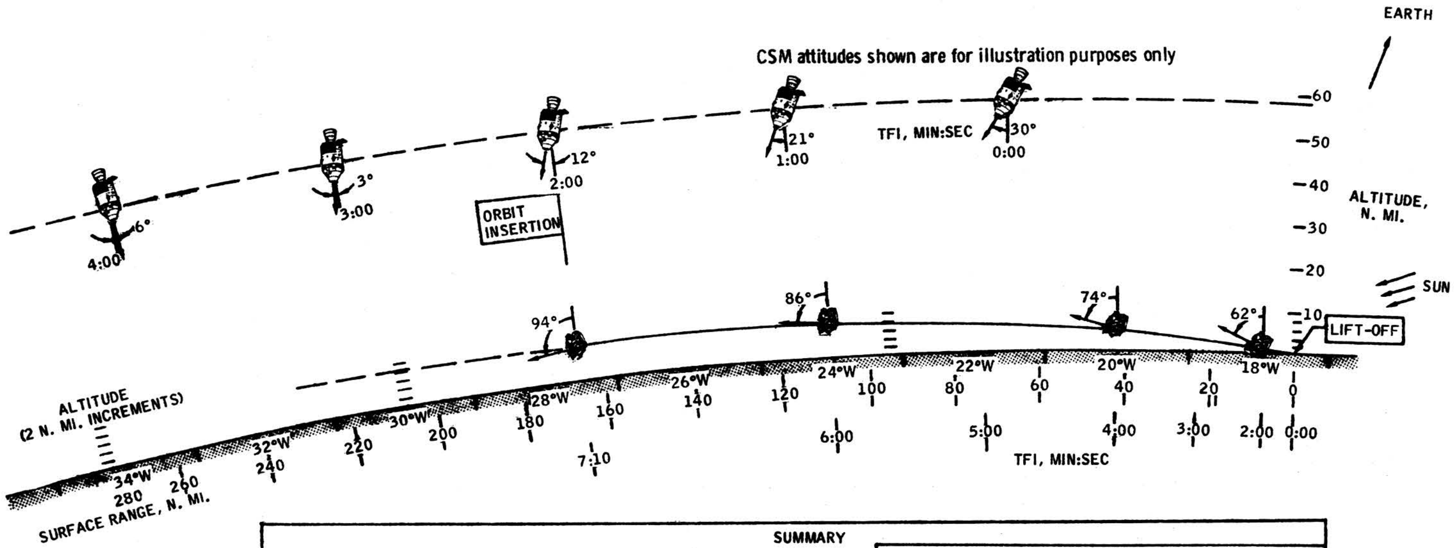
The vertical rise phase is required for the ascent stage to achieve terrain clearance. Vertical rise is maintained until the radial rate exceeds 40 fps. However, the vehicle does not pitch over and begin orbital insertion steering until a radial rate of approximately 50 fps is achieved because of digital autopilot (DAP) steering lag. The vertical rise phase terminates at some 10 seconds after lift-off. During this 10-second period, the LM is rotated to the desired azimuth which is normally in the CSM orbital plane.

At orbital insertion phase initiation, the rendezvous radar antenna locks on to the transponder of the CSM allowing the crew to monitor the CSM/LM relative range and range rate for verification of proper execution of the ascent maneuver. To maintain S-band communications near the end of the orbital insertion phase, a 30° pilot yaw maneuver is initiated at 48 seconds after lift-off. During the 7-minute 10-second burn, the spacecraft covers a range of 165 n. mi. or approximately 10° of arc across the lunar surface. The CSM is leading the LM throughout the burn and at insertion the CSM is almost in front of the LM. A summary chart of the LM powered ascent is shown in figure 7.

The orbital insertion phase is designed to achieve a specified orbital altitude and velocity consistent with subsequent rendezvous requirements, and uses a guidance scheme unique to LM ascent. Yaw steering is used during this phase, if required, to maneuver the vehicle into the CSM orbital plane or into an orbit coplanar with the CSM orbit. The nominal ascent is designed, however, so that no yaw steering is required.

The ascent engine is not gimbaled like the descent engine. That is, it is mounted in a fixed position relative to the LM body. The reaction control system (RCS) jets are automatically pulsed throughout the powered ascent to control the vehicle's attitude. The ascent engine is mounted in such a way as to minimize the RCS activity during the ascent burn.

The primary guidance and control system is backed up by an abort guidance system which is capable of steering the LM to the desired target conditions for the powered ascent. If the primary system should fail, the backup system will assume control of the LM until rendezvous is completed.



SUMMARY								
					LM TO CSM			
EVENT	TFI, MIN:SEC	INERTIAL VELOCITY, FPS	ALTITUDE RATE, FPS	ALTITUDE, FT	RANGE, N. MI.	RANGE RATE, FPS	PHASE ANGLE, DEG	LOOK ANGLE, (LOCAL VERTICAL) DEG
LIFT-OFF	0:00	15	0	0	88	3859	3.9	49.9
END OF VERTICAL RISE	0:10	56	54.1	269	95	4030	4.4	53.6
	2:00	1036	171.5	14 214	169	3843	9.4	75.2
	4:00	2477	184.4	36 567	233	2556	13.4	83.7
	6:00	4278	107.2	54 847	267	795	15.5	87.0
ORBIT INSERTION	7:10	5530.1	35.9	59 957.0	268.9	-448.2	15.6	87.4

h_p = 52 944 FT
h_a = 43.8 N. MI.
η = 21.3°
γ = .37°
ΔV = 6042.8 FPS

Figure 7.- Orbit insertion phase.

LUNAR LANDING MISSION RENDEZVOUS

LUNAR LANDING MISSION RENDEZVOUS

The basic rendezvous maneuver strategy chosen for the Apollo program is the concentric orbits rendezvous scheme which was also used in the Gemini program. This rendezvous scheme consists of three distinct maneuver phases (assuming that the target or passive vehicle, in this case the CSM, is already in orbit). These phases are as follows: (1) launch phase - lift-off of the LM from the moon at the proper time and subsequent insertion into a selected orbit, (2) catchup phase - orbit maneuvers which set up desired conditions for initiation of the final (terminal) phase, and (3) terminal phase - maneuvers which achieve the actual intercept (rendezvous) and docking. The maneuvers in each phase and the time at which they are initiated are all interrelated because they, in turn, control the relative range (distance) and range rate, or the so-called phase angle and closing rate, between the two spacecraft.

The following paragraphs will briefly describe the basic trajectory and operational considerations of the maneuvers included in the three phases. Illustrations are also presented which describe the geometry of the orbits and the maneuver sequences.

After LM ascent from the lunar surface, the LM must perform a series of maneuvers to catch up to and intercept the CSM for rendezvous and docking. Because the LM will be inserted into a lower orbit, it will travel around the moon faster than the CSM. Like two racers going around a track, the one on the inside track (the LM) has less distance to travel and, consequently, would go ahead of the other (the CSM) unless the starting positions are staggered to let the outside racer (the CSM) have a head start. This head start (initial phase angle) is accomplished in the launch phase by choosing the proper lift-off time for the LM such that when it enters its orbit it trails the CSM by a desired distance (about 305 statute miles) determined by the catchup rate between the two vehicles and the time (rendezvous position) at which it is desirable for the LM to finally catch the CSM. A series of LM maneuvers are made during the catchup phase to alter the catchup rate and to raise the LM altitude so that it finally matches that of the CSM. These intermediate maneuvers permit the LM to compensate for propulsion and timing errors that could occur during the powered launch. If an attempt were made to place the LM on a direct intercept course from insertion and such errors were to occur, the rendezvous could become very expensive in propellant consumption, and the final approach of the LM to the CSM could become very difficult in terms of velocity match and visual maneuvering.

The first orbital LM phasing, or catchup, maneuver normally will occur about 50 minutes after LM insertion, when the LM has reached the high point of its initial orbit (apolune), about 52 miles above the moon. The catchup phase maneuvers are scheduled at the high and low points (apses) of the orbit to minimize the velocity (propellant) requirements as radial velocity corrections are minimized at these points. This phasing maneuver adjusts the catchup rate by raising the LM's low point (perilune) to an altitude (normally also 52 miles) at which a second maneuver will be performed that will make the LM orbit essentially concentric with the 17-miles-higher CSM orbit (69 miles circular). Thus, after the second maneuver, the altitude difference (17 miles) between the two vehicles will remain constant as they travel around the moon until the proper time for the LM intercept maneuver. Of course, because the LM is still in the smaller (lower) orbit, it is traveling faster and is gaining on the CSM; for example, at the first maneuver point, the LM is about 167 miles behind, and at the second maneuver (58 minutes later), it trails by only 86 miles. When the LM has caught up to a trailing distance of only 33 miles, it will be at the proper position to execute a third maneuver which will raise its apolune to a height equal to that of the CSM orbit, 69 miles. This terminal phase maneuver further reduces the catchup rate; however, the rate is such that the LM will close the final 33 miles in exactly the time (about 43 min) it takes to reach the CSM's altitude and thus will intercept the target. At intercept, a series of LM maneuvers must be made to exactly match the velocity of the CSM. These are termed braking maneuvers because relative to the CSM the LM is braking, or slowing down, its approach for final rendezvous and docking. Relative to the LM orbit around the moon, however, the LM actually must add energy (velocity) to raise its perilune, which was 17 miles below that of the CSM; thus, the braking thrust is applied essentially in the direction of motion (east to west around the moon). The final rendezvous and docking of the LM with the CSM occurs about 3.5 hours after the LM has lifted off from the surface. A more detailed explanation of the rendezvous sequence described above will now be given to better illustrate the procedures, maneuvers, and time required to accomplish the essential task of returning the two lunar explorers to the vehicle which must take them back to earth.

LAUNCH PHASE - LM LIFT-OFF AND ORBIT INSERTION

The lift-off time is chosen primarily to satisfy a particular phase angle, or relative range, requirement between the LM and CSM at LM insertion. The initial relative range desired depends on the nominal sequence of maneuvers and the catchup rate planned in the concentric sequence after LM orbit insertion. Specifically, the lift-off time of the

LM is chosen such that after the 7.5-minute powered ascent the LM will enter its initial 60 000-foot by 52-mile orbit trailing the CSM by about 305 miles, as shown in figure 1.

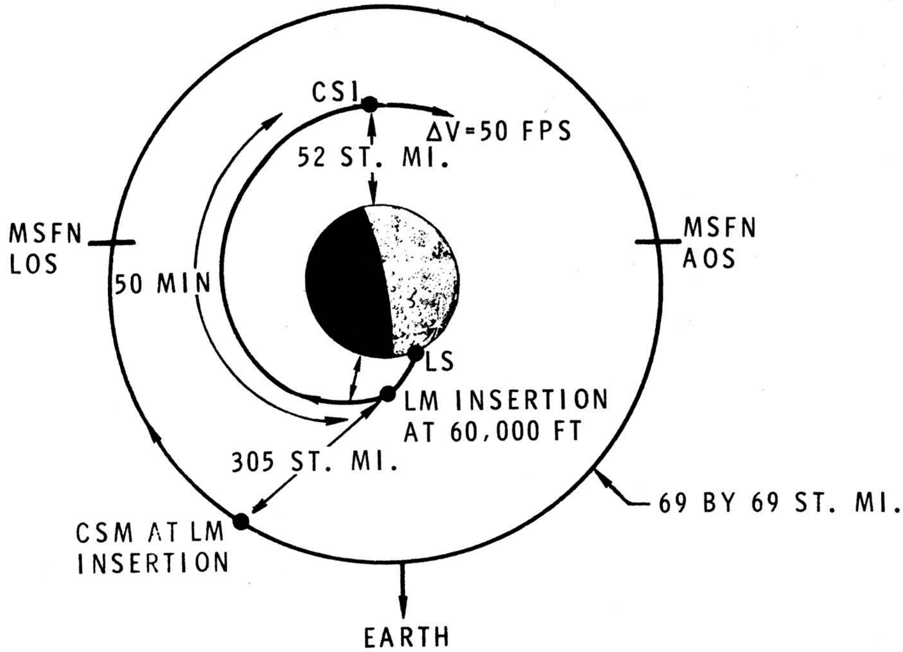


Figure 1.- LM insertion to concentric sequence initiation (CSI).

This lift-off time (calculated by ground control) occurs about 70 seconds after the CSM passes over the landing site. The insertion orbit of 60 000 feet by 52 miles is dictated primarily by the LM ascent engine propellant available and by the requirement to insert the LM high enough to avoid any mountains; however, the 52-mile apolune is chosen to fit desirable rendezvous characteristics. Another factor, the plane orientation of the two orbits, is satisfied by a CSM plane change prior to LM lift-off so that the two vehicles are in the same plane for the rendezvous sequence.

After insertion, the onboard navigation procedures are initiated on both the LM and the CSM to compute and prepare for the first maneuver (phasing) 50 minutes later. The LM acquires rendezvous radar lock-on to the CSM and takes in data to determine both orbits and the required phase maneuver which, along with the concentric maneuver, establishes the desired terminal phase conditions 1.5 hours later. The CSM, likewise, tracks the LM both optically (with the sextant) and with VHF ranging to determine a backup solution for the rendezvous sequence in case the LM could not maneuver and a rescue were necessary.

CATCH PHASE - ORBIT MANEUVERS PRECEDING TERMINAL PHASE

The purpose of the orbit maneuvers between orbit insertion and the terminal phase is to place the LM into a nominal 52-mile circular orbit which is concentric with the 69-mile circular CSM orbit. This nominal orbit provides the proper catchup between the CSM and LM so that the terminal phase initiation (TPI) maneuver is performed at a prescribed position in orbit with respect to darkness.

Phasing Maneuver (CSI)

As the LM reaches its apolune about 58 minutes after lift-off, it executes the proper phasing maneuver to adjust the catchup rate so that the LM will arrive at a trailing distance of about 33 miles as the vehicles pass the midpoint of the moon's shadow. Normally, the phasing requires an addition (ΔV) of about 50 feet per second (34 mph) to the LM's velocity, which raises the 60 000-foot perilune to about 52 miles. If the lift-off were slightly early or late or if the initial insertion orbit were not correct, the phasing maneuver ΔV would vary to account for such a nonnominal catchup rate. The phasing maneuver takes into consideration the fact that one-half orbit (58 min) later the concentric maneuver will be performed to make the LM orbit remain at a constant differential height below the CSM's orbit; thus, the phasing maneuver begins the concentric sequence and is generally called the concentric sequence initiation or CSI for technical identification. The concentric maneuver is termed the constant differential height maneuver, or CDH. The resultant orbital geometry after CSI and CDH is shown in figure 2.

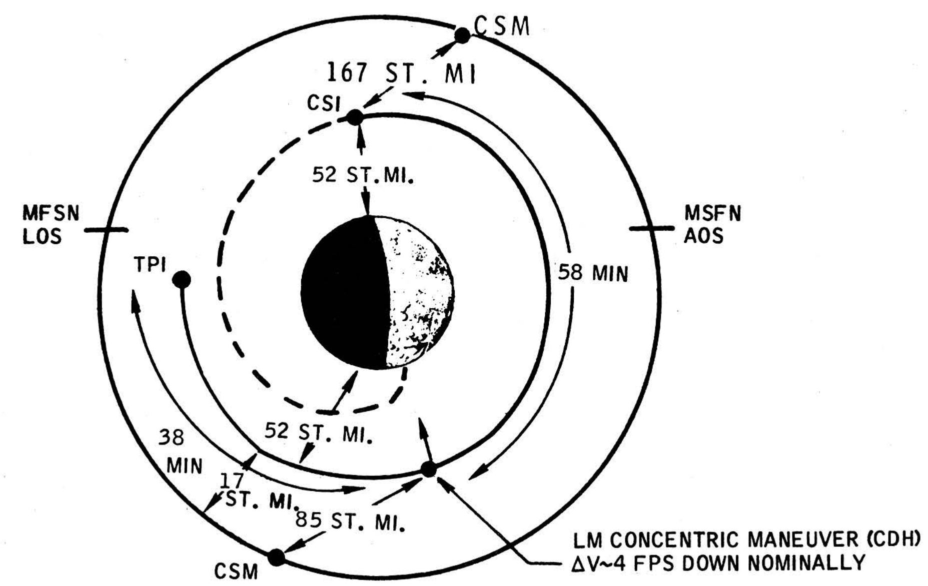


Figure 2.- Concentric sequence to terminal phase.

Constant Differential Height Maneuver (CDH)

About 2 hours after lift-off, the LM should be trailing by about 86 miles if the CSI maneuver has been performed correctly. At this point, the LM orbit is made almost circular by the CDH maneuver such that the catchup rate (about 1.4 miles per minute) is constant around the orbit. This characteristic is desirable from an onboard tracking viewpoint, especially if backup techniques (such as visual sightings and use of simple charts) are required to compute the required intercept maneuver. In the normal case, the CSI maneuver has already circularized the LM orbit about 17 miles below that of the CSM; thus, CDH becomes almost zero. However, if the CSI were varied to adjust for off-nominal phasing, then the CDH could become a sizable maneuver because it would circularize the LM orbit at whatever CDH altitude (not necessarily 52 miles) had resulted from CSI.

TERMINAL PHASE MANEUVERS

The terminal phase, as illustrated in figure 3, consists of the terminal phase initiation maneuver to begin intercept; two small corrections to the intercept trajectory, if necessary; and the final braking maneuver to match velocity at target approach.

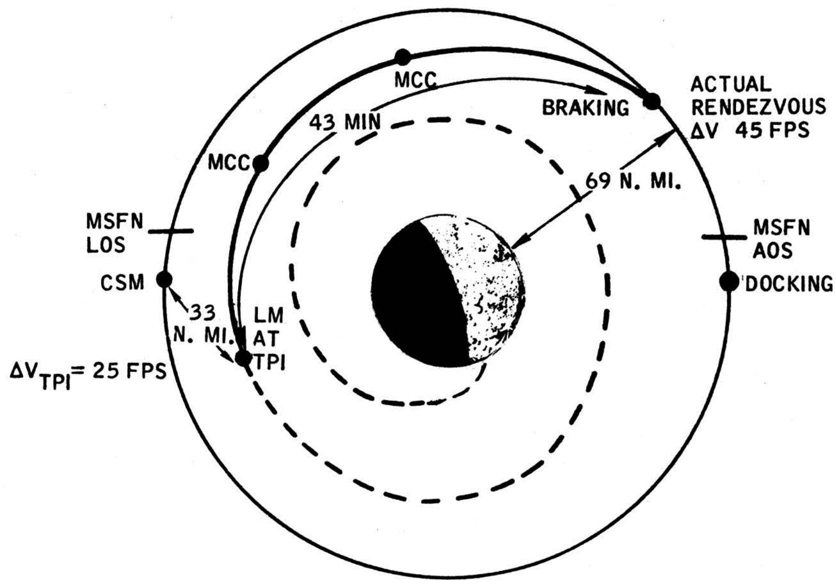


Figure 3.- Terminal phase.

Terminal Phase Initiation (TPI) Maneuver

The third LM maneuver is performed about 38 minutes after CDH when the LM reaches a trailing displacement of about 33 miles behind and 17 miles below the CSM. This particular position has been chosen so that when the maneuver has been executed to start the LM up toward the CSM, the catchup rate becomes about 0.8 mile per minute, and during the 43 minutes of climb to the CSM altitude the range is decreased to zero; that is, the LM arrives at almost 69 miles altitude just as the CSM passes by. The period from the third maneuver to intercept is called the terminal phase of the rendezvous sequence; therefore, the third maneuver is known as terminal phase initiation or TPI. The location of TPI relative to the moon's shadow has been chosen to satisfy both pre-TPI optical tracking requirements as well as visual braking requirements. TPI has been located at the midpoint of the 46-minute darkness period of the CSM orbit so that not only does the CSM have adequate time and proper lighting after CDH to track and compute its backup solution for TPI but also the LM approaches the CSM in sunlight for the braking maneuvers. Thus, even if the CSM acquisition light is out, the LM can visually acquire the target in daylight before the distance is too close so that the braking maneuvers can be easily performed. TPI is normally a 25 foot per second (17 mph) maneuver directed essentially at the CSM; this selected characteristic again facilitates backup visual rendezvous procedures. After TPI has been executed, two small corrective maneuvers (denoted as MCC's for midcourse correction) may be performed, if necessary, to place the LM on a more precise intercept trajectory.

Braking and Docking

As the LM intercepts the CSM from below and very slightly ahead for better visual approach, the LM must slow down (brake) its velocity relative to the CSM. Beginning at about a 1-mile range when the closing rate is about 20 mph, a series of ΔV corrections are applied away from and perpendicular to the direction of the CSM. These maneuvers raise the LM's 52-mile perilune and put the LM in the CSM's 69-mile orbit while matching velocities. The total propellant normally expended is equivalent to a ΔV of 45 feet per second (30 mph). As the range decreases to a few feet, the relative closing rate is decreased to almost zero, and the LM maneuvers to a docking position. The rendezvous sequence is completed about 3.5 hours after lift-off with the physical latchup of the two vehicles and the subsequent crew transfer to the CSM. The unmanned LM ascent stage is then jettisoned.

REENTRY

REENTRY

Atmospheric reentry includes the passage of the command module (CM) through the earth's atmosphere and the safe arrival of the CM at a predetermined geographical location. The passage of the CM through the earth's atmosphere has associated with it problems involving crew safety and accurate trajectory control. The first problem (crew safety) can be divided into three areas: (1) heating, (2) excess deceleration (g) forces, and (3) skipping out of the atmosphere. Aerodynamic heating is produced when the CM, moving at supersonic speeds, penetrates the atmosphere. The air cannot move fast enough to make way for the moving body and is therefore compressed, causing the air to be heated up. Enough heat is produced to ionize the air around the CM, thus producing a barrier to radio communication. The problem of heating is solved by the use of a protective covering of ablative material around the CM. The ablative material thus dissipates heat by melting and vaporizing at the surface. The heat is carried away by the loss of the vaporized portion of the heat shield.

The second and third problems concerning crew safety may be best explained by looking at figure 1. This figure depicts the CM penetrating the top of the earth's atmosphere at specified angles to a local

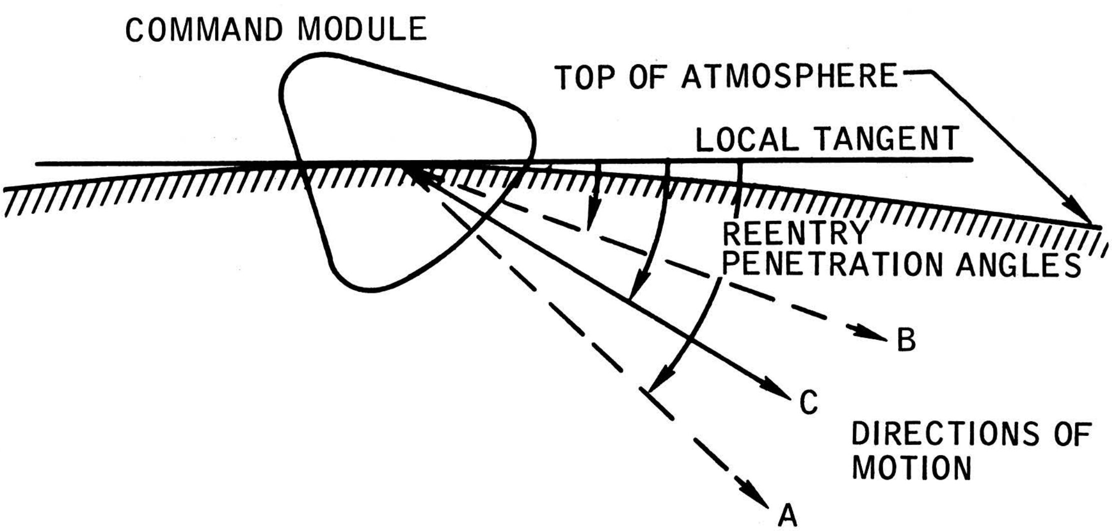


Figure 1.- Command module penetrating the earth's atmosphere.

tangent line. If the CM's direction of motion were along line A the spacecraft would be penetrating the atmosphere at a steep angle and would pull excessive g-forces, thus endangering the crew. If the CM's direction of motion were along line B or shallower, the spacecraft would literally skip out of the atmosphere. This condition is analagous to skipping a stone across a body of water by throwing the stone at a low or shallow angle with respect to the surface of the water. The desired reentry penetration angle lies between lines A and B, such as line C. When the spacecraft enters at this angle, it is subjected to neither excessive g-forces nor skips out of the atmosphere. The reentry penetration angle is controlled along the return to earth leg by making midcourse corrections to the trajectory. The width of the angle between lines A and B is about 2° and is known as the reentry corridor.

The second basic problem, once the spacecraft has safely penetrated the earth's atmosphere, is to guide the spacecraft to a predetermined landing point. This is accomplished by controlling aerodynamic lifting forces on the CM by rolling the spacecraft about an axis parallel to the direction of motion through the use of small reaction control system thrusters. By modulating this lifting force the CM's lateral and horizontal direction of motion is controlled. The direction in which to roll the spacecraft is determined by an onboard computer which in turn automatically maneuvers the spacecraft to that roll attitude. The CM's motion is eventually slowed by the atmospheric drag to a point where parachutes are deployed and the spacecraft floats to a splash landing at the target point.

Figure 2 shows the landing point capability (footprint) of the CM on the surface of the earth. The recovery ship is placed within the cross-hatched area where the CM is targeted to land. The maneuver footprint is approximately 2500 nautical miles long and 300 nautical miles wide.

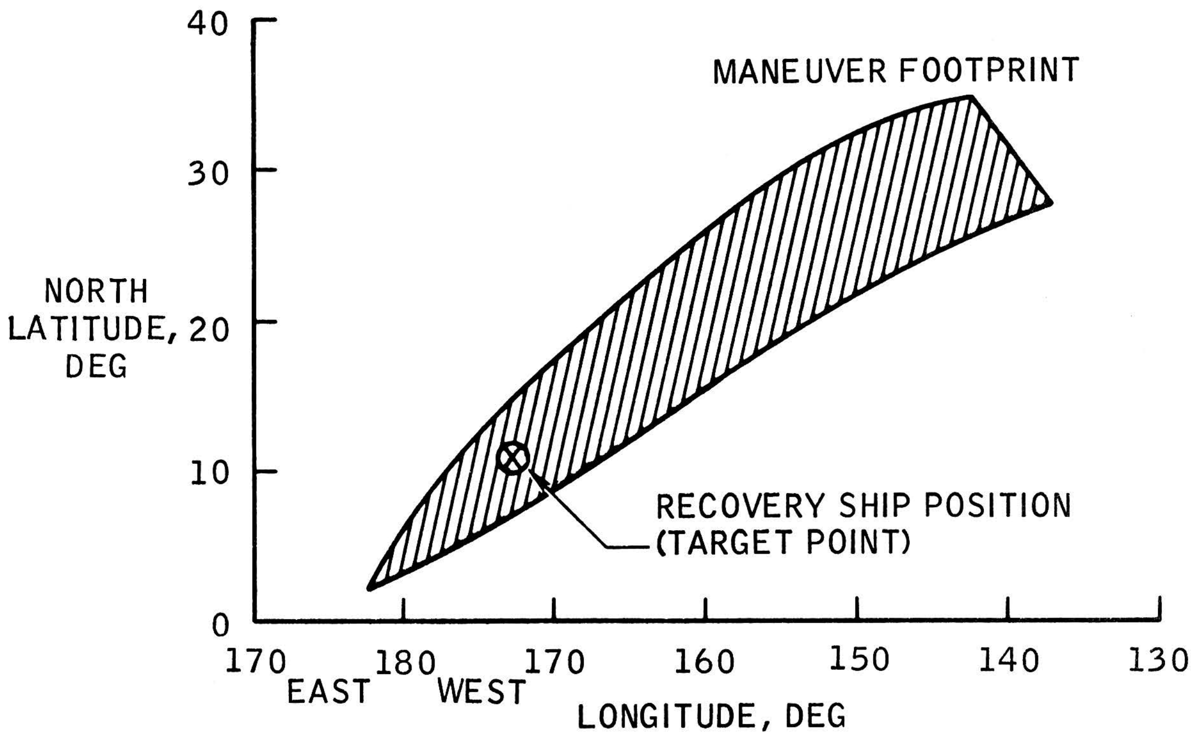


Figure 2.- Command module maneuver capability.

EMERGENCY RETURN TO EARTH

(ABORTS)

INTRODUCTION

If an emergency occurs at some time during the mission which endangers the safety of the crew, it might be necessary to discontinue the flight and to bring the crew back to earth. The decisions to terminate the mission early and to return the crew to earth are collectively called an abort. Before each mission is flown, an operational abort plan is formulated to demonstrate that adequate techniques have been developed and that enough supporting data are available to bring the crew safely back to earth if an abort is required.

There are two general classes of return-to-earth aborts; those from powered flight and those from coasting phases. During powered flight, the situation is changing so quickly that reaction time is critical to crew safety. Aborts from powered flight can be performed automatically, such as during the launch phase, or they can be performed by the crew with onboard data, such as during translunar injection and lunar orbit insertion. Because of the time critical nature of aborts from powered flight, these phases are carefully rehearsed before the mission. During coasting phases, reaction time has a lesser effect on crew safety, and usually abort solutions can be computed during the mission if needed.

Abort techniques are generally designed to return the crew to earth as quickly as possible, and to land near one of the recovery ships. Figure 1 shows a summary of the quickest return times possible during the coasting phases of a lunar mission. The return times depend not only on the distance of the spacecraft from the earth, but also on which engine is used, how much the spacecraft weighs, and how much fuel is available.

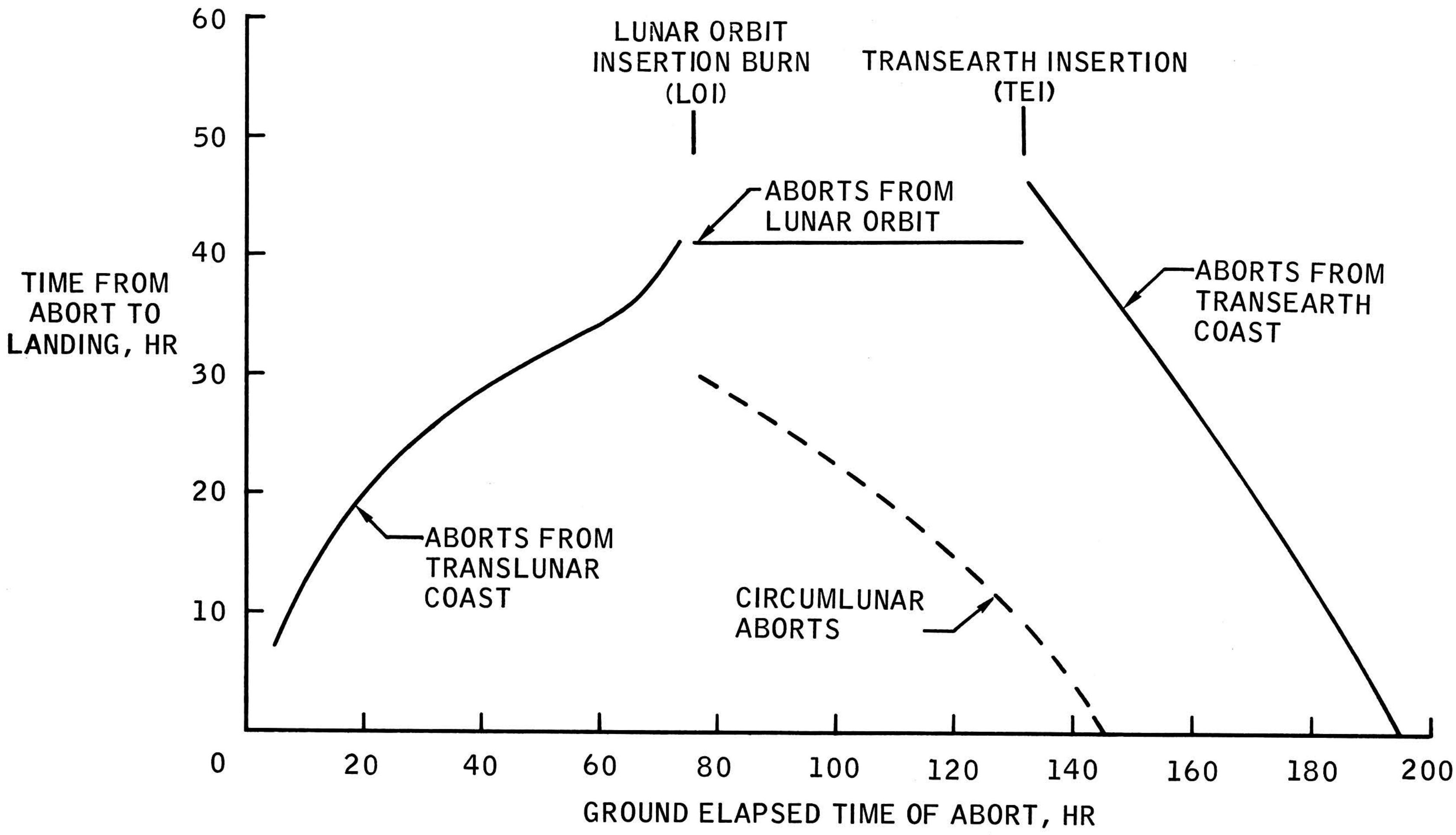


Figure 1.- Typical quickest return times.

The abort plans for three major mission phases (1) launch, (2) translunar coast, and (3) lunar vicinity are described below.

LAUNCH PHASE

The probability of an abort in the first three minutes of the launch phase is more likely than in any other phase of the mission because of possible vehicle breakup and explosion due to aerodynamic forces. Therefore, this region of flight is protected with an automatic abort and emergency detection capability and the launch escape rocket system. For the remainder of the launch (that portion outside the sensible atmosphere) until safe orbital velocity is achieved, booster malfunctions could result in deviations from the planned flight path. To detect such deviations, this phase of flight is closely monitored by both the ground controllers and the crew using displays onboard the spacecraft and in the Houston Control Center. These displays indicate both the current and projected status of the launch trajectory (spacecraft position and speed) for comparison with preflight predictions. Any deviations from the planned launch trajectory can be quickly spotted, and the crew can be advised of impending problems. Also, each launch vehicle and spacecraft system has associated limits for which safe flight is assured. If these limits are violated, an abort would be performed. To avoid a false abort, there must be two related cues that confirm the malfunction prior to abort initiation by the crew.

Once an emergency has occurred, one of four abort modes or techniques must be used for safe return of the crew and spacecraft depending upon the current flight conditions. These abort modes are the following.

a. A mode I abort may occur during that portion of the launch within the atmosphere (0 to 3 minutes). The crew will use the launch escape system to separate the spacecraft from the launch vehicle and land in the Atlantic Ocean continuous recovery area. A mode I abort sequence is shown in figure 2.

