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TRAJECTORY AND RENDEZVOUS ASPECTS OF MANNED LUNAR LANDINGS

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INTRODUCTION

Within the general theme of this first Space Congress, "Where are we going in space," this paper considers some of the factors involved in the design of trajectories and associated orbital maneuvers for accomplishing manned lunar exploration. The emphasis here is on missions using the Lunar Orbit Rendezvous technique. The discussion is primarily concerned with aspects of trajectory design which are of general interest for the whole spectrum of such missions and which are independent of any particular spacecraft or launch vehicle.

Figure 1 presents a typical mission profile based on the Lunar Orbit Rendezvous technique. The overall mission has been divided into a number of phases. The vehicle is launched from a fixed launch site (Cape Kennedy). The main requirement on the launch vehicle is that it have the capability of boosting the spacecraft to earth orbital velocity at some median earth altitude. The earth parking or coasting orbit phase is designed to provide mission flexibility by allowing launch at any time of the month, as will be discussed in more detail later. A precise definition of the injection conditions for the earth-moon transfer trajectory is required because these injection conditions are directly related to the geometric characteristics of the lunar orbits that can be established. The midcourse corrections are designed to eliminate errors in the transfer trajectory resulting from off-nominal injection conditions. In order to establish a close circular orbit about the moon, it is necessary to perform a "braking maneuver" upon close approach to the lunar surface. The retrovelocity increment that must be applied is equal to the difference between the approach speed (hyperbolic) and the circular satellite speed at a given lunar altitude. Other items of interest in establishing the lunar orbit are the orientation of this orbit and how the orbit orientation affects the lunar landing sites that can be attained. Once the lunar orbit is defined attention is focused on the lunar descent phase, which is performed by a landing module which is detached from the orbiting vehicle. There are several possible methods of performing the landing maneuver; however, particular attention is given here to the maneuver that employs a small plane change capability during landing. The time available for surface exploration is closely related to both the landing and the take-off maneuvers.

The end result of the lunar ascent maneuver is to effect a rendezvous with the orbiting vehicle. After the rendezvous is completed, preparations are made for the return to earth, and the vehicle is injected into the moon-earth return trajectory. The midcourse corrections adjust the return trajectory for proper reentry conditions into the earth's atmosphere and the reentry is followed by final descent to the surface.

The essential feature of the Lunar Orbit Rendezvous technique for manned lunar landings is that the actual landing phase of the mission, parts 6 and 7 of the mission profile, is performed by a relatively small component of the overall vehicle. This landing module contains only the engines, fuel, men, and equipment required for the landing, ascent, and rendezvous phase of the mission, while the portion of the vehicle remaining in the lunar orbit contains the more extensive equipment required for all other phases of the mission. As compared with a more direct procedure, in which the whole vehicle would similarly be established in a close lunar orbit for safety and reconnaissance purposes, but in which the whole vehicle would be landed on the lunar surface, the Lunar Orbit Rendezvous technique gives a considerable weight advantage.

This weight advantage is primarily due to the savings in fuel required to perform the descent and ascent maneuvers. The descent and ascent maneuvers each require a velocity increment of the order of 6000 ft/sec. The weight of fuel required to obtain a given velocity increment is directly proportional to the weight of the vehicle. Since the weight of the landing module in the Lunar Orbit Rendezvous technique is considerably less than the weight of the overall vehicle, a considerable reduction in fuel weight is obtained for this phase of the mission. Proportional reductions in fuel tankage, engines, and associated components also contribute to the weight advantage.

This weight advantage is illustrated in Figure 2, which presents a ratio of the weight required at injection into the earth-moon transfer trajectory for the direct landing method to that required for the Lunar Orbit Rendezvous method, as a function of propulsive efficienty (specific impulse). The main factor to be considered here is that as the propulsive efficiency increases, the Lunar Orbit Rendezvous weight advantage decreases. However, even for the most efficient present day fuels, the Lunar Orbit Rendezvous technique has a weight advantage by a factor of two or more. This means that a smaller booster vehicle can be used for a given mission, or that more efficient use can be made of the capabilities of a given booster vehicle. The efficiency increase is accompanied by the somewhat more complex operational requirements of performing the rendezvous in lunar orbit, but this is overshadowed by the weight advantage.

Before proceeding to a discussion of trajectory analysis for the various phases of the mission, it should be mentioned that just about any lunar mission involves a large number of variables, all of which affect the trajectory design to some extent. Figure 3 lists some of the primary considerations which will be included in the present discussion. Most of these factors have already been mentioned in connection with the mission profile. The problem in trajectory design for a manned lunar mission is to define the trajectory parameters for obtaining minimum mission complexity and for obtaining minimum interference between the various requirements in the mission profile.

LAUNCH-TO-INJECTION PHASE

The first phase of the mission which must be considered is the phase involving lift-off, boost into a nearly circular parking or coasting orbit about the earth, and an additional boost to give the vehicle the velocity increment required for the earth-moon transfer trajectory. This launch-to-injection phase has the overall requirements of bringing the velocity of the spacecraft up to a speed within about 1 percent of the escape velocity, and positioning the spacecraft so that the velocity vector will result in a trajectory that passes close to the moon.

The constraints which influence the launch-to-injection phase are primarily geometrical. Even in this first phase of the mission, however, consideration must be given to desirable conditions and flexibility in the later phases. For example, it is desirable to retain the flexibility of arriving at the moon with a specified lighting condition on the lunar surface. From month to month, a specified lighting condition will occur at different positions of the moon in its orbit about the earth, and thus for different declinations of the moon. Therefore, this flexibility requires the capability for injection conditions which give a close approach to the moon for any lunar declination. It will be shown that this requirement can be satisfied by use of a variable coasting time in the earth parking orbit before final injection into the translunar trajectory.

Specification that launches be initiated from Cape Kennedy, at a latitude of about 28-1/2° north, influences the geometry of the earth-moon trajectory. The orbit of the moon about the earth has an inclination to the earth equatorial plane that varies from about 18-1/2° to 28-1/2° over a 19-year cycle. Trajectories to the moon which lie in the earth-moon plane can be established only when the launch site latitude is less than the inclination of the earth-moon plane, if propulsively-expensive plane-change maneuvers are to be avoided. It happens that a situation exists, in 1968, in which the latitude of Cape Kennedy is equal to the inclination of the earth-moon plane and it would be possible, in theory, to establish lunar trajectories in the earth-moon plane. However, making use of this situation would involve severe restrictions from planning and operational standpoints. Thus, the earth-moon transfer trajectory will not in general lie in the earth-moon plane, and consideration must be given to the orientation of the trajectory plane with respect to the earth-moon plane.

The next figure (Fig. 4) shows the relationship between the transfer trajectory plane and the earth-moon plane. The earth-moon plane is the plane of the moon's orbit about the earth, and the transfer trajectory plane is defined by the injection conditions and i_o which is the angle between these two planes. Launch constraints that influence the injection conditions of the translunar trajectory are the location of launch site (latitude and longitude) which is fixed at Cape Kennedy and the launch azimuth, which is restricted by the range safety requirements. Within these constraints, it has been shown that the angle i_o between the trajectory plane and the earth-moon plane is a function of the launch azimuth and the declination of the moon. A later phase of the mission influences the desirable values of i_o in that establishment of low inclination lunar orbits, with a minimum expenditure of fuel, requires that this angle between the trajectory plane and the earth-moon plane be as small as possible. Minimizing the angle between these two planes is thus advantageous in establishing low inclination orbits, and does not appreciably restrict the establishment of high inclination orbits. This angle, i_o, is also important from the guidance standpoint because the guidance requirements for establishing close lunar satellite orbits become more stringent as the angle i_o increases, and the overall accuracy decreases somewhat as the angle between these planes increases.

The injection phase of the trajectory design problem has been analyzed with due consideration given to the various desirable conditions and constraints mentioned above. The analysis indicates that introduction of a variable coast phase in the earth orbit before injection into the lunar trajectory gives the desired flexibility of initiating lunar missions on any day throughout the month.

Figure 5 shows some typical results of this analysis. The ordinate is the length of the coasting arc and the abscissa represents the angular position of the moon for a month. The bounding curves represent the range safety restrictions, with allowable azimuths between 40° and 115° east of north. The azimuth limits separate the figure into two regions; the top region is for relatively long coasting arcs and the bottom is for short arcs. Small values of i_o occur for certain positions of the moon for long coasting arcs and for certain different positions for short coasting arcs. The result is that reasonably small values of i_o can be obtained for any day in the month by proper choice of azimuth and coasting arc.

Figure 6 is another plot of these results. It shows that i_o can be maintained at values less than about 20° for any declination of the moon, within the range safety requirements, by using the pertinent coast arc length.

Figure 7 is an end result of the injection phase analysis and shows the loci of points for injection into the lunar trajectory for various azimuths and lunar declinations. The coast arc lengths are drawn at 30° intervals from the launch site. This particular plot is for a typical trajectory with an earth-moon transit time of about 60 hours and with earth-moon plane inclination of 23°. The loci of injection points is modified somewhat for different transit time and plane inclinations.

To quickly summarize the injection phase of lunar mission, we can say that for properly chosen coasting arcs, the azimuth safety limits and launch site location produce no restrictions on the possible launch days during the month, and the angle between the trajectory plane and the earth-moon plane can always be kept less than about 20° throughout the entire month.

EARTH-MOON TRAJECTORY PHASE

The next phase of the lunar mission is the earth-moon transfer trajectory phase. The purpose of this phase is to place the spacecraft at a proper point relative to the moon for initiation of a retromaneuver to establish a close lunar orbit.

