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MANNED LUNAR LANDING STRATEGY OF SYSTEM MANAGEMENT AND OPERATIONS

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ABSTRACT

The Apollo Lunar Landing Mission planning involves the development of both a spacecraft system and a strategy of operation of that system. Considerable developmental flight testings will be accomplished prior to the Lunar Landing Mission. Although the development flight tests will exercise the lunar module system the lunar landing approach will be the first time that the lunar module system will have been operated in other than a simulated landing approach. This landing approach will also be man's initial face-to-face encounter with certain of the lunar surface environmental problems. This paper discusses the development of a strategy for the lunar landing approach. This strategy relates the spacecraft systems management to the lunar landing objectives and to anticipated system performance and lunar environment problems.

INTRODUCTION

The landing of the lunar module (LM) upon the surface of the moon will be a monumental milestone of the Apollo mission. The fact that this landing will be the first time that the complete LM system will have been operated in the lunar environment further emphasizes this milestone. This will also be man's initial face-to-face encounter with the exact nature of the terrain in the landing area and with the problems of visibility as they may affect the ability to land the LM. To insure success of the landing mission, the nature of the problems that will be countered during the landing approach must be anticipated and a concept of system management that affords highly favorable conditions for the approach must be developed. The purpose of this paper is to discuss the problems of the lunar landing mission, and to present how the Apollo LM system design has been combined with an operation strategy to achieve a high probability of mission success.

BASIC MISSION APPROACH

The Apollo lunar landing mission concept calls for the Apollo spacecraft consisting of the command and service modules (CSM) together with the LM to be injected into an orbit about the moon. From this orbit, the LM will separate from the CSM and descend to the lunar surface.

Considering the entire LM descent after separation from the command module (CM) in lunar orbit, a theoretical landing maneuver could consist of a Hohmann transfer impulse on the back side of the moon with a change in characteristic velocity (ΔV) of 109 ft/sec, followed 180° later by an impulsive velocity change of about 5622 ft/sec as the LM approaches the lunar surface (figure 1). The flight-path angle in the final portion of the approach would be 0°.

Such a theoretical approach would require infinite thrust-to-weight ratio by the descent engine. This, of course, is an impossible and impractical approach. A finite thrust-to-weight ratio of the descent engine must be used and the approach path must account for lunar terrain variations and uncertainties in the guidance system.

Since lunar terrain variations of as much as ±20 000 feet could be expected and since uncertainties in the value of the lunar reference radius, coupled with guidance dispersions, could add another 15 000 feet to the uncertainty, a conservative safe value of 50 000 feet was chosen as a pericynthion altitude. From a performance standpoint, the choice of 50 000 feet as opposed to either 40 000 or 60 000 feet was quite arbitrary since the difference from the standpoint of fuel requirements was very slight, as indicated in figure 2. The initial thrust-to-weight ratio of the LM descent engine will be about 0.3.

Combining this thrust-to-weight ratio with a pericynthion altitude of 50 000 feet leads to the descent profile, as shown in figure 3. The separation and Hohmann transfer maneuver requires slightly less ΔV due to the pericynthion altitude increase. The powered descent portion approaching the landing area, however, requires a ΔV of 5925 ft/sec, which is a considerable increase over the infinite thrust requirement.

A scaled trajectory profile of this theoretical LM powered descent is shown in figure 4, indicating that the entire descent takes approximately 220 nautical miles. The LM velocity and attitude is shown periodically along the flight profile. This trajectory has the predominant characteristics of a low, flat profile terminating with a flight-path angle of about 9°. An obvious feature is that the crew, considering the location of the LM window, never have the opportunity to see where they are going. They can look either directly up, or, if the LM is rotated about its thrust axis, can look down at the surface, but they are never able to see in the direction they are going.

If the crew is to perform any assessment of the landing area or out-the-window safety of flight during the approach, it is obvious that the latter portion of the trajectory must be shaped so that a different attitude of the LM can be used during the approach. Shaping the trajectory away from the fuel optimum approach will result in a penalty in fuel requirements. Both the amount of time the crew will require to assess the landing area and the range from which the landing area can be adequately assessed must be traded off against the amount of fuel involved in the penalty of the shaping. It soon becomes obvious that a strategy is needed that will trade off the system capabilities of the spacecraft and the crew capabilities against the unknowns of the lunar environment encountered during the descent from the orbit, in order to insure that proper utilization of the onboard systems can be made to greater advantage.