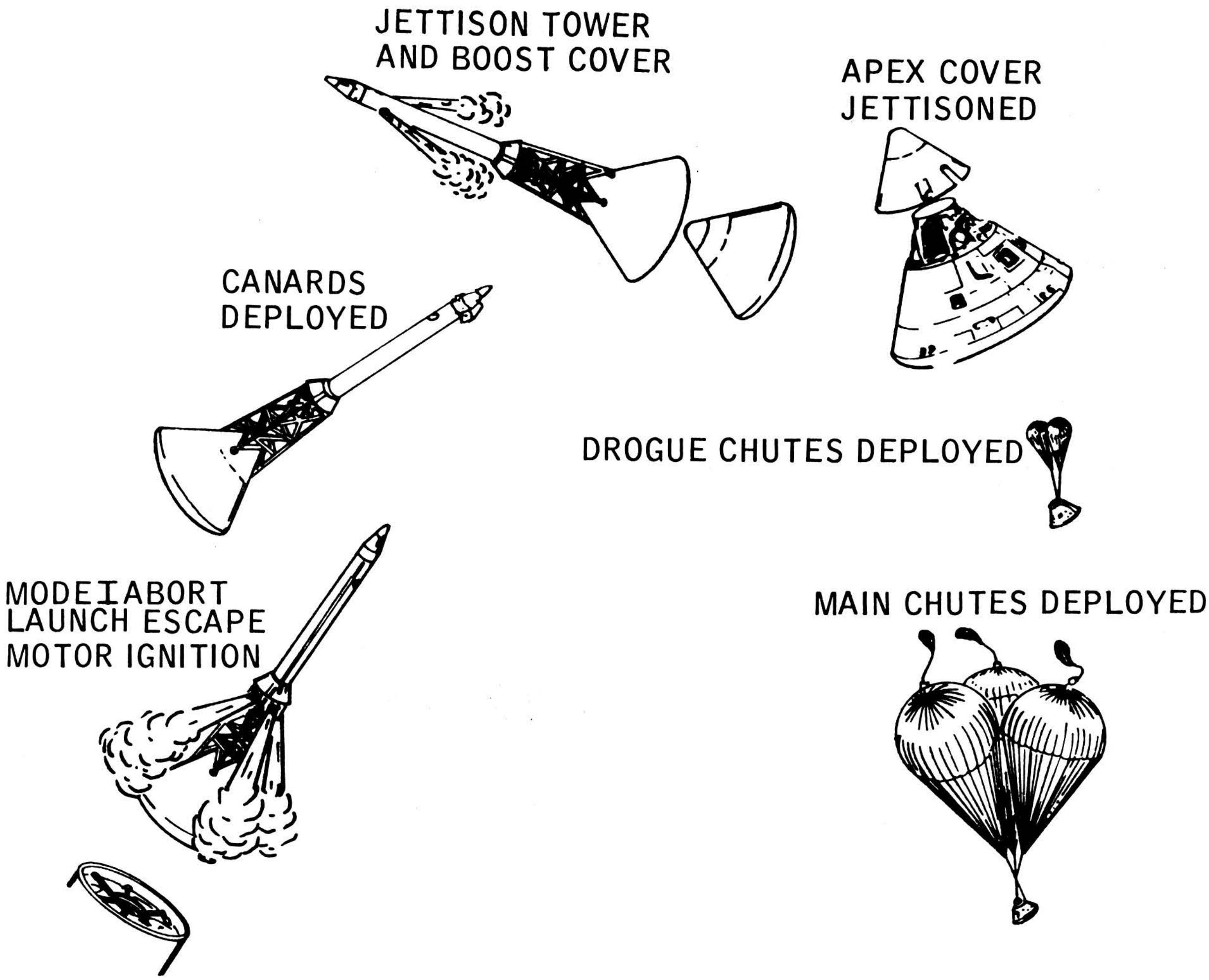


Figure 2.- Launch escape tower abort sequence.

b. A mode II abort may occur during that portion of the launch outside the atmosphere prior to orbital capability (3 to 10 minutes). The crew will separate the spacecraft from the launch vehicle and land in the Atlantic Ocean continuous recovery area.

c. A mode IV abort may occur if the launch vehicle shuts down too soon just prior to orbit insertion (9 to 12 minutes). The crew will separate from the launch vehicle, and, using the spacecraft's rocket engine (SPS), continue to a safe earth orbit insertion.

d. A mode III abort is a secondary technique used only when it is undesirable to continue to orbit. The crew will separate from the launch vehicle, and, using the spacecraft's rocket engine (SPS), retrograde to achieve a landing at the discrete recovery area in the Atlantic Ocean.

These launch abort modes are depicted in figure 3 and are summarized in table I.

TABLE I.- APOLLO LAUNCH ABORT MODE SUMMARY

ABORT MODE	ABORT REQUIREMENT	ABORT PROCEDURE	ABORT RESULT
MODE I	TO PROVIDE RAPID SEPARATION FROM LAUNCH VEHICLE DURING ATMOSPHERIC FLIGHT	AUTOMATIC OR CREW INITIATION OF LAUNCH ESCAPE SYSTEM SPACECRAFT/LAUNCH VEHICLE (SC/LV) SEPARATION ENTRY AND LANDING	SUBORBITAL TRAJECTORY LANDINGS IN THE CONTINUOUS RECOVERY AREA
MODE II	TO PROVIDE RAPID RETURN OF CREW AND SC TO EARTH FOLLOWING AN IN FLIGHT MALFUNCTION	MANUAL SC/LV SEPARATION ENTRY AND LANDING	SUBORBITAL TRAJECTORY LANDINGS IN THE CONTINUOUS RECOVERY AREA
MODE IV	TO ACHIEVE A SAFE ORBIT FOLLOWING AN IN FLIGHT MALFUNCTION	MANUAL SC/LV SEPARATION SC POSIGRADE BURN TO ORBIT SUBSEQUENT DEORBIT ENTRY AND LANDING	CONTINGENCY ORBIT INSERTION WITH LANDINGS AT A PLANNED LANDING AREA AND/OR AN ALTERNATE MISSION
MODE III (BACKUP)	TO PROVIDE RAPID RETURN OF CREW AND SC TO EARTH FOLLOWING AN IN FLIGHT MALFUNCTION	MANUAL SC/LV SEPARATION SC RETROGRADE BURN ENTRY AND LANDING	SUBORBITAL TRAJECTORY LANDINGS AT THE DISCRETE RECOVERY AREA

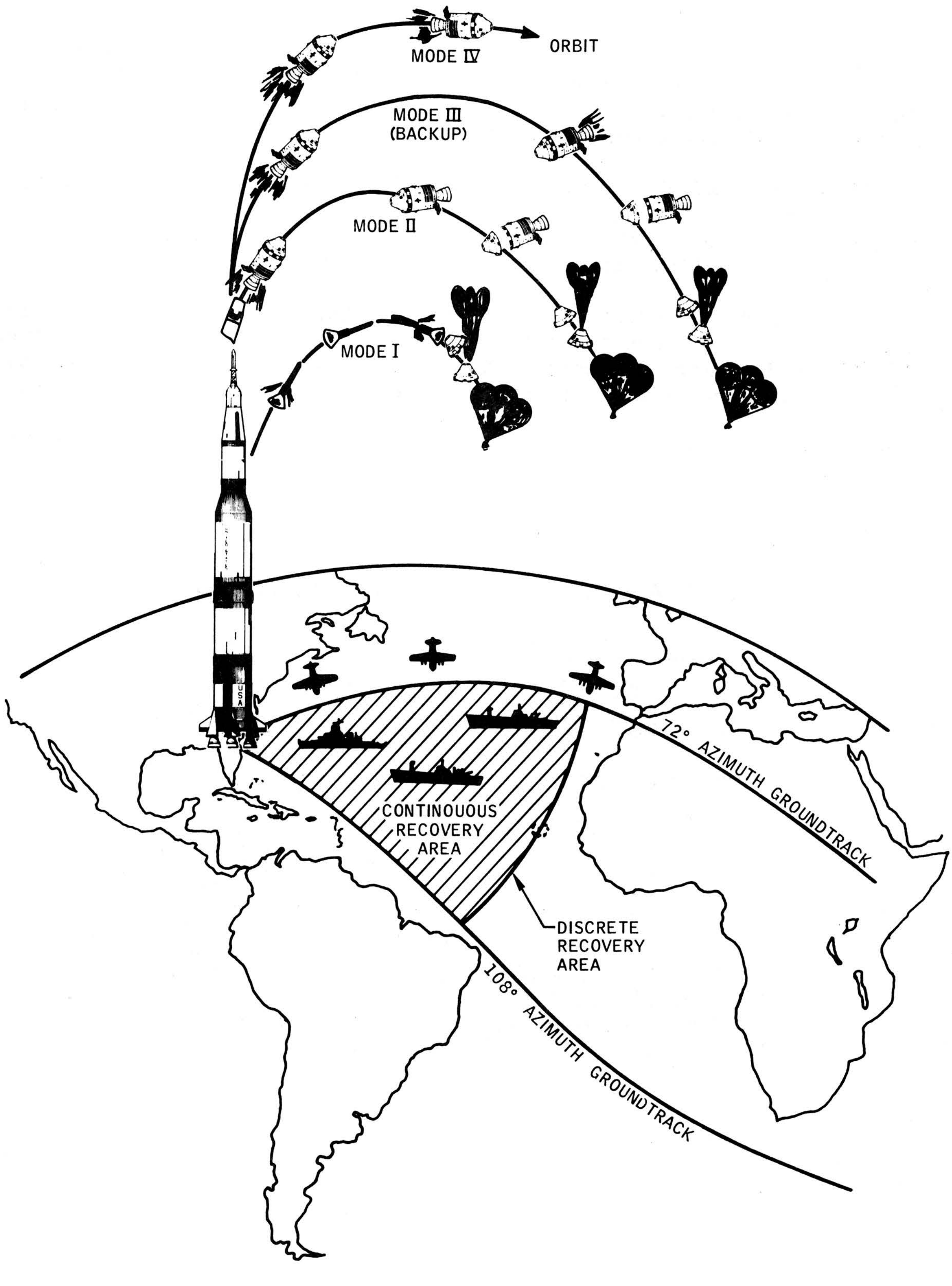


Figure 3.- Apollo launch abort modes and recovery areas.

TRANSLUNAR PHASE ABORTS

The maneuver performed in earth orbit by the third stage of the Saturn rocket places the spacecraft on a translunar trajectory. In lunar mission planning, the term translunar refers to a trajectory outbound from the earth and the term transearth means an inbound trajectory.

If an emergency develops during the translunar phase, the abort problem can be stated simply as how do we change the state of the trajectory from translunar to transearth. The method selected to change the state would depend upon the cause for abort but in general would involve either very small corrective maneuvers to place or maintain the spacecraft on a free-return trajectory around the moon (circumlunar abort) or a very large maneuver which results in a transearth trajectory that does not pass by the moon (direct abort). The circumlunar aborts are similar to midcourse corrections, the difference being aborts are performed to allow earth landing near a recovery ship. These two types of returns to earth are depicted in figure 4.

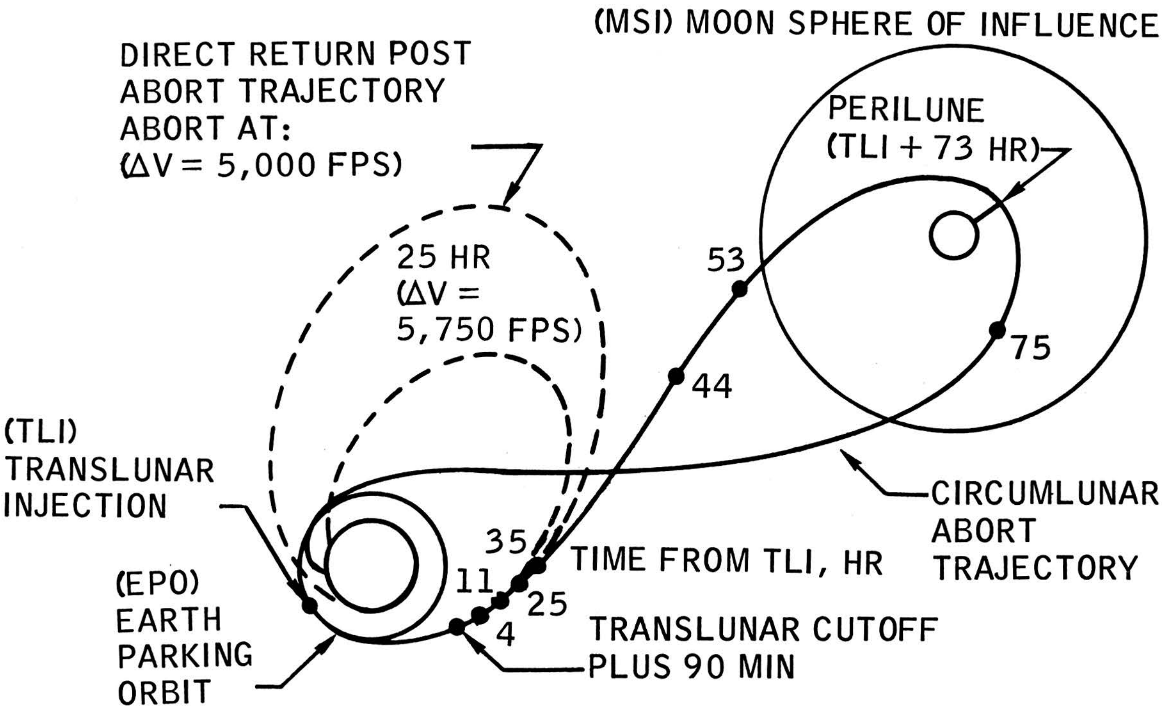


Figure 4.- Abort geometry during translunar coast (TLC).

As an example of the direct abort problem, suppose the spacecraft is on a translunar trajectory 32 hours from translunar injection and an abort decision is made requiring that the spacecraft return to earth before some system completely fails, for example, within 67 hours. Also, suppose it is desired to land at a preselected geographic landing site (site X). The problem simply stated is to determine the maneuver magnitude and direction that will return the spacecraft to landing site X within 67 hours. First consider the maneuver direction: after several hours on translunar coast the spacecraft follows a near rectillinear (straight line) path along a line which passes through the center of the earth and the point at which the spacecraft will enter its first lunar orbit (the earth-moon line). The most economical means of altering the flight path of a spacecraft is to perform a maneuver in the direction opposite or toward the direction of travel. Because the purpose of the direct return abort is to go back to earth without passing the moon, the trajectory needs to be turned away from the moon; therefore, a maneuver is needed in the direction opposite the direction of travel. Hence, the direct return maneuvers are performed in the general direction of the earth (fig. 5). A similar view as seen from the spacecraft is shown in figure 6.

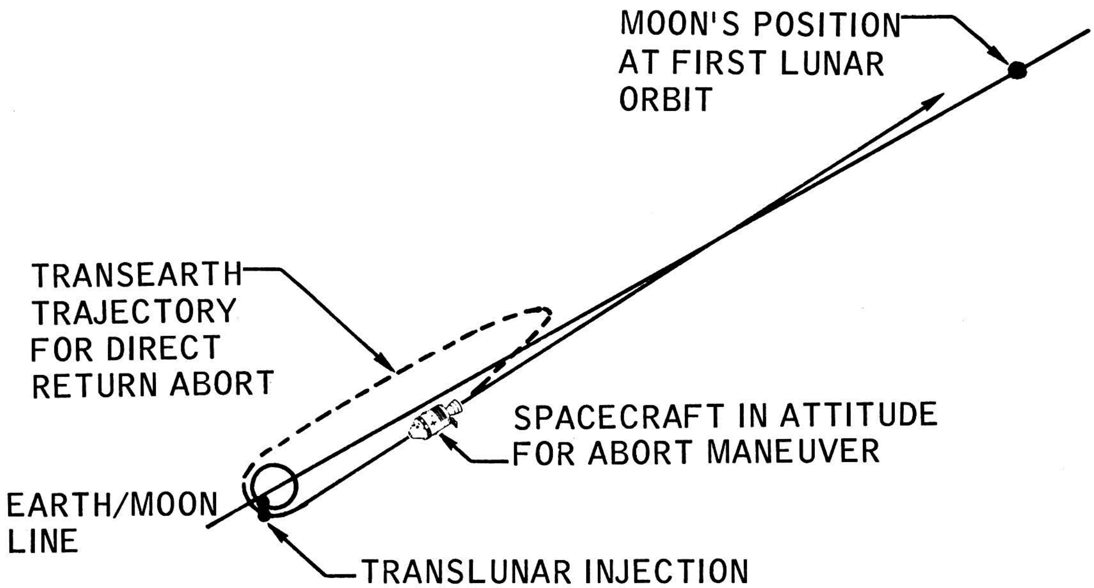


Figure 5.- Spacecraft abort attitude.

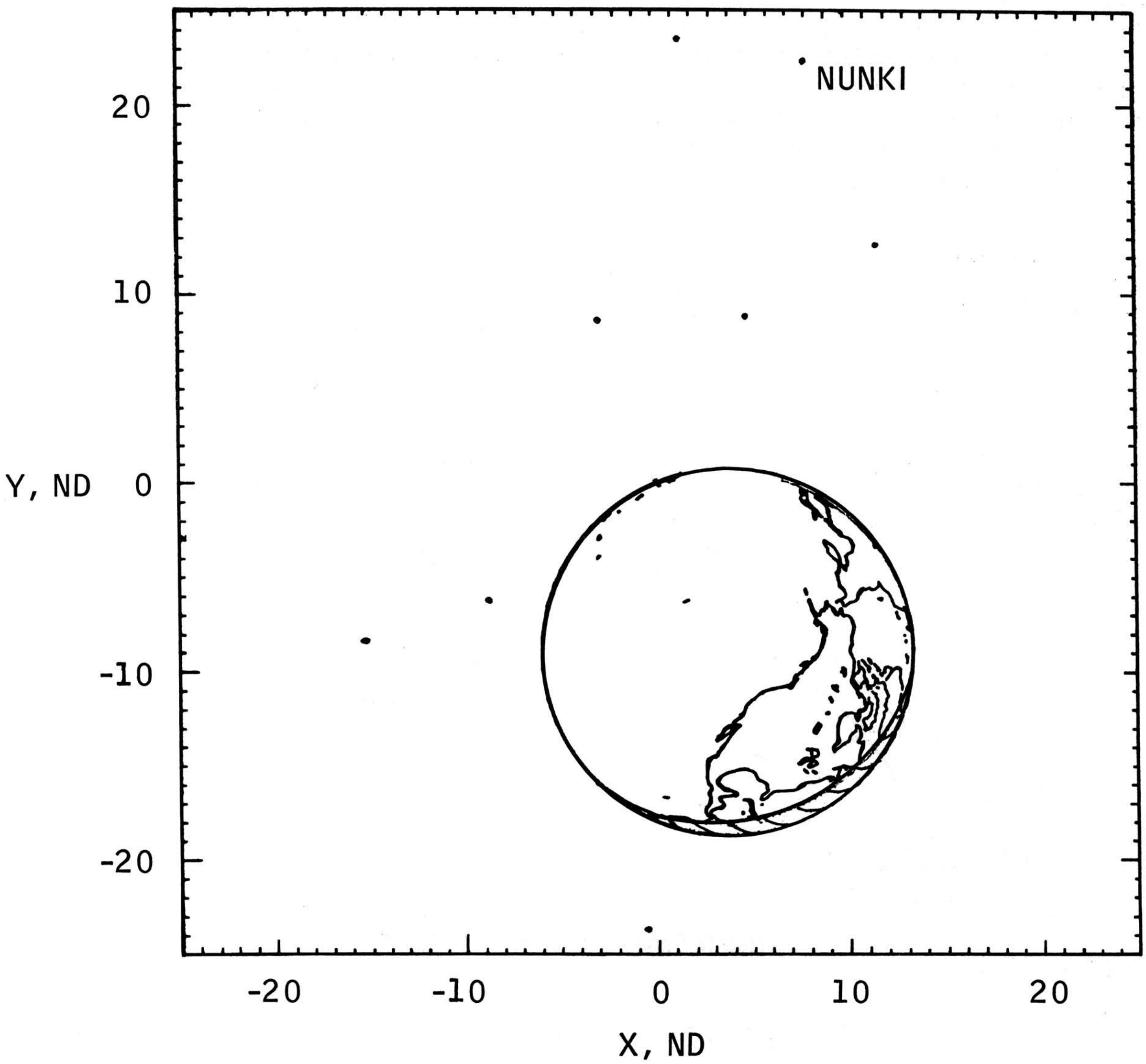


Figure 6.- View from spacecraft at abort altitude (TLI plus 4 hours).

With the maneuver direction generally fixed, it can be said the maneuver magnitude determines the transearth trajectory: its return time and its landing position. Regardless of the magnitude of the maneuver (disregarding the amount the orbit is rotated by the maneuver), the resulting transearth trajectory will pass through two common space-fixed points. One of these points is the spacecraft's position at the time of abort and the other point is where the transearth trajectories intersect the earth (inertial landing point). This inertial landing point and its relationship to the previously mentioned earth-moon line is shown in figure 7. This discourse should not be interpreted as meaning all the transearth trajectories will land at the same geographic point.

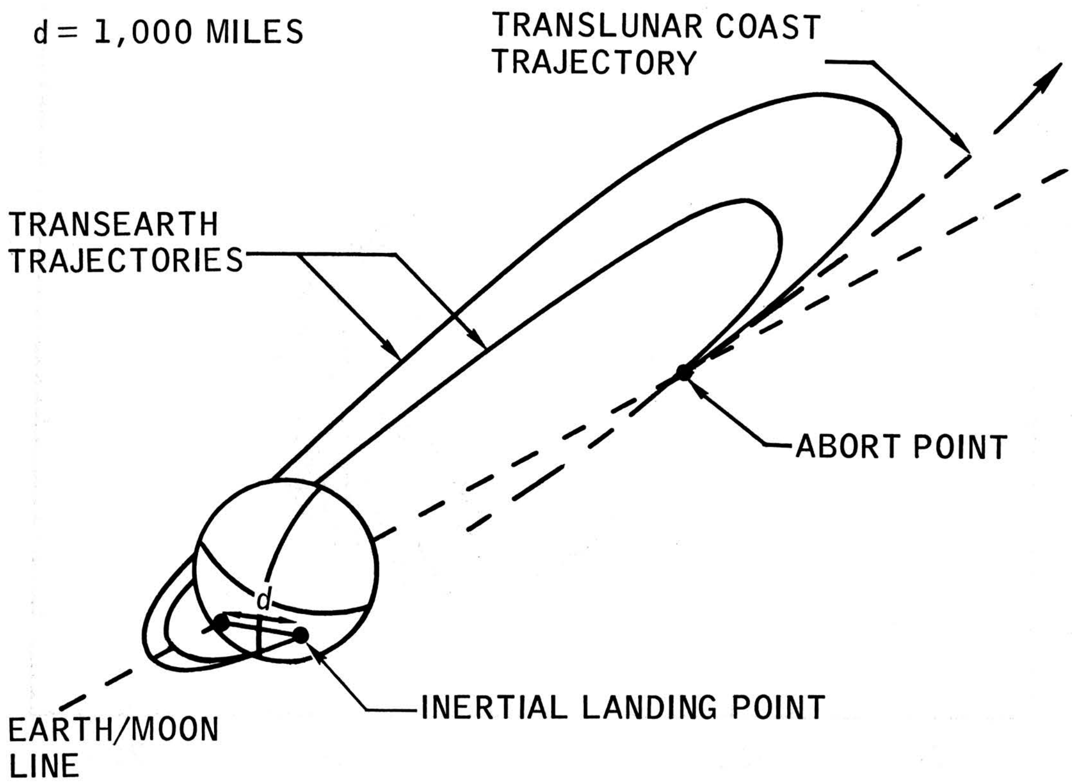


Figure 7.- Transearth trajectories.

At this point, an intricate problem dealing with flight mechanics enters the picture. That problem is as follows. Given a point on translunar coast, how fast can the spacecraft return to earth by using all the fuel available to the main engine? That problem can be eliminated by applying the rule of thumb that if we jettison the lunar module the Apollo spacecraft has sufficient fuel to achieve a return time equal to or faster than the current time from translunar injection. For this example, sufficient fuel is available to return to earth within 32 hours. In other words, it is possible to return before the system completely fails at 67 hours, but is it possible to return to site X?

Suppose at the abort time the geographic landing site (site X) is about 90° east of where the inertial landing point will be. For the spacecraft to land at site X, it must arrive at the inertial landing point at the same time site X rotates to the inertial landing point. (This is similar to the rendezvous of two spacecraft in inertial space.) The earliest time site X will arrive at the inertial landing point is 18 hours (time required for the earth to rotate 270°) (fig. 8).

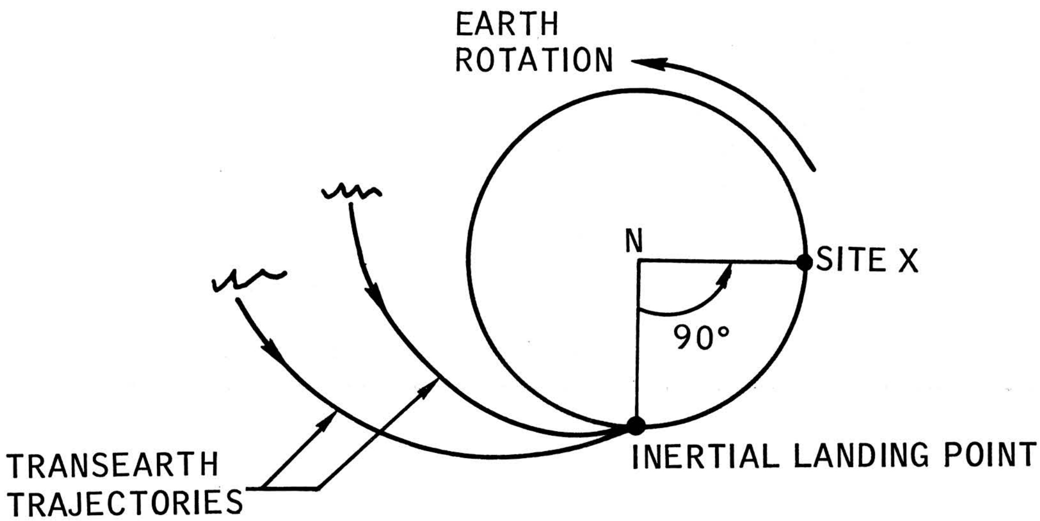
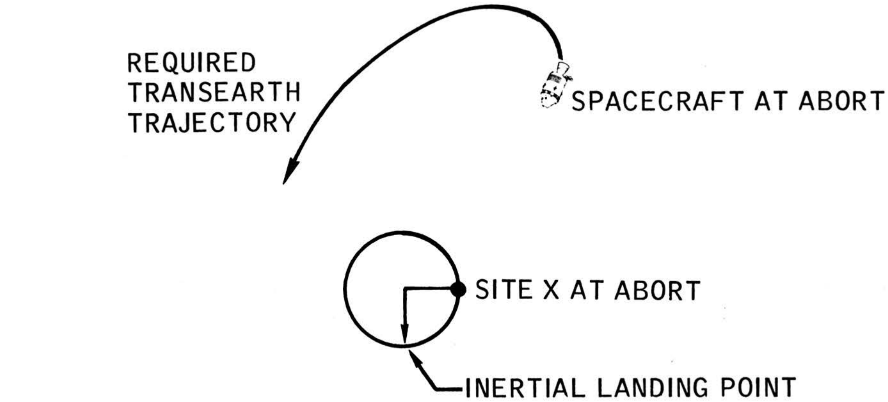
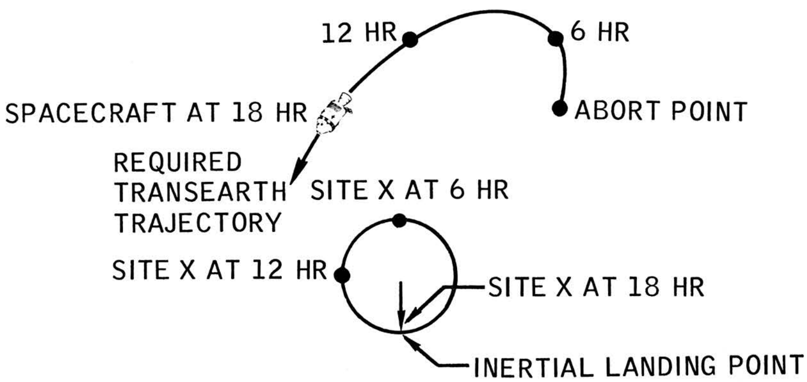


Figure 8.- Inertial landing point/geographic landing site relationship.

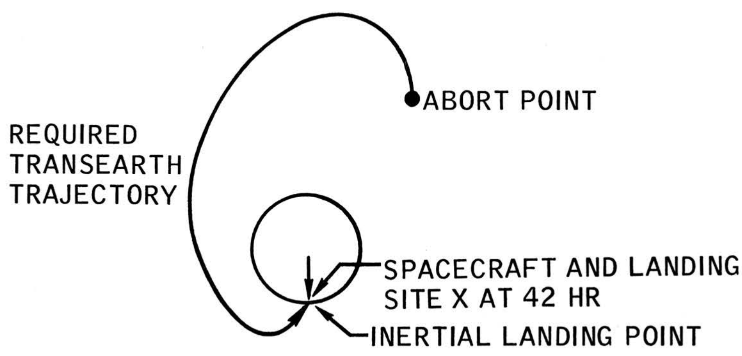
Through the previous examination of the problem, the fastest known time of spacecraft arrival was determined to be 32 hours following abort. Therefore, the spacecraft cannot rendezvous with site X the first time site X arrives at the inertial landing point. However, 24 hours from site X's first arrival it will be back at the inertial landing point (42 hours from the time of the abort). Therefore, the transearth trajectory that would solve this abort problem is the one with a return time of 42 hours from abort to landing, and the maneuver magnitude would be that required to achieve that particular transearth trajectory (fig. 9).



(A) AT ABORT.



(B) FIRST 18 HOURS OF TRANSEARTH TRAJECTORY.



(C) AT LANDING (ABORT PLUS 42 HOURS).

Figure 9.- Typical translunar coast abort geometry.

During the actual mission, the real-time computer at mission control has the capability to compute, as does the spacecraft computer, solutions to the abort problems if and when a need arises. This can be accomplished without having any premission trajectory information available. However, a large amount of this type information is provided preflight: (1) to familiarize the flight crew, flight controllers, and support groups with the data they will be observing during mission time, (2) to provide information to cockpit simulators to verify the performance of the spacecraft's guidance and control systems, and (3) to verify and establish confidence in the performance of the real-time computer abort programs.

Among the data provided preflight and updated and passed to the crew during mission time are several selected abort solutions which result in landing in the primary landing area (the mid-Pacific area) near Hawaii. The data passed to the crew include the abort ΔV, the time of the abort, and the resulting landing point. If an abort is required, these data are entered into the spacecraft's computer, and the computer points the spacecraft in the proper maneuver direction to burn the required abort ΔV. Table II includes some of the aforementioned parameters that were provided to flight crew and flight controllers for Apollo 11.

TABLE II.- TRANSLUNAR PHASE ABORT DATA

Ground elapsed time of abort, hr:min:sec	Abort ΔV, frs	Time from abort to 400 000 ft, hr:min:sec	Landing point	
			Latitude, deg	Longitude, deg
07:00:00	5832	18:07:16	3.4S	165W
14:00:00	4831	35:17:20	1.3S	165W
28:00:00	5331	45:08:30	2.0S	165W
38:00:00	7994	34:37:55	6.4S	165W
47:00:00	6058	49:57:21	3.0S	165W
56:00:00	4948	65:07:44	1.0S	165W

LUNAR PHASE ABORTS

Once the spacecraft begins the lunar orbit insertion (LOI) burn, it is no longer on a trajectory that will return it to earth. For the spacecraft to leave lunar orbit, the velocity of the spacecraft must be increased so the spacecraft can escape the gravitational pull of the moon.

Normally this acceleration is obtained by firing the service propulsion system (SPS) engine in the service module. However, techniques have been developed so that the LM descent engine can return the spacecraft to earth. These techniques could be used if the SPS engine fails during the LOI burn which is the first large burn of the engine.

The abort procedures are of three basic modes which are illustrated in figures 10, 11, and 12. Based on the length of the LOI burn, one of the three modes can be used to leave lunar orbit.

DOI Phase Aborts

Because the SPS engine is being used to perform the DOI maneuver, a 1-second overburn can result in an impacting trajectory. To return the spacecraft to a non-impacting trajectory if such an overburn occurs, the crew has been provided with an onboard contingency maneuver. The contingency maneuver is a 120-fps fixed attitude burn that occurs at DOI ignition plus 35 minutes (AOS plus 12 min).

Once the lunar module separates from the command module and lands on the moon, it can no longer be used to accelerate the spacecraft and return the crew to earth because very little propellant remains. However, since the SPS engine will have already performed satisfactorily during the lunar orbit insertion burn, confidence would increase that it will fire again to leave lunar orbit.

Following the transearth injection burn, the spacecraft will coast until earth entry. If a quicker return or change in landing location is desired, the SPS engine would be used to provide the necessary midcourse corrections.

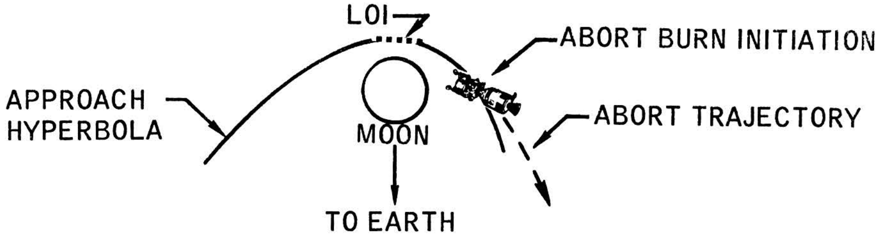


Figure 10.- Lunar orbit insertion (LOI) mode I abort.

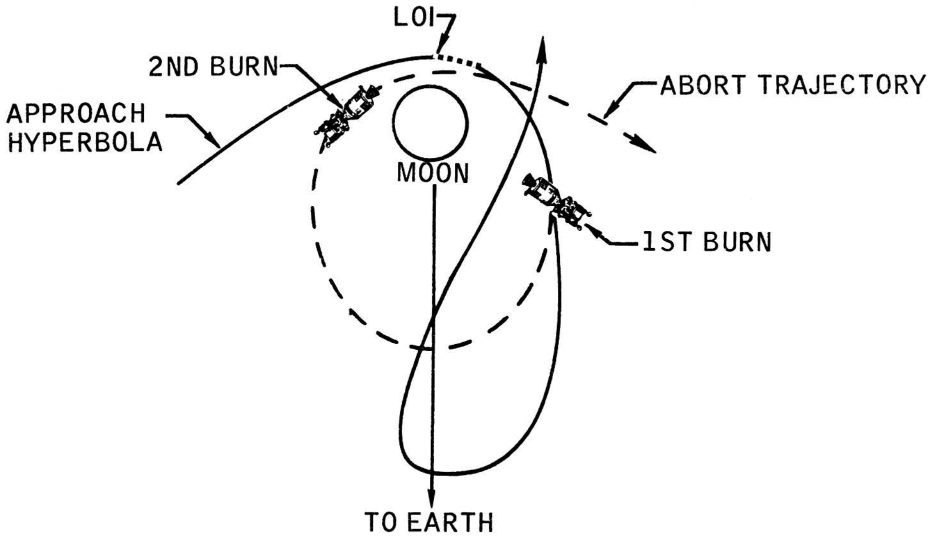


Figure 11.- Lunar orbit insertion (LOI) mode II abort.

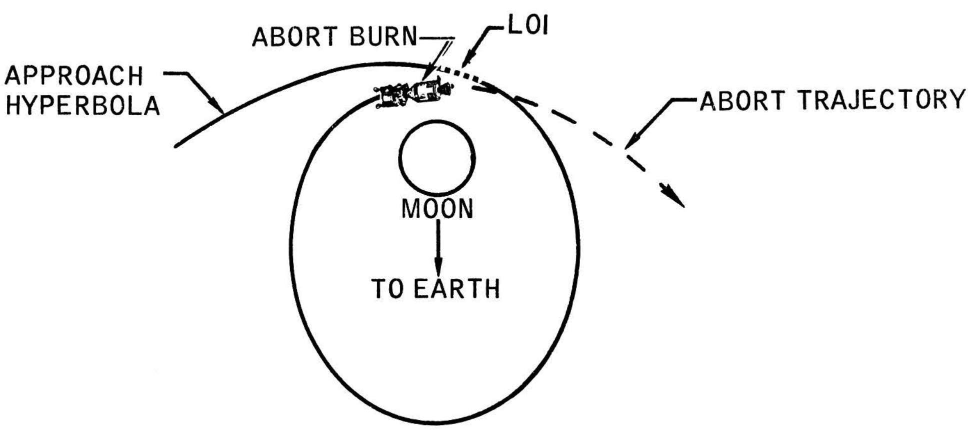


Figure 12.- Lunar orbit insertion (LOI) mode III abort.

III. APOLLO 13 NOMINAL MISSION SUMMARY

APOLLO 13 MISSION SUMMARY

The lift-off of the Apollo 13 spacecraft from Pad A Complex 39 of the Merritt Island Launch area is planned to occur at 13:13 a.m. c.s.t. on April 11 with a flight azimuth of approximately 72°. The major events of this mission are shown in figure 1. A mission summary of the earth-moon positions is presented in figure 2.