In order to study general trends, such as the overall characteristics of transfer trajectories and the lunar orbits that can be established from a given transfer trajectory, a series of "two-body" approximations have been made. The approximation scheme is known as the "patched-conic" or "sphere of influence" technique. Figure 8 gives an overall illustration of the "patched-conic" technique. It shows the earth, the moon and its imaginary selenocentric "sphere of influence." The assumptions involved are:

(1) The moon moves in a circular orbit about the earth at the mean earth-moon distance.

(2) The spacecraft is subjected to only the earth's gravitational force when it is outside the lunar sphere of influence.

(3) The spacecraft is subjected to only the moon's gravitational force when it is inside the lunar sphere of influence.

The so-called "patch-conic" technique got its name from the discontinuity which exists at the sphere of influence. At this point an earth-centered two-body conic section is "patched" or "matched" to a selenocentric two-body conic section. The advantage of using the patched-conic method is that closed-form analytical expressions are available for two-body orbits, as for motion of a spacecraft with respect to the earth or for motion with respect to the moon, whereas no such analytical expressions exist for motion involving a spacecraft influenced by the gravitational fields of both the earth and moon simultaneously; the so-called three-body problem. Calculation of trajectories using the three-body problem involves numerical integration of the equations of motion on high-speed digital computers and is inconvenient for generalized studies. The patched-conic method is also programed for digital computation, and many trajectories can be computed in a small amount of machine time.

The analytical expressions for the two-body problem give the vehicle position and velocity components with respect to the earth as a function of the injection conditions. At the sphere of influence, the motion of the moon is taken into account, and the position and velocity components of the vehicle relative to the moon can be determined. In this manner the two parts of the earth-moon transit trajectory can be matched, and the characteristics of trajectories in the vicinity of the moon can be determined from the injection conditions and the points of intersection with the sphere of influence. In general, the earth-centered position of the trajectory is a highly elongated ellipse with focus at the earth, and the moon-centered portion is a hyperbola with focus at the moon. This simplified approach is sufficiently accurate to define transfer trajectory characteristics for generalized studies, and provides the necessary data for first estimates of trajectory parameters to be used in precise calculations.

For trajectories which pass close to the moon, and from which close lunar orbits can be established, the target point on the sphere of influence is defined for a given set of earth injection conditions. That is, if the energy $(V/VP)_o$ and the inclination (i_o) of an earth-moon transfer trajectory is specified, then the target point on the sphere of influence is defined. Figure 9 shows the locus of target points on the sphere of influence for a range of transfer trajectory inclinations (i_o) and earth-injection velocity ratios $(V/VP)_o$. The ratio of injection velocity to the escape velocity defines the transit time of the trajectory where $(V/VP)_o = 0.992$ corresponds to a flight time of approximately 70 hours and $(V/VP)_o = 1.0$ corresponds to a flight time of approximately 40 hours. The selenocentric sphere of influence is shown in the upper portion of the figure. The radius of the sphere is approximately 36,000 miles and the coordinates of the target point are given by η (latitude) and ξ (longitude) with positive directions as shown. Here ξ is referenced to the earth-moon line and η is referenced to the earth-moon plane. The shaded region defines the location of the entry points on the sphere of influence. The lower part of this figure presents a plot of the coordinates of the target point (ξ, η) on the sphere of influence for earth-moon injection conditions that range from 0.992 to 1.0 for injection velocity ratios and 0° to $\pm 90^\circ$ for transfer trajectory inclinations. As an example, if the energy of the transfer trajectory is specified as $(V/VP)_o = 0.994$ (transit time from earth to moon of about 60 hours) and the inclination of the transfer trajectory is specified as $(i_o) = 30^\circ$, then the coordinates of the associated entry point are $\xi = 39^\circ$, $\eta = 4.5^\circ$.

Thus, for a given transfer trajectory there exists a specified target point on the sphere of influence that the vehicle must pass through prior to establishing a close lunar orbit. Conversely, the target point defines the characteristics of the trajectory in the vicinity of the moon, particularly the distance of closest approach to the moon and the orientation of the trajectory plane with respect to the moon. Here again, a later phase of the mission influences the desired approach trajectory orientation, so at this point the next mission phase should be considered.

LUNAR ORBIT, LANDING, AND EXPLORATION PHASE

The next phase of the mission is the lunar orbit, landing, and exploration phase, in which consideration must be given to three pertinent items.

(1) Establishment of a lunar orbit.
(2) The possible lunar landing sites.
(3) Time available for exploration.

An important factor in the choice of landing sites and exploration times is that these must be compatible with performing a rendezvous maneuver of the landing vehicle with the vehicle which remains in the lunar orbit, so these three items are closely interconnected. Under these conditions the geometric characteristics (inclination i and nodal position Ω) of the established lunar orbit are the same as those of the approach hyperbola from which the orbit was established.

Now consideration must be given to what landing sites are accessible from a given orbit and how the geometric orientation of the established orbit may affect the possible landing sites on the lunar surface. In view of the prohibitive expense involved in making large orbital plane changes, it is expected that the exploration module will land nearly in the orbital plane of the command vehicle. With this consideration in mind, the landing sites which can be achieved form a band on the surface of the moon directly beneath the lunar orbit, as illustrated in Figure 10. The width of this band is determined by the amount of onboard fuel available for making small orbital plane changes during the landing maneuver. Therefore, it can be seen that the location of the possible landing sites are strongly dependent on the geometric characteristics (inclination i and ascending node Ω) of the lunar orbits that can be established.

Figure 11 shows how the orientation of an orbit is defined by its inclination i, and nodal position Ω. Shown here is the lunar sphere of influence with a target point (ξ, η) obtained from a given transfer trajectory. Also shown is a lunar orbit which has been established from this transfer trajectory. An expression has been derived which relates the geometric properties (i, Ω) of the lunar orbits to the target point (ξ, η) on the sphere of influence (or to the earth injection parameters, since a specified set of earth injection conditions define a unique target point). This expression is shown on the figure. It should be noted that i and Ω are not independent parameters, That is, if i is specified, then Ω is defined by the expression shown. Therefore, a lunar orbit can be established that will pass over any point on the lunar surface by specifying the proper orbit inclination or nodal position.

The type of landing and take-off maneuvers that are employed during a lunar orbit operation have a considerable effect on the landing sites that are accessible from an established lunar orbit, and the corresponding exploration time allowed on the lunar surface. There are several different approaches that can be taken in selecting the landing and take-off maneuvers and consideration has been given to the following three basic maneuvers:

(1) Inplane landing and take-off maneuver.
(2) Inplane landing with plane change capability on take-off.
(3) Plane change capability during both landing and take-off maneuvers.

The inplane landing and take-off maneuver is rather restrictive with respect to exploration time. For the present discussion, plane change capability during both landing and take-off maneuvers will be considered.

Now we would like to consider the allowable stay times on the lunar surface as defined by lunar orbit rendezvous considerations. It is not anticipated that there will be many problems associated with short term lunar landings with stay times ranging up to 18 hours or so; however, the longer periods of exploration may involve some restrictions. Figure 12 illustrates the landing and take-off maneuvers and the exploration times allowable on the lunar surface. Now suppose we wish to land at some latitude (l) on the lunar surface. One method of doing this is by establishing a lunar orbit with an inclination i greater than the required latitude by the offset angle δ $(i = l + \delta)$. The angle δ is a function of the plane-change capability of the vehicle and is directly related to the amount of fuel required for the plane change. When the exploration vehicle starts its descent to the lunar surface a small out-of-plane impulse is applied so that the path is along the dashed line and the landing site is δ^{o} out of the orbital plane. Because of the moon's rotation on its axis, there will be a relative motion between the landing site and the orbital plane. On the figure the landing site will appear to move to the right along a parallel of latitude at about 13.2^{o}/day. After a certain interval $\Delta\theta$ the landing site will again have an offset of δ^{o} and the "return to orbit" phase of the mission must be initiated. The exploration time on the lunar surface, which is a function of the landing site latitude, is given in the upper right-hand corner. At any time during this period, the vehicle can return to the orbiting satellite with not more than an offset of δ^{o} required. This provides a built-in safety factor in the event an immediate return to the orbiting vehicle is required during the exploration period. It should be noted that the maximum exploration time exists when the landing is made δ^{o} out of the orbit plane; however, the spacecraft is not restricted to landing at point A. By shortening the length of time available for exploration of the lunar surface, the spacecraft could land at any point along the selected lunar latitude between points A and B.

The allowable exploration time is a function of the landing site on the lunar surface and of the offset angle δ. Figure 13 shows a typical plot of landing sites and corresponding exploration times for a landing and take-off maneuver with an offset $\delta = 5^{o}$. These data were obtained from a median energy transfer trajectory. The shaded regions on the left denote landing sites that cannot be attained efficiently with a median energy transfer trajectory unless a wait period in orbit is considered. The other curves define the attainable landing areas and their corresponding stay times in days. Also shown on this slide are regions of indefinite stay time. These are regions in which a vehicle can land, explore the lunar surface indefinitely, and never be more than δ^{o} out of the orbit plane. These indefinite stay regions consist of circles at the poles for landings made from a polar orbit, and a band about the lunar equator for landings made from equatorial orbits. The figure seems to indicate that very little of the western sector is available for exploration. However, it should be pointed out that this figure is based on performing the landing maneuver shortly after establishing the lunar orbit. If the combined vehicle remains in the lunar orbit for some time before the landing maneuver is performed, the effect is to rotate the landing areas defined here to the left by about 13^{o} for each day in orbit. Thus, by using a lunar parking phase, the shaded regions are also available as landing areas.

From the typical landing areas and associated exploration times shown on Figure 13, it is seen that landing areas extending over a considerable part of the central lunar disk are available from low or moderately inclined lunar orbits. These landing areas provide reasonably long exploration times and are compatible with the operational and safety requirements of the Lunar Orbit Rendezvous technique.