STRATEGY CONSIDERATIONS

The LM landing strategy can be defined as the science and art of spacecraft mission planning exercised to meet the lunar environmental problems under advantageous conditions. In order to plan strategy, the objectives, the problems to be faced, and the characteristic performance of available systems need to be well known.

The objectives of the LM landing planning strategy are to anticipate the lunar environmental problems and to plan the landing approach so that the combined spacecraft systems, including the crew, will most effectively improve the probability of attaining a safe landing. The major factors that must be considered in this strategy are the problems brought about by the orbital mechanics of the landing maneuver, the limitations of the spacecraft systems (including limitations in fuel capacity and payload capability), and the constraints of the lunar environment (including terrain uncertainties, visibility, and determination of suitable landing positions).

The orbital mechanics aspects have been discussed in the preceding section. The lunar environmental constraints will be discussed in a subsequent section. The remainder of this section is concerned with descriptions of the spacecraft systems and the mission landing position requirements.

Although all of the LM systems are important to attain the lunar landing, those affecting the strategy are (a) the guidance and control system, (b) the landing radar, (c) the spacecraft window, and (d) the descent propulsion system.

Spacecraft Systems

Guidance and control system.- The guidance and control system is important to the landing strategy in that it is the means whereby the flight plan is executed. The performance uncertainties of this system determine the accuracy with which each segment of the flight plan is accomplished as well as the area of the lunar surface that must be considered for a possible landing site.

Two sources of lunar orbit navigation information will be available to initialize the LM guidance system. The Manned Space Flight Network (MSFN) will be the prime source of this information. The second source will be the navigation performed onboard the CSM prior to the separation of the LM. The CSM system used for this navigation function, as well as the LM guidance system, is described by Chilton.[1] The accuracy of the lunar orbit navigation, whether performed by the MSFN or by the spacecraft onboard system, determines the position and velocity uncertainties at the start of the LM descent.

Assuming that the guidance system will be updated by landing radar to eliminate the altitude dispersions, the landing dispersions will be a function of the initial condition uncertainties brought about from lunar orbit navigation coupled with the inertial system drift during the powered descent. A summary of the navigation uncertainties for both the LM initializations; and at the termination of the orbit transfer and the powered landing approach, is presented in figure 5.

The 3σ landing dispersion ellipses are shown graphically in figure 6 for cases where the lunar orbit navigation was done by the MSFN and also onboard the CSM. The ellipses are quite similar. In the case of the MSFN, the major axis is slightly shorter and the minor axis is slightly longer than those for the case utilizing CSM-onboard navigation.

LM control system.- The control of the LM during the descent to the surface can be provided automatically through steering commands generated by the guidance system and manually by the crew by inputs through an attitude controller.

A description of the LM control system is presented by Trageser and Hoag.[2] The primary control system stabilization utilizes a digital autopilot mode of the guidance computer. Figure 7 shows the attitude thruster firing combinations to create control moments. The engines are located on an axes system rotated about the LM descent engine thrust area 45° from the spacecraft axes. They are operated as control couples for three-axis attitude control.

Two control couples are available for each axis. The method of providing translational control while in the hovering condition is to tilt the spacecraft by means of the attitude control system. This produces a lateral component of acceleration from the descent engine thrust in the desired direction which is stopped by returning to vertical and reversed by tilting in the opposite direction.

During the descent the attitude control system is also coupled to a slow-moving gimbal actuator system of the descent engine to enable a means of trimming the descent engine thrust direction so

Superior numbers refer to similarly numbered references at the end of this paper.

that it passes through the LM center of gravity. The trimming system reduces undesirable torques from the descent engine in order to conserve reaction control system (RCS) propellant.

Landing radar system.- The LM landing radar system is important in landing strategy. As indicated earlier, it is used to eliminate the guidance system altitude dispersions and, also, the uncertainties of knowing the altitude from the lunar surface prior to beginning the descent.

The LM landing radar is a four-beam dopple system with the beam configuration shown in figure 8. The center beam measures the altitude, and the other three beams measure the three components of velocity.