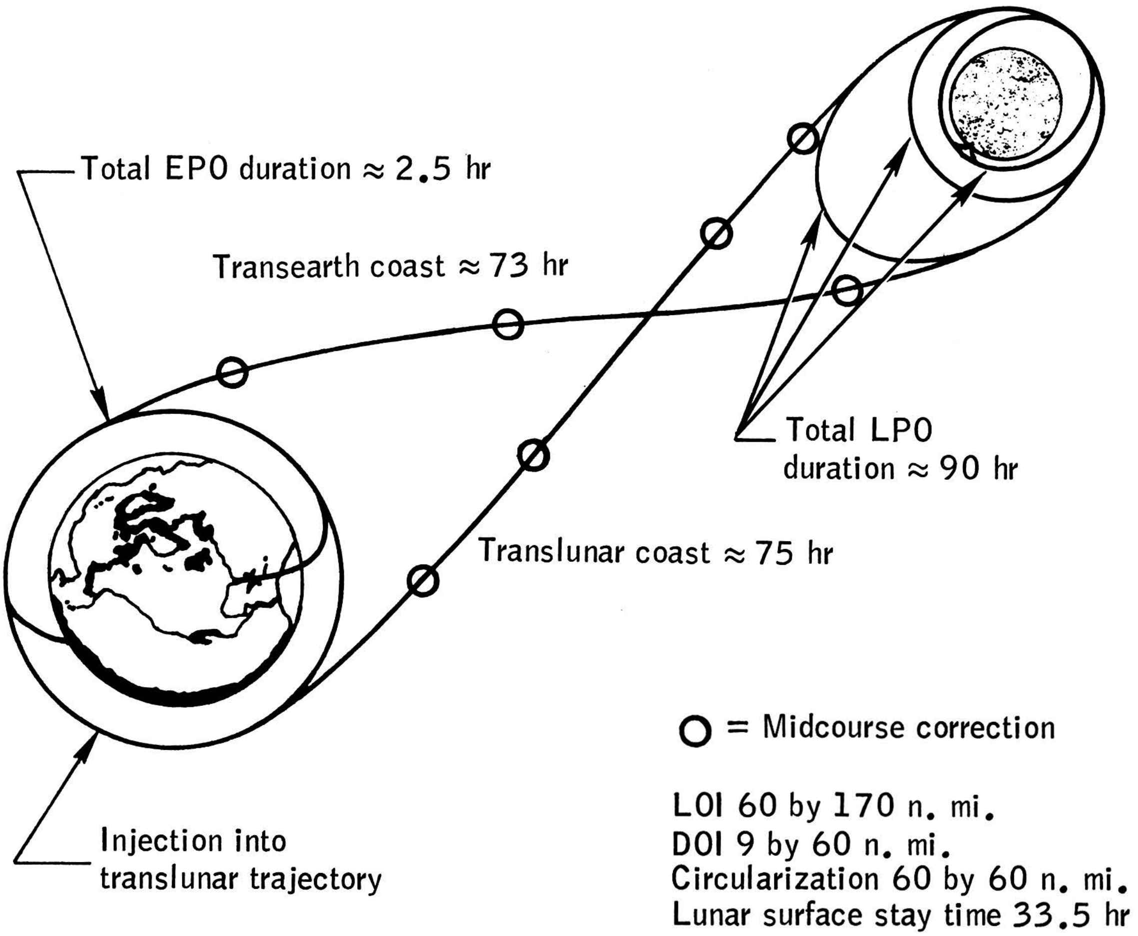


Figure 1.- Mission profile.

The mission description is divided into the following 18 phases.

a. Earth launch

b. Earth parking orbit

c. Translunar injection

d. Free-return circumlunar trajectory

e. Posttranslunar injection events

f. Translunar coast

g. Lunar orbit insertion (LOI)

h. Descent orbit insertion (DOI)

i. CSM/LM coast from LOI to undock

j. CSM undock to LM landing
   CSM/LM undock and separation
   CSM circularization
   Powered descent

k. Lunar surface stay

l. First CSM lunar orbit plane change

m. LM ascent

n. Rendezvous sequence

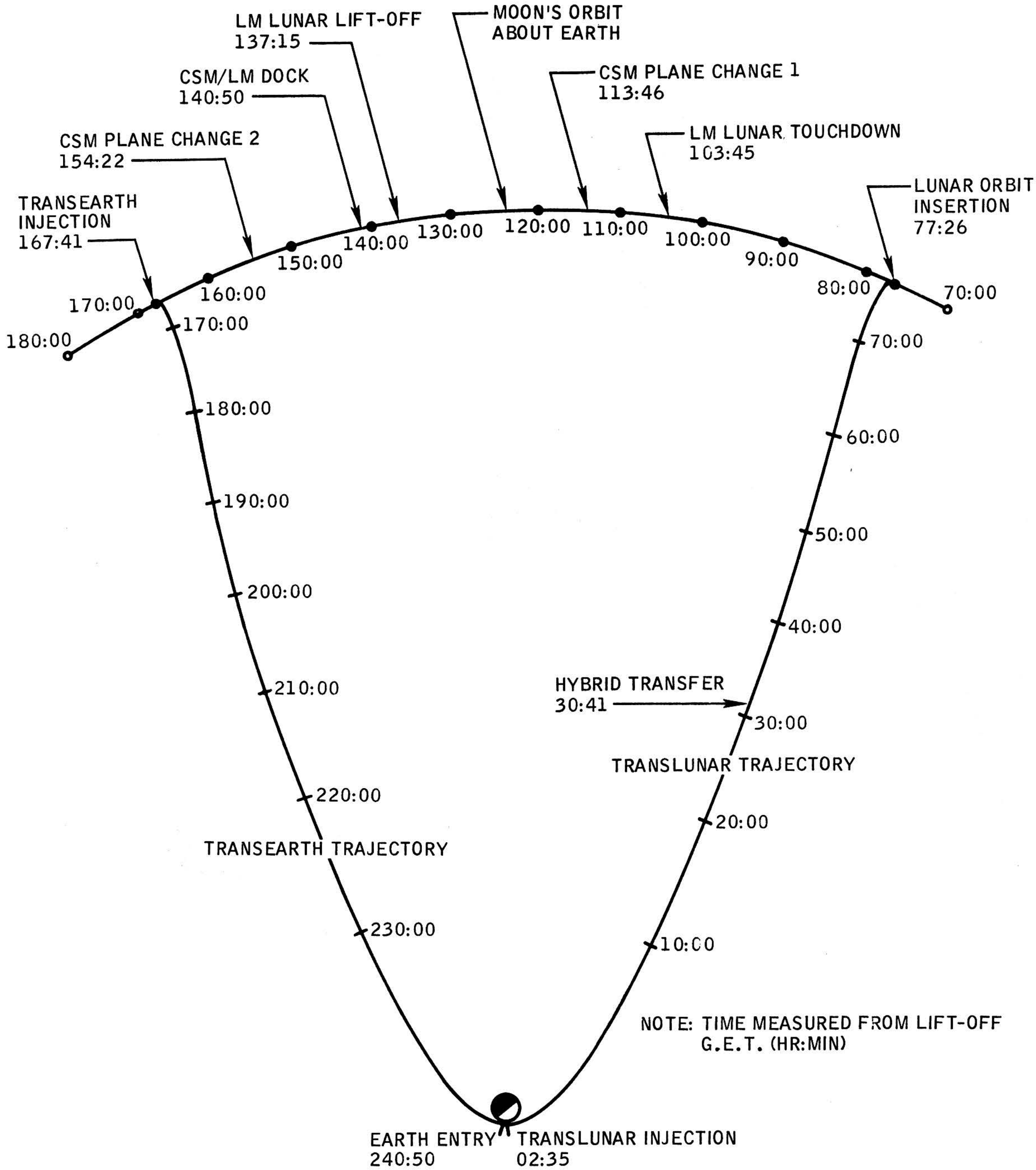
o. CSM/LM separation, LM jettison, and second CSM lunar orbit plane

p. Transearth injection

q. Translunar coast

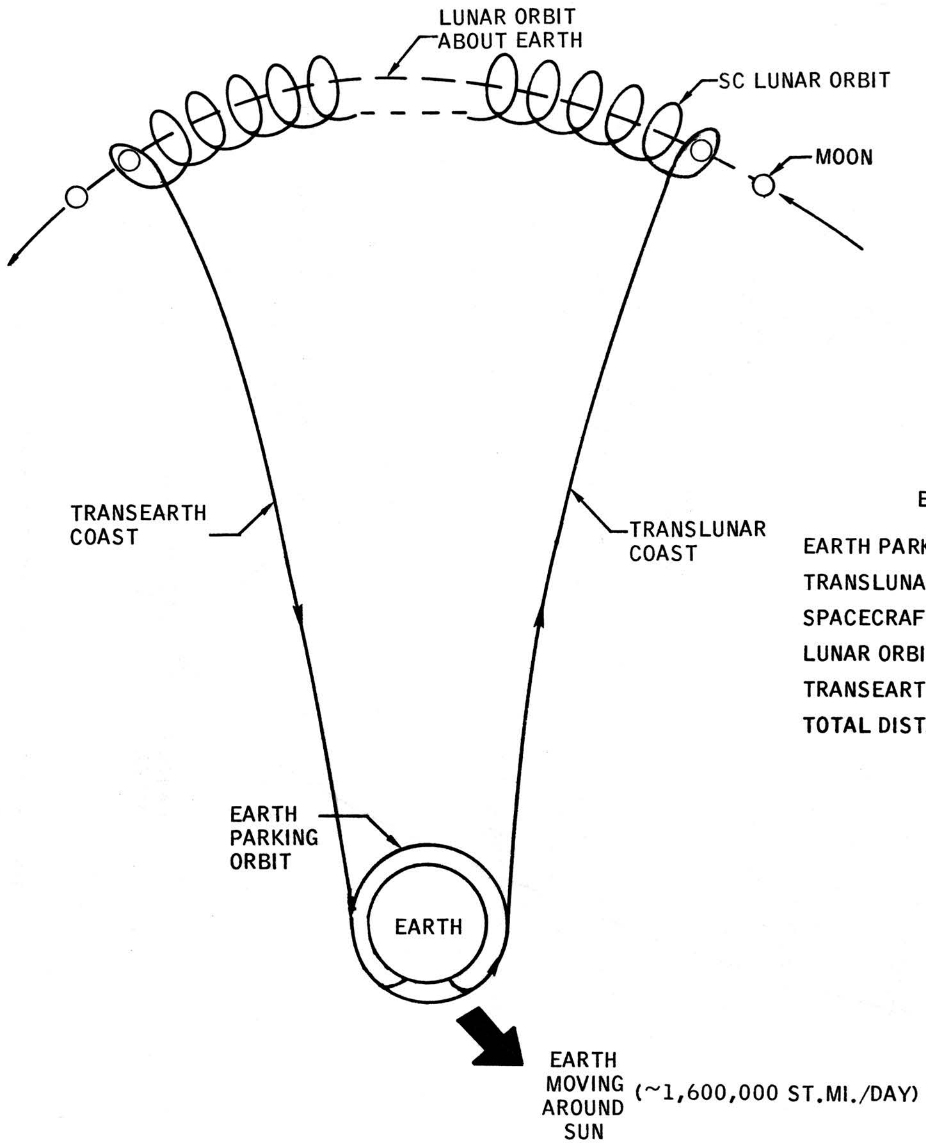
r. Entry

The duration of the Apollo 13 mission is approximately 10 days 1 hour; lunar orbit staytime is approximately 90 hours, and the translunar and transearth flight times are 75 hours and 73.5 hours, respectively. The sequence of major events is presented in table I(a), and the planned TV coverage for the Apollo 13 mission is presented in table I(b).



(a) Vehicle and moon positions relative to the earth.

Figure 2.- Mission summary.



EVENT	APPROXIMATE SPACECRAFT DISTANCE TRAVELLED WITH RESPECT TO THE EARTH, ST. MI.
EARTH PARKING ORBIT (EPO)	42,181
TRANSLUNAR COAST	293,703
SPACECRAFT LUNAR ORBIT	331,327
LUNAR ORBIT ABOUT EARTH	196,219
TRANSEARTH COAST	287,656
TOTAL DISTANCE TRAVELED	1,151,086

(b) Path of vehicle with respect to earth (spacecraft distance traveled).

Figure 2.- Mission summary.- Concluded

TABLE I.- MISSION EVENTS FOR AN APRIL 11 LAUNCH

(a) Sequence of major events

Event	Time, hr:min:sec, g.e.t.	Time, hr:min:sec, c.s.t.	Data summary	
Launch	00:00:00	April 11, 1970 13:13:00	Azimuth, deg	~72
			Launch complex	39A
EPO insertion	00:11:39.6	13:24:39.6	Geodetic latitude, deg	32.7
			Longitude, deg	-53.3
			Geodetic altitude, n. mi.	103.3
			Velocity, fps	25 567.7
Translunar injection				
Burn initiation	02:35:23.5	15:48:23.5	Geodetic latitude, deg	-22.2
			Longitude, deg	143.2
			Velocity, fps	25 593.6
			Apogee altitude, n. mi.	106.4
			Geodetic altitude, n. mi.	98.9
Burn termination	02:40:49.0	15:53:49.0	Geodetic latitude, deg	-9.7
			Longitude, deg	166.1
			Burn duration, sec	325.4
			Plane change, deg	.7
			Apogee altitude, n. mi.	308 693.
			Geodetic altitude, n. mi.	164.0
Post-TLI events				
CSM/S-IVB separation	03:05:49.0	16:18:49.0		
Docking	03:15:49.0	16:28:49.0		
CSM/LM ejection	04:00:49.0	17:13:49.0		
Evasive maneuver (performed by S-IVB)	04:13:49.0	17:26:49.0	ΔV, fps	9.4
Translunar coast, midcourse correction maneuvers				
MCC-1	TLI plus 9 hr	April 12, 1970 00:53:49	Geodetic altitude, n. mi.	~57 000
MCC-2 (hybrid transfer)[a]	30:40:49.0	19:53:49	Geodetic altitude, n. mi.	121 437.
			ΔV, fps	14.7
			Burn duration, sec	127.4
			SM RCS propellant used, lb	164.4
			Plane change, deg	.1
MCC-3	LOI minus 22 hr	April 13, 1970 20:39:05.3	Geodetic altitude, n. mi.	~176 400
MCC-4	LOI minus 5 hr	April 14, 1970 13:39:05.3	Altitude above mean lunar radius, n. mi.	~12 400

[a]Mission rules state that if the first maneuver is less than 3 sec it will be performed by RCS; however, a decision has been made to perform this maneuver using SPS.

TABLE I.- MISSION EVENTS FOR AN APRIL 11 LAUNCH - Continued

(a) Sequence of major events - Continued

Event	Time, hr:min:sec, g.e.t.	Time, hr:min:sec, c.s.t.	Data summary	
Lunar orbit insertion (LOI)				
Burn initiation	77:26:05.3	18:39:05.3	Mass, lb	95 530.4
			Altitude above LLS, n. mi.	69.2
			Selenographic latitude, deg	2.4
			Selenographic longitude, deg	-169.9
			Perilune altitude above LLS, n. mi.	58.0
			Selenographic inclination, deg	5.9
			Velocity, fps	8251.6
Burn termination	77:32:02.4	18:45:02.4	Altitude above LLS, n. mi.	60.1
			Selenographic latitude, deg	4.2
			Selenographic longitude, deg	166.6
			Selenographic inclination, deg	5.4
			Burn duration, sec	357.1
			Inertial burn arc, deg	23.4
			Plane change, deg	.5
			ΔV, fps	2821.3
			SPS propellant used, lb	23 321.8
			Velocity, fps	5480.4
			Orbital period, hr:min:sec	02:08:41
			Perilune altitude above LLS, n. mi.	58.9
			Apolune altitude above LLS, n. mi.	170.1
S-IVB predicted lunar impact	77:48:32.0	19:01:32.0	Selenographic latitude, deg	-3.0
			Selenographic longitude, deg	-30.0
Descent orbit insertion (DOI)				
Burn initiation	81:45:49.9	22:58:49.9	Mass, lb	72 154.0
			Altitude above LLS, n. mi.	58.9
			Selenographic latitude, deg	3.6
			Selenographic longitude, deg	174.9
			Perilune altitude above LLS, n. mi.	58.9
			Apolune altitude above LLS, n. mi.	170.2
			Velocity, fps	5486.5
Burn termination	81:46:12.7	22:59:12.7	Altitude above LLS, n. mi.	58.8
			Selenographic latitude, deg	3.7
			Selenographic longitude, deg	173.7
			Selenographic inclination, deg	5.4
			Burn duration, sec	22.8
			Inertial burn arc, deg	1.2
			Plane change, deg	0.0
			ΔV, fps	210.5
			Velocity, fps	5278.5
			SPS propellant used, lb	1490.9
			Orbital period, hr:min:sec.	01:54:12.5
			Perilune altitude above LLS, n. mi.	8.9
			Apolune altitude above LLS, n. mi.	59.2
First pass over Censorinus	84:24:55.0	April 15, 1970 01:37:55.0		

TABLE I.- MISSION EVENTS FOR AN APRIL 11 LAUNCH - Continued

(a) Sequence of major events - Continued

Event	Time, hr:min:sec, g.e.t.	Time, hr:min:sec, c.s.t.	Data summary	
Descent				
CSM/LM undock and SEP	99:18:28.7	16:31:28.7	Revolution number	12
Circularization (CSM)				
Burn initiation	100:37:22.0	17:50:22.0	Mass, lb	36 592.1
			Selenographic latitude, deg	1.2
			Selenographic longitude, deg	-166.2
			Selenographic inclination, deg	5.4
			Altitude above LLS, n. mi.	58.3
			Perilune altitude above LLS, n. mi.	9.0
			Velocity, fps	5297.9
			Revolution number	12
Burn termination	100:37:25.9	17:50:25.9	ΔV, fps	70.3
			Burn duration, sec	3.9
			Altitude above LLS, n. mi.	58.3
			Perilune altitude above LLS, n. mi.	54.3
			Apolune altitude above LLS, n. mi.	63.6
			Velocity, fps	5366.5
			SPS propellant consumed, lb	254.8
			Burn arc, deg	0.2
			Period, hr:min:sec	01:58:37
PDI (DPS ignition time)	103:33:11.4	20:46:11.4	Altitude above LLS, ft	51 488
			Velocity, fps	5573.8
			Revolution number	14
High gate (P63 to P64)	103:41:43.4	20:54:43.4	Altitude above LLS, ft	7181
			Velocity, fps	475.9
Low gate (500 ft)	103:43:21.4	20:56:21.4	Altitude above LLS, ft	592
			Velocity, fps	72.3
Vertical descent (P64 to P65)	103:44:03.4	20:57:03.4	Altitude above LLS, ft	140
			Velocity, fps	9.3
LM landing	103:44:36.2	20:57:36.2	ΔV, fps	6621.4
			Burn duration, min:sec	11:24.8
			DPS propellant consumed, lb	16 738.3
			Revolution number	14
CSM first pass over LLS	103:45:13.6	20:58:13.6	Revolution number	14

TABLE I.- MISSION EVENTS FOR AN APRIL 11 LAUNCH - Continued

(a) Sequence of major events - Continued

Event	Time, hr:min:sec, g.e.t.	Time, hr:min:sec, c.s.t.	Data summary	
CSM plane change (LOPC-1)				
Burn initiation	113:46:02.7	April 16, 1970 06:59:02.7	Mass, lb	36 286.3
			Selenographic latitude, deg	-5.0
			Selenographic longitude, deg	-48.1
			Altitude above LLS, n. mi.	56.2
			Perilune altitude above LLS, n. mi.	55.7
			Apolune altitude above LLS, n. mi.	61.7
			Velocity, fps	5360.6
			Revolution number	19
Burn termination	113:46:12.6	06:59:12.6	Mass, lb	35 639.1
			ΔV, fps	181.4
			Burn duration, sec	10.0
			Selenographic latitude, deg	-5.0
			Selenographic longitude, deg	-48.6
			Altitude above LLS, n. mi.	56.2
			Perilune altitude above LLS, n. mi.	55.7
			Apolune altitude above LLS, n. mi.	61.7
			Plane change, deg	1.9
			Selenographic inclination, deg	5.0
			Velocity, fps	5360.1
			Revolution number	19
Second pass over Censorinus	129:04:59.6	22:17:59.6	Revolution number	27
CSM second pass over LLS	137:14:05.1	April 17, 1970 06:27:05.1	Revolution number	31
Ascent				
LM lift-off	137:15:22.5	06:28:22.5	Mass, lb	10 774.9
			Selenographic latitude, deg	-3.7
			Selenographic longitude, deg	-17.5
			Revolution number	31
LM insertion	137:22:32.4	06:35:32.4	Mass, lb	5843.1
			ΔV, fps	6042.8
			Burn duration, sec	430
			Latitude, deg	-4.2
			Longitude, deg	-27.5
			Altitude above LLS, ft	59 957.0
			Perilune altitude above LLS, ft	52 944.4
			Apolune altitude above LLS, ft	266 267.9

TABLE I.- MISSION EVENTS FOR AN APRIL 11 LAUNCH - Continued

(a) Sequence of major events - Continued

Event	Time, hr:min:sec, g.e.t.	Time, hr:min:sec, c.s.t.	Data summary	
Rendezvous				
CSI	138:12:10.1	07:25:10.1	Revolution number	32
			Burn duration, sec	44.7
			ΔV, fps	49.5
			Propellant used, lb	31.9
			Resultant h_a/h_p, n. mi.	44.9/43.2
			Range at cutoff, n. mi.	149.0
			Range rate at cutoff, fps	-121.4
			Propulsion system	LM RCS
CDH	139:10:28.9	08:23:28.9	Burn duration, sec	0.0
			ΔV, fps	0.0
TPI	139:51:44.6	09:04:44.6	Burn duration, sec	22.4
			ΔV, fps	25.0
			Propellant used, lb	16.0
			Resultant h_a/h_p, n. mi.	60.7/43.6
			Range at cutoff, n. mi.	32.0
			Range rate at cutoff, fps	-132.6
			Propulsion system	LM RCS
			Revolution number	32
Braking	140:33:05.0	09:46:05.0	Burn duration, sec	27.9
			ΔV, fps	31.1
			Propellant used, lb	19.9
			Range at final braking, n. mi.	0.01
			Range rate at final braking, fps	-0.2
			h_a/h_p at final braking, n. mi.	59.4/58.0
			Propulsion system	LM RCS
			Revolution number	33
Docking	140:50:00.0	10:03:00.0		
LM jettison	143:10:59.8	12:23:59.8	Selenographic latitude, deg	-3.9
			Selenographic longitude, deg	-24.8
CSM/LM separation				
Burn initiation	143:15:58.4	12:28:58.4		
Burn termination	143:16:04.5	12:29:04.5	Mass, lb	35 766.8
			ΔV, fps	1.0
			Burn duration, sec	6.1
			Selenographic latitude, deg	-4.6
			Selenographic longitude, deg	-40.3
			Altitude above LLS, n. mi.	59.4
			Perilune altitude above LLS, n. mi.	57.2
			Apolune altitude above LLS, n. mi.	59.5
			Plane change, deg	0.0
			Selenographic inclination, deg	5.0
			Velocity, fps	5342.7
			Revolution number	34

TABLE I.- MISSION EVENTS FOR AN APRIL 11 LAUNCH - Continued

(a) Sequence of major events - Continued

Event	Time, hr:min:sec, g.e.t.	Time, hr:min:sec, c.s.t.	Data summary	
LM deorbit	144:39:12.6	13:52:12.6	Mass, lb	5329.1
			ΔV, fps	185.5
			Burn duration, sec	78.5
			Selenographic latitude, deg	3.2
			Selenographic longitude, deg	66.3
LM impact	144:55:31.1	14:08:31.1	Mass, lb	5220.8
			Selenographic latitude, deg	-3.0
			Selenographic longitude, deg	-19.7
			Velocity, fps	5503.9
			CSM latitude, deg	-0.7
			CSM longitude, deg	17.8
			CSM revolution number	35
LOPC-2				
Burn initiation	154:21:19.6	23:34:19.6		
Burn termination	154:22:02.8	23:35:02.8	Mass, lb	32 892.9
			ΔV, fps	827.6
			Burn duration, sec	43.2
			Selenographic latitude, deg	4.4
			Selenographic longitude, deg	89.7
			Altitude above LLS, n. mi.	55.5
			Perilune altitude above LLS, n. mi.	55.4
			Apolune altitude above LLS, n. mi.	60.8
			Plane change, deg	8.8
			Selenographic inclination, deg	11.6
			Velocity, fps	5363.2
			Revolution number	40
Pass over Descartes	158:42:43.9	pril 18, 1970 03:55:43.9	Revolution number	42
Pass over Davy crater chain	158:49:46.0	04:02:46.0	Revolution number	42
Transearth injection				
Burn initiation	167:38:39.1	12:51:39.1	Mass, lb	32 824.2
			Altitude above LLS, n. mi.	57.8
			Selenographic latitude, deg	9.9
			Selenographic longitude, deg	-177.7
			Perilune altitude above LLS, n. mi.	54.4
			Selenographic inclination, deg	11.5
			Velocity, fps	5351.2
			Revolution number	46

TABLE I.- MISSION EVENTS FOR AN APRIL 11 LAUNCH - Continued

(a) Sequence of major events - Concluded

Event	Time, hr:min:sec, g.e.t.	Time, hr:min:sec, c.s.t.	Data summary	
Burn termination	167:40:53.8	12:53:53.8	Altitude above LLS, n. mi.	63.2
			Selenographic latitude, deg	11.4
			Selenographic longitude, deg	173.8
			Perilune altitude above LLS, n. mi.	56.3
			Selenographic inclination, deg	17.8
			Burn duration, sec	134.7
			Inertial burn arc, deg	8.5
			Plane change, deg	9.3
			ΔV, fps	3143.9
			SPS propellant used, lb	8794.8
			Velocity, fps	8270.6
Transearth coast, midcourse correction maneuvers				
MCC-5	TEI plus 15 hr	April 19, 1970 03:53:53.8	Geodetic altitude, n. mi.	~186 600
MCC-6	EI minus 22 hr	April 20, 1970 16:02:49.6	Geodetic altitude, n. mi.	~104 000
MCC-7	EI minus 3 hr	April 21, 1970 11:02:49.6	Geodetic altitude, n. mi.	~2560
Entry interface	240:49:49.6	14:02:49.6	Transearth coast time, hr	73.1
			Inertial velocity, fps	36 129.4
			Geodetic altitude, n. mi. (ft)	65.8(400 249)
			Inertial flight-path angle, deg	-6.5
			Geodetic latitude, deg	-15.1
			Longitude, deg	-173.5
			Equatorial inclination (ascending), deg	40.
CM landing	241:03:40.8	14:16:40.8	Geodetic latitude, deg	-1.575
			Longitude, deg	-157.497

TABLE I.- MISSION EVENTS FOR AN APRIL 11 LAUNCH (Concluded)

(b) Planned TV coverage

SUBJECT	GET	DAY/DATE	CST	DURATION	VEH
EARTH SURFACE	01:36	SAT/APR 11	2:49 PM	05 MIN	CSM
TRANSPOSITION & DOCKING	03:15	SAT/APR 11	4:28 PM	1 HR 08 MIN	CSM
SPACECRAFT INTERIOR (MCC-2)	30:15	SUN/APR 12	7:28 PM	30 MIN	CSM
SPACECRAFT INTERIOR & IVT TRANSFER	58:00	MON/APR 13	11:13 PM	30 MIN	CSM
FRA MAURO	95:50	WED/APR 15	1:03 PM	15 MIN	CSM
LUNAR SURFACE ACTIVITIES (EVA-1)	108:10	THURS/APR 16	1:23 AM	3 HR 52 MIN	LM
LUNAR SURFACE ACTIVITIES (EVA-2)	127:50	THURS/APR 16	9:03 PM	6 HR 35 MIN	LM
FINAL RENDEZVOUS PHASE	140:23	FRI/APR 17	9:36 AM	12 MIN	CSM
LUNAR SURFACE	166:10	SAT/APR 18	11:23 AM	40 MIN	CSM
POST TEI - LUNAR SURFACE	168:00	SAT/APR 18	1:13 PM	25 MIN	CSM
EARTH & SPACECRAFT INTERIOR	221:45	MON/APR 20	6:58 PM	15 MIN	CSM

The launch window summary for April is presented in table II.

TABLE II.- APRIL LAUNCH WINDOW SUMMARY

Launch date	April 11, 1970
Site	Fra Mauro
Launch azimuth, deg	72 to 96
Launch time, hr:min, c.s.t.	13:13 to 16:36
Free-return h_p, n. mi.	210 to 60
Hybrid transfer ΔV, fps	14.7
Translunar flight time, hr:min	74:45
Lunar orbit inclination, deg	5.3
Approach azimuth at landing, deg	-93.9
Sun elevation at landing, deg	9.9 to 11.4
Goldstone landing coverage, hr:min	7:15
Goldstone plus Parks landing coverage, hr:min	10:10
Lunar surface staytime, hr	~34
Photo sites	Censorinus Davy crater chain Descartes
Total lunar orbit staytime	90 hr, 46 revs
Transearth flight time, hr	73 to 71
Total mission time, hr.	241 to 238

EARTH LAUNCH

The Saturn V launch phase consists of complete burns of the S-IC and S-II stages and a partial burn of the S-IVB stage. The basic launch vehicle flight configurations used during flight from lift-off through LV/SC final separation are shown in figure 3.

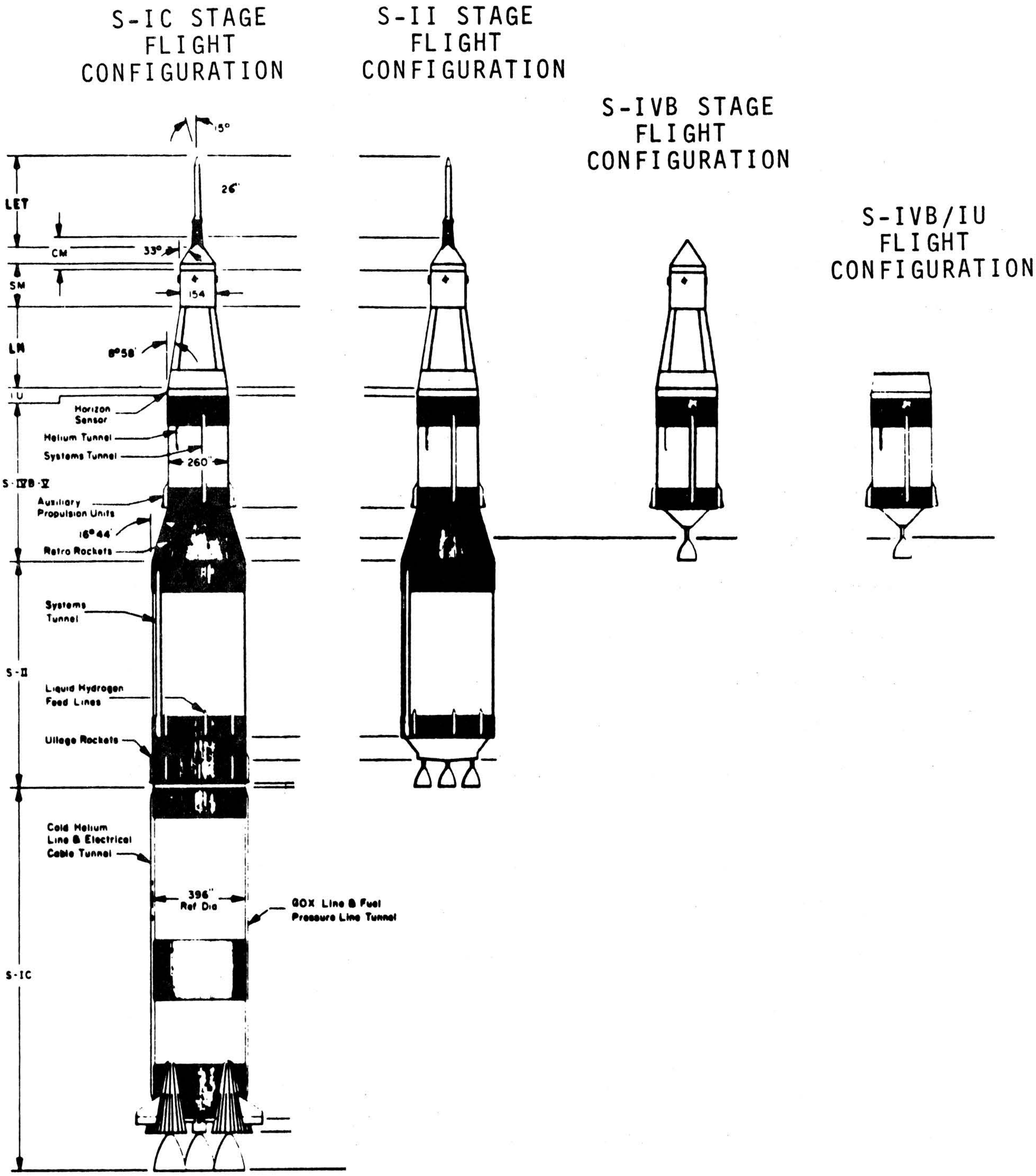


Figure 3.- Launch vehicle flight configurations.

The first three flight configurations are identified by the main booster stage that is used; that is, S-IC stage configuration, S-II stage configuration, and S-IVB stage configuration. The final LV flight configuration is the S-IVB stage with the Instrument Unit. Configuration dimensional data, initial mass characteristic for the first three configurations are summarized in table III.

TABLE III.- NOMINAL AS-508 FLIGHT CONFIGURATION

	S-IC stage flight configuration	S-II stage flight configuration	S-IVB stage flight configuration
Length, ft	363.02	195.77	114.27
Diameter, ft	33.0	33.0	2.17
Initial mass of flight configuration, lb	6 426 089.	1 468 302.	366 687.
Initial propellant mass of operating booster stage,[a] lb	4 663 400.	996 366.	235 562.
Initial inert mass of operating booster stage,[a] lb	2 922 272.	78 644.	26 159.
Inertial velocity at engine cutoff, fps	[b]8 990.525	[b]22 830.948	25 562.447
Altitude at engine cutoff, ft	[b]218 278.	[b]615 501.	627 980.

[a]Initial characteristics are for ground lift-off.

[b]72.080° flight azimuth.

The S-IC stage configuration is powered by five clustered F-1 engines that turn liquid oxygen (LOX) and RP-1 (kerosene) at an approximate flowrate mixture ratio of 2.3.

The S-II stage configuration is propelled by five clustered J-2 engines that consume LOX and liquid hydrogen (LH_2). Two basic propellant mixture ratios are used: a LOX/LH_2 flowrate mixture ratio of approximately

TRANSLUNAR INJECTION (TLI)

The second S-IVB burn or translunar injection burn (TLI) is initiated over Australia on the second revolution in earth orbit, and the burn is totally in sunlight. The TLI tracking ships are not used for this mission; however, four ARIA support the burn.

The TLI burn places the spacecraft on a free-return trajectory with a high perilune altitude (~210 n. mi.). The evasive maneuver after CSM/LM ejection is made with the S-IVB APS rather than with the SM SPS. The characteristics of the TLI burn are as follows.

Burn initiation[a]

Time, hr:min:sec, g.e.t.	2:35:23.5
Geodetic latitude, deg	-22.2
Longitude, deg	143.2
Velocity, fps	25 594
Apogee altitude above mean radius, n. mi.	106.4
Perigee altitude, n. mi.	98.9

Burn termination[a]

Time, hr:min:sec, g.e.t.	2:40:49.0
Geodetic latitude, deg	-9.7
Longitude, deg	166.1
Velocity, fps	35 632
Burn duration, sec	325.4
S-IVB propellant used, lb	157 798.
Plane change, deg	0.7
Inclination to earth-moon plane, deg	4.8
Apogee altitude above mean radius, n. mi.	308 693
Perigee altitude above mean radius, n. mi.	164.0

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[a]The parameters for this phase are approximate and are presented for information only. The official source for this phase is the MSFC LV operational trajectory.