EARTH RETURN PHASE

After completion of the lunar ascent and rendezvous maneuver, attention is turned to the Earth Return phase of the mission. Analysis of the earth return problem can be conducted along the same lines as the establishment of the lunar orbits. Hence, the lunar sphere of influence is utilized and exit points are defined, similar to the target points which were defined in the entry case. Figure 14 shows the lunar sphere of influence with both the target areas and exit areas illustrated. The region on the left of the earth-moon line is the target region that was used for establishing lunar orbits as discussed earlier and the region on the right of the earth-moon line is the exit region through which the vehicle must pass on its return to the earth. The latitude and longitude of the exit points are defined by ξ and η, respectively, and are positive as shown in the figure. The exit region has a width of approximately $\pm 10^\circ$ which includes inclinations of the moon-earth transfer trajectory between $\pm 90^\circ$. Longitudinal limits were applied to the allowable exit region for manned lunar return missions. These limits were obtained by restricting the return flight time to less than 100 hours(right-hand boundary) and a maximum allowable velocity increment for injecting the vehicle into the moon-earth transfer trajectory of approximately 3100 ft/sec (left-hand boundary). This restricted region is shaded in Figure 14. For a successful return mission within the defined limits of velocity and flight time, then the lunar trajectory characteristics must be such that the vehicle exits from the sphere of influence within this shaded region. The exact location of the exit point within this region will be determined by the return-trajectory inclination desired and the velocity required to achieve a specific longitude. If the exit point on the lunar sphere of influence has a positive latitude the vehicle will land in the earth's southern hemisphere and if the exit point has a negative latitude the landing site will be in the northern hemisphere.

In the previous section we discussed the geometric characteristics of lunar orbits, that is, the orbit inclination i and the orbit nodal position Ω, which define the orientation of a lunar orbit. Once the lunar orbit is established, the inclination (i) is fixed and the nodal position (Ω) of the orbit precesses at a constant rate (approximately 13.2°/day).

Figure 15 shows the exit region on the lunar sphere of influence as defined by the velocity increment (3100 ft/sec) and the return flight time (100 hours) limits. Superimposed on the figure are three representative lunar orbital planes which are inclined 90°, 20°, and 6.5°, respectively. Although only three orbit planes are shown, it should be noted that any number of arbitrarily inclined orbit planes could be established that would pass through the exit region. Therefore, it is possible to return to earth from an orbit that has almost any inclination desired, however, once the inclination of the lunar orbit is specified it is no longer possible to choose an arbitrary nodal position so that the orbit plane will pass through the exit area at all times. For example, if the inclination of the established lunar orbit is 20° and the longitude of its node is 120°, as shown by the dashed line in Figure 15, then a return flight is not possible at this time without implementing a plane change because the orbit plane does not pass through the exit area. Due to the motion of the moon, the trace of the established orbit will appear to precess to the left and in approximately 4 days the longitude of the node will be at about 80° and the orbit plane will pass through the exit area so a return flight can be initiated.

Now consider the lunar orbit plane in Figure 15 whose inclination is 6.5°. It can be seen that this orbit plane always passes through the exit region regardless of the nodal position, so a return to earth flight can be initiated at any time. Since the exit region on the lunar sphere of influence is bounded between the latitudes of about $\pm 10^{\circ}$, it is possible to return to the earth at any time if the inclination of the lunar orbit is less than about 10°. For lunar orbits with inclinations somewhat greater than 10°, mandatory wait periods may exist before the earth return flight can be initiated.

Figure 16 shows the relation between the geometric characteristics of the lunar orbit planes for the orbit establishment and return phases of the mission for the more highly inclined orbits. First, for orientation purposes, consider the sketch of the moon with a vehicle in an arbitrary circular orbit shown in the upper right-hand portion of the figure. The shaded areas in this sketch show the required nodal positions for lunar orbits having inclinations between 20° and 90°. If this area is viewed from the north lunar pole, the results shown in the major portion of the figure will be obtained.

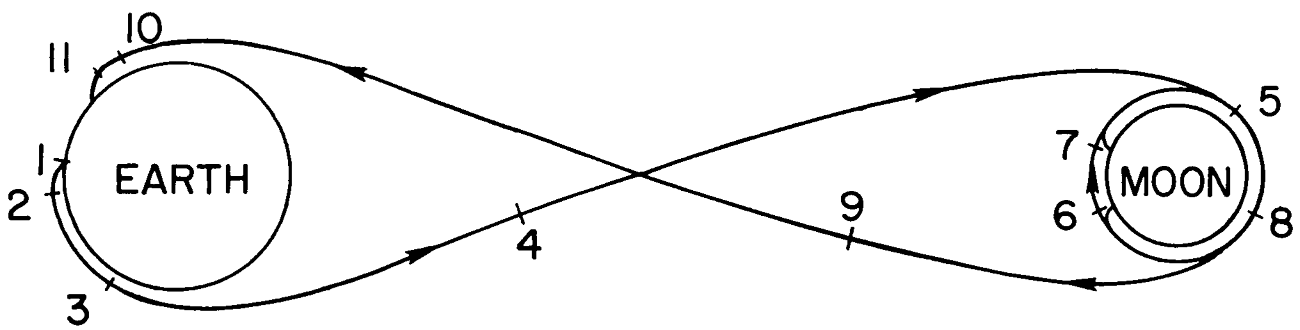
The hatched area lying between 135° and 145° represents the nodal-line positions of those lunar orbits established from a typical earth-moon transfer trajectory and having inclinations between 20° and 90°. The stippled area lying between 30° and 100° represents the range of nodal-line positions required for a return flight to either the northern or southern hemisphere of the earth. The two areas are not compatible, so a return flight cannot be initiated immediately from any lunar orbit with an inclination greater than 20°. However, after a specified wait period the return can be initiated.

As the orbital inclination is decreased below 20°, both regions begin to expand, and below 10° they overlap; thus, a return flight may be initiated at any time after orbit establishment. Therefore, in general it can be said that low inclination lunar orbits provide landing areas that have reasonably long exploration times with the capability of an earth return at any time after the orbit is established and are compatible with the Lunar Orbit Rendezvous technique.

CONCLUSIONS

It has been shown that requirements on some of the later phases of the lunar mission profile have a decided effect on some of the variables associated with earlier phases of the mission. Within the constraints of the Lunar Orbit Rendezvous technique, it is possible to land at any point on the lunar surface and return to the established lunar orbit for exploration periods on the order of 1/2 a day or less. However, longer exploration periods impose constraints on the accessible landing sites and their corresponding stay times, and for high-inclination lunar orbits there are times during the exploration period in which an immediate earth return flight cannot be initiated efficiently due to the orientation of the established lunar orbit.

Low inclination lunar orbits avoid these problems. Low inclination orbits permit landings throughout most of the lunar equatorial region, such that the exploration module can stay on the surface for reasonably long periods and have the capability of an immediate return to the lunar orbit and an immediate earth return at any time during the exploration period.



1. LAUNCH FROM CAPE KENNEDY
2. ESTABLISHMENT OF EARTH ORBIT(COASTING PHASE)
3. INJECTION INTO EARTH-MOON TRAJECTORY
4. MIDCOURSE CORRECTION
5. ESTABLISHMENT OF LUNAR ORBIT
6. LUNAR DESCENT MANEUVER
7. LUNAR ASCENT MANEUVER AND RENDEZVOUS
8. INJECTION INTO MOON-EARTH TRAJECTORY
9. MIDCOURSE CORRECTION
10. REENTRY
11. EARTH TOUCHDOWN

NASA

Figure 1.- Typical profile for a lunar orbit rendezvous mission.