Two positions of the landing radar antenna provide both altitude and velocity measurements over a wide range of spacecraft attitudes. In the first position, the antenna is tilted back from the thrust axis by approximately 24° so that the altitude beam will have a reasonably steep incidence angle to the surface during the early portions of the descent and, hence, will still provide accurate altitude information. As the LM approaches the landing maneuver, the antenna is physically switched to the second position making the altitude beam parallel to the X-axis of the LM.

The landing radar will begin to provide altitude measurements at an approximate altitude of 30 000 feet. These altitude measurements will be used to update the inertial system starting at an altitude of about 25 000 feet. The radar velocity updates will begin at approximately 10 000 feet. The landing radar desired accuracy is given in table I.

LM window system.- The LM window, although perhaps not normally considered a system, is a very important part of the landing strategy because it is through this window that the crew must observe the landing area to confirm the adequacy of the surface for touchdown. The physical configuration of the LM window is shown in figure 9. This drawing is a view from within the LM cockpit showing the left hand, or the command pilot's, window. The window is triangular in shape and skewed so that it provides maximum viewing angles for the landing approach maneuver.

Although the window is not large in size, the pilot's eye position is normally very close to the window so that the angular limits provided are quite wide. These angular limits are displayed in figure 10, showing the limits as viewed from the commander's design-eye position. The plot shows the azimuth and elevation variations of possible viewing limits referenced from a point where the pilot would be looking straight ahead, with respect to LM body axes (parallel to the Z-body axis), for the zero point. It is possible for the pilot to see downward at an angle of about 65° from the normal eye position and to the left side by approximately 80°. If the pilot moves his head either closer to the window, or further back, these limitations change slightly.

The guidance system is coupled with the window system through grid markings so that the pilot can observe the intended landing area by alining his line-of-sight with the grid marking according to information displayed from the guidance system. Figure 9, in addition to showing the window system, shows the location of the display and keyboard, which among other things provides digital readout information from the guidance system. The procedures for utilizing these integrated systems for landing site designation and redesignation will be discussed later in this paper.

Descent propulsion system.- The descent engine is an extremely important system to the design of the LM descent strategy. Initially, the descent engine was capable of being throttled over a range from 10 to 1. Design considerations, however, have made it necessary to limit the throttle capability to that shown in figure 11. This figure shows that at the start of powered flight, there is an upper fixed position of the throttle which would nominally provide approximately 9700 pounds of thrust. As long as the throttle is maintained in this fixed position, thrust magnitude will vary according to the nominal solid line.

At the start of the powered flight, approximately ±1.5 percent uncertainty in the thrust is expected at this fixed-throttle setting. The uncertainty grows up to ±2.5 percent after approximately 300 seconds of fixed-throttle usage.

The descent engine is always throttleable, in the region of 6300 pounds of thrust, to approximately 1050 pounds of thrust. The change from a fully throttleable engine in the upper region of the thrust level to a fixed-throttle position affects the guidance procedures during the initial powered descent, as will be explained later.

Mission Landing Position Requirement

Important strategy considerations are the types of requirements that are placed on the landing position. The first consideration is a requirement to land at any suitable point within a specified area, with the implication that the area could be quite large. Obviously, if the area is large enough, the requirements on the guidance system would be diminished considerably.

The second type of requirement is that of landing at any suitable point within a reasonably small area, constrained in size primarily by the guidance dispersions. This would, of course, dictate that the size of the area chosen will be compatible with the capabilities of the guidance and navigation system.

The third consideration is that of landing at a prespecified point, such as landing within 100 feet of the position of a surveyor spacecraft, or perhaps another type of spacecraft. It is obvious that this latter consideration imposes the greatest requirements on the strategy and also the guidance system, and would require some means of establishing contact with the intended landing position during the approach.

The present strategy is primarily based upon the second consideration---that of landing in areas of the size compatible with the guidance system dispersions. If, however, the landing area can be increased in size to the point such that downrange position control is not of primary importance, the associated strategy is not greatly different from that for the requirement assumed since the trajectory-shaping requirements would be the same for the terminal portion of the trajectory. The subsequent discussions of this paper will be based primarily upon a landing area size compatible with guidance system dispersions.