5.5 is used for the first 322 seconds of the S-II main stage operations, and a LOX/LH_2 ratio of approximately 4.2 is used during the remainder of the S-II stage flight. An S-II center engine shutdown is planned at approximately 299.0 seconds into this burn to reduce the S-II stage pogo or longitudinal vibration effects noted during earlier flights.

The S-IVB stage configuration is powered by a single J-2 engine with LOX and LH_2 used as propellants. The LOX/LH_2 flowrate mixture ratio used during the boost to parking orbit (S-IVB first burn) is approximately 4.9.

After launch, the vehicle rises approximately 43 feet vertically to clear the launch umbilical tower. During the vertical rise, a yaw maneuver is executed to increase the lateral distance between the vehicle and the tower. After the tower is cleared vertically, the pitch and roll programs are initiated. The roll program alines the vehicle body axis with the computed flight azimuth. The pitch program provides a trajectory that satisfies vehicle performance, heating, and load requirements. Maximum dynamic pressure is encountered approximately 80.0 seconds after first motion. The launch escape tower (LET) is jettisoned by crew command after assurance that the S-II ignition and thrust buildup has occurred. The S-II engine cutoff command is initiated at approximately 558.0 seconds after first motion.

The vehicle insertion into a circular 100-n. mi. altitude (referenced to earth equatorial radius) parking orbit occurs at the end of the S-IVB first burn period. The S-IVB stage cutoff is commanded by the guidance cutoff conditions are achieved.

The earliest possible time for parking orbit insertion to occur is about 711.5 seconds and corresponds to a flight azimuth of 90°. The latest possible time for parking orbit insertion is about 715.5 seconds and corresponds to a flight azimuth of approximately 72.080°.

EARTH PARKING ORBIT

The vehicle coasts in earth parking orbit for up to three revolutions while a subsystems checkout is performed. During coast in parking orbit, the vehicle orbit is continually perturbed by auxiliary propulsion system (APS) burns, by aerodynamic drag, and by thrust from the propulsive hydrogen vents. The maximum and minimum parking orbit inclinations associated with a 72° to 96° flight azimuth range are approximately 32.6° and 28.4°. The LV maintains local horizontal attitude throughout the EPO phase except for an inertial hold of approximately 10 seconds immediately after EPO insertion. The total time spent in EPO is approximately 2 hours 35 minutes. The groundtrack for the EPO is shown in figure 4.

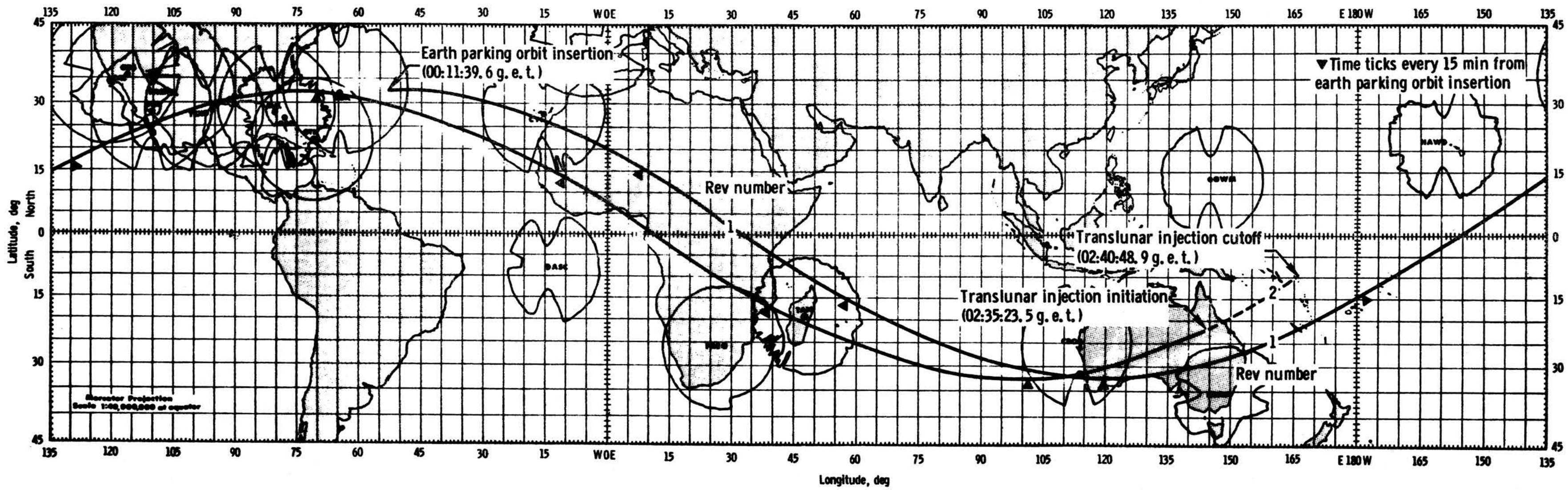


Figure 4. - Earth parking orbit groundtrack.

FREE-RETURN CIRCUMLUNAR TRAJECTORY

The free-return trajectory of the hybrid profile has the following characteristics.

Perilune	
Time, hr:min:sec, g.e.t.	77:40:21.9
Altitude above LLS, n. mi.	211.7
Selenographic latitude, deg	3.4
Selenographic longitude, deg	-178.9
Return vacuum perigee altitude, n. mi.	22.3
Transit time from TLI	
Burnout to entry interface, hr:min:sec . . .	150:30:33
Earth entry interface	
Time, hr:min:sec, g.e.t.	153:11:22
Altitude, n. mi.	65.8
Geodetic latitude, deg	-27.9
Longitude, deg	-174.2
Inclination, deg	31.4
Earth landing	
Time, hr:min:sec, g.e.t.	153:25:22
Geodetic latitude, deg	-20.5
Longitude, deg	39.9
Time prior to sunset, hr:min	10:24

POSTTRANSLUNAR INJECTION EVENTS

The summary of the major events from TLI cutoff through S-IVB LOX blowdown is given in table IV. To determine the separation attitude maneuver (TB7 plus 900 sec), the sun was constrained to between 32° and 90° of the LV +X-axis. This constraint provides over-the-shoulder lighting and avoids any CSM shadow on the S-IVB for the docking phase.

Upon ground command, the S-IVB performs an APS evasive maneuver approximately 12 minutes after CSM/LM separation. The S-IVB attitude for this maneuver will be the same as the separation attitude except that the sign of the middle gimbal angle (yaw gimbal) will be reversed. The APS burn will be 9.4 fps.

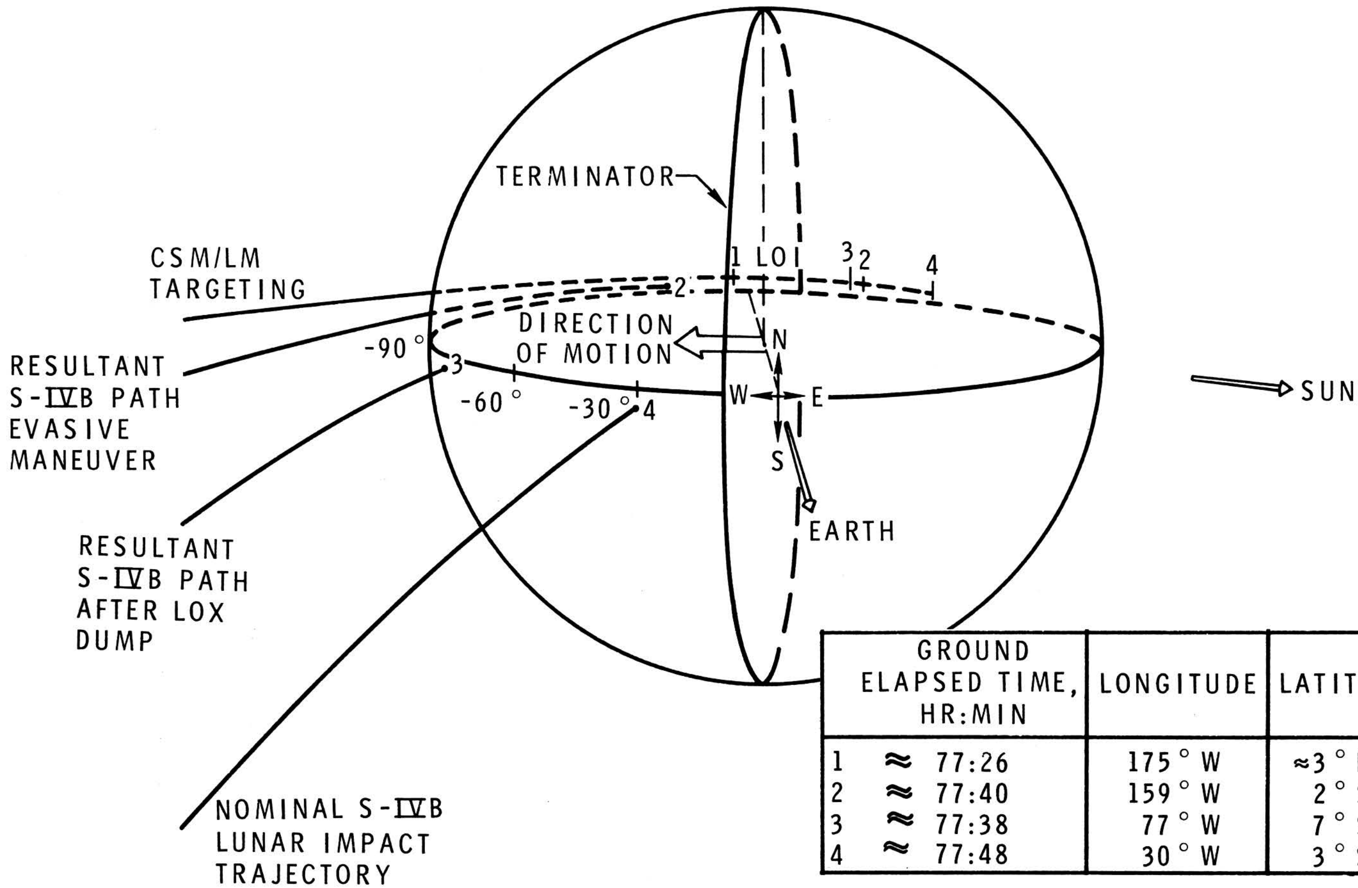
The S-IVB performs a LOX dump 21 minutes 20 seconds after the APS evasive maneuver. The local horizontal attitude for this dump is pitch 183°, yaw -5°, roll 180°.

The LOX dump maneuver is designed to reduce the probability of SC recontact with the S-IVB and also to achieve S-IVB impact with the moon. Nominally, an S-IVB APS lunar impact burn will be executed after the APS evasive and LOX dump maneuvers (lift-off plus 6 hr) to target the S-IVB for a lunar impact at 30°W and 3°S which is near the Apollo 12 landing site. The relative spacecraft/S-IVB arrival locations are shown in figure 5.

TABLE IV.- SUMMARY OF EVENTS FROM TLI

CUTOFF THROUGH LOX DUMP

Time from TB7, hr:min:sec	Event	ΔV, fps	Comments
00:00:00	Hold cutoff attitude		
00:02:31	Command and hold local horizontal		
00:15:00	Initiate maneuver to separation attitude		Local horizontal attitude pitch = 120° yaw = -40° roll = 180° .8 deg/sec
00:20:00	Freeze separation attitude inertially		Latest time for maneuver to be completed
00:25:00	Begin SC separation/SLA jettison	0.5	
00:25:15	AUTO maneuver to docking attitude		2.0 deg/sec
	Null rates and translation		
	Initiate 0.7-fps closing rate		4-sec RCS +X
	Null 0.7-fps closing rate	0.7	-X RCS
00:35:00	Begin dock		
01:20:00	LM/CSM ejection from S-IVB	1.2	Spring ejection and 3-sec -X RCS
01:33:01	S-IVB APS evasive maneuver	9.4	Start TB8
01:54:20	S-IVB LOX dump		Local horizontal attitude pitch = 183° yaw = -5° roll = 180°



	GROUND ELAPSED TIME, HR:MIN	LONGITUDE	LATITUDE
1	≈ 77:26	175° W	≈3° N
2	≈ 77:40	159° W	2° S
3	≈ 77:38	77° W	7° S
4	≈ 77:48	30° W	3° S

Figure 5.- Relative spacecraft/S-IVB arrival locations.

TRANSLUNAR COAST

A groundtrack for the translunar coast phase is given in figure 6. The time history of attitude for the first 10 hours of translunar coast is provided in figure 7.

After TLI, the spacecraft is on a free-return circumlunar trajectory with a perilune altitude of 210 n. mi. At TLI plus 28 hours, a transfer maneuver is performed which results in a non-free-return translunar trajectory (fig. 8) with a 58.8-n. mi. perilune altitude. The transfer trajectory is restricted such that, in the event that an SPS maneuver could not be made, an earth return would be possible within the DPS docked ΔV capability. It is assumed that the LM maneuver could be made at a minimum of 2 hours after perilune. The characteristics of the burn are the following.[a]

Time, hr:min:sec, g.e.t.	30:40:49.0
Geodetic altitude, n. mi.	121 437
ΔV, fps	14.7
Burn duration, sec	127.4
SM RCS propellant used, lb	164.4
Plane change, deg	0.1

Passive thermal control attitude will be maintained throughout most of the translunar coast phase. Four midcourse correction maneuver points have been defined at the following times.

a. TLI plus 9 hours (MCC-1)

b. TLI plus 28 hours (MCC-2) (the transfer maneuver)

c. LOI minus 22 hours (MCC-3)

d. LOI minus 5 hours (MCC-4)

The SC enters the lunar umbra approximately 43 minutes prior to LOI ignition. There is continual MSFN coverage from TLI cutoff until lunar occultation.

---

[a]Mission rules state that if the first maneuver is less than 3 sec it will be performed by RCS; however, a decision has been made to perform this maneuver using SPS.

The duration of the translunar coast phase is 74 hours 25 minutes. Note that this duration violates the highly desirable ground rule of an even integer number of days for translunar flight to provide a more favorable crew work/rest cycle. However, a thorough review of mission design scans indicated that no even-integer day was possible without violation of other mandatory ground rules.

Geodetic altitude from translunar injection to LOI is shown in figure 9. Altitude above the lunar surface during the last 10 hours of translunar coast is shown in figure 10.

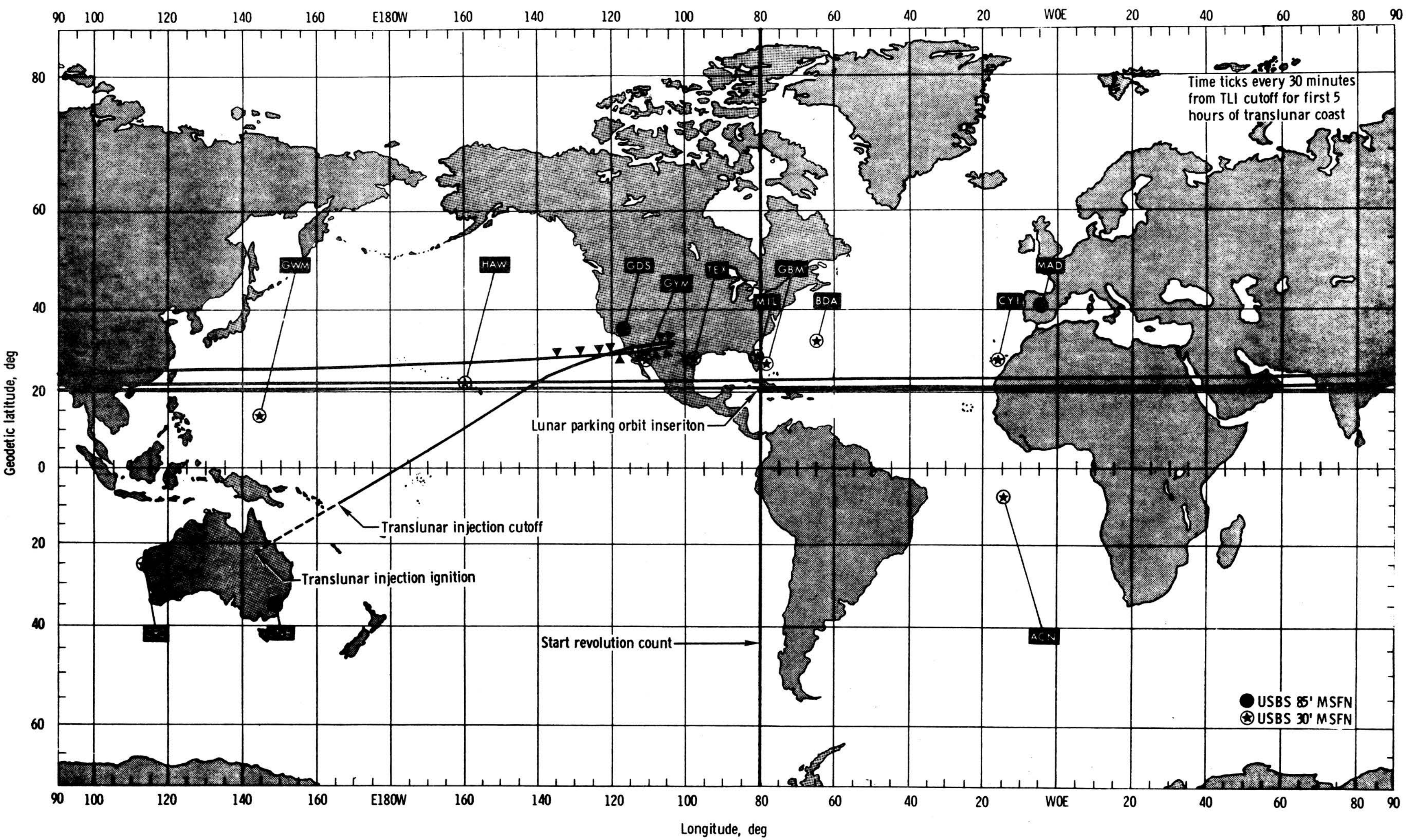


Figure 6. - Translunar coast groundtrack.

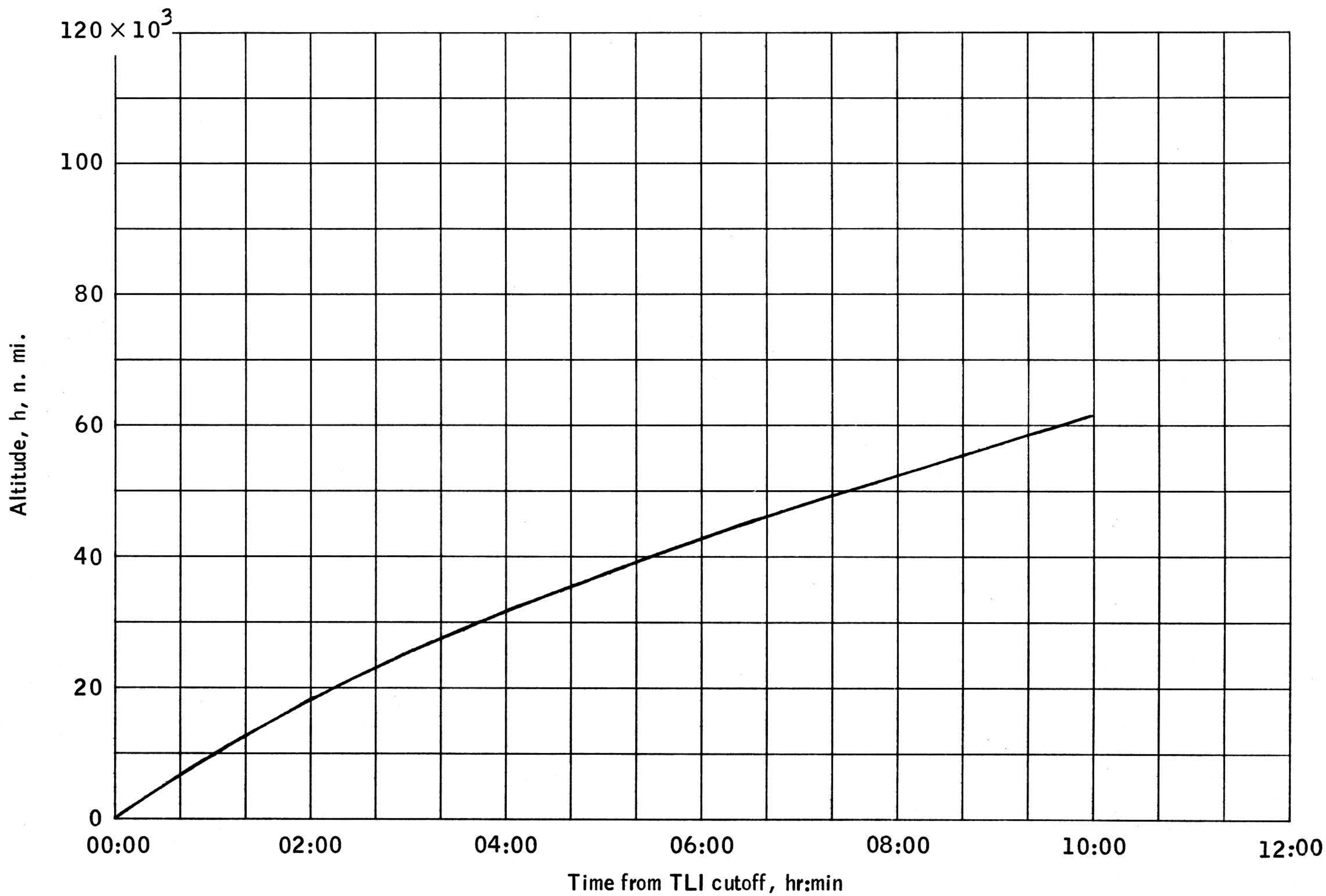


Figure 7.- Time history of altitude for first 10 hours of translunar coast.

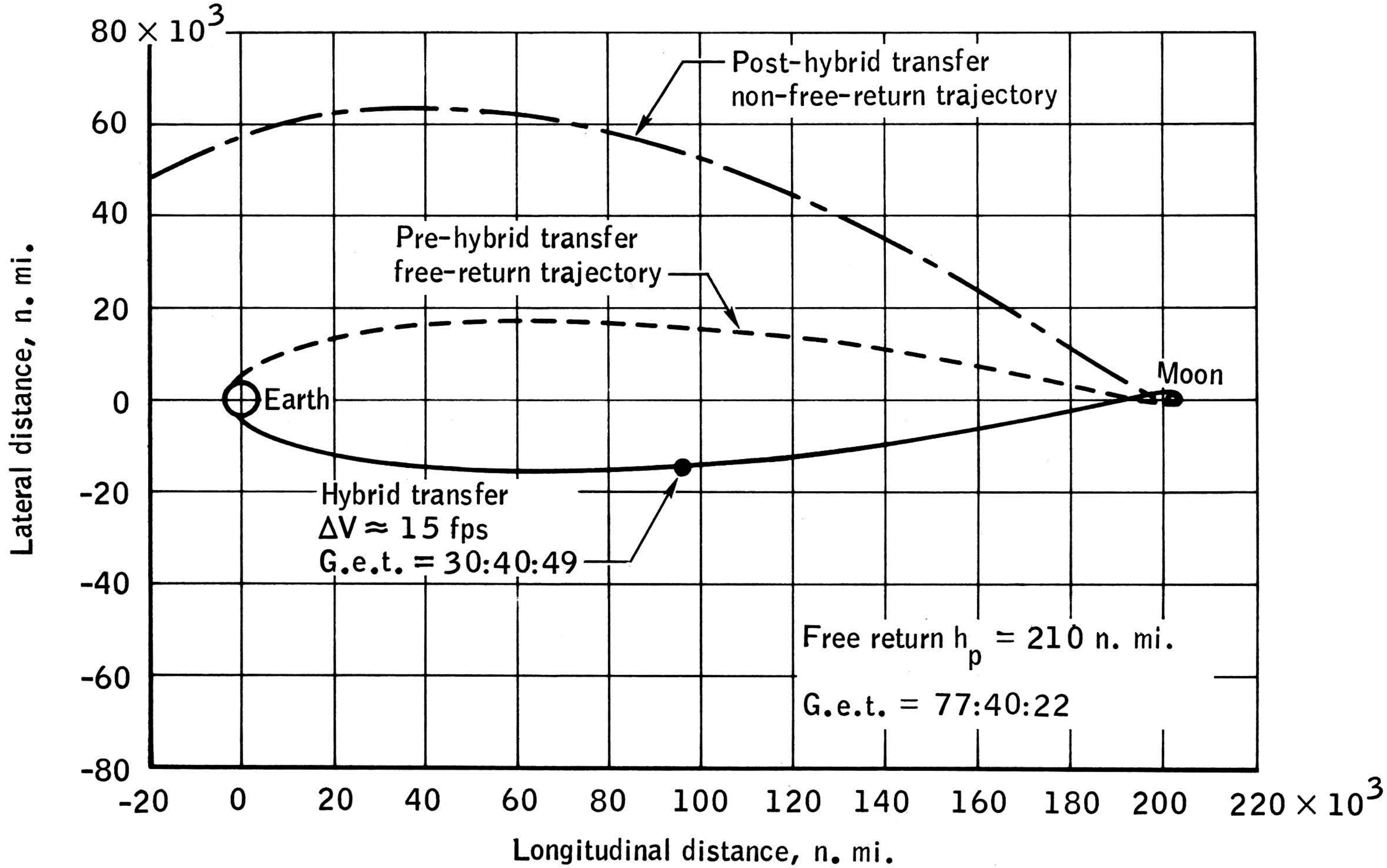


Figure 8. - H-2 hybrid mission profile.

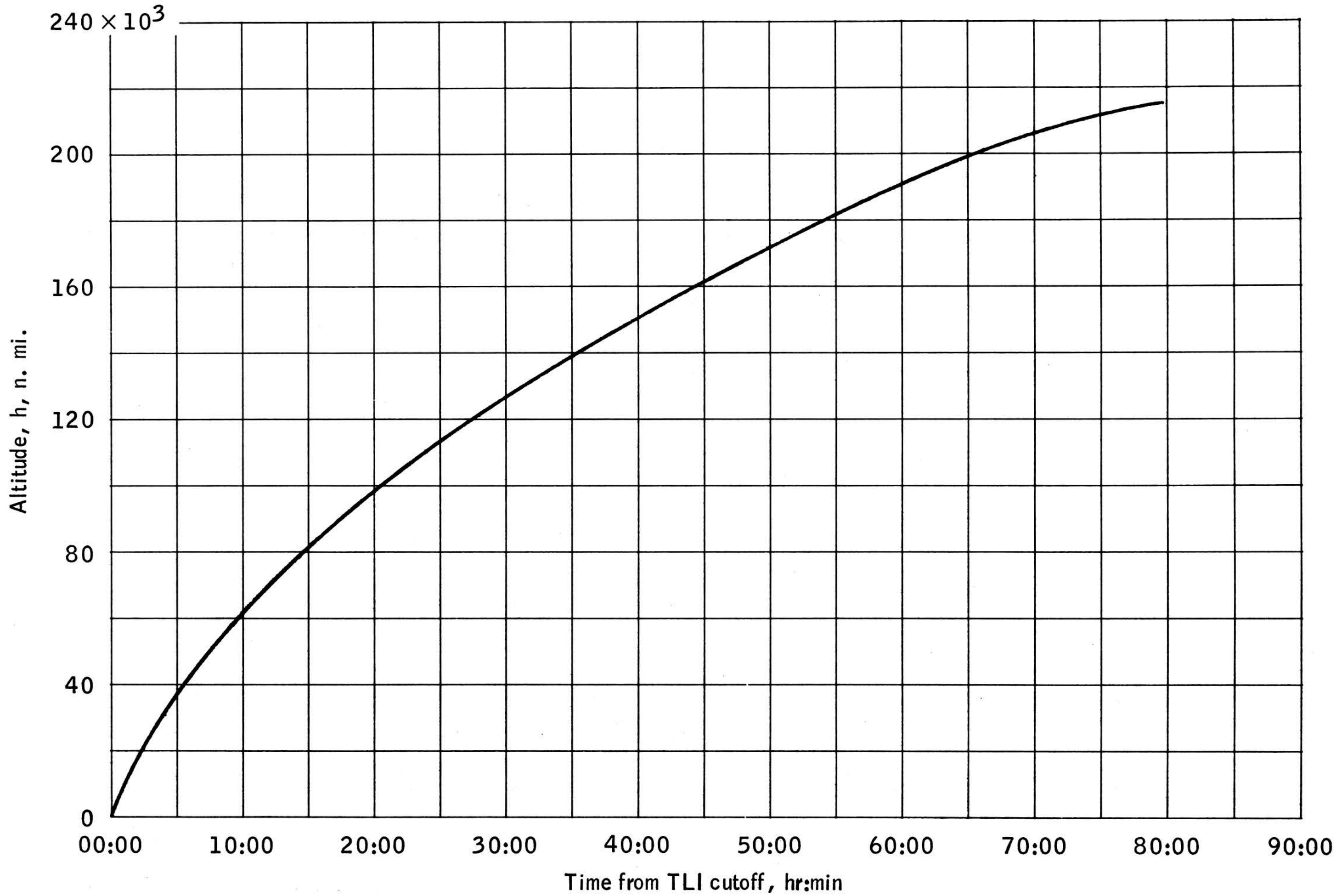


Figure 9.- Altitude time history from translunar injection to lunar orbit insertion.

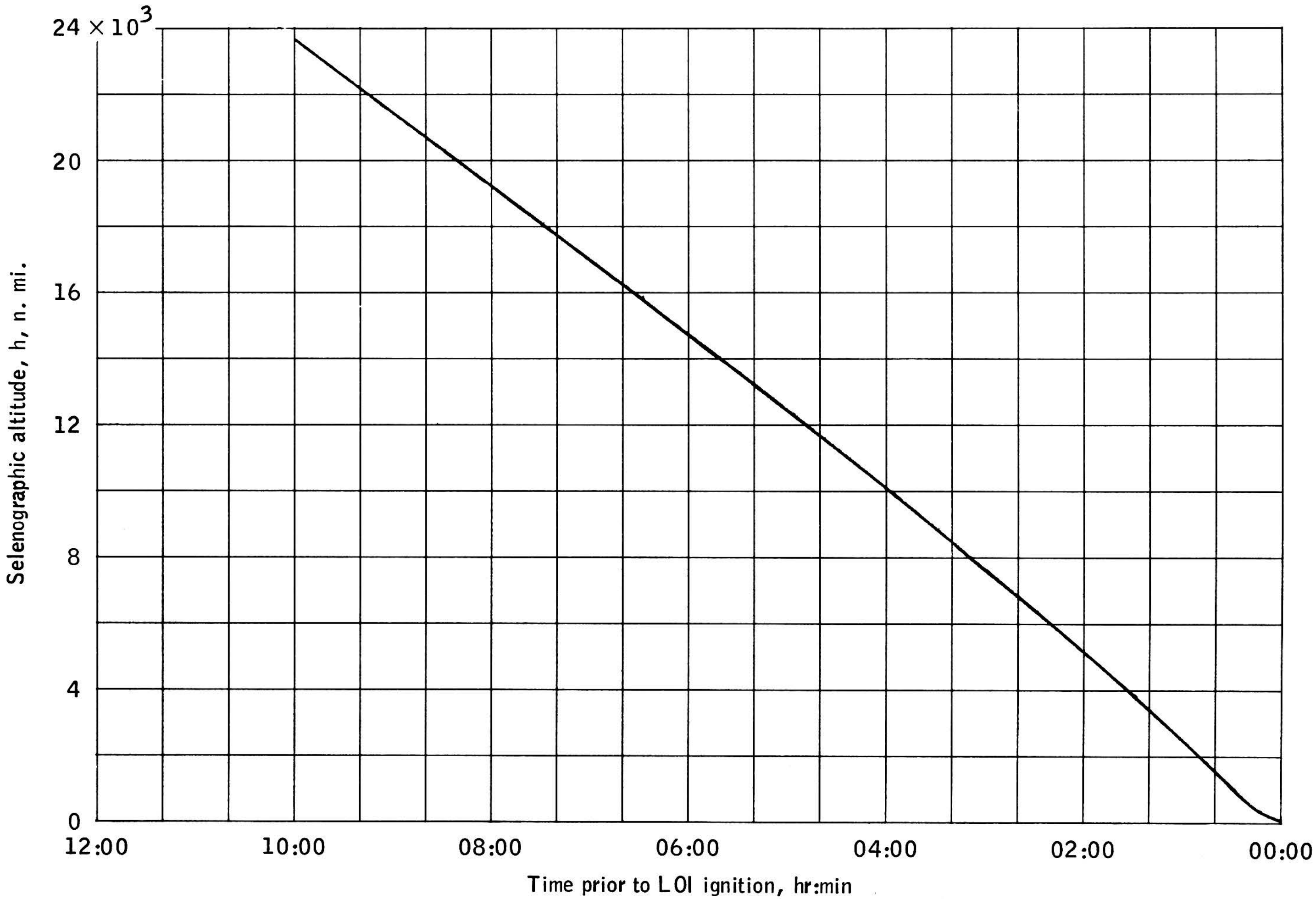


Figure 10.-Time history of altitude 10 hours prior to LOI ignition.

LUNAR ORBIT INSERTION (LOI)

The LOI maneuver is designed to insert the CSM into approximately a 58- by 170-n. mi. lunar parking orbit (LPO). A summary of the lunar orbit activities is shown in table V. The lunar orbit insertion geometry is shown in figure 11, and the CSM lunar orbit groundtrack from LOI to LM landing is shown in figure 12. A description of the burn is as follows.