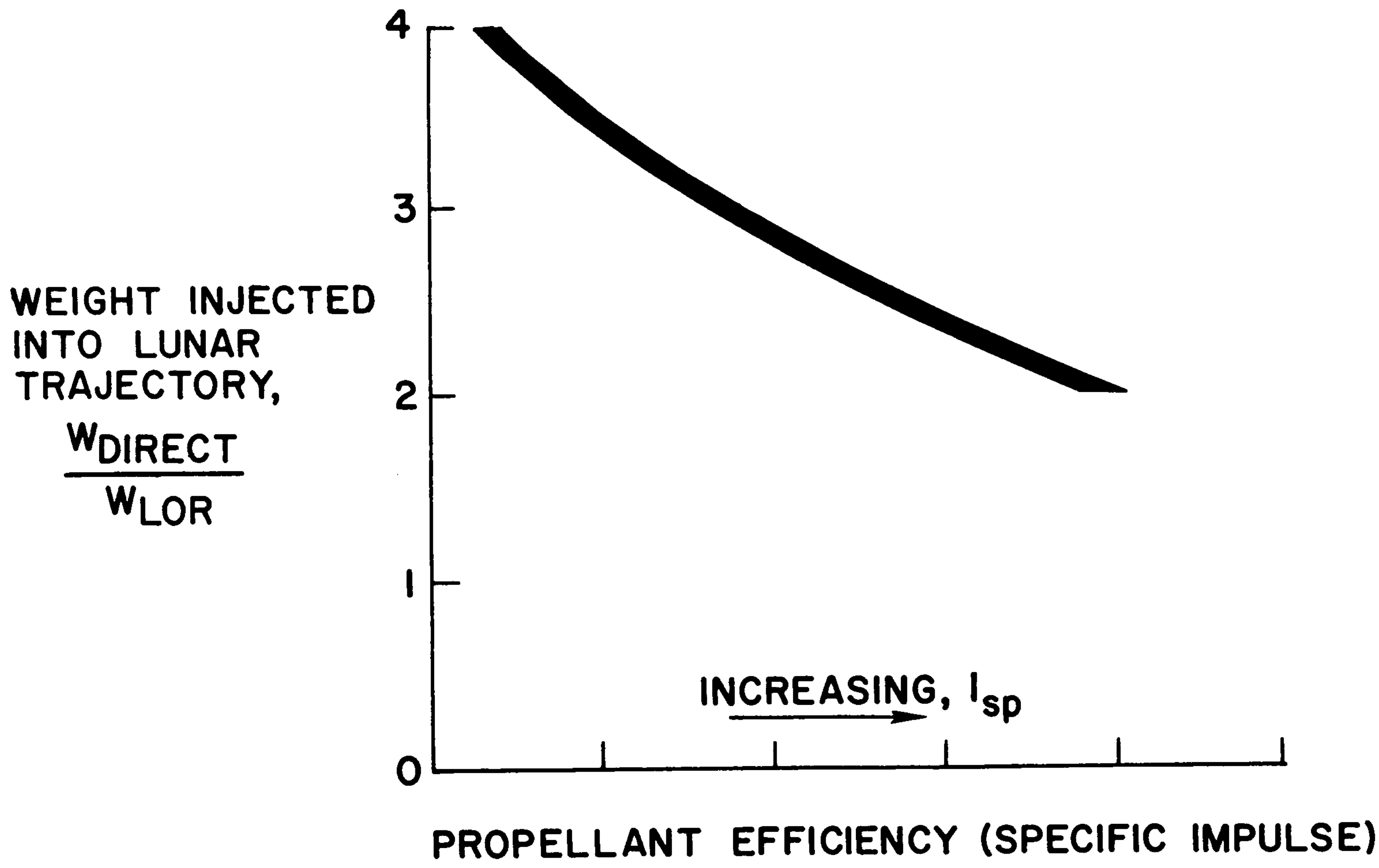


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Figure 2.- Weight advantage of lunar orbit rendezvous technique.

1. LAUNCH SITE LOCATION
2. RANGE SAFETY REQUIREMENTS
3. INJECTION CONDITIONS REQUIRED FOR CLOSE APPROACH TO MOON
4. ESTABLISHMENT OF DESIRABLE LUNAR ORBIT
5. DESIRED LANDING SITE LOCATION AND SURFACE EXPLORATION TIME
6. SURFACE LIGHTING CONDITIONS (TIME OF MONTH)
7. CAPABILITY FOR RENDEZVOUS WITH ORBITING VEHICLE
8. INJECTION CONDITIONS FOR RETURN TO EARTH FROM LUNAR ORBIT

NASA

Figure 3.- Mission considerations for lunar orbit rendezvous technique.

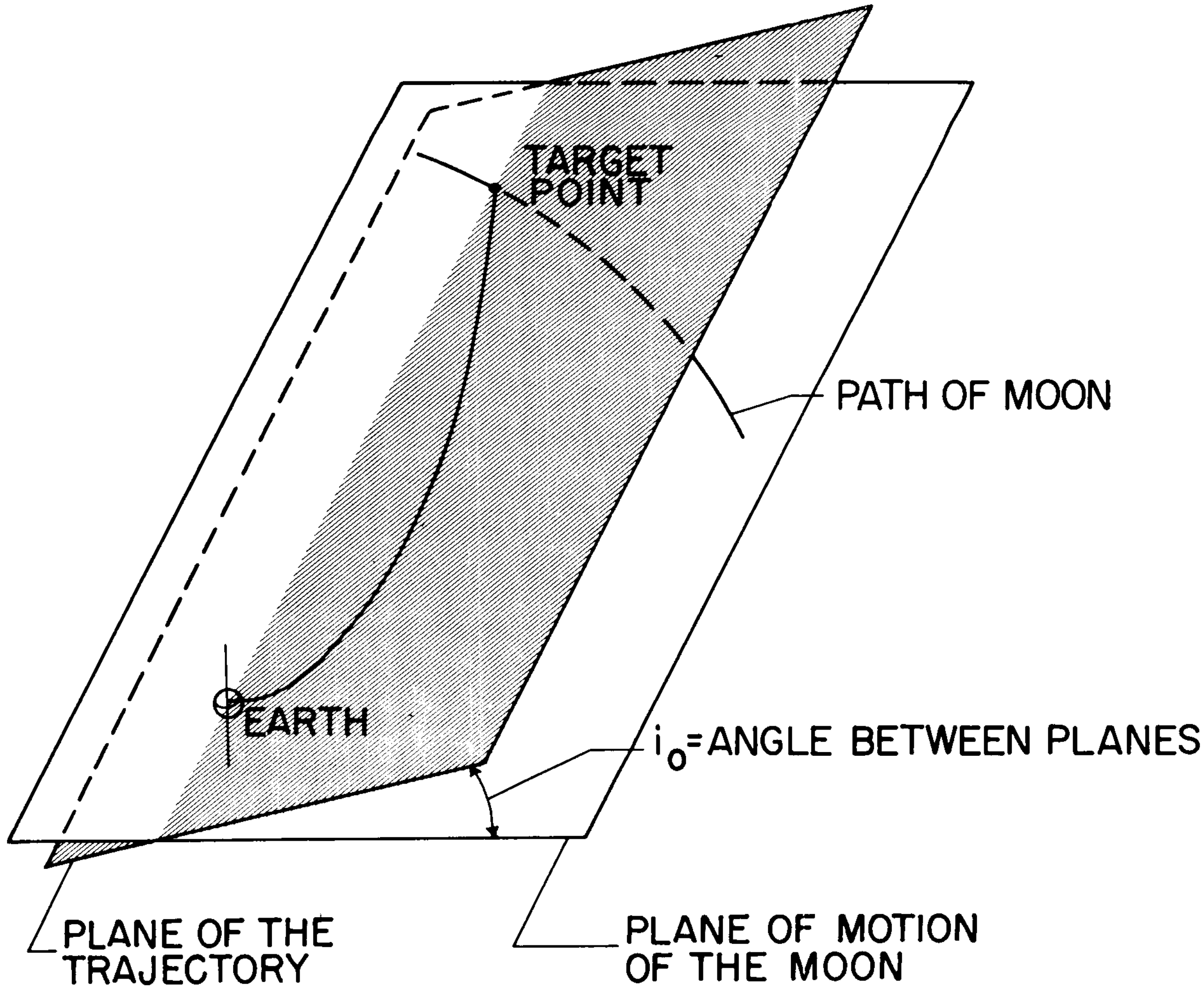

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Figure 4.- Relation between the earth-moon plane and the trajectory plane.

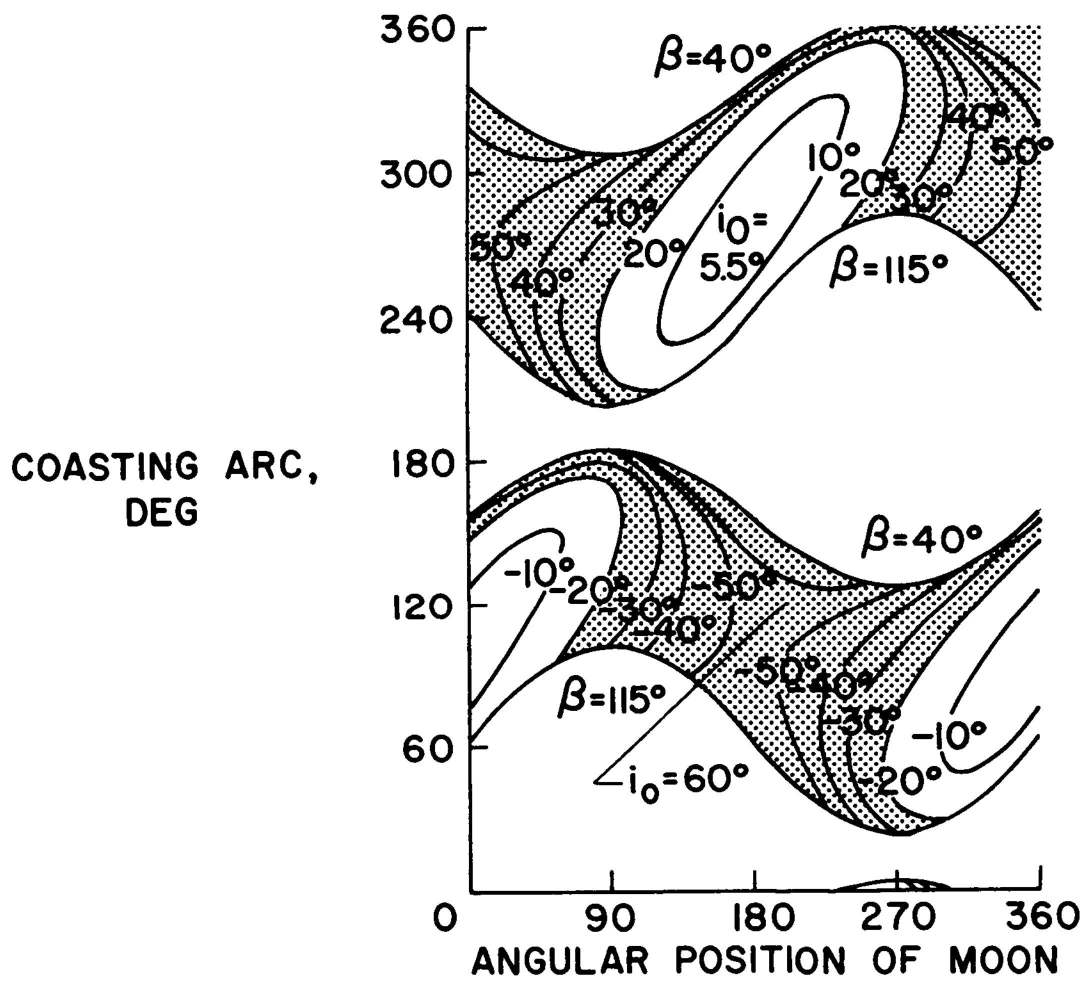

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Figure 5.- Relation between trajectory plane inclination and coasting arc.