POWERED DESCENT DESIGN

After consideration of all the tradeoffs that could be identified as worthy of consideration during the LM powered descent, a three-phase trajectory design logic was chosen. The logic of this design will be discussed in the subsequent sections; however, the general logic is indicated in figure 12. The first phase following powered descent initiation at 50 000 feet is termed the braking phase. This phase is terminated at what is called a "Hi-gate" position. The second phase is termed the final approach phase, and is terminated at what is called the "Lo-gate" position, which is the start of the landing phase. The landing phase is terminated at a point referred to as "Touchdown." The total trajectory covers approximately 250 nautical miles. The logic of the braking phase is designed for efficient velocity reduction. That is, since there is no necessity for pilot visualization of the landing area during this phase, the attitudes may be chosen so that the spacecraft would have efficient utilization of descent engine thrust for reducing velocity.

During the final approach phase, the trajectory is shaped to allow an attitude from which the pilot can visually assess the landing site. An additional requirement met by this phase is provision of a view of the terrain at such a time that the pilot can confirm the flight safety of the trajectory prior to landing commitment. The landing phase is flown very much as a VTOL type of aircraft would be flown on earth, to allow the pilot vernier control of the position and velocities at touchdown. The attitude chosen is flown so as to provide the crew with a detailed visualization of the landing site.

The scaled profile of the design descent trajectory is shown in figures 13(a) and (b), and includes an indication of the spacecraft attitude at various points along the trajectory. The final approach and landing phases together cover only about 2 percent of the total trajectory range, although the time spent within these phases is about 30 percent of the total time. The logic of the design of the three phases and a summary of the ΔV budget for the descent are discussed in the following sections.

Braking Phase

Objectives and constraints.- The objective of the braking phase is to provide efficient reduction of the horizontal velocity existing at the initiation of the powered descent. During most of this phase, the altitude is great enough so that the pilot does not have to worry about the terrain variations and he can conduct the reduction in velocity at attitudes that allow great efficiency. The major constraint of this trajectory phase is imposed by the fixed throttle position thrust of the descent engine. It is desirable to use the maximum thrust of the descent engine as long as possible in order to provide efficient utilization of the fuel. There is, however, an initial part of the powered descent which is flown at reduced throttle to insure that the descent engine gimbal trim mechanism has nulled out-of-trim moments caused by center-of-gravity offsets.

Ignition logic.- The logic for ignition of the descent engine for initiation of the braking phase is as follows. First, the LM state (position and velocity) is integrated forward in time. Next, the guidance problem for the braking phase is solved (but not implemented) continuously with the advanced LM states as initial conditions. When the guidance solution requires the level of thrust equal to the expected thrust of the fixed-throttle position (figure 14), that solution is chosen for initiation of braking. Finally when the LM reaches the position and velocity state that yielded the proper thrust solution, the guidance computer sends the engine on signal to the descent propulsion solution. In order to prevent large moments caused by center-of-gravity offset, the engine is ignited at the 10 percent level, instead of maximum thrust. This level is maintained for some 28 seconds to trim the engine gimbal through the center of gravity before increase of thrust to the maximum, or fixed-throttle setting. This low level of thrusting is accounted for in the ignition logic.

Guidance with limited throttle.- The general approach of the braking phase routine, from the standpoint of the guidance system, is to utilize the same type of guidance equations that are appropriate for the throttled phases which follow. Thus, modifications in the targeting are required to allow for the utilization of the fixed throttle position during this phase. It is still desired to vary the state vector of the LM from its value at the start of powered descent to the state specified at the Hi-gate position of the trajectory. The guidance equations would normally determine the thrust level or acceleration level and attitude required in order to make an efficient change in the state. Knowledge of the initial thrust-to-weight value of the descent engine allows choice of initial conditions and guidance equations to be utilized in such a way as to select a time to go for the entire phase that will use the approximate thrust-to-weight of the upper limit of the descent engine. In actual operation, the LM system during this phase will respond to commands of attitude change, but as long as the guidance system is calling for a thrust above 6300 pounds, the descent engine will remain in its fixed (or upper limit) position. If the thrust variation of the descent engine at this fixed throttle position was known exactly, the trajectory could be preplanned to obtain the desired Hi-gate state vector. In view of the uncertainties of the descent engine, however, the trajectory must be planned so that the guidance system will begin calling for thrust values in the region in which the descent engine can be throttled.