LOI burn initiation

Time, hr:min:sec, g.e.t.	77:26:05.3
Mass, lb	95 530.4
Altitude above LLS, n. mi.	69.2
Selenographic latitude, deg	2.4
Selenographic longitude, deg	-169.9
Velocity, fps	8251.6
Perilune altitude above LLS, n. mi. . . .	58.0
Selenographic inclination, deg	5.9

LOI termination - start LPO

Time, hr:min:sec, g.e.t.	77:32:02.4
Altitude above LLS, n. mi.	60.1
Selenographic latitude, deg	4.2
Selenographic longitude, deg	166.6
Velocity, fps	5480.4
Selenographic inclination, deg	5.4
Burn duration, sec	357.1
Inertial burn arc, deg	23.4
Plane change, deg	0.5
ΔV, fps	2821.3
SPS propellant used, lb	23 321.8
Orbit period, hr:min:sec	2:08:41
Perilune altitude above LLS, n. mi. . . .	58.9
Apolune altitude above LLS, n. mi.	170.1

TABLE V.- LUNAR ORBIT ACTIVITIES

Activity	Revolution number	Time, hr:min:, g.e.t.
LOI	Beginning of 1	77:32
DOI	Beginning of 3	81:46
IVT to LM for housekeeping activities	3	83:10
IVT to CSM	4	85:00
Begin 8.5-hour rest period	5	86:10
End rest period	9	94:40
Begin IVT to LM	10	96:45
Undocking and separation	12	99:18
CSM circularization	12	100:37
PDI	14	103:33
Landing	14	103:45
Begin first EVA	16	108:10
Terminate first EVA	18	112:02
First CSM plane change	19	113:46
Begin ~9.5 hour rest period	20	114:50
End rest period	25	124:20
Begin second EVA	26	127:50
Terminate second EVA	28	134:25
LM ascent	31	137:15

TABLE V.- LUNAR ORBIT ACTIVITIES - Concluded

Activity	Revolution number	Time hr:min:, g.e.t.
Docking	33	140:50
IVT to CM	34	142:37
LM jettison	34	143:11
CSM separation	34	143:16
LM deorbit burn	35	144:39
LM lunar impact	35	144:56
Begin ~8-hr rest period	35	145:15
End rest period	39	153:15
Second CSM plane change	40	154:22
Photography of Descartes and Davy crater chain	41, 42, 43	158:43 158:50
Transearth injection	end of 46	167:39

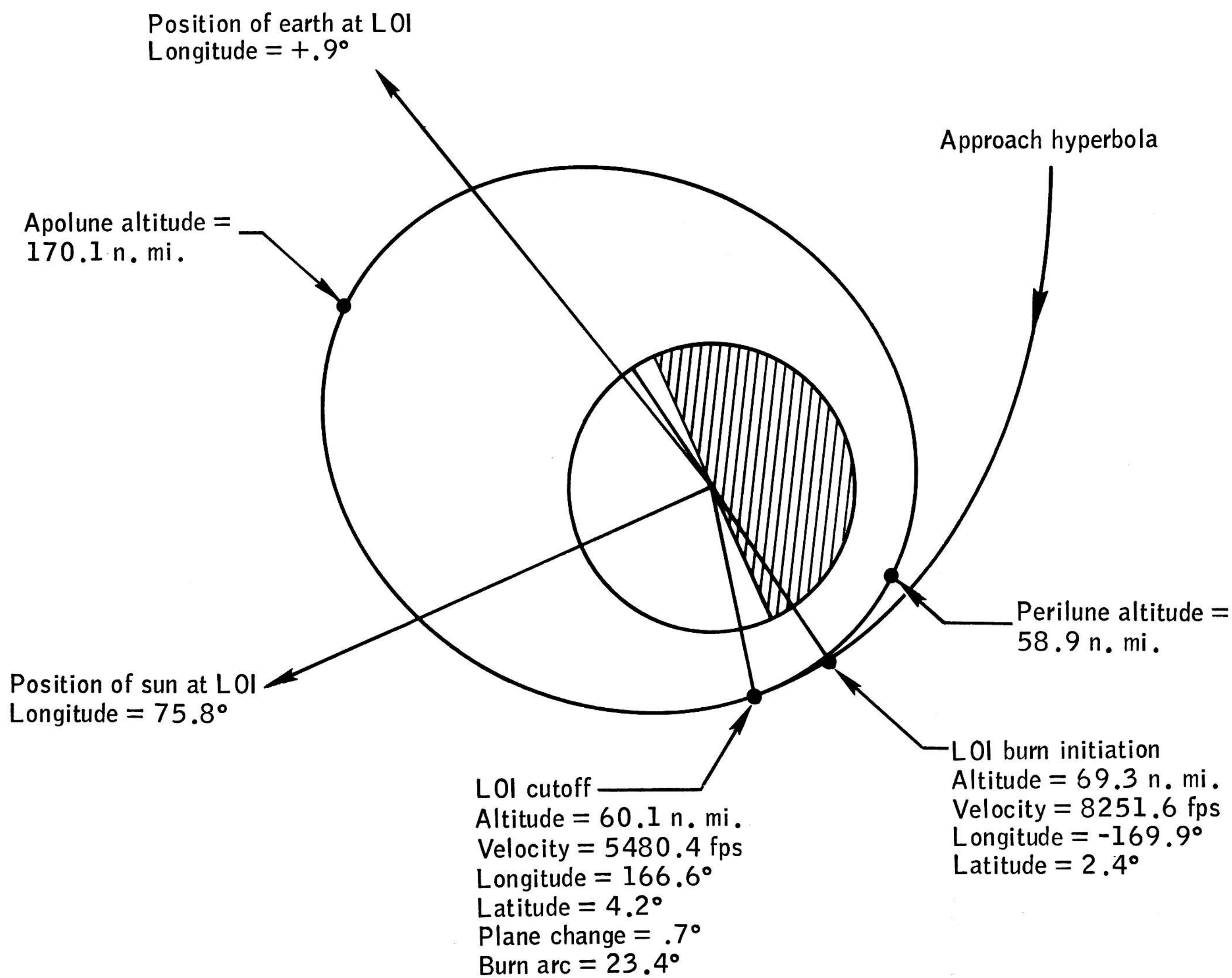


Figure 11.- LOI geometry.

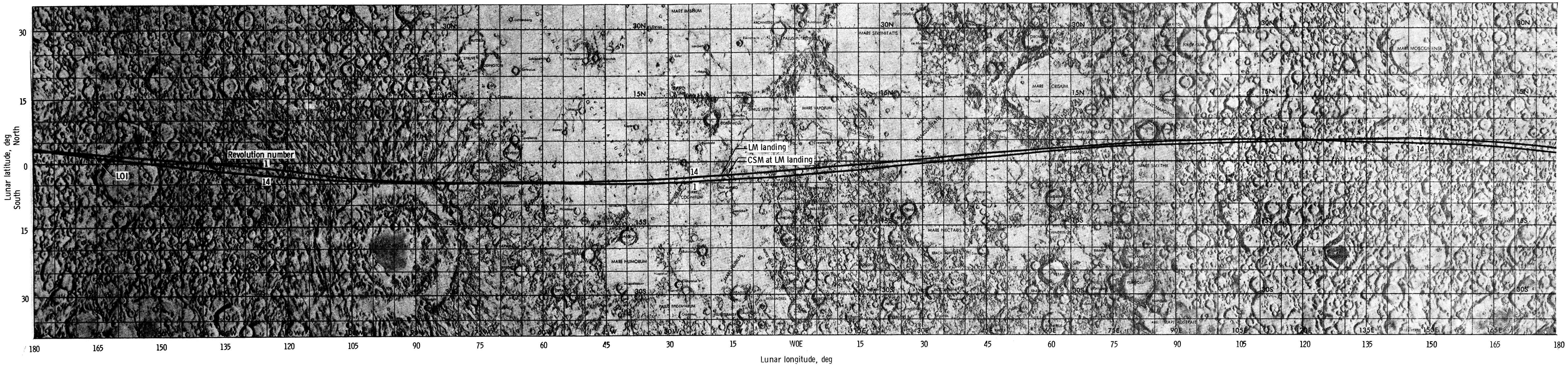


Figure 12.- Lunar groundtrack from LOI to LM landing.

DESCENT ORBIT INSERTION (DOI)

The DOI maneuver which is performed by the SPS on this mission occurs after two revolutions in the 59- by 170-n. mi. orbit. The targeting is biased to place the spacecraft in an initial 9- by 59.2-n. mi. lunar orbit. This is done so that the predicted lunar potential perturbation will place the spacecraft in a 50 000-foot by 60-n. mi. orbit at PDI on rev 14. The orbit is designed so that the perilune is at 2.9°W selenographic longitude at PDI. The altitudes are relative to the landing site radius. The burn is initiated near perilune of the second revolution in the 59- by 170-n. mi. orbit. The characteristics of the burn are the following.

DOI burn initiation

Time, hr:min:sec, g.e.t.	81:45:49.9
Mass, lb	72 154.0
Altitude above LLS, n. mi.	58.9
Selenographic latitude, deg	3.6
Selenographic longitude, deg	174.9
Velocity, fps	5486.5
Perilune altitude above LLS, n. mi.	58.9
Apolune altitude above LLS, n. mi.	170.2

DOI termination

Time, hr:min:sec, g.e.t.	81:46:12.7
Altitude above LLS, n. mi.	58.8
Selenographic latitude, deg	3.7
Selenographic longitude, deg	173.7
Velocity, fps	5278.6
Selenographic inclination, deg	5.4
Burn duration, sec	22.8
Inertial burn arc, deg	1.2
Plane change, deg	0.0
ΔV, fps	210.5
SPS propellant used, lb	1490.9
Orbit period, hr:min:sec	1:54:12
Perilune altitude above LLS, n. mi.	8.9
Apolune altitude above LLS, n. mi.	59.2

CSM/LM COAST FROM LOI TO UNDOCK

The main activities in lunar orbit prior to undock include the CSM DOI maneuver at the beginning of the third revolution, LM housekeeping activities, and an 8.5-hour rest period. A summary of the major activities is given in table VI.

TABLE VI.- LUNAR ORBIT ACTIVITIES PRIOR TO UNDOCK

Activity	Revolution number	Time, hr:min, g.e.t.
LOI	Beginning of 1	77:26
DOI	Beginning of 3	81:46
IVT to LM for housekeeping activities	3	89:30
IVT to CSM	4	91:10
Begin 9-hr rest period	5	92:30
End rest period	9	101:30
IVT to LM for LM checkout	11	103:55
Undock	12	108:01

CSM/LM UNDOCK TO LANDING

The period from undock to landing includes the following maneuvers: CSM/LM undock and separation, CSM circularization, and the powered descent [fig. 13(a)]. The undocking and separation, accomplished as a single maneuver, occurs during the twelfth orbit. Approximately 79 minutes later, the CSM performs a circularization maneuver, necessary for rendezvous purposes and landmark tracking. This maneuver changes the CSM orbit from the 59-n. mi. by 8.2-n. mi. LM descent orbit to a 63- by 56-n. mi. parking orbit. The powered descent maneuver is initiated near the 8.2-n. mi. perilune of the LM descent orbit. The powered descent maneuver [fig. 13(b)] consists of three operational phases: braking, for efficiency; approach, for crew visibility; and landing, for transition from automatic manual control for landing on the lunar surface. The transition from braking to approach phase is termed high gate, and the transition from approach to landing phase is termed low gate.

The lunar groundtracks for the spacecraft are included in figure 14(a); the same track is shown in figure 14(b) for the CSM. The LM track during the powered descent is shown in figure 14(c); the locations of powered descent initiation (PDI), high gate, and landing are indicated. The CSM track during powered descent is shown in figure 14(d). The MSFN tracking summary is given in table VII. The LM-CSM relative motion from undocking to landing is shown in figure 15.

More details of each maneuver from undocking to landing are given in the following subsections.

CSM/LM Undock and Separation

During the twelfth orbit and 4.5 hours prior to landing, the LM and CSM undock in preparation for descent. Stationkeeping for LM inspection purposes is no longer required; therefore, immediately after undocking, the CSM performs a manual maneuver of 1.0 fps directed radially downward toward the center of the moon. The maneuver provides an increase in the LM-CSM separation distance.

CSM Circularization

The DOI maneuver as described in another section was performed by the SPS and placed the docked CSM and LM into the 59- by 8.2-n. mi. LM descent orbit. Approximately 79 minutes after undocking or at the apolune of the LM descent orbit, the CSM performs a circularization maneuver. This manever places the CSM in a parking orbit of 63 by 54 n. mi. This maneuver is necessary for two reasons: (1) the rendezvous phasing is improved for a near circular CSM orbit; (2) the relative motion of the CSM to the surface is nearly constant, which makes landmark tracking less difficult.

The maneuver is a 3.9-second posigrade burn made with the SPS.

The characteristics of the burn are the following.

Circularization burn initiation

Time, hr:min:sec, g.e.t.	100:37:22.0
Mass, lb	36 592.1
Altitude above LLS, n. mi.	58.3
Selenographic latitude, deg	1.2
Selenographic longitude, deg	-166.2
Velocity, fps	5297.9
Perilune altitude above LLS, n. mi.	9.0
Apolune altitude above LLS, n. mi.	59.3

Circularization termination

Time, hr:min:sec, g.e.t.	100:37:25.9
Altitude above LLS, n. mi.	58.3
Selenographic latitude, deg	1.3
Selenographic longitude, deg	-166.4
Velocity, fps	5366.5
Selenographic inclination, deg	174.6
Burn duration, sec	3.9
Inertial burn arc, deg	0.2
Plane change, deg	0.0
ΔV, fps	70.3
SPS propellant used, lb	254.8
Orbit period, hr:min:sec	1:58:37
Perilune altitude above LLS, n. mi.	54.3
Apolune altitude above LLS, n. mi.	63.6

Powered Descent

The powered descent maneuver (PDI) which consists of braking phase, approach phase, and landing phase [fig. 12(b)] is initiated at perilune of the LM descent orbit (51 000 ft) and at a central angle of 16.1° from the landing site. The ullage maneuver and DPS engine gimbal angle trim periods are performed at the constant inertial attitude required by PGNCS guidance at DPS ignition. The braking phase is designed for efficient reduction of orbital velocity and, therefore, uses maximum thrust from the DPS for most of the phase; however, the DPS is throttled during the final 2 minutes of this phase for guidance control. The spacecraft is in a windows-up attitude so that the landing radar can be used. The braking phase guidance is based on quadratic acceleration equations. The targets for this phase are not actually achieved because the guidance automatically switches when time to go (to achieve targets) becomes less than 62 seconds from braking phase targets to approach phase targets.

The approach phase, which begins at approximately a 7500-foot altitude (high gate), provides for pilot visual monitoring of the approach to the lunar surface. That is, the guidance is targeted to provide spacecraft attitudes which permit crew visibility of the landing area through the forward window throughout this phase. The same quadratic acceleration guidance law used in the braking phase is also used here. The altitude target is approximately 80 feet. However, the approach phase is operationally considered to be terminated at an altitude of 500 feet (low gate) where the landing phase begins. The landing phase is designed to provide continued visual assessment of the landing site and to provide compatibility for pilot takeover from the automatic control. There is no change in guidance law or targets at this point because the approach phase targets have been selected to satisfy these additional constraints. Under automatic guidance, a vertical descent is initiated when time to go becomes less than 12 seconds. The vertical descent portion of the landing phase starts at an altitude of 100 feet and terminates at landing on the lunar surface. The guidance is a velocity error nulling routine. Normally, a 3-fps rate of descent is used throughout the vertical descent.

TABLE VII.- LM LIGHTING AND MSFN TRACKING SUMMARY -

CSM/LM UNDOCK TO LANDING

Station	Time, hr:min:sec g.e.t.	Azimuth, deg	Range, n. mi.
Station acquisition			
Goldstone	a		
Guaymas	a		
Corpus Christi	a		
Merritt Island	a		
Grand Bahama	a		
Bermuda	a		
Grand Canary	a		
Madrid	a		
Ascension	a		
[Enter penumbra at $99^h55^m24^s$; enter umbra at $99^h55^m33^s$.]			
Station termination			
Goldstone	100:17:13	90.7	216 773
Guaymas	100:17:15	89.2	216 506
Corpus Christi	100:17:15	95.7	215 965
Merritt Island	100:17:19	109.1	215 460
Grand Bahama	100:17:20	108.1	215 385
Bermuda	100:17:20	142.2	215 228
Madrid	100:17:40	-105.9	216 199
Grand Canary	100:17:42	-101.1	215 703
Ascension	100:17:52	-57.3	216 060
[Enter penumbra at $100^h43^m59^s$; enter sunlight at $100^h44^m09^s$.]			
Station acquisition			
Madrid	101:05:56	-96.5	216 653
Hawaii	101:05:58	73.1	218 440
Goldstone	101:06:03	98.4	216 307
Guaymas	101:06:05	95.4	216 033
Corpus Christi	101:06:10	103.8	215 589
Grand Bahama	101:06:12	126.7	215 186
Merritt Island	101:06:13	126.3	215 239
Bermuda	101:06:16	179.8	215 179
Grand Canary	101:06:29	-93.7	216 121
Ascension	101:06:31	-64.0	216 530
[Enter penumbra at $101^h49^m30^s$; enter umbra at $101^h49^m40^s$.]			
Station termination			
Goldstone	102:10:56	110.9	215 727
Hawaii	102:11:20	78.8	217 564
Guaymas	102:11:30	106.7	215 499
Corpus Christi	102:11:32	122.7	215 215
Grand Bahama	102:11:37	-171.1	215 072
Merritt Island	102:11:38	176.98	215 095
Bermuda	102:11:40	-136.31	215 257
Madrid	102:11:45	-86.6	217 261
Grand Canary	102:11:50	-86.9	216 782
Ascension	102:11:55	-69.7	217 271
[Enter penumbra at $102^h38^m05^s$; enter sunlight at $102^h38^m15^s$.]			

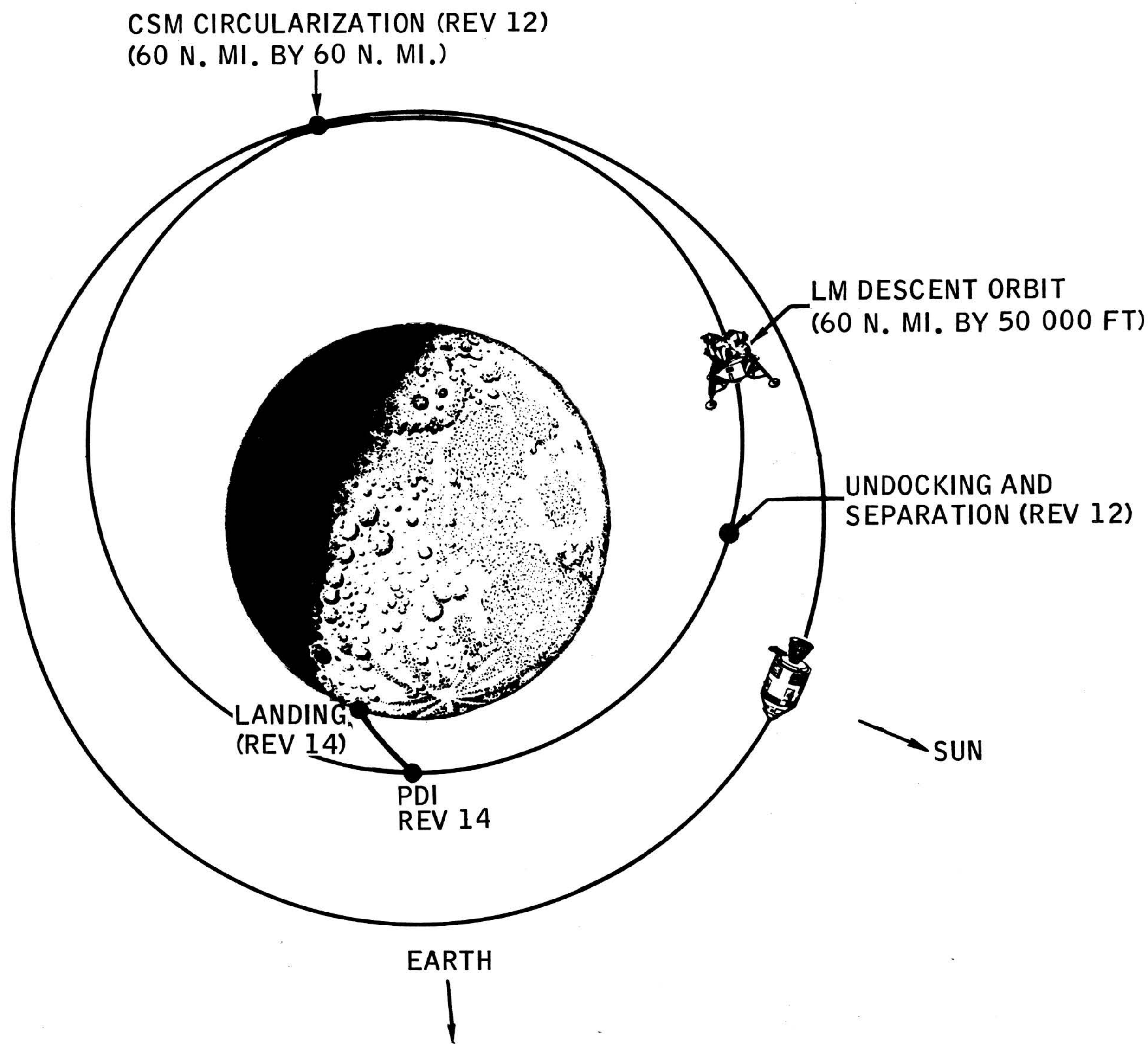
TABLE VII.- LM LIGHTING AND MSFN TRACKING SUMMARY -

CSM/LM UNDOCK TO LANDING - Concluded

Station	Time, hr:min:sec g.e.t.	Azimuth, deg	Range, n. mi.
Station acquisition			
Hawaii	102:59:55	82.7	216 945
Goldstone	103:00:00	125.2	215 454
Guaymas	103:00:03	121.5	215 233
Corpus Christi	103:00:04	157.0	215 087
Merritt Island	103:00:11	-137.4	215 140
Bermuda	103:00:13	-116.8	215 462
Grand Bahama	103:00:14	-126.8	215 144
Madrid	103:00:18	-79.0	217 770
Grand Canary	103:00:21	-81.6	217 343
Ascension	103:00:28	-72.5	217 897
Station termination			
Ascension	103:37:54	-74.1	217 202
Hawaii	b		
Goldstone	b		
Guaymas	b		
Corpus Christi	b		
Merritt Island	b		
Grand Bahama	b		
Bermuda	b		
Madrid	b		
Grand Canary	b		

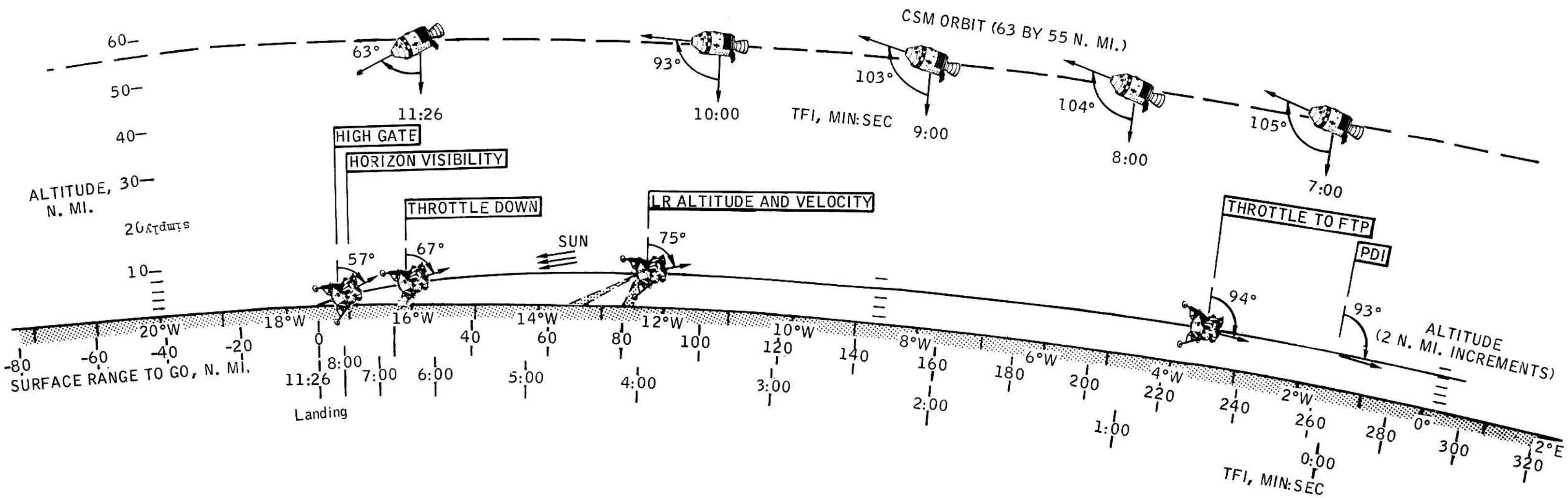
[a]Stations tracking at undock. Undock in sunlight.

[b]Stations tracking at landing. Land in sunlight.



(a) LM descent orbital events.

Figure 13.- LM descent.

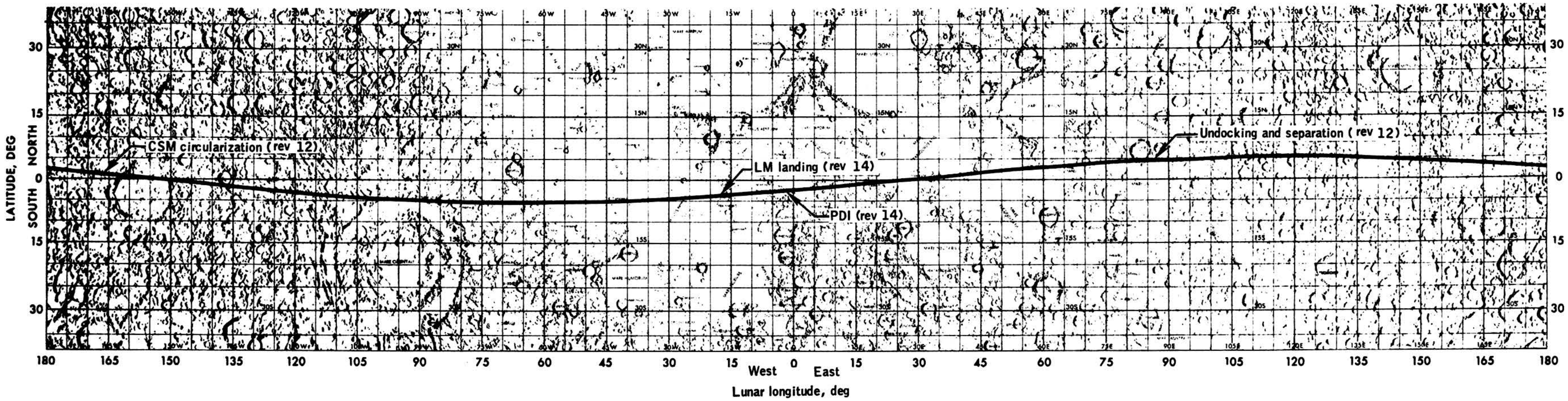


SUMMARY					
EVENT	TFI, MIN:SEC	V_i, FPS	$\dot{H}$, FPS	H, FT	ΔV, FPS
POWERED DESCENT INITIATION	0:00	5558	-3	51 488	0
THROTTLE TO MAXIMUM THRUST	0:26	5526	-3	51 410	33
LANDING RADAR ALTITUDE AND VELOCITY UPDATE	4:00	3309	-84	40 071	2289
THROTTLE RECOVERY	6:32	1422	-84	23 720	4291
HORIZON VISIBILITY	8:10	641	-184	11 411	5193
HIGH GATE	8:30	478	-182	7 464	5396
LOW GATE	10:10	57 (71)*	-13	498	6201
LANDING	11:25	-15 (0)*	-3	5	6620

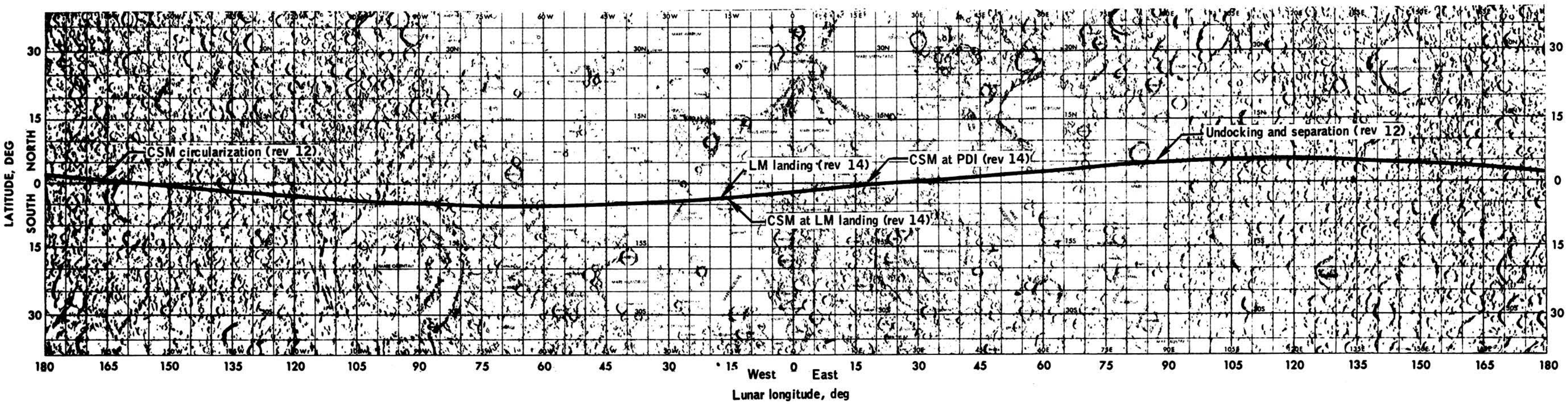
* (HORIZONTAL VELOCITY RELATIVE TO SURFACE)

(b) Powered descent profile.

Figure 13. - Concluded.

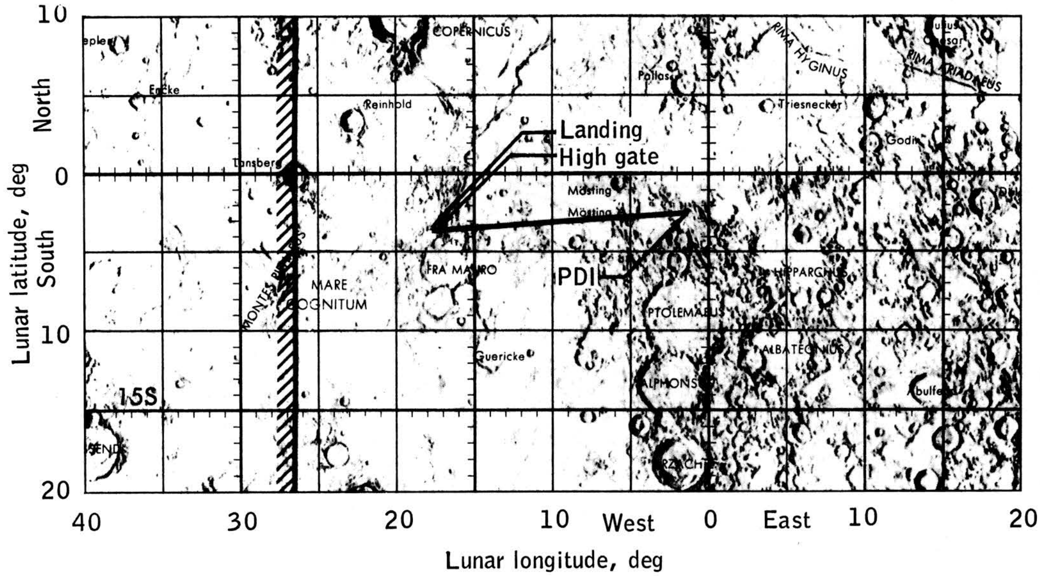


(a) LM groundtrack from undocking to landing.

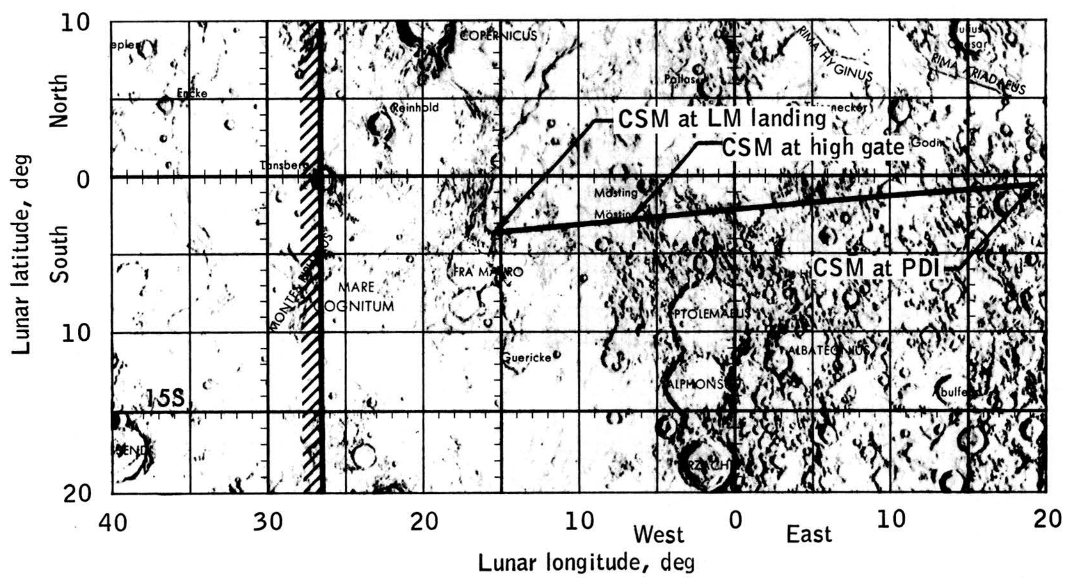


(b) CSM groundtrack from undocking to landing.

Figure 14.- LM and CSM groundtracks during powered descent.



(c) LM groundtrack during powered descent.



(d) CSM groundtrack during powered descent.

Figure 14.- Concluded.

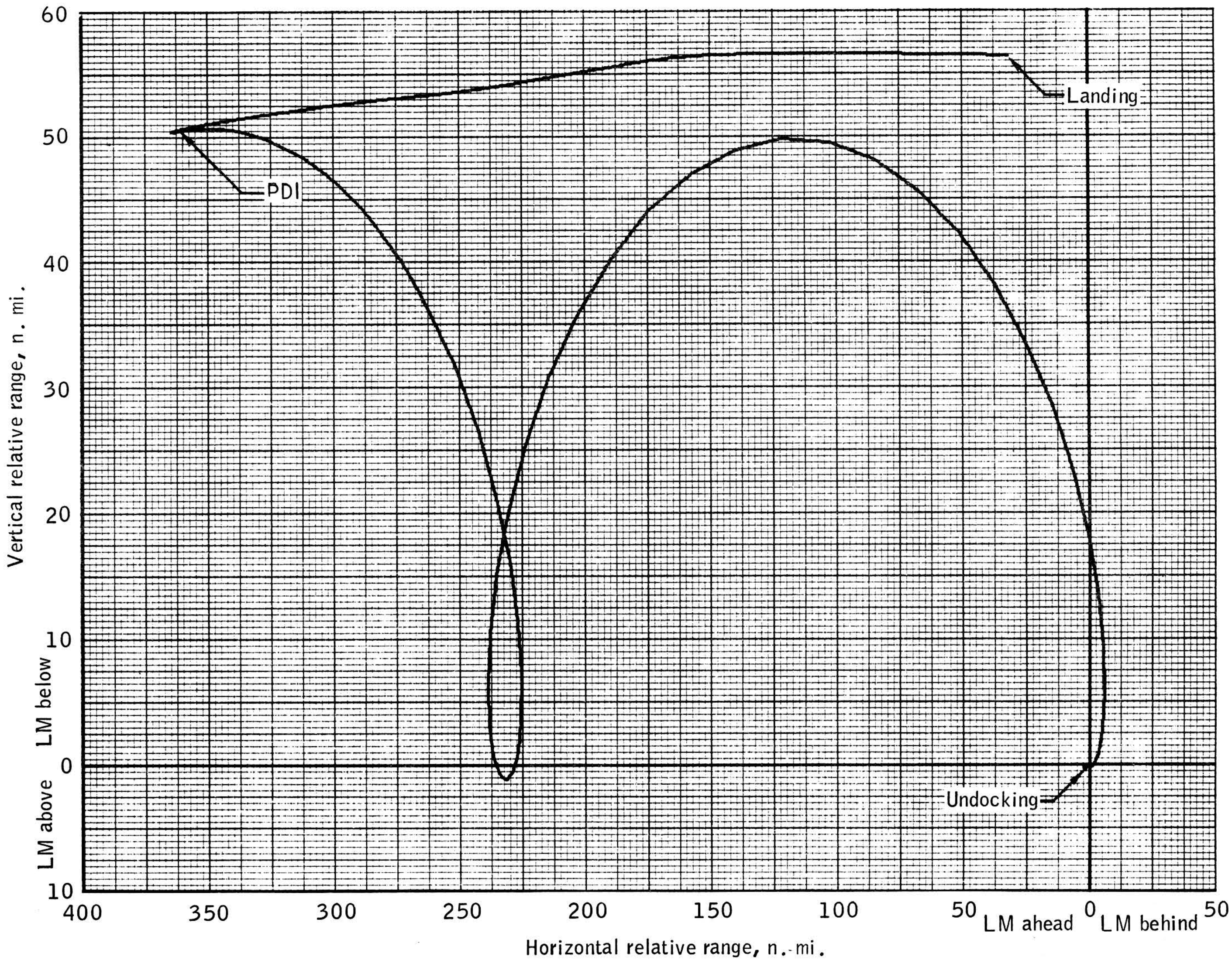


Figure 15.- LM and CSM relative motion from undocking to landing.

NOMINAL LUNAR SURFACE EVA

The nominal plan is for the two LM crewmen to spend 8 hours out on the lunar surface in their EMU's, or 16 manhours of EVA time. This is divided into two periods of 4 hours each, separated by a housekeeping, sleep, and eat period of about 14 hours. The nominal landing configuration for the LM is with the ladder on the +Z landing strut down sun, facing west.

The Goldstone or Parks 210-foot antennas are not available for LM-earth communications throughout all of EVA-1. This situation requires that primary consideration be given to early deployment and activation of the erectable S-band antenna.