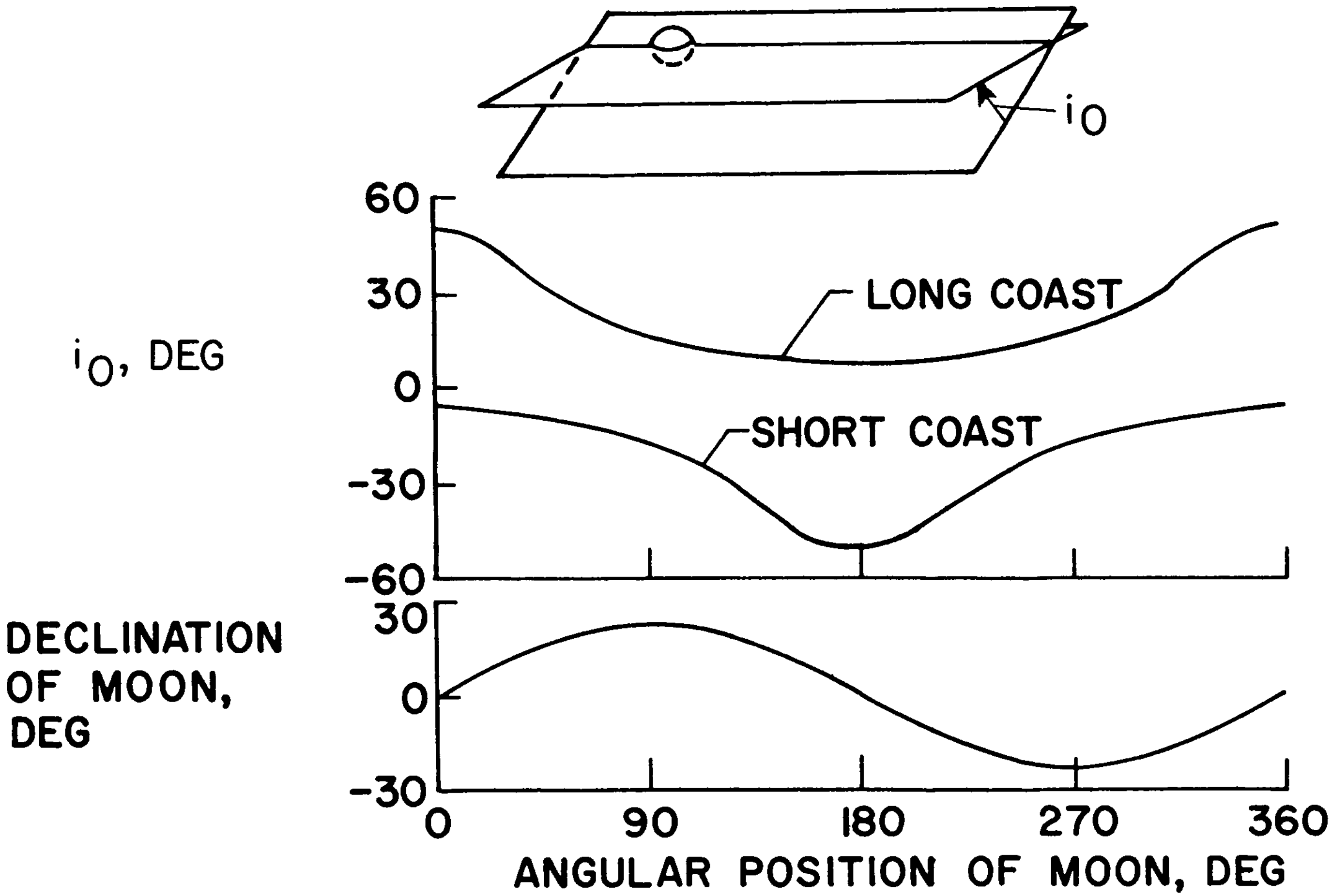

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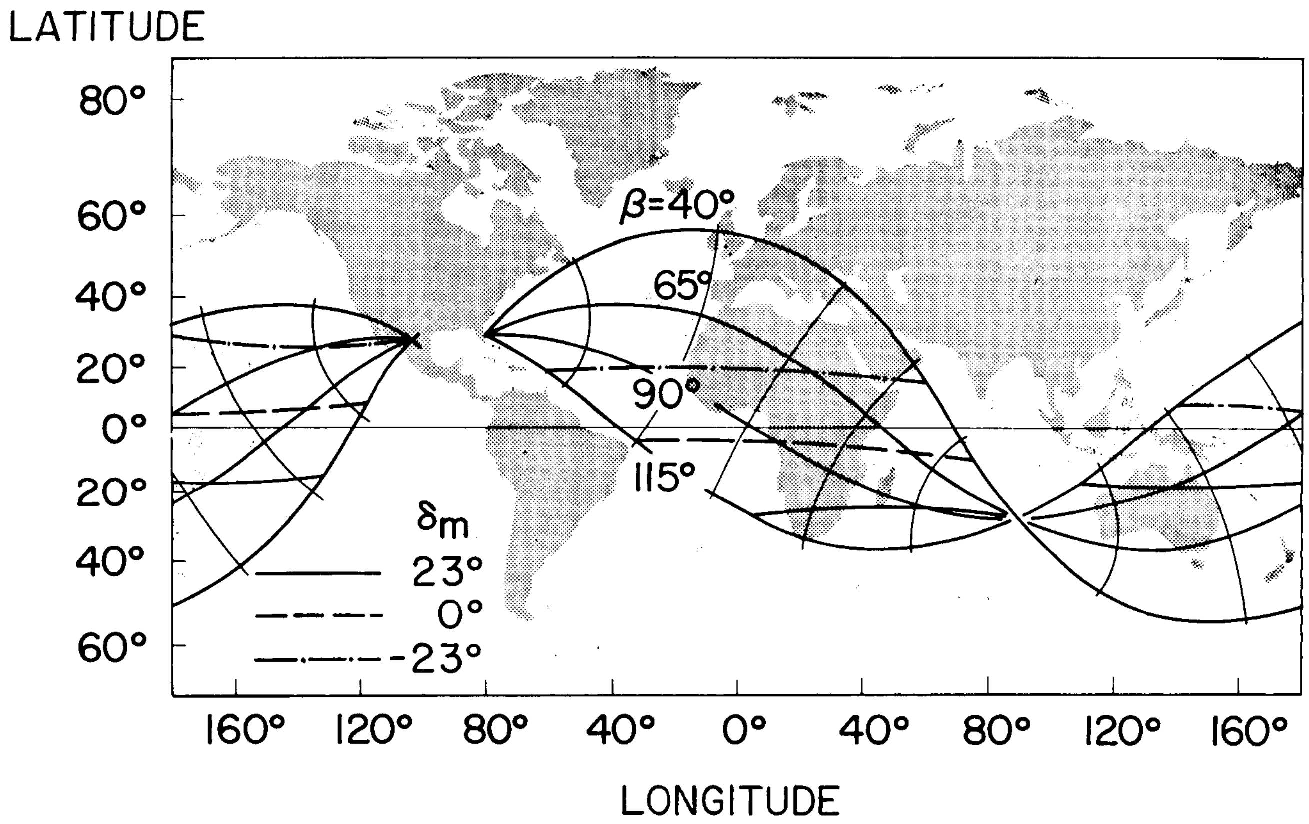
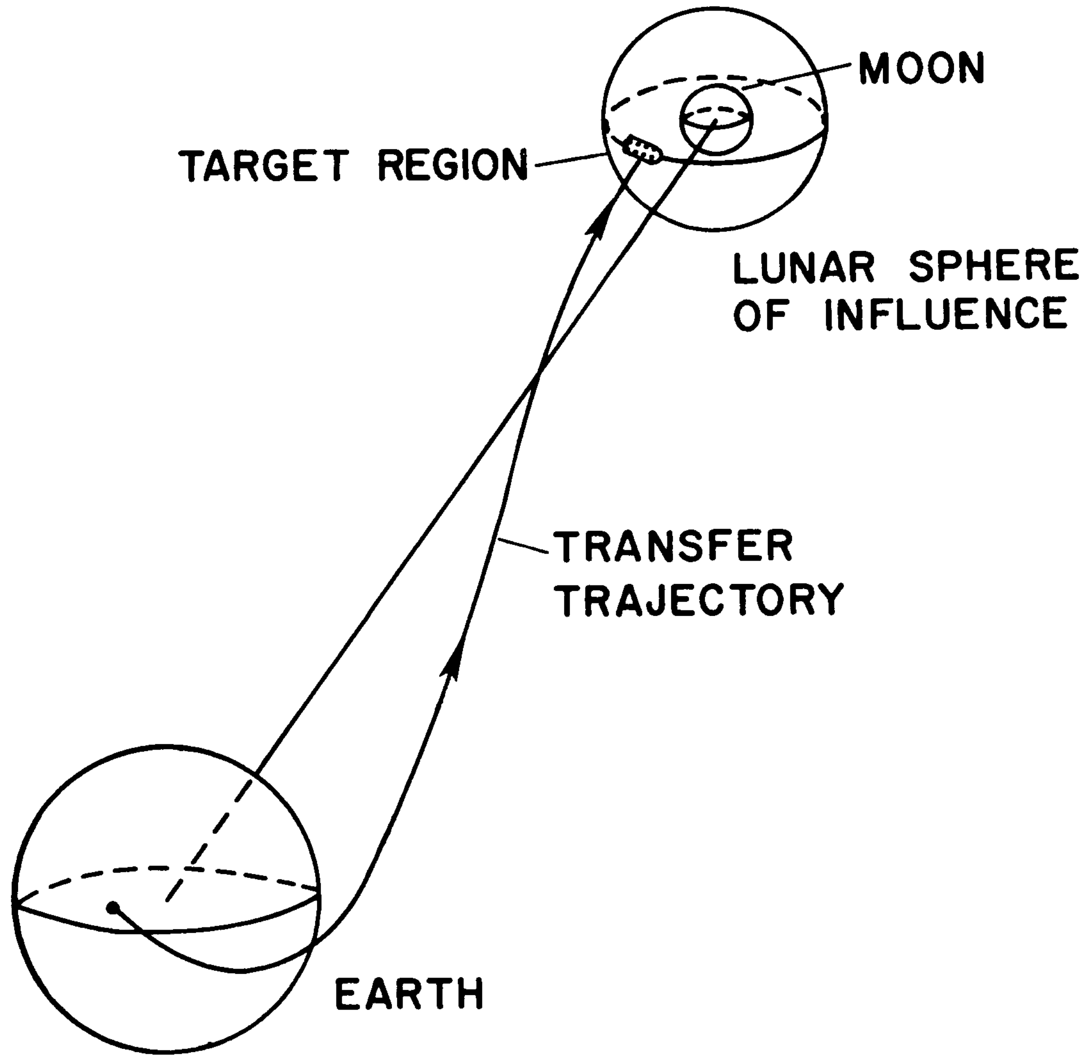
Figure 6.- Angle between initial trajectory plane and earth-moon plane.