(below 6300 pounds) prior to reaching Hi-gate position. This provides control over the velocities when the Hi-gate position is attained. The logic of this guidance scheme is shown in figure 15. The figure shows the profile of the trajectories as a function of range, and also a profile of the descent engine thrust (both the nominal value and that commanded by the guidance system as a function of range). The nominal thrust-to-weight case is shown first, and the trajectory is essentially preplanned by flying backward from the Hi-gate position, first using a thrust in the throttleable range to go back for a period of time, the period of time being determined by the possible magnitude of the uncertainty of the descent engine. This, in effect, determines the fictitious target that can be used in the guidance system in the first portion of the trajectory. The fictitious target is based upon the nominal thrust profile when the descent engine is in the fixed thrust position. The logic of the guidance is perhaps best explained by comparison of the actual value of thrust with that commanded by the guidance system, even though, in the upper thrust region, the descent engine is not responsive to these commands. Initially, the guidance system is directed to a fictitious target upstream from the Hi-gate state. The nominal thrust-to-weight variation follows the solid line (figure 15), and the guidance system computes the commanded variation of thrust-to-weight shown. At an intermediate position, the guidance targeting is switched from that of the fictitious target to that of the Hi-gate target. The discontinuity seen in the commanded position has no effect on the system, since, in this region, the descent engine throttle is not responsive to the guidance system. If the thrust-to-weight value remains nominal, the commanded thrust-to-weight magnitude will gradually decrease until it is within the region in which the descent engine can be throttled. This will nominally occur at the fictitious target position. Then, the guidance system has a number of seconds before the LM attains the Hi-gate position, in which to match the velocity and position desired at the Hi-gate position. From Hi-gate onward, the commanded thrust will be at or less than the maximum in the throttleable range. Figure 16 illustrates the thrust profiles (commanded and actual) for low and high thrust-to-weight ratios. In the case of the low thrust-to-weight ratio, where the actual value of the thrust is less than that of the expected nominal, the initial commanded thrust has the same type of variation as the nominal prior to the switchover point. But after the switchover point, there is a delay in time and range in getting to the region where the commanded thrust reaches the throttleable region. This point is thus only a few seconds prior to attainment of Hi-gate position. The extreme low thrust-to-weight would be that in which the commanded thrust would reach the throttleable region thrust exactly at the time the Hi-gate position was reached. For the case wherein the thrust-to-weight is higher than nominal, the commanded thrust will attain the throttleable position a number of seconds before that for nominal thrust. This allows a much longer time to effect the desired velocity condition at the Hi-gate position. However, this means that the region ahead of the Hi-gate is being flown at a much lower thrust-to-weight ratio for a longer period of time than would be desirable from a standpoint of fuel efficiency. This case involves the greatest fuel penalty.

Figure 17 shows the ΔV penalty variation due to fixed-thrust uncertainties. The left-hand scale indicates the ΔV penalty, the horizontal scale, the bias time of the fictitious target back from the Hi-gate target, and the right-hand scale the thrust-to-weight uncertainty expressed in ± percentages. The figure indicates that the ±2 percent uncertainty of the descent engine will require a bias time of approximately 65 seconds, and will invoke a bias ΔV penalty on the order of 45 ft/sec.

Landing radar updating.- The effect of landing radar updating on the guidance commands is important from the standpoint of elimination of altitude uncertainties, and the resulting changes in attitude and throttle settings required by the change in solution of the guidance equations. The effect of landing radar update is a continuing effect throughout the trajectory once the initial update altitude is reached; therefore some aspects of the following discussions will involve the final approach phase and the braking phase.

The altitude update is initiated at 25 000 feet, as determined by the primary guidance system, and is continued at each 2-second interval for the remainder of the approach. Velocity updates are initiated at about 9000 feet. The velocity is updated a single component at a time, in 2-second intervals (6 seconds required for a complete update). The altitude updating is continued with the velocity components. After each complete (3 components) velocity updating, an altitude update only is performed, after which the velocity updating is continued. The weighting factors for landing radar altitude and velocity updating are illustrated (figure 18) as linear functions of the parameter being updated. These are linear approximations to optimum weighting based upon least-squares estimation.