EVA-1 Timeline Description and Rationale

The first lunar surface excursion on Apollo 13 commences with depressurization of the LM ascent stage cabin. The forward hatch is opened by the LMP, following which the CDR assumes a kneeling posture facing away from the hatch. He backs out of the hatch in a prone posture. The lunar equipment conveyor (LEC) is dropped by the CDR. The CDR checks for adequate voice communication and telemetry transmission as soon as he is clear of the hatch opening. The LMP then hands the jettison bag to the CDR, who tosses it off to his right, between the +Z and -Y struts. The CDR completes the platform procedures by deploying the modularized equipment stowage assembly (MESA). He does this by pulling a release loop. The CDR ascertains proper MESA deployment and then descends the ladder to the ground.

As soon as the LMP has handed the jettison bag to the CDR, he begins the photography using his 70mm electric data camera and the 16mm surface sequence camera. The CDR's 70mm electric data camera has previously been readied for transport to the surface.

The CDR begins a familiarization procedure as soon as he reaches the surface. As soon as he feels confident to perform a transport procedure with the LEC, the 70mm data camera is transferred to the surface. This camera is the only still camera on the surface during EVA-1. The CDR commences his preliminary photography of the surface as part of the familiarization.

Following this, the CDR takes the contingency sampler out of his suit pocket, deploys it, and scoops up 1 or 2 pounds of lunar material. The contingency sampler is temporarily placed on the +Z secondary struts.

The LMP egresses at this point, going through the same steps as the CDR, except for the MESA deploy and jettison bag discard. He leaves the LM hatch slightly ajar and leaves a 70mm camera by the door as backup to the surface camera.

As soon as the CDR assures himself that the LMP has safely egressed, he proceeds to quad I to unstow the S-band erectable antenna. This unit is carried to the vicinity of the +Z strut for deployment.

The LMP goes through a familiarization sequence which includes taking the 70mm data camera from the CDR. He documents some of the early S-band antenna deployment procedures, then begins his first task, unloading and assembling the TV camera. The MESA blanket is peeled off the MESA, and the tripod is removed. The TV camera stowage box is opened and the camera removed for attachment to the tripod. The 100-foot camera cable is unreeled from the right side of the MESA, following which the camera is placed approximately 50 feet off the +Y strut. The LMP places the 70mm camera on the now empty TV bracket and begins the preparation for transfer of portable life support system (PLSS) expendables to the ascent stage. This involves unfolding the table from the front of the MESA, and hanging the equipment transfer bag (ETB) from it. The ETB is stowed under the folded-up table before launch. The ETB contains two extra weigh bags and a packaged 100-foot safety line. These are stowed on the MESA, and two PLSS LiOH canisters and two PLSS batteries are loaded into the ETB, together with the contingency sample bag, which is removed from its handle at this point. The LMP is on call during this sequence to assist the CDR in S-band antenna alinement by steadying it during that procedure. As soon as the S-band erectable antenna is deployed on its tripod, connected to its 30-foot cable, and alined, the LMP ingresses the LM. If any part of the PLSS expendables preparation is unfinished at this point, the CDR picks this up after the LMP is inside the LM ascent stage. The LMP turns the antenna switch to S-band erectable antenna position, checks receiver signal strength on the AGC display, and checks the TV camera circuit breaker. Television transmission is verified from MCC. If signal strength is not up to par, the CDR may be requested to experiment with changing the antenna alinement.

The CDR then picks up the LEC and connects it to the packed ETB. The bag is transferred to the ascent stage on the LEC. The LMP pulls down on the strap of the LEC as it goes through the pulley secured to the alinement optical telescope (AOT) bar. The LMP receives the ETB and unloads the PLSS expendables and contingency sample bag. These are temporarily stowed, and the ETB is passed back to the surface on the LEC. The 16mm camera, two extra magazines, a backup black and white TV camera, and a traverse map are transferred to the surface in the ETB. The CDR unstows the 16mm sequence camera from the ETB and photographs the LMP egress, if possible. The LMP egresses once again, closes the hatch, and descends to the ground. Both men participate in deployment of the American flag. The LMP may find it necessary to re-aim the TV camera to cover the flag deployment ceremony. The sequence camera may also be used during this period.

The two crewmen then proceed to the scientific equipment (SEQ) bay (quad II). The CDR goes by way of the +Y strut, picking up the TV and repositioning it to cover SEQ bay activities. He notes and photographs any anomalies or unusual features on the LM, and documents LM foot pad penetration, DPS erosion, and bell clearance. The LMP does the same kind of inspection, but around the -Y side and with the sequence camera.

The crewman who reaches the SEQ bay first opens the bay door by manipulating two lanyards. The door opens like an overhead garage door to reveal packages 1 and 2 of the Apollo Lunar Scientific Experiment Package (ALSEP). A small horizontally moving door swings out to shield the crewmen from the hot radioisotope thermoelectric generator (RTG) cask on the left of the SEQ bay. The LMP removes package 2 as soon as the door is open. The packages can be removed like a suitcase from a shelf, or they can be removed on an extendable boom and lowered to the surface on a cable by means of a ratchet device. The LMP takes the package to the vicinity of the fuel cask and unstows the hand tool carrier (HTC) and sets it up. This becomes a receptacle for the ALSEP tools which are next removed from package 2. Package 2 contains the HTC, the Apollo lunar surface drill (ALSD), the ALSEP tools, and the RTG. Meanwhile, the CDR removed package 1 from the SEQ bay. This package contains all of the experiment units: the Passive Seismic Experiment (PSE), the Cold Cathode Gage Experiment (CCGE), the Charged Particle Lunar Experiment (CPLE), and the Heat Flow Experiment (HFE).

The CDR aids the LMP in unstowing the tools from package 2 as soon as package 1 is on the ground. The ALSD is also on package 2. It comes off and is placed on a foot pad (probably -Z), following which a universal hand tool (UHT) is inserted in a socket in package 2, and the package is lowered by the CDR to a horizontal position on the surface.

The LMP meanwhile lowers the RTG fuel cask on the left side of the SEQ bay to a horizontal position and uses a special tool to remove the dome of this cask. Another special tool then engages and releases the RTG fuel capsule inside the cask. The capsule is transferred to the RTG of package 2 and heat up of the generator begins. As soon as he fuels the RTG, the LMP picks up the ALSD and the HTC and proceeds back around the LM to the MESA.

The CDR pushes the deployment booms back inside the SEQ bay, closes the SEQ bay doors, and returns package 2 to an upright position. He has previously taken two mast sections from the tool holder on package 2 and locked them together. These will ultimately be the ALSEP antenna mast, but their first purpose is a carry handle. The two ALSEP packages are joined barbell fashion by the mast. The CDR then performs a TV panorama sequence and his three photographic panoramas.

Meanwhile, the LMP has unloaded the first sample return container (SRC) from the MESA, placed it on the table, and opened it. Inside are six core sample drill stems, caps for the stems, and supplies for the selected sample the crew will perform after ALSEP is deployed. These items, plus tools from the MESA, are loaded into the HTC. The solar wind composition experiment is deployed, and the close-up stereo camera (to be used in EVA-2) is placed in the sun to preserve its battery capability.

The crewmen meet at the MESA and each installs the other's sample collection bag (tote bags). These were stowed in the MESA. Both crewmen then proceed 300 feet or more in as westerly a direction as possible to a suitable ALSEP deployment area.

Once at the ALSEP deployment site, the LMP deposits his burden at the approximate spot of the Heat Flow Experiment deployment. He aids the CDR in emplacing the RTG-bearing package 2 and helps connect the power cable to package 1. Package 1 is then placed on the surface in a north-south alinement, and the LMP removes the HFE. He withdraws to the HFE deployment area unreeling the cable as he goes. He places the electronics package along an east-west line and disassembles the HFE probe packages. By deploying the probe cables, he can ascertain where to place the bore holes in which the probes will be placed. He then assembles the ALSD and commences the implantation of the two bore strings.

The CDR assists the LMP in removal of the HFE package, then prepares for PSE deployment. The PSE rests on a small stool which the CDR places 10 feet east of package 1. A small hole is gouged out of the surface under the stool. The PSE is removed from package 1 and placed on the stool. All of these experiments are secured to their ALSEP package by special quick-release fasteners called boyd bolts. The boyd bolts are released by the UHT. The CDR places the PSE on the stool, alines it to the east, and deploys a thermal skirt. The completely deployed PSE resembles a sombrero, with the thermal skirt forming the hat brim. The CDR levels the PSE with reference to a spirit level mounted on the top, and reports a sun compass reading to Houston. He returns to the central station (package 1) and removes the CCGE. This unit must be isolated from the rest of the ALSEP experiments because it is a delicate atmosphere sensor; therefore, it is deployed some 60 feet SW of the central station.

The upside-down (as stowed) CPLEE is next to come off the central station. This solar partical sensor is placed on smoothed ground 10 feet south of the central station. This leaves the central station base clear save for the antenna aiming mechanism, the antenna, and the dust detector.

The CDR releases the boyd bolts that hold down the top of the central station and assists the top into its raised configuration which reveals the central station sunshield. He mounts the antenna mast, places the aiming mechanism (a gimbal device with leveling and alinement provision) on the mast, and the antenna on the aiming mechanism. The antenna is leveled and alined to predetermined values.

The CDR completes ALSEP deployment by pushing a button which releases a dummy load across the RTG leads and permits power up of central station electronics. The station requires several minutes to come up to full power for transceiver operation. The CDR turns a special switch at ground request to initiate operation. If necessary, the CDR can also switch on the auxiliary ALSEP transmitter B and can cycle the four experiments with a third switch. The switches are actuated by manipulation of the UHT.

During the pause for power up, the CDR photographs the various experiments, the general layout, and the area. The last is accomplished with a photographic panorama. If the LMP has encountered difficulties in drilling the HFE bore holes, the CDR will deploy the HFE probes (short tubes of sensors and heaters jointed in the center) placing them down the holes as soon as the LMP has finished implanting the bore stem sections of which the holes are composed. The CDR photographs the bore stem ends with the HFE probe wires in them, and the HFE electronics box to complete ALSEP documentation.

During the CDR activities described above, the LMP has been drilling the two bore holes into the lunar surface. The ALSD is a rotary-percussive drill. The drill has a quick release chuck which drives sections of hollow Fiberglas-boron stems down into the regolith. The sections are added one at a time (after an initial two which include the closed bit) to a depth of 3 meters, or nearly 10 feet, for each of the two bore holes. The stems remain empty because the bit is closed, and the material is pushed aside or passes up the outside of the stem string to the surface on helical threads on the outside of each section. The twelve sections (six per hole) are stowed in a special rack, part of the ALSD package. The two HFE heat probes are pushed to the bottom of these holes with the extendable emplacement tool. Special covers and sunshields are also placed in each hole with this tool. The tool has graduations on it so that the crewman can report the depth of the probe and the height above the surface of the topmost (final section) bore stem.

The LMP then removes the quick-release chuck from the ALSD and couples two of the core sample stems onto the drill drive shaft. These stems feature an open bit to permit taking a core sample. The core stems are made of molybdenum and couple with threaded ends, unlike the bore stems which are friction-fit. When coupled in a string of six and driven into the surface, a sample nearly 8 feet deep can be taken. The core sample string is drilled through an orifice in a treadle which is also part of the ALSD package. This treadle permits clockwise rotation of the string; counter-clockwise string rotation results in a clutch engaging the stem which prevents further rotation in that direction. The LMP steps on the treadle as he drills down through it. The treadle holds the emplanted stem while the drill is removed by manual counter rotation (powered rotation is clockwise only) to allow addition of another stem to the string. If possible, the CDR will film the core stem drilling procedure with the sequence camera.

When the required six core stems are in the surface, the CDR assists the LMP in withdrawing the string from the surface. The two crewmen attempt to raise the string by tugging upward on the drill. Failing that, the drill power is actuated sufficient to break the soil cohesion as the string is raised. When two stems are visible, the CDR takes a special Stilson wrench from the ALSD rack and engages the topmost stem section such that the drill head can be removed and discarded. The treadle then is engaged, and the joint between the topmost and fifth section is loosened (but not separated) using the wrench. The wrench is then used to twist the fifth section clockwise and thus release the treadle. The string is then raised until the next section joint is within reach, and the joint loosening procedure is repeated until the string is out of the ground and all five joints have been broken loose. If special difficulties such as the treadles jamming with rock fragments are encountered, an extra wrench (stowed on the MESA) is available in the HTC.

The drill end is capped as soon as practicable after the drill is removed. The bit end is likewise capped as soon as it comes out of the surface. These caps are Teflon and friction-fitted. They are marked alphabetically, with A going on top and B on the bit end of the string. The crewmen then start at the upper end of the string, either manually or with the wrenches separating the stems one by one, capping each as it comes loose from the string, and stowing all in the HTC pouch. The cap markings and order are reported to Houston during this procedure so that the core structure can later be restored in the Lunar Receiving Laboratory at MSC.

The crewmen remove all drill equipment from the vicinity of the HFE area and request Houston for a prediction of EVA time left and EMU operational status from telemetry data. As the status is being uplined, the crewmen take a last look at the ALSEP area and rest preparatory to the return traverse. The CDR takes the 70mm camera; the LMP, the sequence camera.

The return traverse to the LM affords opportunity to collect selected samples (interesting rocks, glassy fragments, fine material) as they are encountered during the return trip. If additional time is available over the 4 hours nominal, a westerly traverse will be made, the traverses being a function of time. Samples are documented prior to and after being gathered. Rocks are placed in the tote bags or in the HTC pouch. (The LMP places samples in the CDR's tote bag; the CDR places samples in the LMP's bag.) If fine-grain or rock fragments are gathered, a dispenser of small sample bags is available to hold these samples.

When the two crewmen regain the LM, the TV camera is repositioned to record closeout activities and cover the second EVA egress. This will be done by the CDR because the LMP carries the HTC. The LMP takes his tote bag off the CDR and holds it for offload by the CDR. The crew will examine some samples between EVA's and relay advance geological information, which may have a bearing on the second EVA Lunar Field Geology Experiment. These samples are left in the CDR's tote bag. Both cameras and all magazines go into the ETB for transfer into the ascent stage. The CDR unloads the samples from the tote bags into two special weigh bags placed on a scale. These weigh bags and scales were stowed in SRC-1. He seals and places the weigh bags in SRC-1. He may fill one of the weigh bags with rocks and fine material from around the LM, if weight is short of maximum. Finally the core stems are placed in the SRC and this rock box is sealed. During this period, the LMP removes the second SRC from the MESA and places it on the +Y pad in the sun. He covers it with a thermal shroud salvaged from the S-band erectable antenna. This procedure ensures a proper between-EVA thermal environment for the SRC to protect the integrity of the Indium seal.

The CDR assists the LMP to clean off his EMU with a special brush; the LMP mounts the ladder and ingresses the ascent stage. The CDR checks the ETB for proper contents and readies the ETB for transfer. The ETB was left hooked to the LEC at the beginning of the EVA. The LMP tugs the ETB into the ascent stage guided by the CDR. The hook end of the LEC is returned to the CDR, who fastens the hooks onto the tote bag. The tote bag is transferred to the ascent stage, where it is stowed by the LMP. In like manner, SRC-1 is transferred to the ascent stage. The CDR then cleans his EMU as best he can and mounts the ladder.

When he is on the platform, the LMP hands him the pulley end of the LEC which he secures to the platform rail, ready to support EVA-2. The CDR moves through the hatch, the hatch is closed, and the repressurizing procedure initiated, thus concluding EVA-1.

EVA-2 Timeline Description and Rationale

The second EVA period commences with depressurization, following which the LMP opens the forward hatch for CDR egress. The CDR assumes a kneeling position, legs out the hatch, and moves through the hatch to lie prone on the LM platform. As soon as he is secure, he hands the LMP the hook/pulley end of the LEC and descends the ladder to the surface.

The LMP hangs the LEC pulley from the overhead handhold and attaches the ETB to the hooks on the LEC preparatory to lowering the bag to the lunar surface. The bag contains the three cameras (two 70mm data cameras and the surface sequence camera), extra magazines, and the lunar traverse map. As soon as this transfer is effected, the LMP joins the CDR on the surface. As before, the hatch is left slightly ajar precluding pressure buildup forcing the hatch closed.

The CDR proceeds to retrieve the second SRC from the pad where it was left on the previous EVA. It is placed on the MESA SRC table, clamped, and opened. The LMP brings the hand tool carrier close by the MESA as the CDR unpacks another 35 bag dispenser, a set of three core tubes, three special sample containers, a new pair of weigh bags, and a solar wind experiment bag. The weigh bags and the solar wind bag go on the MESA; the dispenser and core tubes go on the HTC.

Next the HTC receives its complement of tools, including a large two-piece hoe/shovel for use in digging trenches. The vertical-seeking photography reference tool, the gnomon, is also taken off the MESA and stowed on the HTC.

Both crewmen proceed to install the PLSS-mounted packs, called tote bags, on each other. The LMP's carry pouch (a pocket attached to the tote bag) contains the 100-foot safety tether, a filter, and the special sample containers. The CDR's pouch receives the extra 70mm and 16mm film magazines.

While the CDR was loading the HTC, the LMP retrieved the closeup stereo camera (CSC) from its place in the sun where it rested between EVA's, deployed the skirt, or light shroud, and took some trial pictures in the vicinity of the LM. The CDR will nominally hand-carry this unit out on the geology traverse.

Both crewmen check the HTC and their packs for completeness. The surface sequence (16mm) camera is stowed in the HTC pouch. The traverse map is placed in a special pocket on the HTC. The HTC is usually carried by the LMP.

The crewmen then proceed to assemble the large scoop to the extension handle, take one of the two spare weigh bags (stowed in the ETB at launch), and go to a representative area of ground near the LM. The large scoop has a 0.5-cm sieve attachment. The crewmen sieve material for 5 minutes and deposit residual rocks and chips in the weigh bag. This bag is sealed and stowed in the ETB. The crew proceeds to calibrate their film (black and white SO-267) and obtain photometric data by taking a series of photographs of a special contrast chart carried on the HTC.

The next sample taken is the so-called contaminated sample under the LM itself. This is a sample of fine material scooped by the CDR with the small scoop. The sample is photographically documented and placed in a small bag, the number of which is reported to Houston. This is rebagged inside another small bag and the sample is also deposited in the ETB.

The events of the next 3 hours or so are dependent upon the sites and traverse selected between the crew and the geology team supporting the flight. During the traverse, the crew carries on a running commentary on what they are seeing and doing. They report all movements between samples, noting directions and distance with regard to the LM. Occasionally, the LMP and CDR read each other's film counter to CAPCOM to permit those keeping track of the film budget and records to update and correct their records. Changes of direction or an advance to a new leg of the traverse allow a 12- to 14-picture photo panorama, which satisfies the backsite-to-the-LM requirement as well.

When an area is encountered in which obstructions exist between the two crewmen and the LM, the CDR (who has the relay link with the LM) attempts to test the communication capability of the EMU EVCS system by moving behind or into this obstruction. This could be a fault escarpment, crater wall, large boulder, hill, or other surface feature.

During the traverse, the crewmen dig several small holes of trenches to gain an understanding of surface structure and mechanical properties and to obtain subsurface samples. They also take several core samples. These samples are taken by attaching a core tube to the extension handle and driving the tube into the ground by striking the extension handle end with the hammer. All samples are taken in a prescribed manner. When the candidate sample site is identified by either the CDR or the LMP, the CDR places the gnomon in proximity to the sample. He takes a stereo pair of photographs cross-sun at 5 feet while the LMP walks up to the sample site. Either before or after sampling, the LMP takes a picture at a distance of approximately 15 feet of the horizon or a prominent landmark in the background with the camera focused at 74 feet. The LMP will endeavor to be within 45° of a cross-sun orientation for this photograph. He takes a second down-sun picture focused on the sample at 5 feet. The CDR or LMP? picks up the sample by tongs, scoop, or hand. If the sample is small enough or is fine material, it goes into a sample bag, and the bag number is reported. Otherwise, it is placed unbagged in one of the tote bags. The CDR finishes the documentation by taking a cross-sun photo of the sample site at 5 feet. Where characteristics of the sample in situ or material/surface conditions near the sample justify it, the closeup stereo camera (CSC) is also used. The documentation photographs provide the required CSC picture localization information. The CSC user reports the frame number and the orientation of the camera (there is a sun compass on top) each time the camera is used. The CDR then picks up the gnomon and walks ahead to the next sample site. If an area larger than that covered by one photograph at a distance of 5 feet is to be sampled (e.g., the bottom of a crater), a series of stereo photographs will be taken at 15 feet, as required, while the LMP takes one or more down-sun. These photos are supplemented by 5-foot pictures as required to document individual samples. The criteria whereby samples are taken are described in the Lunar Field Geology Detail Test Objective in the Mission Requirements.

The CDR performs a photographic experiment using a special polarizing filter which he attaches to his camera. He takes pictures of different kinds of rocks and distant surfaces at various sun phase angles and filter settings, co-varying both. The crew takes samples of some of the rocks that are so photographed. One of these areas will be designated as the outpost or most distant spot from the LM. The crewmen rest briefly and then begin to perform several experiments. They collect a selected variety of rocks from the surface for the Gas Analysis Sample. Two or three surface microbreccias and crystalline rocks are picked up for the magnetic sample. Both of these samples go into special mini-SRC's, small can-like containers with their own sealing capabilities.

The crew digs a 2-foot-deep trench using their special hoe/shovel. This is for the soil mechanics experiment found in the Mission Requirements. The trench site is carefully documented by comments and photos before, during, and after the deep trench is dug. The crewmen take documented samples from the top, bottom, sides, and any areas where discontinuities or contacts between dissimilar materials, textures, or hues occur. To complete this experiment, the LMP makes a boot imprint in the middle of the piled-up material removed from the trench, and this imprint is photographed and discussed. The closeup camera is freely used here to document the deep trench. Fines are collected in the bottom of the trench for the special environmental sample, the third of the special samples. This sample also goes in a can-like sealed container. The CDR will dig the deep trench while the LMP makes a motion picture of this operation with the sequence camera. The sequence camera has been used during the traverse as required to film rocks rolling down crater walls, astronaut movement over the surface, sampling techniques, and special problems which lend themselves to motion picture documentation, such as HTC carrying difficulties, if any. The sequence camera has three magazines available, which yield approximately 23 minutes of movies at 12 fps. The camera is discarded when the third magazine is expended. The closeup camera has also been used on targets of opportunity during the traverse, according to the criteria set forth in its Detail Test Objective. The film cassette is extracted from the camera and stowed in a tote bag pouch for return to the LM. When the film capacity (100 pairs) is reached, the camera is left on the lunar surface.

With visits to the rest of the sites on the traverse following the same sampling procedures as described for the outbound part of the traverse, the crewmen make their way back to the LM.

The LMP deposits the HTC near the MESA, and then pauses while the CDR removes his loaded tote bag. The LMP performs this service for the CDR, then holds the bags open so the CDR can extract the samples and place them in a weigh bag. The first weigh bag is filled with bagged samples. If any space remains, unbagged samples are put in. The second weigh bag is similarly filled. Larger rocks or other samples not placed in the SRC are stowed in one of the tote bags and the bag is readied for transfer. The special environment and gas analysis sample containers are placed in the SRC. The core tubes complete the offload from the geology traverse into the SRC. The LMP takes down and rolls up the solar wind experiment. This device is bagged and placed in the ETB. The CDR closes and seals the SRC. Both crewmen place their data cameras, all film magazines with the sieve, contaminated, and extra samples in the ETB.

The CDR cleans the LMP's EMU as much as possible, and the LMP climbs the LM ladder. He moves through the hatch and hooks up the LEC to the overhead hand hold. The CDR checks the ETB contents for a final time and then supports the LMP's hauling the ETB into the ascent stage. The ETB is detached and placed out of the way; then the LMP returns the empty hooks of the LEC to the CDR, who attaches them to the tote bag. The bag is transported into the cabin and placed on the ascent engine cover. The SRC is then transported on the LEC to the ascent stage. The CDR cleans his EMU, while the LMP detaches the LEC and makes ready for CDR ingress.

The CDR climbs the ladder to the platform, tosses away the end of the LEC that the LMP hands him through the hatch, and enters the cabin. The LMP closes the hatch door, and the repressurization cycle is started to end the second and final EVA on Apollo 13.

FIRST CSM LUNAR ORBIT PLANE CHANGE (LOPC-1)

Approximately 36 hours 17 minutes after LOI, the CSM performs a plane change to place the LM in the CSM orbital plane at the time of LM lift-off. The maneuver is performed approximately 5 revolutions after LM landing.

The burn characteristics are the following.

Burn initiation

Time, hr:min:sec, g.e.t.	113:46:02.7
Mass, lb	36 286.3
Altitude above LLS, n. mi.	56.2
Selenographic latitude, deg	-5.0
Selenographic longitude, deg	-48.1
Velocity, fps	5360.6
Perilune altitude above LLS, n. mi.	55.7
Apolune altitude above LLS, n. mi.	61.7

Burn termination

Time, hr:min:sec, g.e.t.	113:46:12.6
Altitude above LLS, n. mi.	56.2
Selenographic latitude, deg	-5.0
Selenographic longitude, deg	-48.6
Velocity, fps	5360.1
Burn duration, sec	10
Inertial burn arc, deg	0.5
Plane change, deg	1.9
ΔV, fps	181.4
SPS propellant used, lb.	647.2
Orbit period, hr:min:sec	1:58:34
Perilune altitude above LLS, n. mi.	55.7
Apolune altitude above LLS, n. mi.	61.7

LM ASCENT

LM ascent begins after a lunar staytime of approximately 33.5 hours. Time of LM lift-off is determined by the LM-CSM phase requirements for rendezvous. A sketch of the powered ascent phase is shown in figure 16. The powered ascent is divided into two phases. The first phase is the vertical rise phase which is required for the ascent stage to achieve terrain clearance. A description of trajectory parameters and vehicle attitudes during the vertical rise phase and the transition to the orbital insertion phase is shown in figure 17. Guidance switches to the orbital insertion phase for radial rates greater than 40 fps, but because of DAP steering lag the pitchover does not begin until a radial rate of approximately 50 fps is achieved. This means the vertical rise phase is actually terminated at 10 seconds after lift-off. During the vertical rise, the LM Z-body axis is rotated toward the desired azimuth which is normally in the CSM orbital plane.

A description of the orbital insertion phase in shown in figure 18. Included is a description of total ascent phase performance, insertion orbital parameters, and onboard displays at insertion. The onboard display values reflect LGC estimated values. Yaw steering is used during the orbital insertion phase, if required, to maneuver the LM into the CSM orbital plane or into a plane that is coplanar with the CSM orbit. In the nominal case, no yaw steering is required. A groundtrack for the LM and CSM during LM ascent is shown in figure 19. LM ascent radar data are shown in table VIII.

TABLE VIII.- MSFN TRACKING SUMMARY FOR LM ASCENT

Stations tracking at LM lift-off			
Station	Time,[a] hr:min:sec, g.e.t.	Azimuth, deg	Range, n. mi.
Carnarvon	137:15:22	25	213 092
Guam	137:15:22	-106	212 569
Hawaii	137:15:22	-89	214 717
Honeysuckle	137:15:22	-28	213 566
Parks	137:15:22	-28	213 454

[a]LM lift-off time.

Stations tracking at LM insertion			
Station	Time,[b] hr:min:sec, g.e.t.	Azimuth, deg	Range, n. mi.
Carnarvon	137:22:32	23	213 252
Guam	137:22:32	-104	212 781
Hawaii	137:22:32	-88	214 989
Honeysuckle	137:22:32	-30	213 776
Parks	137:22:32	-30	213 664

[b]LM insertion time.

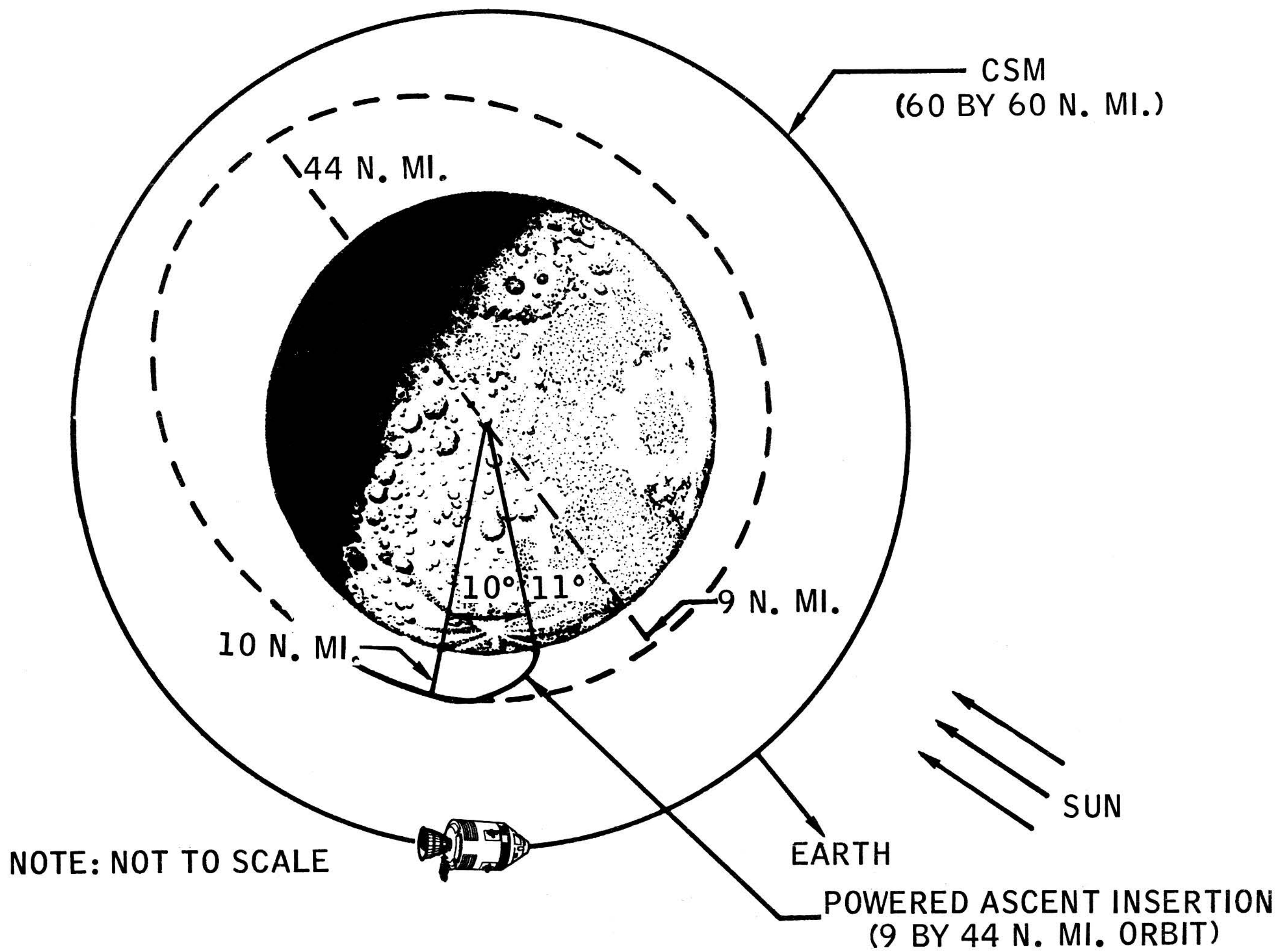


Figure 16.- LM ascent.

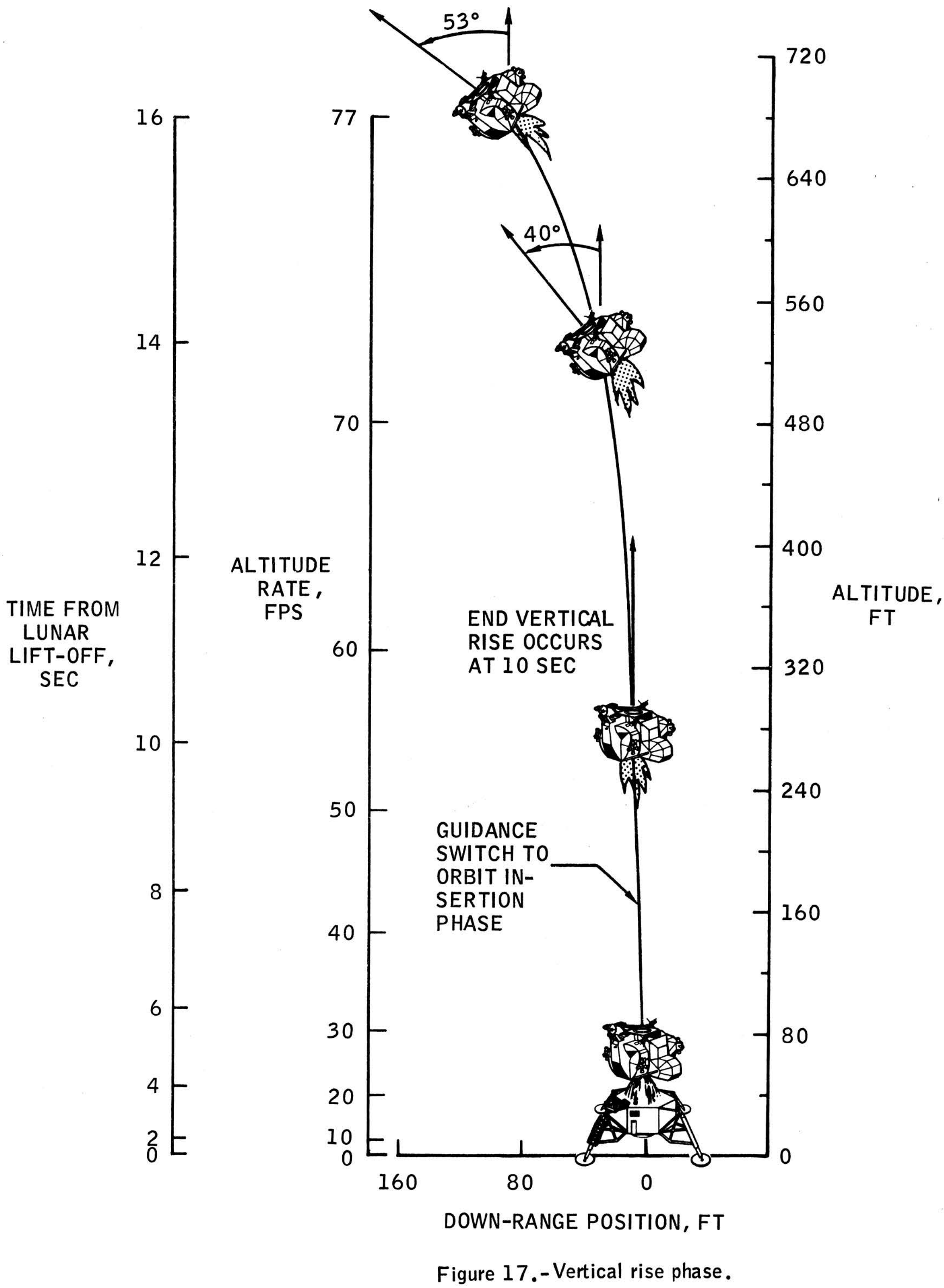
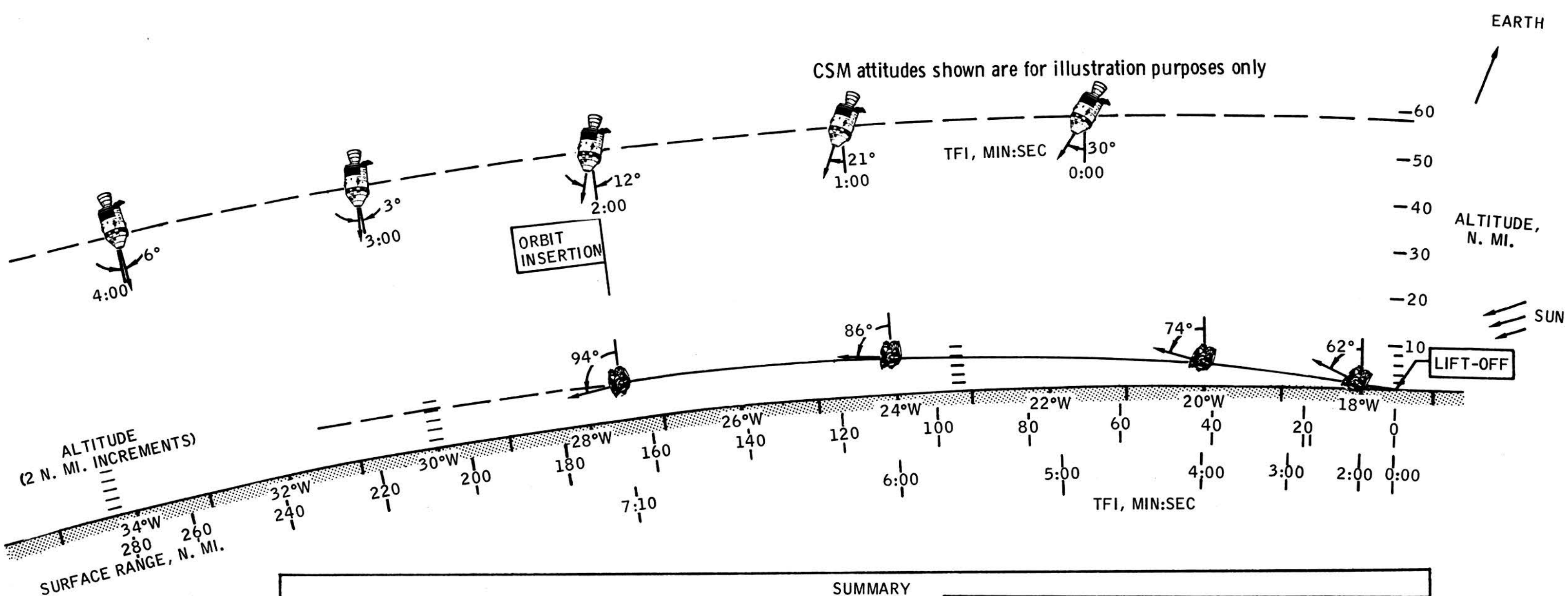


Figure 17.- Vertical rise phase.