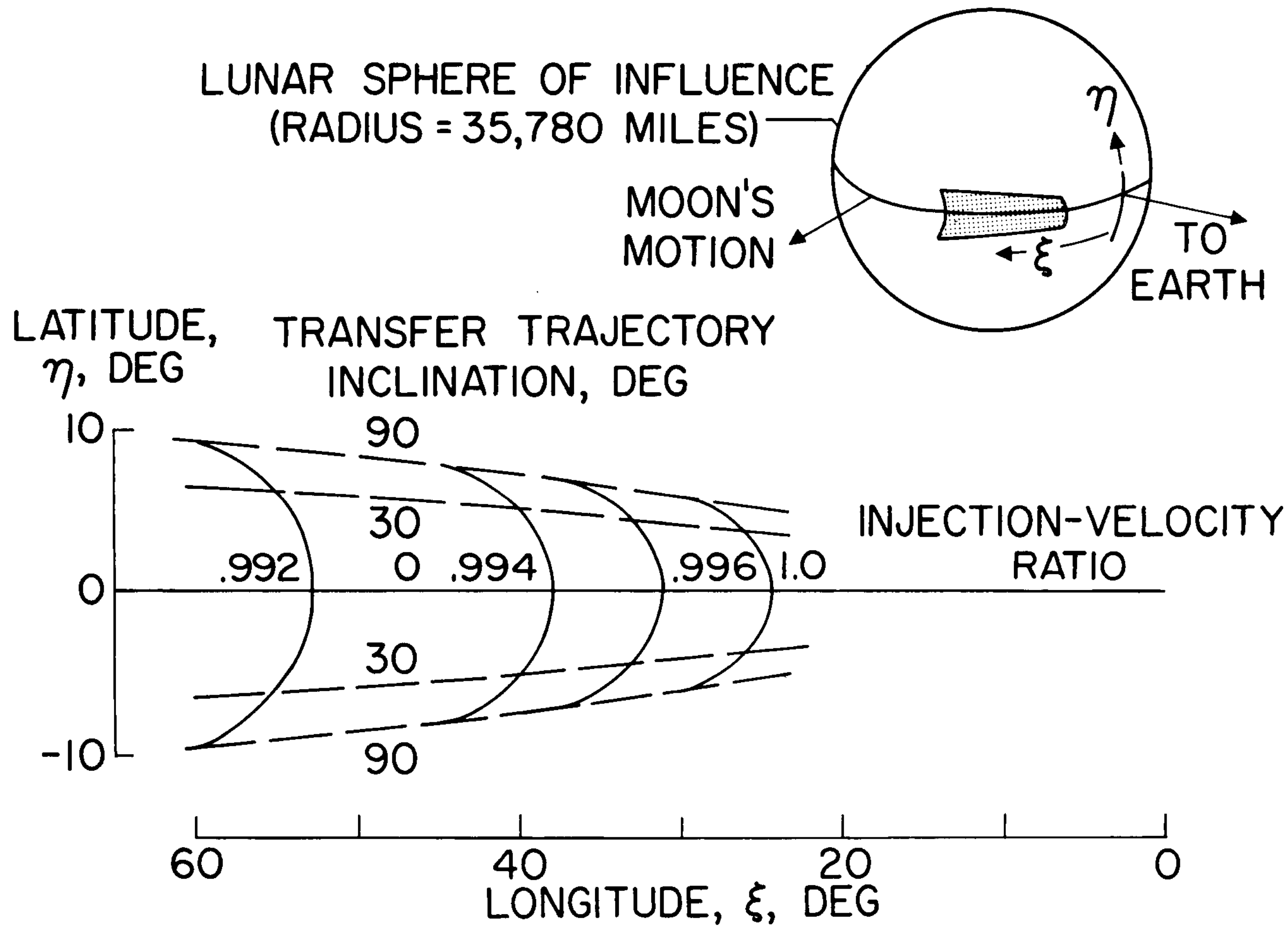
NASA

Figure 7.- Injection-point locations for approaching the moon.



NASA

Figure 8.- Patched conic approximation.



NASA

Figure 9.- Target points on lunar sphere of influence.

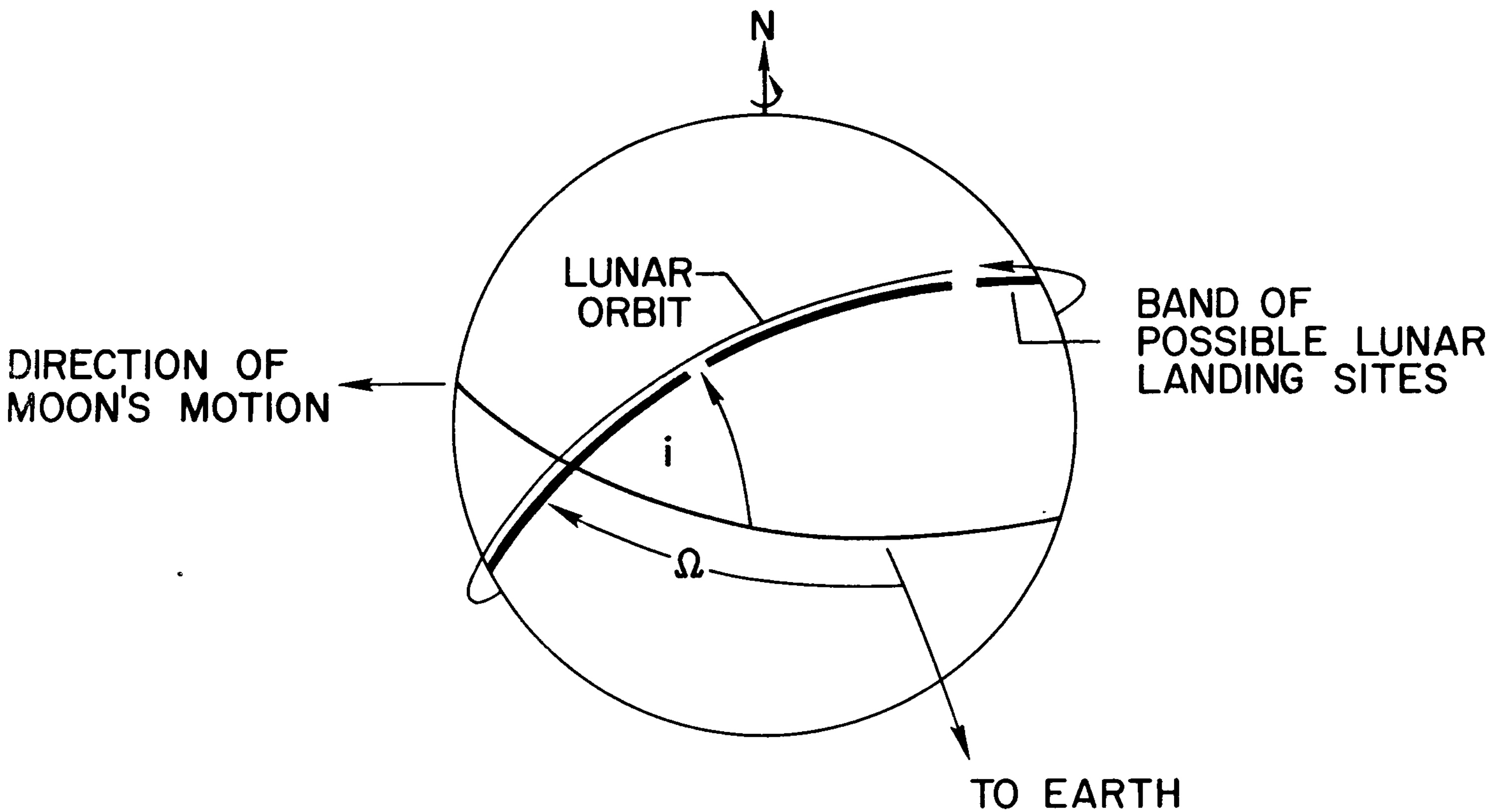


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Figure 10.- The possible lunar landing sites.