The comparable guidance commands for an ideal descent (no initial condition errors, no IMU errors, no landing radar errors, no terrain variations, no DPS uncertainties) and those encountered with an assumed set of errors and terrain variation are shown in figure 19. The ideal condition pitch profile exhibits a slope discontinuity at the fictitious target point (TF) for engine throttling. At the Hi-gate target point (HG), the pitch angle undergoes rapid pitchup to the constant attitude desired for final near constant (about 35° from the vertical). At the Lo-gate target (TLG, about 500 feet altitude), the attitude begins to change in order to satisfy the near vertical attitude desired just prior to the vertical descent. The case for the assumed terrain and navigation errors shows that the pitch angle deviates from the ideal case by slightly more than 10° at a maximum prior to Hi-gate, and is about equivalent to the ideal after Hi-gate. The thrust level shows generally the same level of command. The pitch angle deviations are of concern because of possible effect upon landing radar operation and because of increased expenditure of descent engine fuel.

Because of the interdependence of the guidance system and the landing radar, it is necessary to apply approach terrain criteria by which lunar landing sites may be chosen. The present status of the approach terrain criteria is shown in figure 20, and indicates primarily the deviation limits of terrain elevation which are considered acceptable to the closed-loop guidance solution. In addition to the terrain deviation allowance, a general slope of up to ±1° 30 nautical miles back from the landing site is also allowed. The basis for the criteria was to limit the pitch angle deviations caused by landing radar update within ±10° of the nominal variation for the period of altitude update prior to attainment of the Hi-gate target. After attainment of the Hi-gate target, the most important parameter becomes the angular margin of the landing position line-of-sight above the LM lower window limit.

Descent guidance monitoring.- An important crew function during the braking phase is to monitor the performance of the guidance system onboard. This is done by checking the solution of the primary guidance system with the solution of position and velocity obtained from the abort guidance system. This is accomplished by periodic differencing of the primary and abort guidance solutions of altitude, altitude rate, and lateral velocities. The altitude rate parameter is perhaps the most significant parameter for monitoring, because this parameter can result in trajectory that violates safety criteria. It has been shown, however, that it will require greater than the extremes of 3σ performance of the abort and primary guidance solutions to lead to an unsafe trajectory prior to attainment of the Hi-gate position. Because the MSFN will be very effective in spacecraft altitude rate measurement, it also will be very effective in providing an independent "vote" in the event that onboard differencing indicates the possibility of a guidance failure.

Summary of braking phase.- The braking phase, lasting about 450 seconds, occurs over about 243 nautical miles, during which the velocity is reduced from 5500 ft/sec to approximately 600 ft/sec, and the altitude from 50 000 feet to about 9000 feet. The attitude during the phase is normally such that the thrust vector is close to being alined with the flight path angle. In this attitude, the pilot is not able to look in the direction of the intended landing area. In the first portion of this phase, the LM could assume any desired roll attitude about the thrust axis. Mission planning will determine if the initial attitude will allow the crew to look down on the lunar surface to check the progress over the terrain. As the LM approaches the position at which landing radar will become operative, the roll attitude will be such that the windows will be oriented away from the surface in order to provide a more favorable attitude for landing radar operation and to prepare for the pitchup maneuver, at the Hi-gate position, that will allow a view forward to the landing area.

Final Approach Phase

Objectives and constraints.- The final approach phase is perhaps, from the standpoint of the strategy, the most important phase. It is primarily in this phase that the trajectory is shaped at a cost of fuel, in order to provide the crew with visibility of the landing area. In this phase the crew is first confronted with some of the unknowns of the lunar environment, such as the possibility of reduced visibility. The first objective is to provide the crew with out-the-window visibility, and to provide adequate time for assessment of the landing area. The second objective is to provide the crew with an opportunity to assess the flight safety of the trajectory before commitment to the continuation of the landing. The third is to provide a relatively stable viewing platform in order to best accomplish the first and second objectives. In other words, maneuvering should be kept to a minimum. The primary constraints on the strategy in this phase are again the desire to keep fuel expenditure to a minimum and the limitation of the LM window. In the event that the ascent engine must be used for abort during this approach to the surface the difference in thrust-to-weight between the descent and ascent engines also must be considered as a constraint. The ascent engines thrust-to-weight initially is only about one-half of that of the descent engine in this phase. The altitude loss, as a function of nominal trajectory altitude and velocity, during vertical velocity nulling must be included in the consideration for a safe staged abort. Other constraints that must be considered are the problems of the lunar terrain illumination and its inherent contrast properties which may make it difficult for the pilot to see and assess the terrain features. The primary variables that may be interchanged during this approach phase include the pitch attitude, the altitude at which Hi-gate or transition altitude is chosen, the flight-path angle of the trajectory, and the variation of look angle to the landing area (referenced to the spacecraft thrust axis). This again takes into consideration the limitation of the LM window.