SUMMARY								
					LM TO CSM			
EVENT	TFI, MIN:SEC	INERTIAL VELOCITY, FPS	ALTITUDE RATE, FPS	ALTITUDE, FT	RANGE, N. MI.	RANGE RATE, FPS	PHASE ANGLE, DEG	LOOK ANGLE, (LOCAL VERTICAL) DEG
LIFT-OFF	0:00	15	0	0	88	3859	3.9	49.9
END OF VERTICAL RISE	0:10	56	54.1	269	95	4030	4.4	53.6
	2:00	1036	171.5	14 214	169	3843	9.4	75.2
	4:00	2477	184.4	36 567	233	2556	13.4	83.7
	6:00	4278	107.2	54 847	267	795	15.5	87.0
ORBIT INSERTION	7:10	5530.1	35.9	59 957.0	268.9	-448.2	15.6	87.4

h_p = 52 944 FT

h_a = 43.8 N. MI.

η = 21.3°

γ = .37°

ΔV = 6042.8 FPS

Figure 18.- Orbit insertion phase.

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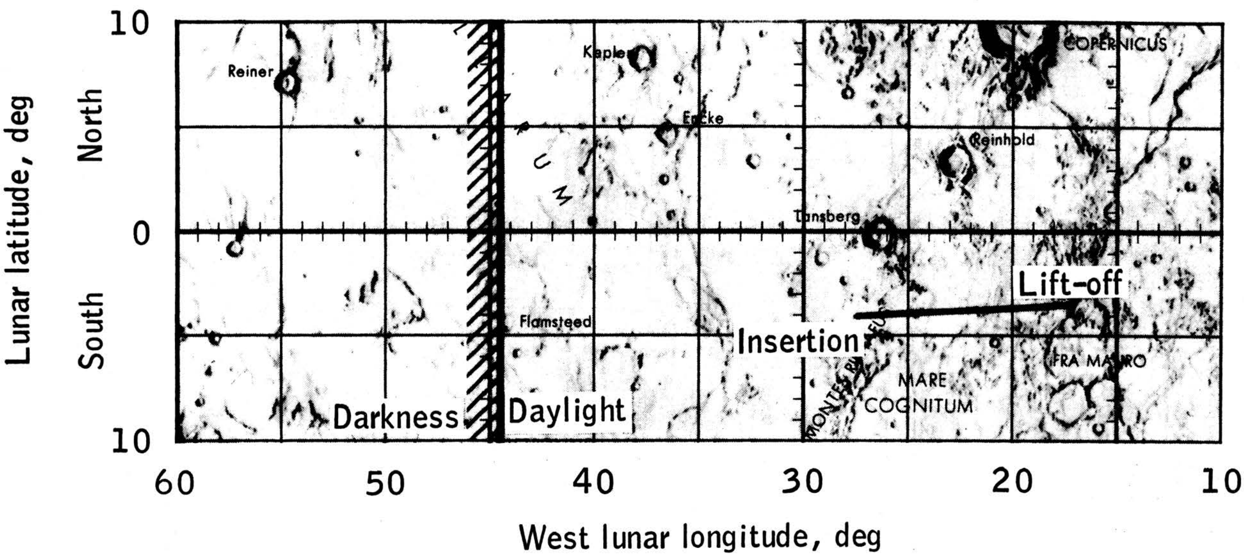
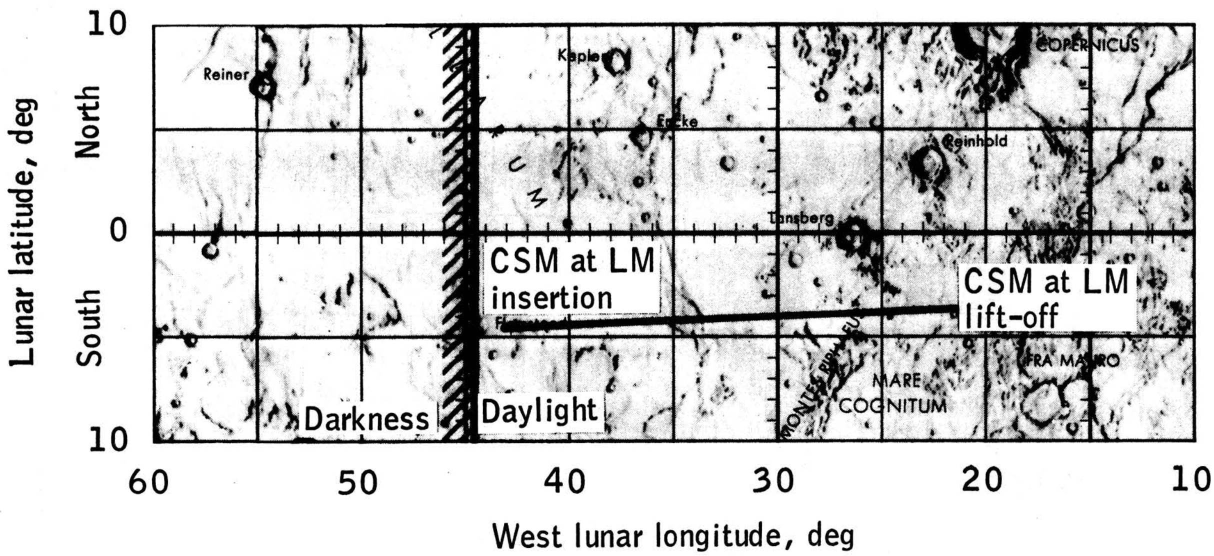


(a) LM groundtrack.



(b) CSM groundtrack.

Figure 19.- LM and CSM groundtracks during powered ascent.

RENDEZVOUS SEQUENCE

Sequence Summary

After insertion of the LM into orbit, the LM will compute and execute the coelliptic rendezvous sequence which nominally consists of four major maneuvers: CSI, CDH, TPI, and TPF. A nominally-zero plane change maneuver will be scheduled between CSI and CDH, and two nominally-zero midcourse correction maneuvers will be scheduled between TPI and TPF; the TPF maneuver is actually divided into several braking maneuvers. The complete maneuver sequence is presented in table IX. All maneuvers will be performed with the LM RCS system with either two jets or four jets of a prescribed quad for thrust vector control. The entire maneuver sequence from lift-off through docking requires approximately 3.5 hours.

Before detailed information is presented about each maneuver, some general mission and trajectory design requirements for the rendezvous plan are noted.

a. CSI will occur impulsively approximately 50 minutes after insertion.

b. CDH will occur one-half period after CSI and will be a zero ΔV maneuver.

c. LM coelliptic Δh will be 15 n. mi. below CSM.

d. Time between CDH and TPI will be approximately 41 minutes.

e. TPI will occur at the midpoint of darkness and on a LM-to-CSM elevation angle of 26.6°.

f. Braking should occur in daylight; stationkeeping should begin approximately 25 minutes after the vehicles enter sunlight.

The orbital schematic for the rendezvous sequence is presented in figure 20. The relative motion of the LM with respect to the CSM is shown in figure 21. The onboard tracking, lighting, and mission events summary are presented in figure 22 and in table X.

TABLE IX.- RENDEZVOUS MANEUVER SUMMARY

Event	Time, day:hr:min:sec, g.e.t.	Propulsion system	Burn time, sec	Total ΔV, fps	Weight at ignition, lb	Propellant used[a], lb	Resultant orbit[b] h_a/h_p, n. mi.	Range at cutoff, n. mi.	Range rate at cutoff, fps
CSI	5:18:12:10.1	RCS +Z (two-jet)	44.7	49.3	5843.1	31.9	44.9/43.2	149.0	-121.4
PC[c]	5:18:41:19.5	RCS ±Y (two-jet)	0.0	0.0	5811.2	0.0	44.9/43.2	114.6	-121.2
CDH	5:19:10:28.9	RCS	0.0	0.0	5811.2	0.0	44.9/43.2	80.1	-120.0
TPI	5:19:51:44.6	RCS +Z (two-jet)	22.4	25.0	5811.2	16.0	60.7/43.6	32.0	-132.6
MC-1[c]	5:20:06:44.6	RCS ±Z (two-jet)	0.0	0.0	5795.2	0.0	60.7/43.6	15.0	-97.5
MC-2[c]	5:20:21:44.6	RCS ±Z (two-jet)	0.0	0.0	5795.2	0.0	60.7/43.6	4.6	-46.2
First braking maneuver	5:20:33:05.0	RCS -Z (two-jet)	10.6	11.8	5795.2	7.6	59.0/48.8	0.5	-19.7
Second braking maneuver	5:20:34:18.4	RCS -Z (two-jet)	8.8	9.8	5787.6	6.3	59.7/53.7	0.2	-9.7
Third braking maneuver	5:20:35:56.8	RCS -Z (two-jet)	4.3	4.8	5781.3	3.0	58.6/56.2	0.08	-4.9
Fourth braking maneuver	5:20:36:16.4	RCS -Z (two-jet)	4.2	4.7	5178.3	3.0	59.4/58.0	0.01	-0.2
Docking	5:20:50:00.0	--	--	--	5775.3	--	59.4/58.0	0.0	0.0

[a]Estimates for completing consumables document will present official propellant usage. Slight changes caused by rigor of analysis are expected.

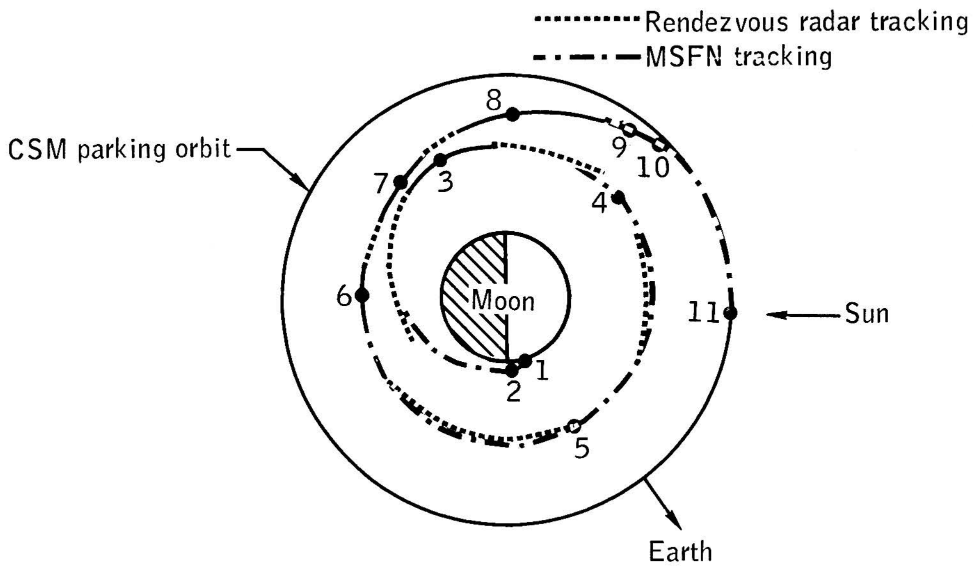
[b]Altitudes measured above landing site.

[c]Nominally a zero ΔV maneuver.

TABLE X.- MSFN TRACKING

AND SHADOW TIMELINE

Event	Time, day:hr:min:sec, g.e.t.
Insertion	5:17:22:32.4
Sunset	5:17:30:42.7
MSFN LOS	5:17:46:22.0
CSI	5:18:12:10.1
Sunrise	5:18:19:14.9
MSFN AOS	5:18:34:39.5
PC	5:18:41:19.5
CDH	5:19:10:28.9
Sunset	5:19:28:15.7
MSFN LOS	5:19:43:20.7
TPI	5:19:51:44.8
MC-1	5:20:06:44.8
Sunrise	5:20:14:40.7
MC-2	5:20:21:44.8
MSFN AOS	5:20:29:20.7
First braking maneuver	5:20:33:05.0
Second braking maneuver	5:20:34:18.4
Third braking maneuver	5:20:35:56.8
Fourth braking maneuver	5:20:36:16.4
Docking	5:20:50:00.0



Event	Time, hr:min:sec g.e.t.	Event	Time, hr:min:sec g.e.t.
1. Lift-off	137:15:22.5	7. MC-1	140:06:44.8
2. LM insertion	137:22:32.4	8. MC-2	140:21:44.8
3. CSI	138:12:10.1	9. Begin braking	140:33:05.0
4. PC	138:41:19.5	10. Begin stationkeeping	140:37:20.6
5. CDH	139:10:28.9	11. Docking	140:50:00.0
6. TPI	139:51:44.8		

Figure 20.- Orbital schematic of ascent through docking.

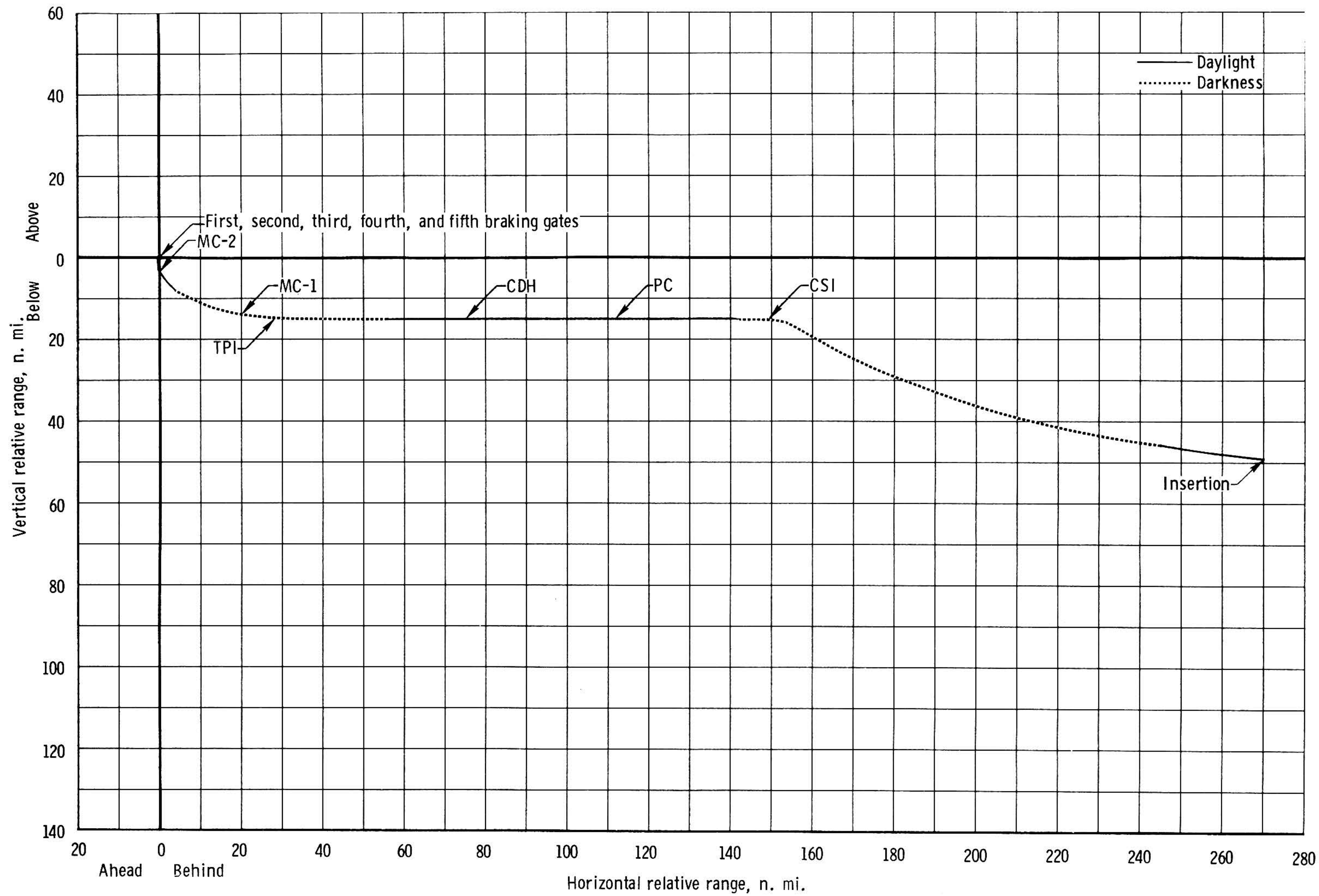


Figure 21. - Vertical relative range versus horizontal relative range.

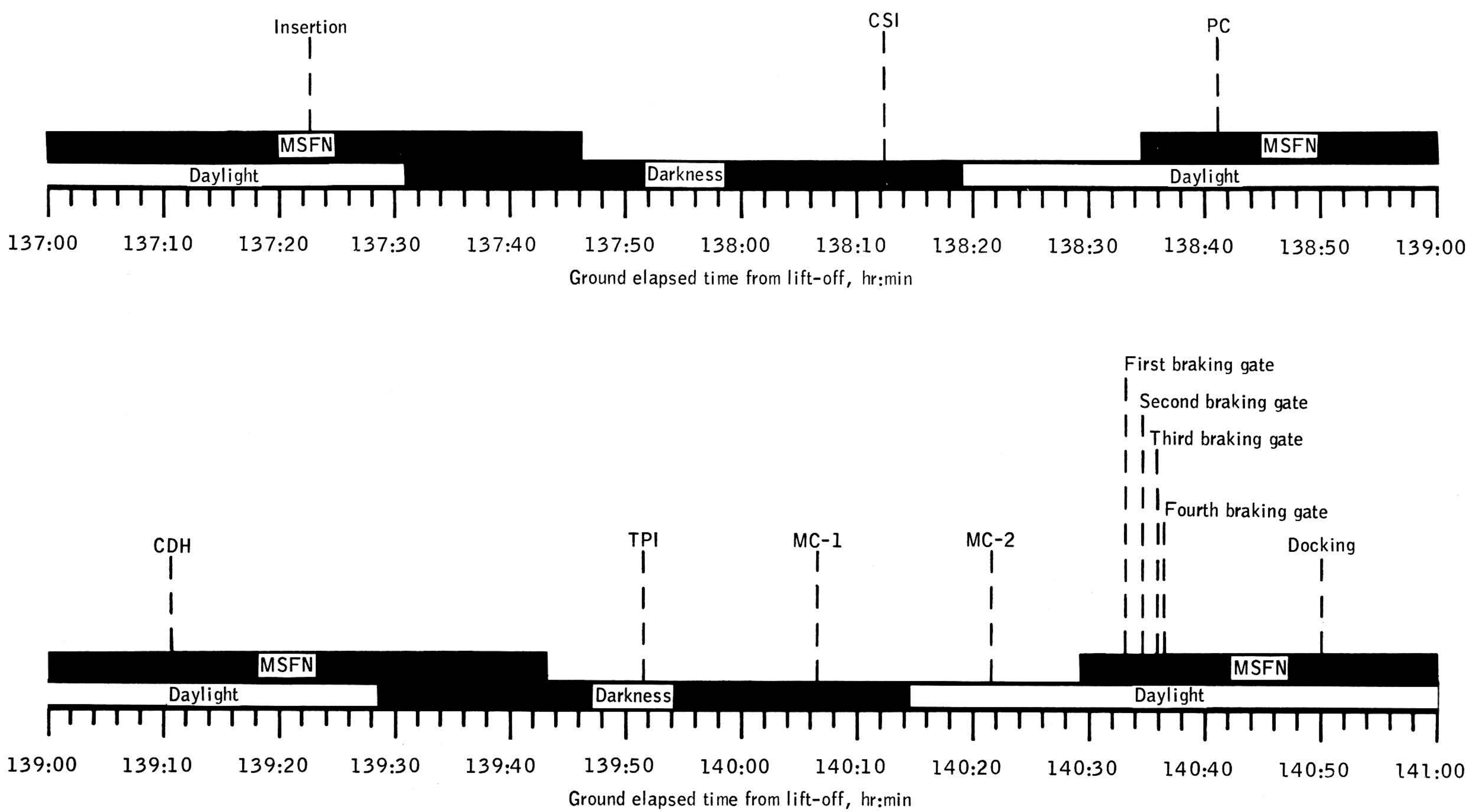


Figure 22.- Mission events and tracking summaries for LM-active rendezvous.

CSI

The purpose of the CSI maneuver is to establish the proper phasing conditions at CDH so that after the CDH maneuver is performed, TPI will occur at the desired time and elevation angle. For this mission, however, the LM insertion orbit is adjusted such that for the dispersion-free case the CSI maneuver essentially yields the desired coelliptic Δh of 15 n. mi. and the CDH maneuver becomes zero. The CSI maneuver will be scheduled approximately 50 minutes after insertion, about 7 minutes prior to sunrise. The 50-minute interval between insertion and CSI is required for LM RR and CSM sextant/VHF ranging navigation, MSFN state vector update, and LM platform realinement. Should the CSM tracking of the LM prior to CSI indicate that an out-of-plane situation exists, an out-of-plane component designed to force a common node 90° after CSI would be incorporated in the CSI maneuver. The characteristics of the horizontal posigrade CSI maneuver which is performed with the +Z (two-jet) RCS thrusters are as follows.

Burn initiation	
Time, hr:min:sec, g.e.t.	138:12:10.1
Latitude, deg N	2.9
Longitude, deg E	174.3
Altitude above landing site, n. mi.	43.3
Phase angle, deg	8.6
Burn termination	
Time, hr:min:sec, g.e.t.	138:12:54.8
Latitude, deg N	3.0
Longitude, deg E	172.0
Altitude above landing site, n. mi.	43.2
Burn duration, sec	44.7
ΔV, fps	49.5
Propellant used, lb	31.9
Resultant h_a/h_p, n. mi.	44.9/43.2
Catchup rate, deg/min	0.071

PC Maneuver

A plane change, nominally not required, would be executed approximately 29 minutes after CSI at the common node of the CSM and LM orbital planes; this common node is forced by the CSI maneuver. At the time of the PC maneuver, the out-of-plane velocity component would be nulled to zero with a coplanar situation being established by the Y-axis RCS thrusters.

If a required plane change is not begun at CSI, the PC maneuver would force a common node at CDH, and the plane change would be completed at CDH. The onboard LM and CSM tracking schedule and navigation plan is resumed immediately after the CSI and PC maneuvers.

CDH

Nominally CDH will be a zero ΔV maneuver. (Proper adjustment of the LM insertion conditions will cause CSI to essentially coellipticize the LM with the CSM orbit at the desired coelliptic Δh of 15 n. mi. and therefore reduce CDH to a zero ΔV maneuver.) If required because of dispersions in the LM insertion orbit, CDH would occur one-half orbital period after CSI and the resulting coelliptic Δh could vary from the nominal value as much as ±5 n. mi. The direction of the CDH ΔV would vary, depending on the dispersion involved. The RCS thrusters used (either X or Z) would be those that would avoid loss of RR lock.

TPI

The TPI maneuver, executed with the +Z RCS (two-jet) thrusters, is initiated on a 26.6° line-of-sight elevation angle to the CSM. The ΔV vector is approximately along the line of sight to the CSM. The maneuver occurs approximately 41 minutes after CDH and approximately 23 minutes before daylight. The Lambert targeting for this maneuver is based on a 130° CSM central travel angle between TPI and TPF. The characteristics of the TPI maneuver are as follows.

Burn initiation

Time, hr:min:sec, g.e.t.	139:51:44.6
Latitude, deg S	1.3
Longitude, deg W	136.1
Altitude above landing site, n. mi.	43.7
Phase angle, deg	1.7

Burn termination

Time, hr:min:sec, g.e.t.	139:52:07.0
Latitude, deg S	1.2
Longitude, deg W	137.3
Altitude above landing site, n. mi.	43.8
Burn duration, sec	22.4
ΔV, fps	25.0
Propellant used, lb	16.0
Resultant h_a/h_p, n. mi.	60.7/43.6
Catchup rate, deg/min	0.032

Midcourse Correction and TPF

The LM RR tracking and CSM sextant and VHF ranging will resume approximately 4 minutes after TPI and will cease approximately 4 minutes prior to MC-1. MC-1 is scheduled 15 minutes after TPI and is nominally zero. LM RR tracking and CM sextant and VHF ranging begin again approximately 4 minutes after MC-1 and end approximately 3 minutes prior to MC-2 which occurs 30 minutes after TPI and is also nominally a zero maneuver. The terminal phase braking is performed with the two-jet -Z RCS thrusters. Final approach and stationkeeping should occur approximately 25 minutes after the vehicle enters sunlight on the backside of the moon.

CSM/LM Separation, LM Jettison, and Second Lunar Orbit Plane Change

Docking occurs at approximately 140^h45^m g.e.t. After equipment transfer and decontamination procedures, the LM ascent stage is jettisoned at 143^h04^m g.e.t. The CSM separates from the ascent stage by performing a +Z RCS translation of 1.0 fps in the inertial deorbit attitude. This separation maneuver occurs approximately 270° prior to the deorbit maneuver and is in a retrograde direction.

The discarded LM ascent stage will be targeted to impact the lunar surface at 3.00°S and 19.67°W approximately 34 n. mi. from the Apollo 13 landing site and approximately 60 n. mi. from the Apollo 12 landing site. The nominal target (fig. 23) lies on a great circle between the two sites, and seismometer readings will be provided from both the Apollo 13 and the Apollo 12 sites. The predicted maximum range of the seismometer is 60 to 65 n. mi.; therefore, if the resultant impact point is between the nominal target and an area 25 to 30 n. mi. west of the nominal target, both seismometers should register. If the resultant impact point is more than approximately 30 n. mi. west of the nominal target, only the Apollo 12 seismometer will register; and if the resultant impact point is significantly east of the nominal target, only the Apollo 13 seismometer will register. The 3σ dispersion ellipse has approximate dimensions of ±19 n. mi. down range and ±1 n. mi. cross range. If the LM lands behind the planned impact point, it cannot land directly on the Apollo 13 site, and if the LM lands in front of the planned impact point, it cannot land on the Apollo 12 site. In either case, a reading from at least one of the sites will be obtained.

The ascent stage deorbit burn duration will be 75.2 seconds beginning at $144^h32^m20.2^s$ g.e.t. This RCS burn (+X, four-jet) will be performed at a yaw attitude of approximately 166°. The burn targets are

ΔV_x = -180.0 fps and ΔV_y = 45.0 fps, which will result in a total velocity of 185.5 fps. After the burn, the ascent stage is in a 59.5- by -63.1-n. mi. orbit. The ascent stage will travel through a central angle of 85° before it impacts at approximately 28 minutes after the deorbit burn. At landing, the ascent stage will have a velocity of approximately 5508 fps (5500 fps horizontal and 365 fps radial). The flight-path angle of the ascent stage at impact will be approximately -4°.

A plane change is performed in revolution 40 at $154^h13^m10.8^s$ g.e.t. to set up the bootstrap photography. The plane change, which is an SPS burn of 824.6 fps, results in an orbit that passes directly over Descartes and Davy Rille two revolutions later. The characteristics of the burn are the following.

Burn initiation

Time, hr:min:sec, g.e.t.	154:21:19.7
Altitude above LLS, n. mi.	55.5
Selenographic latitude, deg	4.7
Selenographic longitude, deg	91.8
Velocity, fps	5363.9
Perilune altitude above LLs, n. mi.	55.4
Apolune altitude above LLS, n. mi.	61.2
Selenographic inclination, deg	4.9

Burn termination

Time, hr:min:sec, g.e.t.	154:22:02.8
Altitude above LLs, n. mi.	55.5
Selenographic latitude, deg	4.4
Selenographic longitude, deg	89.7
Velocity, fps	5363.2
Burn duration, sec	43
Inertial burn arc, deg	2.2
Plane change, deg	8.8
ΔV, fps	827.6
SPS propellant used, lb	2814.6
Perilune altitude above LLS, n. mi.	55.4
Apolune altitude above LLS, n. mi.	60.8
Selenographic inclination, deg	11.6

Photography and landmark tracking are performed during revolutions 40, 41, 42, 43, and 44. The TEI maneuver is performed at the end of revolution 46 at $167^h28^m58.3^s$ g.e.t. The lunar orbit groundtrack from LOPC-2 to TEI is shown in figure 24.

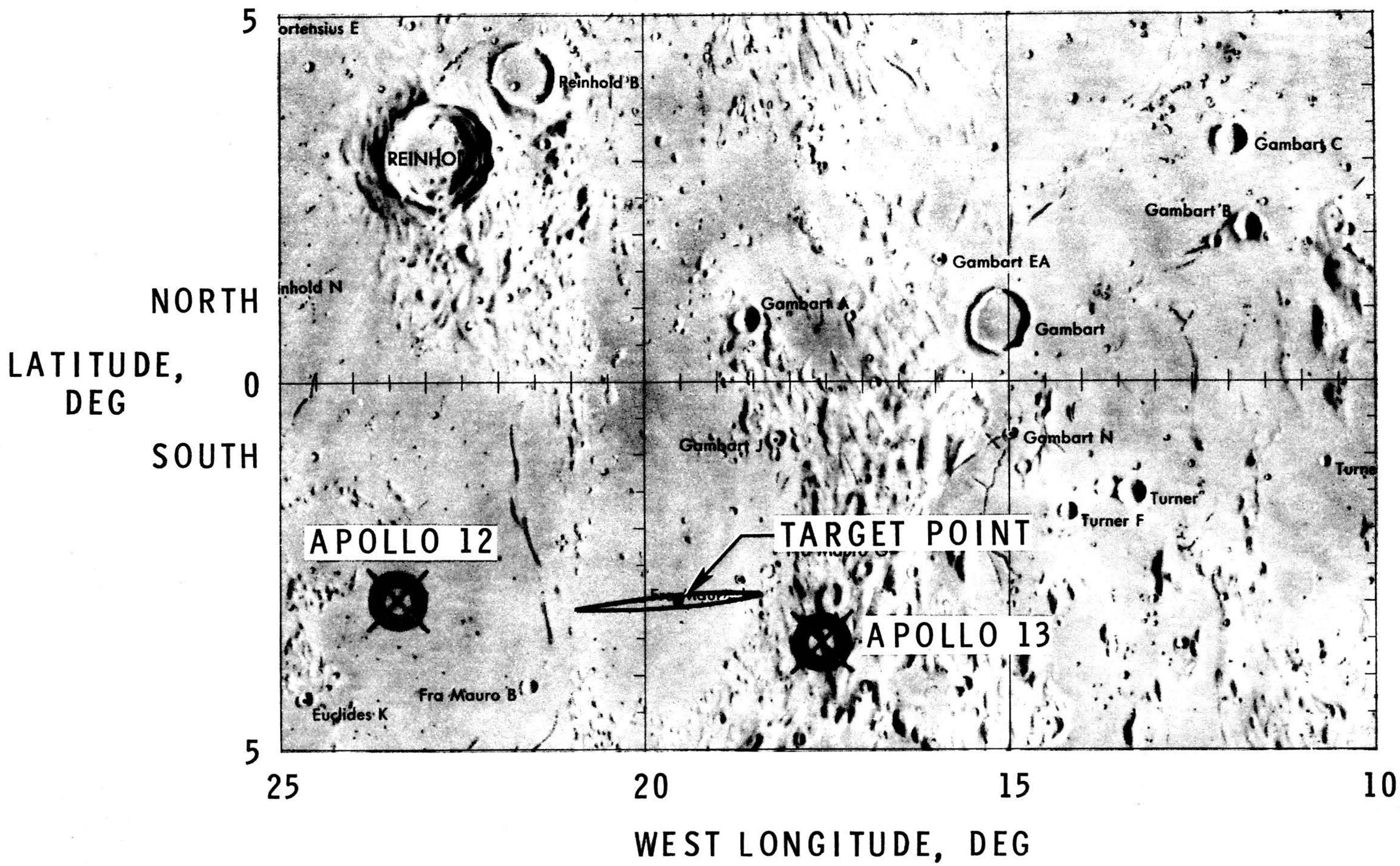


Figure 23.- Three-sigma impact ellipse for Apollo 13 LM ascent stage deorbit maneuver.

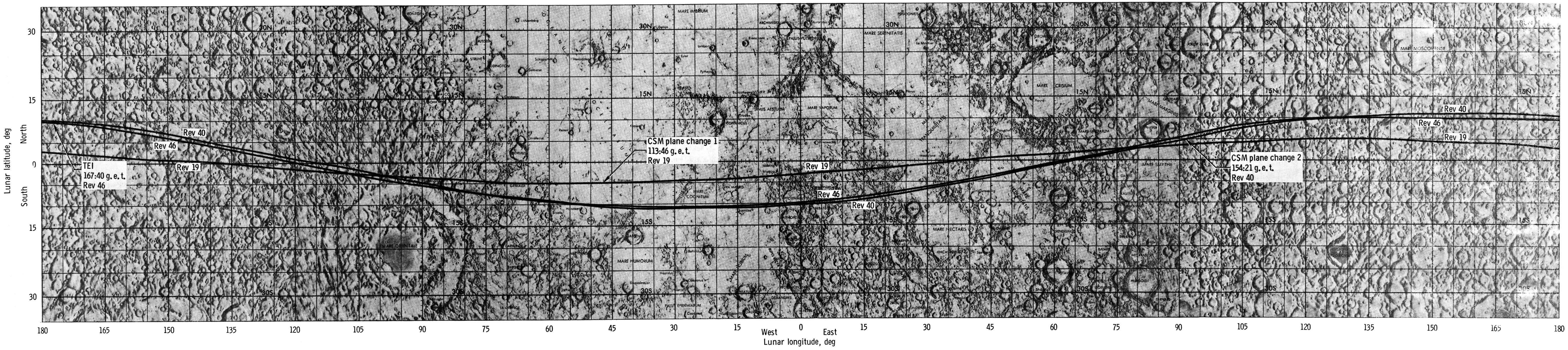


Figure 24. - Groundtrack from LOPC-2 to TEI.

TRANSEARTH INJECTION

The TEI maneuver occurs $3^{d}17^{h}58^{m}$ after LOI. The burn was targeted for a 73.3-hour transearth flight time. The transearth injection geometry is shown in figure 25.

The characteristics of the burn are the following.