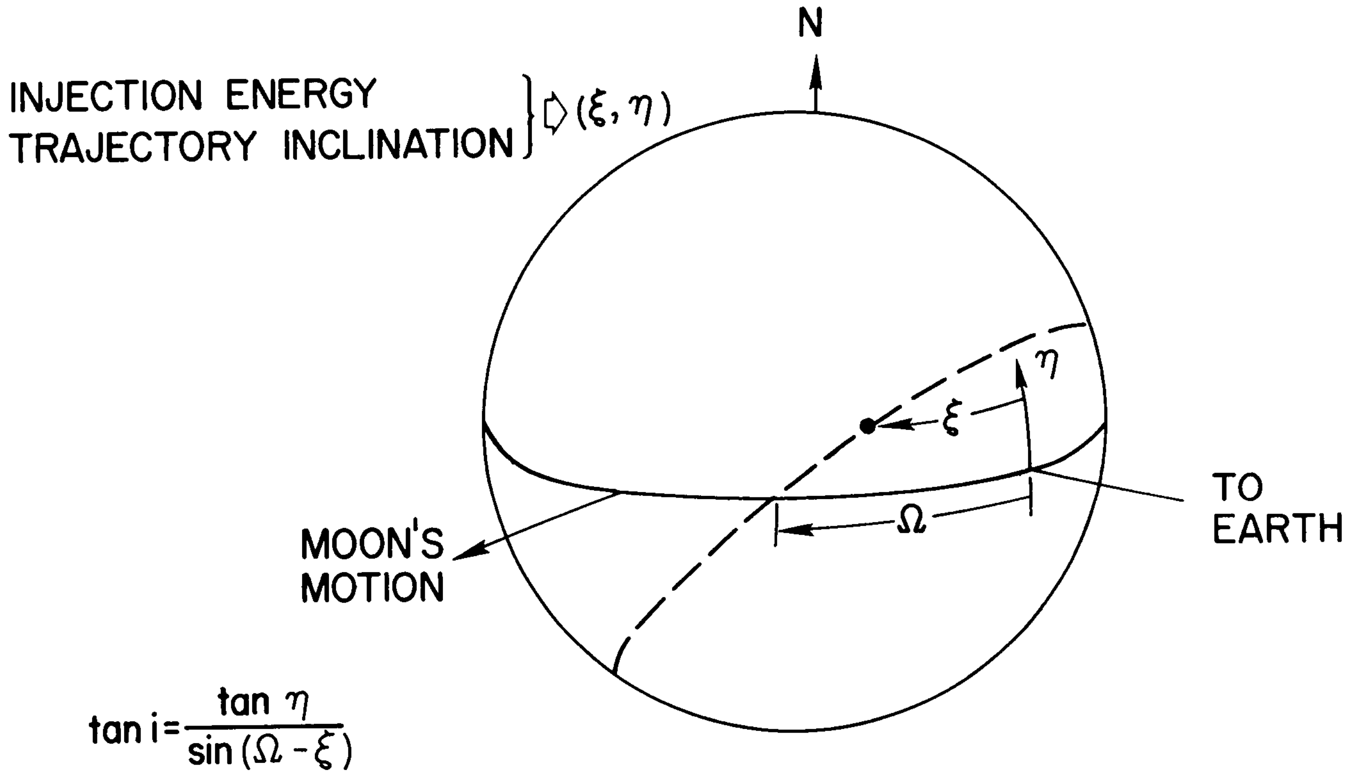

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Figure 11.- Geometric characteristics of lunar orbits.

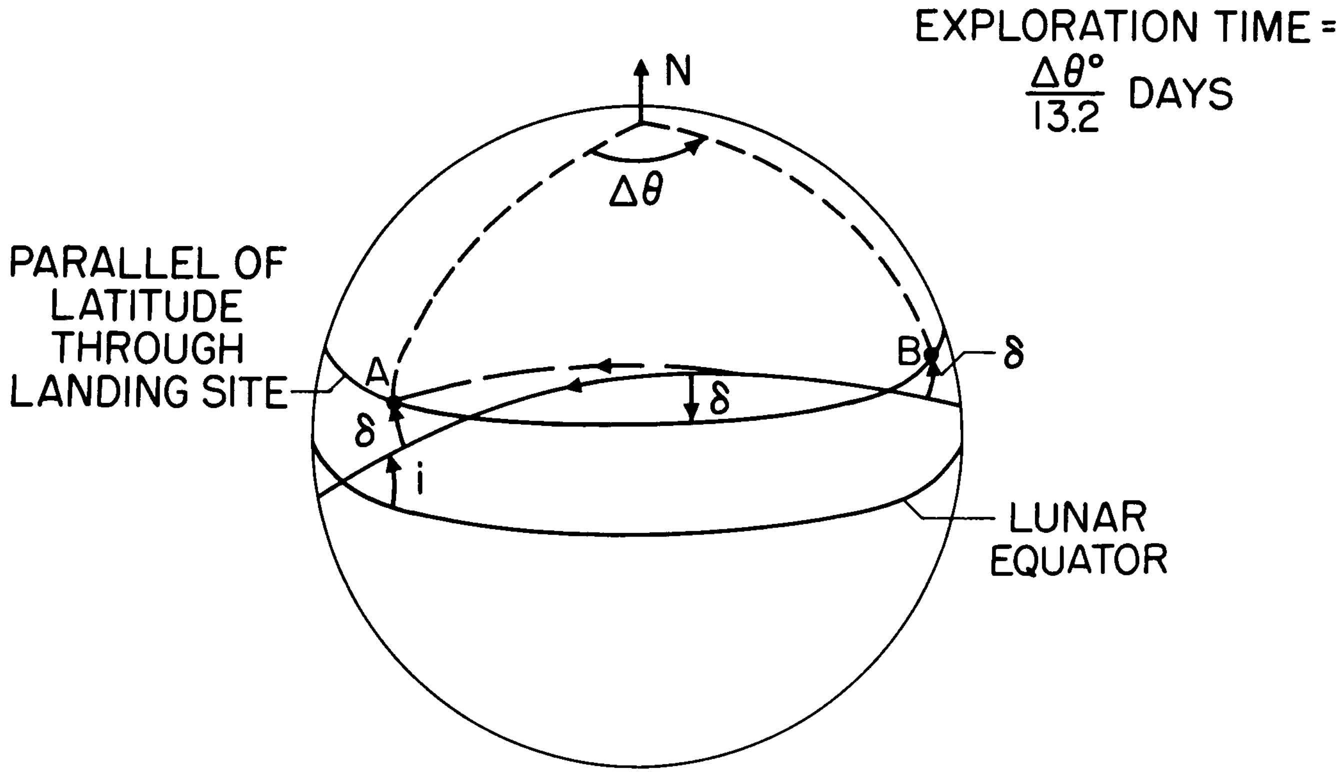

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Figure 12.- Lunar landing and take-off maneuvers from rendezvous considerations.

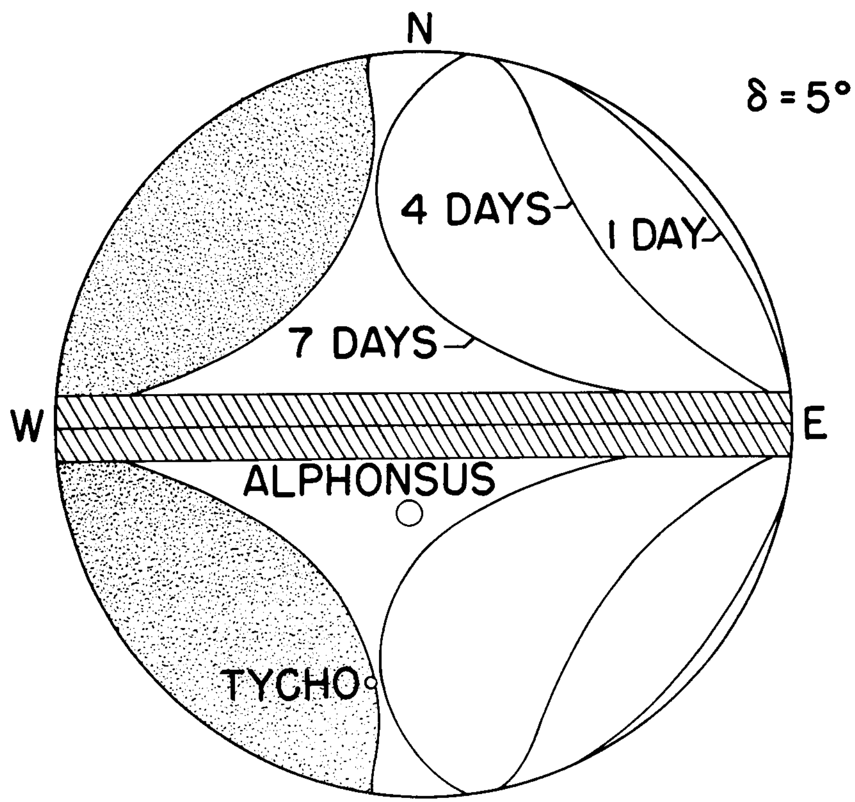

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Figure 13.- Landing sites and exploration times from rendezvous considerations.