Determination of Hi-gate.- Perhaps the first parameter that must be chosen, in order to design the final approach phase, is the Hi-gate altitude. The first factor is the range from which the landing area can be adequately assessed. If this were the only factor to be considered, it would of course be unwise to waste fuel to provide this ability, if the viewing range to the target landing area was so great that the detail of the area could not be observed. The second factor is the time that the crew will require to adequately assess the landing area. A third consideration is that of flight safety requirements with regard to the uncertainties of the terrain altitude considering the operating reliability of the landing radar and its capacity to update the guidance system (the inertial system), and also considering the abort boundaries associated with the ascent engine (figure 21). Preliminary estimates were made of all these factors and considering a desire to be able to get to Hi-gate, even if the landing radar is not updating the guidance system, the third requirement predominates, and flight safety dictates the choice of Hi-gate altitude. If further analysis of the landing radar operations indicates a highly reliable system, then the flight safety requirements will be satisfied and the Hi-gate altitude would be selected on the basis of the first two considerations.

The flight safety of the final approach trajectory will be largely governed by the magnitude of the uncertainties in altitude above the terrain. The present expected uncertainties are listed in figure 22. These uncertainties include that of the guidance and navigation system, which, considering that onboard lunar orbit navigation is accomplished, there will be an approximate 1500 feet of altitude uncertainty on a 1σ basis. If lunar navigation is conducted by the MSFN, the uncertainty will be approximately 500 feet less. At the present time, and largely as a result of some of the data from the Ranger spacecraft missions, there is a large uncertainty in the lunar radius magnitude, both the bias and the random uncertainties. Both of these quantities are established as 1 kilometer or approximately 3200 feet, 1σ basis at this time. Lunar Surface Technology personnel have indicated that their present capability in determining the slopes in the areas of the maria is limited to an uncertainty of approximately ±3° on a 3σ basis. This is equivalent to a 700-foot, 1σ uncertainty, considering the ranges of uncertainty of the landing position. In addition, present mission planning allows for a terrain profile along the approach path limited to a general slope of ±1° with local variations not to exceed the deviations presented in figure 20. This results in altitude biases of 700 to 800 feet (3σ) over the ranges of uncertainty of the landing position.

The minimum Hi-gate altitude can be determined by combining the altitude 3σ uncertainties and biases previously discussed. The manner in which these factors are combined, however, depends upon the navigational updating in orbit (with CSM optics or MSFN) and during the powered descent (with landing radar). Results for the various combinations are given in table II. The first case is based upon MSFN orbit navigation and no landing radar updating, and represents the largest Hi-gate altitude, 32 600 feet. This extreme and impractical Hi-gate altitude results from the fact that no terrain updating occurs anytime during the mission; therefore all of the uncertainties and biases are maximal.

The second case differs from the first only in that two sightings from orbit to a landmark, in the proximity of the landing site, are provided in order to update the position (radius) of the landing site. In this case it is assumed that orbital navigation of the CSM state is accomplished by MSFN, and that landing radar updating during the powered descent is unavailable. The minimum Hi-gate for this case is 6700 feet, a substantial reduction from the value given in case 1. This is because the landing site update eliminates the lunar radius bias and significantly reduces the random uncertainties in radius.

A moderate increase (over case 2 value) in Hi-gate altitude is shown in the third case because of the moderate increase in Primary Guidance, Navigation, and Control System uncertainties from onboard navigation (which includes the landing site update) as opposed to MSFN navigation. The minimum Hi-gate for this case is 7500 feet.

In the preceding analysis it has been assumed that the crew would immediately discern a collision situation and take appropriate action. Allowing a finite time, about 10 seconds, for situation assessment, an operational Hi-gate altitude of approximately 9000 feet fulfills crew safety criteria without landing radar.