Burn initiation

Time, hr:min:sec, g.e.t.	167:38:39.1
Mass, lb	32 824
Altitude above LLS, n. mi.	57.8
Selenographic latitude, deg	9.9
Selenographic longitude, deg	-177.7
Velocity, fps	5351.2
Selenographic inclination, deg	11.5
Perilune altitude above LLS, n. mi.	54.4
Apolune altitude above LLS, n. mi.	62.3

Burn termination

Time, hr:min:sec, g.e.t.	167:40:53.8
Altitude above LLS, n. mi.	63.2
Selenographic latitude, deg	11.4
Selenographic longitude, deg	173.8
Velocity, fps	8270.6
Selenographic inclination, deg	17.8
Burn duration, min:sec	2:15
Inertial burn arc, deg	8.5
Plane change, deg	9.3
ΔV, fps	3144
SPS propellant used, lb	8795
Perilune altitude above LLS, n. mi.	56.3

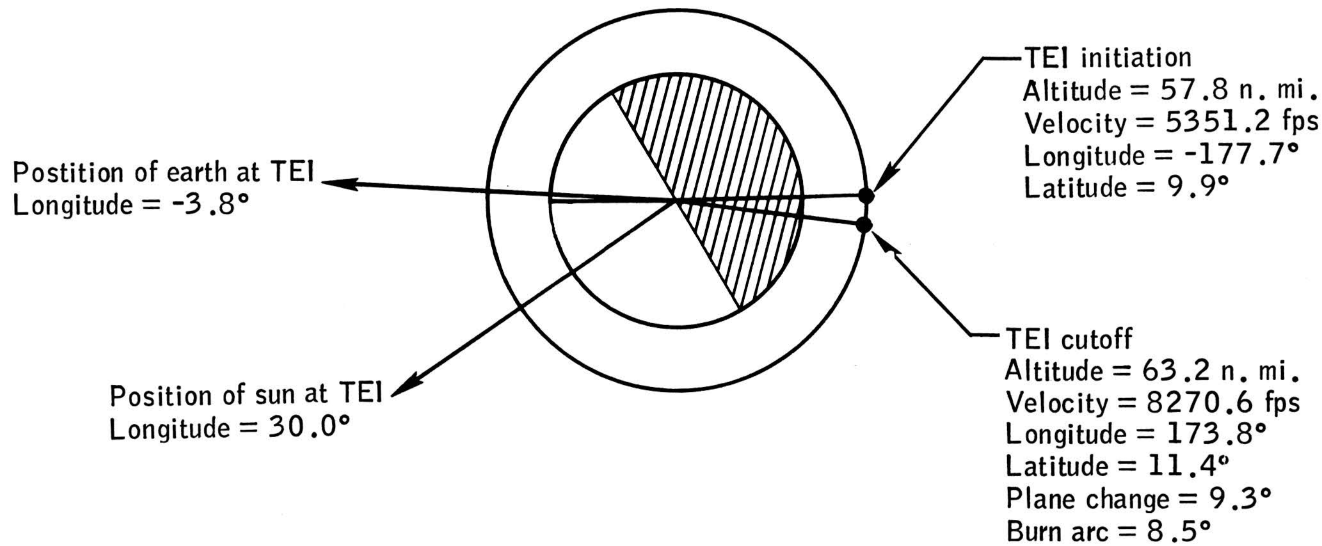


Figure 25.- TEI geometry.

TRANSEARTH COAST

A groundtrack of the transearth coast phase is provided in figure 26. Altitude above the moon is shown for the first 10 hours of coast in figure 27. The altitude time history for TEI to entry is shown in figure 28. Three midcourse decision points have been defined for the transearth phase.

a. MCC-5, TEI plus 15 hours

b. MCC-6, EI minus 22 hours

c. MCC-7, EI minus 3 hours

The maneuvers will be targeted for corridor control only. Altitude is plotted against time for the last 10 hours of transearth coast in figure 29.

The SC remains in the lunar umbra during the entire TEI burn, enters sunlight approximately 5 minutes after cutoff, and remains in sunlight until entry. Passive thermal control will be maintained during most of transearth coast phase.

The SC is acquired by MSFN approximately 11 minutes after TEI cutoff, and the last station to lose signal is Honeysuckle (5° minimum elevation) which loses signal at $240^{h}46^{m}$ g.e.t. or approximately 4 minutes prior to entry.

The primary trajectory parameters at entry interface are the following.

Time, hr:min:sec, g.e.t.	240:49:49.6
Transearth coast time, hr	73
Inertial velocity, fps	36 129.4
Altitude, n. mi.	65.8
Inertial flight-path angle, deg	-6.5
Latitude, deg S	15.1
Longitude, deg W	173.5
Equatorial inclination (ascending), deg	40

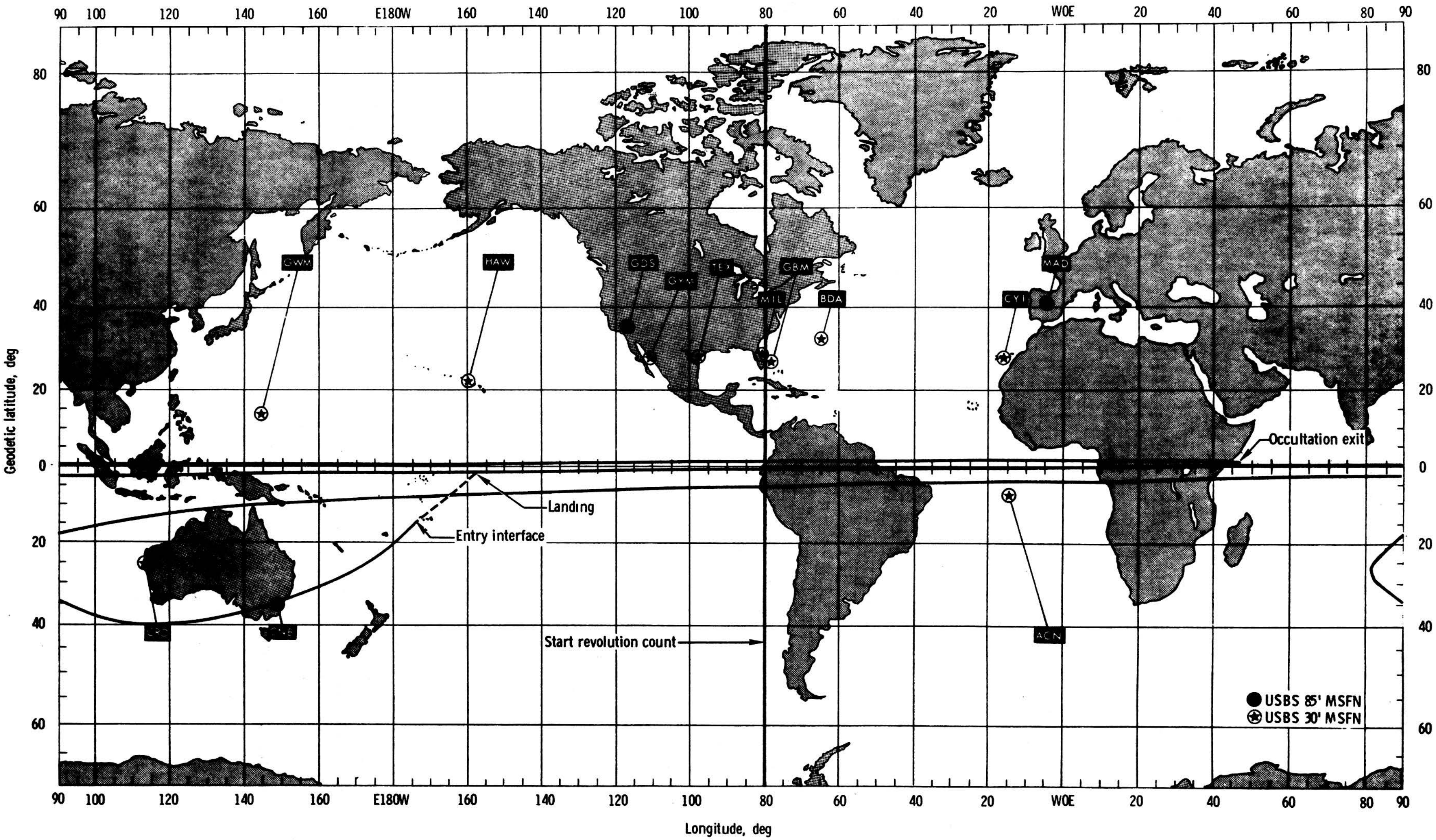


Figure 26. - Transearth coast groundtrack.

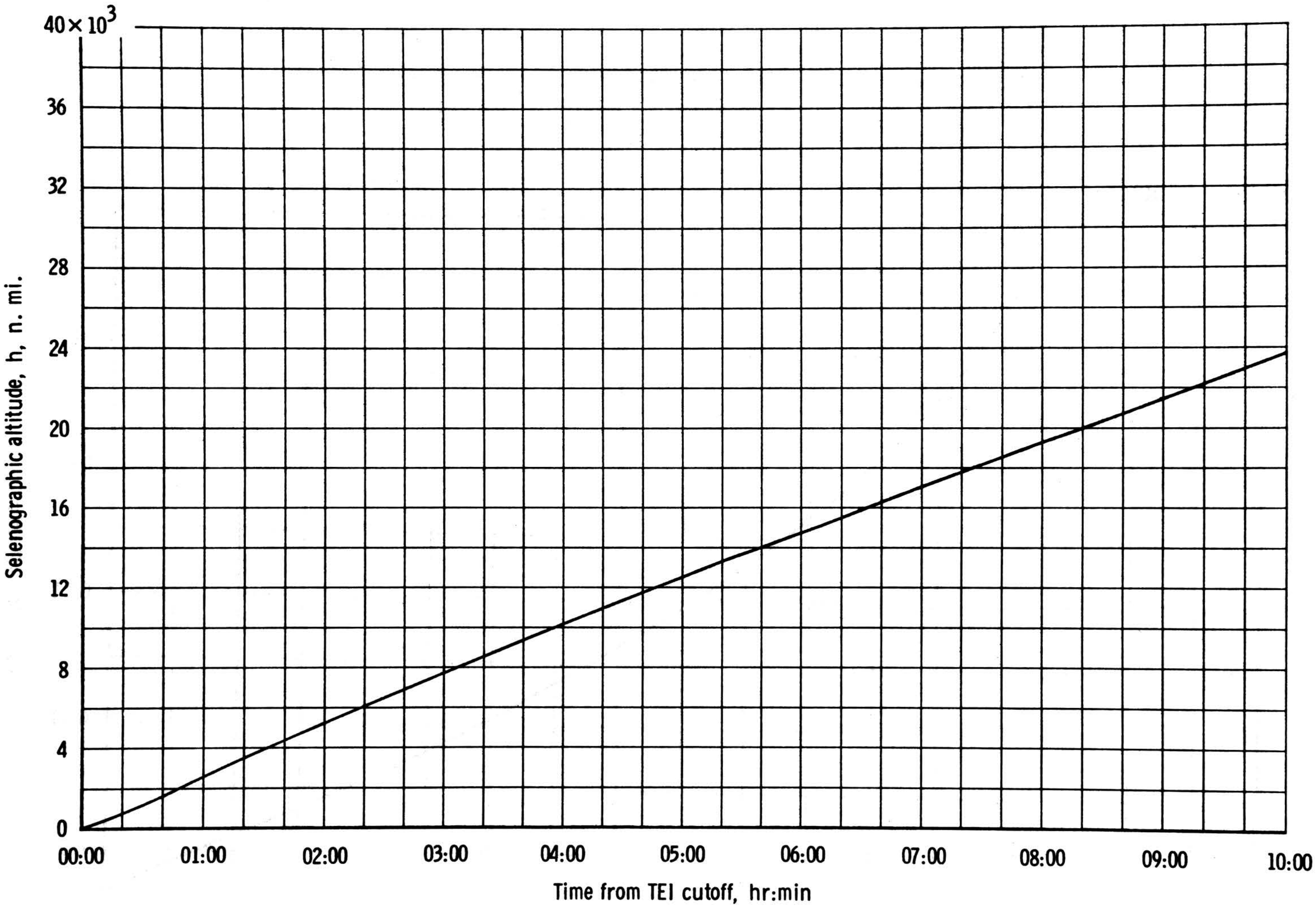


Figure 27. - Altitude time history during the first 10 hours of transearth coast.

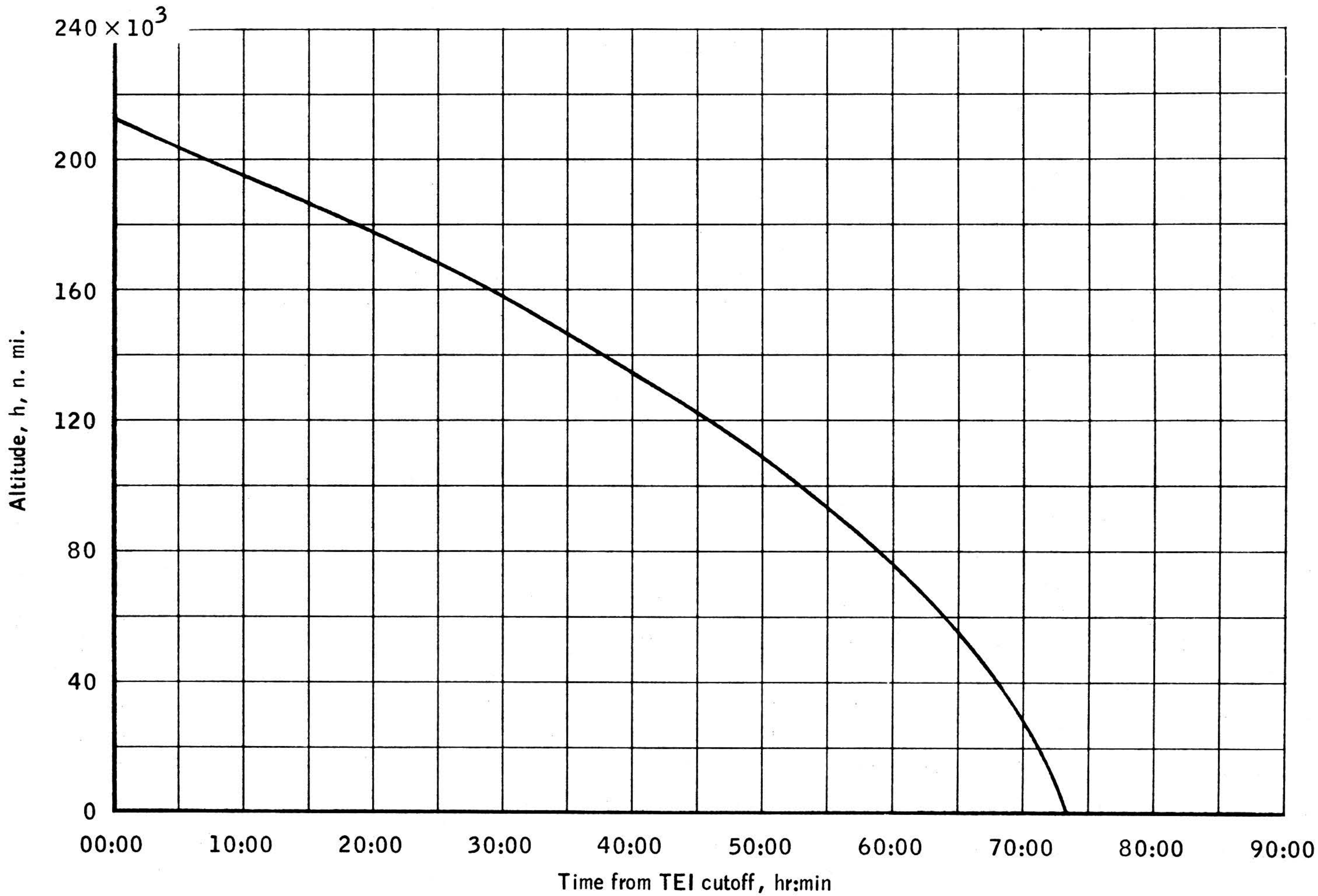


Figure 28.- Altitude time history from transearth injection to entry interface.

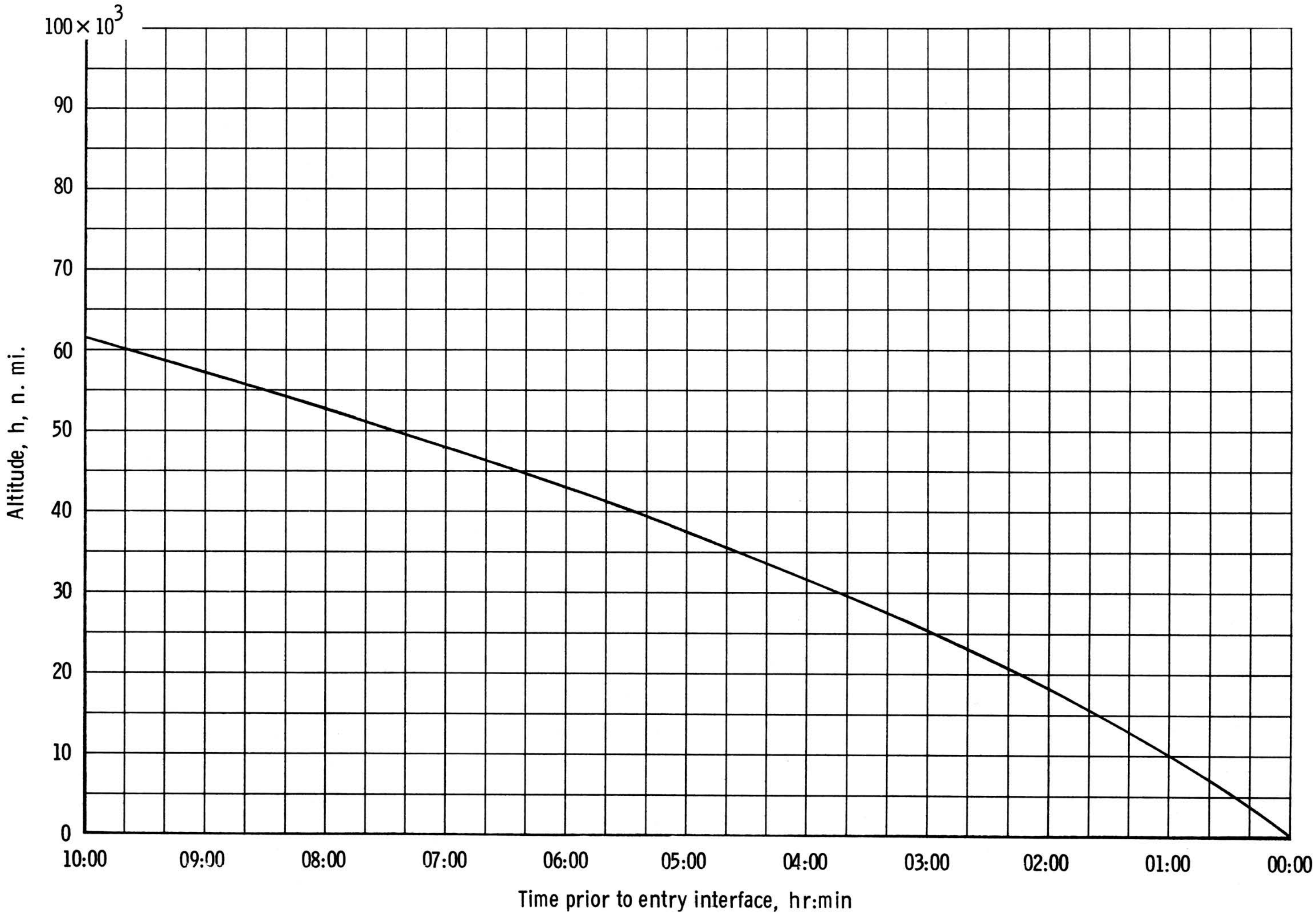


Figure 29. - Altitude time history 10 hours prior to entry interface.

ENTRY

The entry phase of the operational trajectory was simulated with the Apollo Reentry Simulation program with six degrees of freedom. Three-degree-of-freedom trajectories were used to determine the CM maneuver footprint. The entry corridor is presented in figure 30. The CM enters the earth's atmosphere near the Samoan Islands and lands 1250-n. mi. down range at a point about 140 n. mi. south of Christmas Island.

At the nominal EI, $240^h49^m38^s$ after lift-off, the CM is at an altitude of 399 691 feet, and the coordinates are 15.155°S geodetic latitude and 173.521°E longitude. Inertial velocity, flight-path angle, and azimuth at this point are 36 129 fps, 6.50° below the local horizontal, and 52.495°, respectively.

A plot of the CM maneuver footprint and the nominal ground trace on a map of the entry area are presented in figure 31. The footprint is extended to a 3500-n. mi. entry range. The nominal touchdown target location is 1250-n. mi. down range from the EI position, and the coordinates of the target are 157.5°W longitude and 1.559°S geodetic latitude. A sequence of pertinent events is given in table XI and includes the periods of communications blackout which occur along the trajectory. The guidance phases are shown in figure 32, which shows altitude as a function of range to the target. The total heating rate is 312.1 Btu/ft^2/sec, and the total heat load is 27 377.9 Btu/ft^2.

The CM RCS uses 12 pounds of propellant for the separation and attitude hold maneuvers before the spacecraft reaches 400 000 feet. The RCS uses 31.8 pounds of propellant to perform the guidance commands during the remainder of the entry. The load factor at the c.g. reaches a first maximum of 6.49g and a second maximum of 4.18g.

The drogue parachute deployment sequence begins at an altitude of 23 300 feet, which is 8 minutes 6 seconds after EI. The two drogue parachutes are deployed 2 seconds later. At an altitude of 10 500 feet, the low altitude baroswitch closes, and the drogue parachutes are disconnected. The three main parachutes are deployed 1 second after the baroswitch closes. The CM, suspended on the main parachutes, reaches splashdown 13 minutes 48 seconds after EI.

TABLE XI.- ENTRY EVENTS SEQUENCE

Event	Time from lift-off, hr:min:sec	Time from 400 000 ft, min:sec
Entry	240:49:50	00:00
Enter S-band communication blackout	240:50:08	00:18
Guidance changes from P63 to P64 Load factor = 0.05g	240:50:19	00:29
Initiate constant drag control	240:50:42	00:52
Maximum heating rate	240:51:00	01:10
Guidance initiate at R-DOT = -700 fps	240:51:08	01:18
Maximum load factor (FIRST)	240:51:10	01:20
Subcircular velocity	240:51:58	02:08
Guidance change from P64 to P67	240:52:04	02:14
Exit S-band communication blackout	240:53:14	03:24
Maximum load factor (SECOND)	240:55:16	05:26
Termination of CMC guidance	240:57:00	07:10
Drogue parachutes deployment	240:58:02	08:12
Main parachutes deployment	240:58:52	09:02
Splashdown	241:03:41	13:51

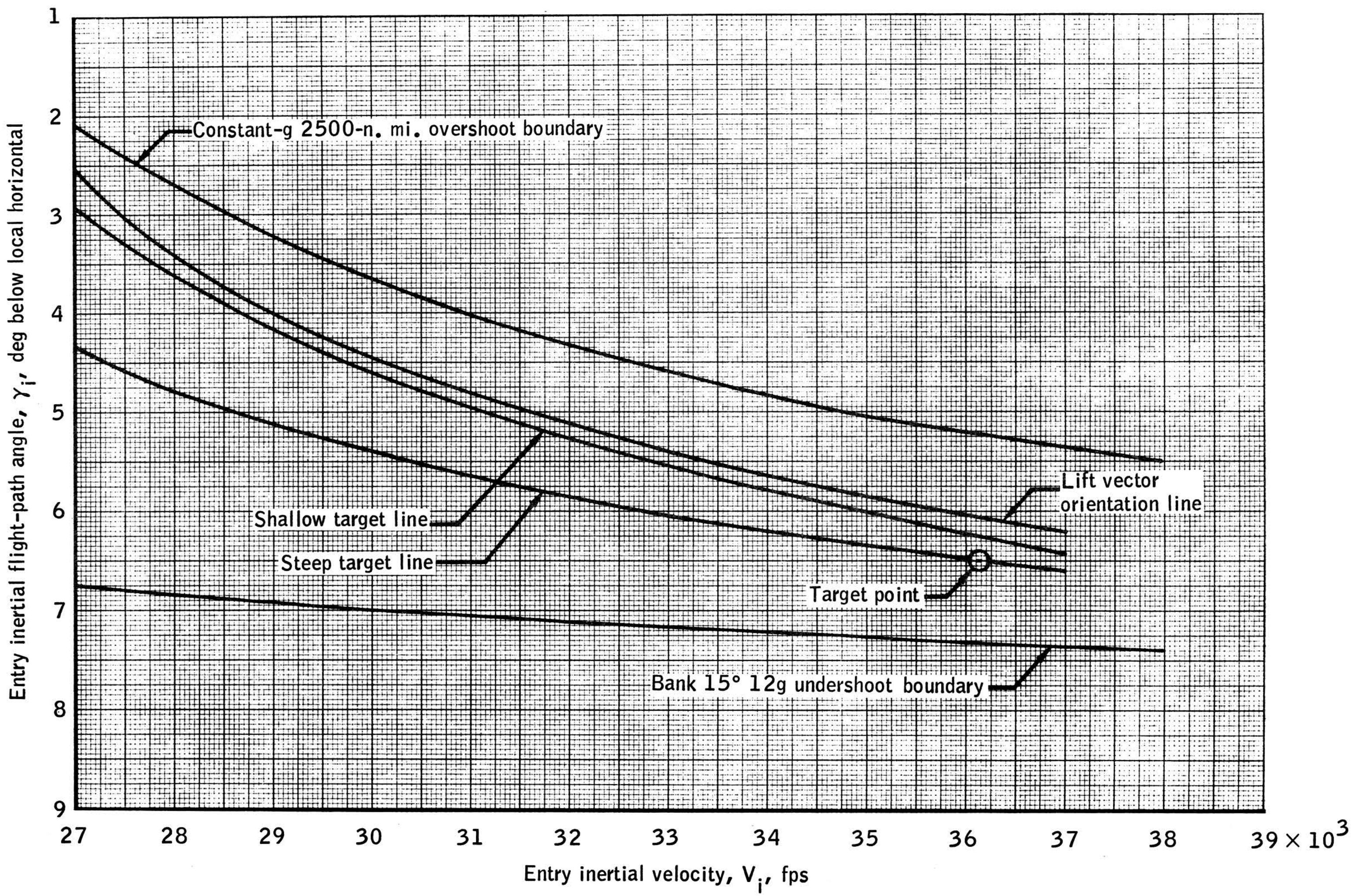


Figure 30.- Entry corridor.

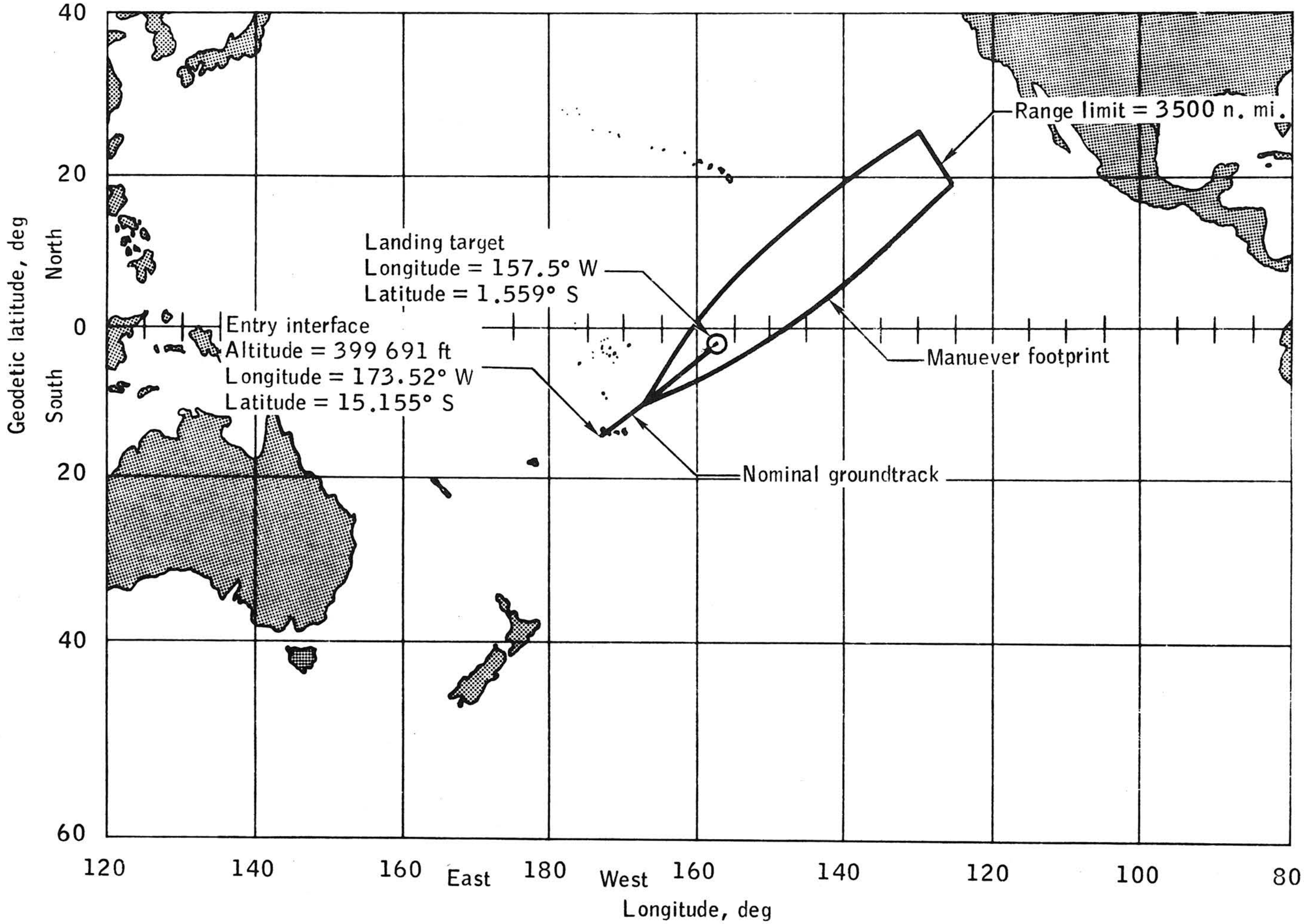


Figure 31.- Maneuver footprint and nominal groundtrack.

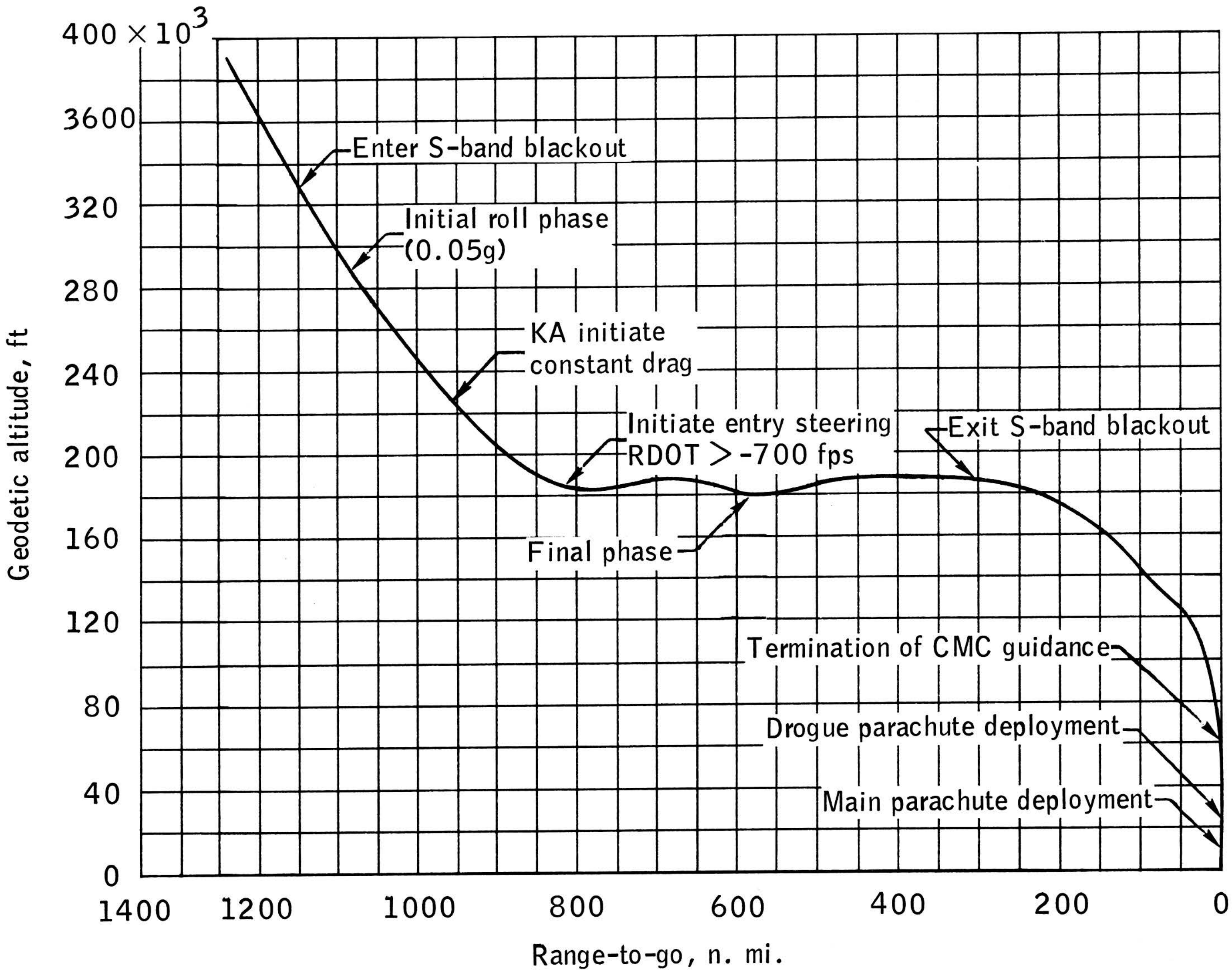


Figure 32.- Geodetic altitude versus range-to-go.

APOLLO MILEAGE CONVERTER

Nautical miles	Statute miles	Nautical miles	Statute miles
1	1.1508	600	690.4800
2	2.3016	700	805.5600
3	3.4524	800	920.6400
4	4.6032	900	1 035.7200
5	5.7540	1 000	1 150.8000
6	6.9048	2 000	2 301.6000
7	8.0556	3 000	3 452.4000
8	9.2064	4 000	4 603.2000
9	10.3572	5 000	5 754.0000
10	11.5080	6 000	6 904.8000
15	17.2620	7 000	8 055.6000
20	23.0160	8 000	9 206.4000
25	28.7700	9 000	10 357.2000
30	34.5240	10 000	11 508.0000
35	40.2780	20 000	23 016.0000
40	46.0320	30 000	34 524.0000
45	51.7860	40 000	46 032.0000
50	57.5400	50 000	57 540.0000
55	63.2940	60 000	69 048.0000
60	69.0480	70 000	80 556.0000
65	74.8020	80 000	92 064.0000
70	80.5560	90 000	103 572.0000
75	86.3100	100 000	115 080.0000
80	92.0640	200 000	230 160.0000
85	97.8180	300 000	345 240.0000
90	103.5720	400 000	460 320.0000
95	109.3260	500 000	575 400.0000
100	115.0800	600 000	690 480.0000
200	230.1600	700 000	805 560.0000
300	345.2400	800 000	920 640.0000
400	460.3200	900 000	1 035 720.0000
500	575.4000	1 000 000	1 150 800.0000

APOLLO SPEED CONVERTER

Ft per sec	Statute mph	Ft per sec	Statute mph
1	.6818	175	119.3150
2	1.3636	200	136.3600
3	2.0454	300	204.5400
4	2.7272	400	272.7200
5	3.4090	500	340.9000
10	6.8180	1 000	681.8000
15	10.2270	2 000	1 363.6000
17	11.5906	3 000	2 045.4000
20	13.6360	4 000	2 727.2000
25	17.0450	5 000	3 409.0000
30	20.4540	10 000	6 818.0000
35	23.8630	15 000	10 227.0000
40	27.2720	20 000	13 636.0000
50	34.0900	25 000	17 045.0000
60	40.9080	30 000	20 454.0000
75	51.1350	35 000	23 863.0000
85	57.9530	40 000	27 272.0000
100	68.1800	45 000	30 681.0000
125	85.2250	50 000	34 090.0000
150	102.2700		