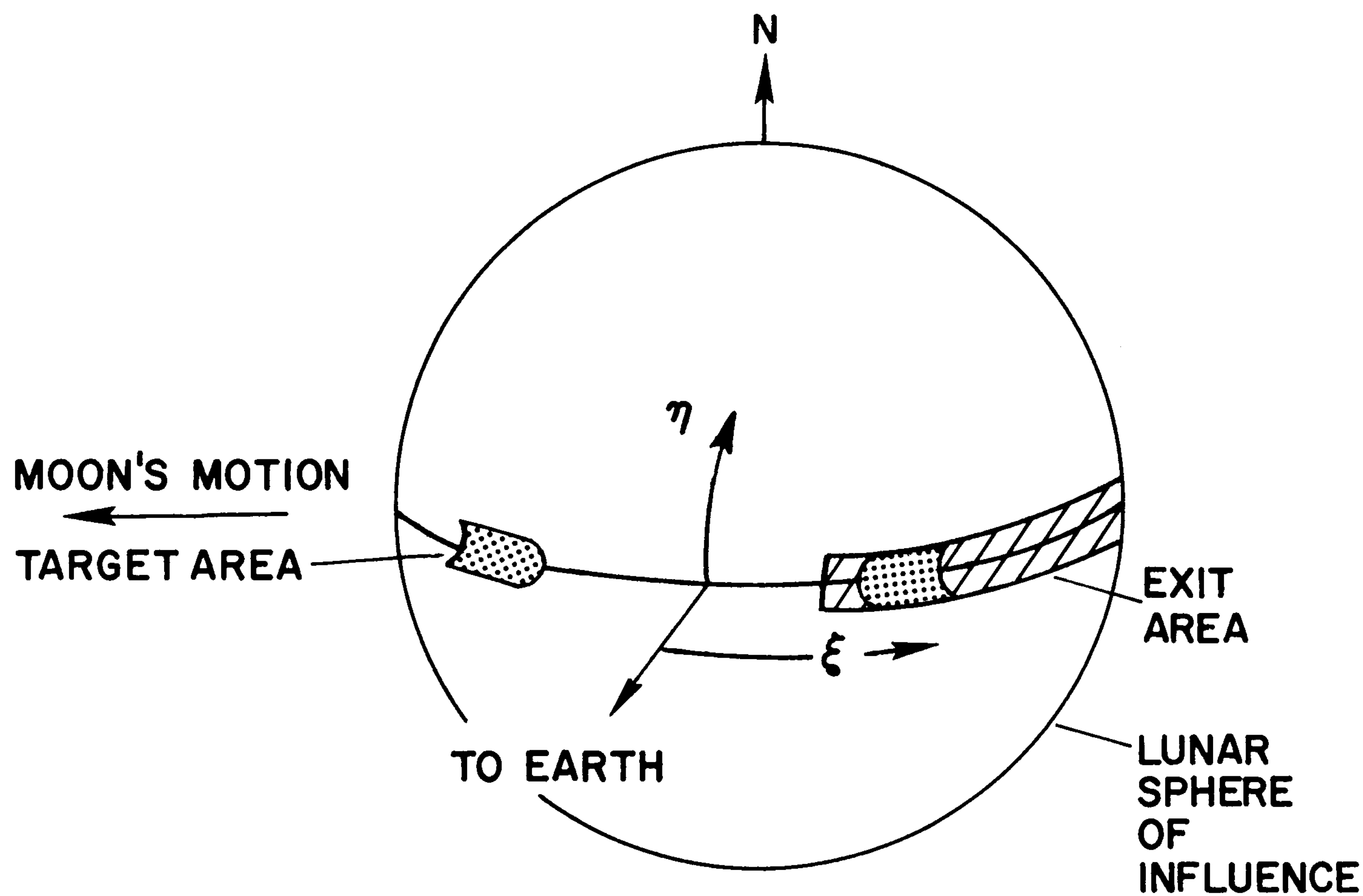

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Figure 14.- Exit and target areas on the lunar sphere of influence.

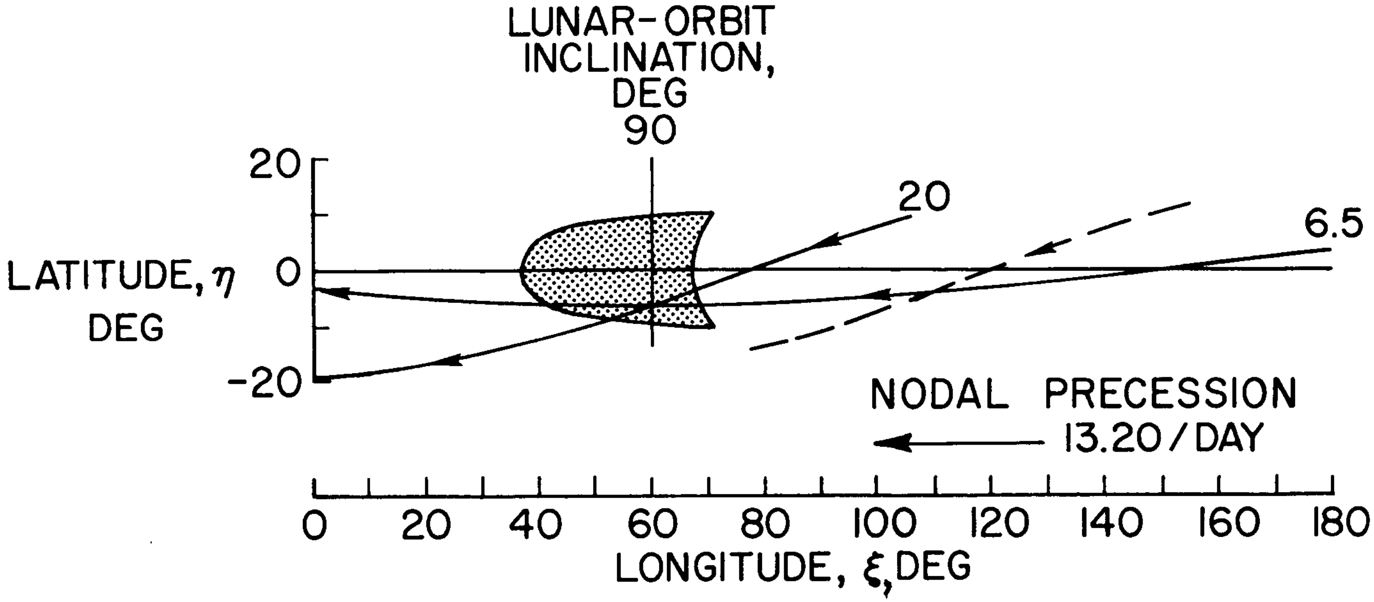

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Figure 15.- Lunar orbit characteristics for earth return.

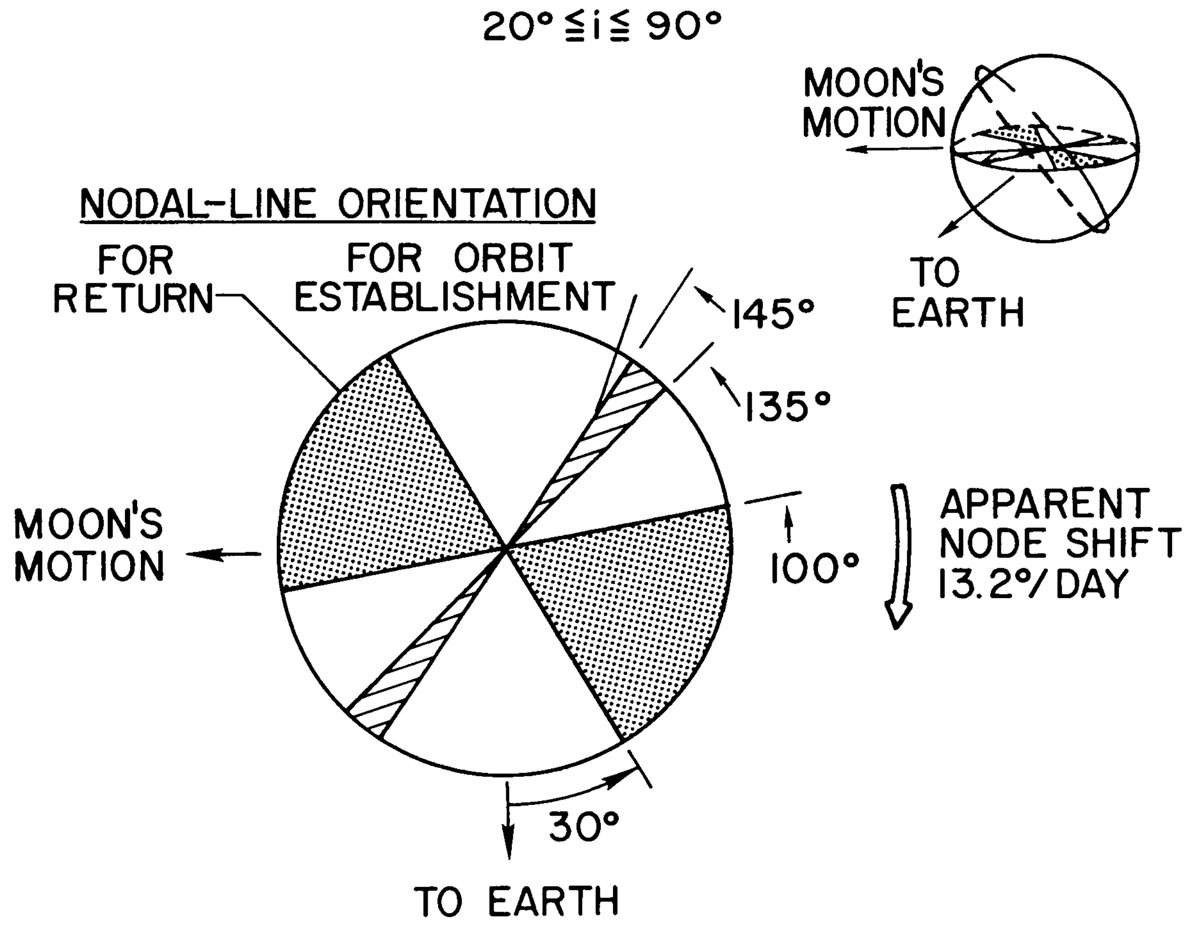

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Figure 16.- Geometric characteristics of lunar orbit planes for return considerations.

NASA-Langley, 1964