Parameter interchanges.- Considering that the Hi-gate altitude requirement has been set at approximately 9000 feet, the major interchanges that still need to be established include the flight path angle, the acceptable look angle to the landing area, and the time required to assess the landing area. Each interchange may affect the state vector that is specified at Hi-gate, and this change must be taken into account in the total landing descent profile planning. Figure 23 shows the penalty of fuel as a function of Hi-gate altitude. The selection of about 9000 feet as the Hi-gate altitude expends about 250 ft/sec of ΔV. Because the LM pilot can only see down 65° from his straight ahead viewing position, it is desirable for the look angle to be greater than 25° above the thrust axis. Considering the variations in attitude, that may come about through the guidance system, caused by flying over variable terrain, a desired look angle of 35° has been chosen providing a margin of 10° over the lower limit of the window.

The flight path angle is also important. The angle must not be too shallow in order to get the proper perspective of the landing area as it is approached, and, conversely, it must not be too steep, purely from the standpoint of the pilot being better able to judge the safety of the approach path. In figure 24 is illustrated the ΔV penalty for variations in flight-path angle for various look angles. As may be seen in the figure, the major ΔV penalty is incurred by increasing the look angle. Small penalty is incurred for varying the flight path angle from 10° up to 20° for a given look angle. The sum of the interchange is that the Hi-gate altitude will be approximately 9000 feet, the look angle to the target approximately 10° above the lower limit of the window, and the flight path angle will be in the order of 13° to 15° throughout the major portion of the final approach phase.

The shaping accomplished in the final approach phase requires approximately 270 ft/sec of equivalent fuel. In order to see what this has provided, a comparison of the selected trajectory with that of the fuel optimum showing the variations of horizontal and vertical velocity as a function of time to go is given in figure 25. The time to go from 9000 feet altitude down to the Lo-gate position has been increased by approximately 45 seconds (see figure 24). In addition, the vertical velocity has been cut by approximately a third for equivalent altitudes; however, the primary difference appears in the comparison of horizontal velocity at equivalent altitudes, noting that at 5000 feet the fuel optimum trajectory has a velocity of about 1000 ft/sec, whereas the selected trajectory has a horizontal velocity of about 450 ft/sec.

Redesignation footprint.- Though an adequate perspective of the landing area and adequate viewing time are provided by the selection of the flight path angle, the line-of-sight angle, and the Hi-gate altitude, it is still pertinent to determine how much of the area the pilot needs to survey. This is a function of how much fuel the pilot will have in order to change his landing site if he decides that the point toward which the guidance system is taking him is unacceptable. Assuming that it will take the pilot a few seconds to get oriented to the view in front, it appears that the maximum altitude from which he could consider a redesignation would probably be less than 8000 feet. The available footprint as a function of fuel required for this purpose is shown in figure 26. The perspective is that of looking directly from overhead the spacecraft perpendicular to the surface where the spacecraft position is at the apex of the lines. The nominal landing point, or that point to which the spacecraft is being guided by the automatic system, is the zero-zero range position. The solid contour lines are the ranges that could be reached provided the indicated amount of fuel could be expended. For a ΔV expenditure of approximately 100 ft/sec, an additional 8000 feet downrange could be obtained, and approximately 10 000 feet in either direction crossrange. The horizontal line at the bottom of the figure indicates the lower window limit, and the second line indicates the position 5° above the lower window limit. The other lines indicate the side window view limitations experienced by the pilot or command pilot. The copilot would have a similar limitation of side vision toward the direction of the pilot; therefore, only the region bounded by the inboard side window limits would be common to the field of view of both crew members.

The variation of footprint capability as the altitude is decreased during the descent is indicated in figure 27. Contours of footprint capability are shown for an expenditure of 100 ft/sec of fuel at altitudes of 8000 feet, 5000 feet, and 3000 feet. The footprint capability shrinks the closer the approach is made to the landing area. However, a given budgeted amount of fuel provides an area that subtends very closely the same angular view from the pilot's viewing position. The present strategy is based upon having a high probability that the intended landing area will be generally suitable. For this reason there will be a low probability of requiring large redesignations of the landing position.

It has been assumed that a maximum capability of designating 3000 feet downrange will be required and this provision of fuel is allotted for redesignation at 5000 feet of altitude. Approximately 45 ft/sec of fuel is required for this redesignation capability. The footprint available for this fuel allotment is shown in figure 28.

The LM pilot does not have the opportunity to see the footprint as viewed here, but sees it from the perspective provided by the approach flight path angle. The pilot view from the Hi-gate altitude is indicated in figure 29. During this phase, the spacecraft is pitched back approximately 40°, thus the horizon is very near the -40° elevation depression angle. The landing site is at approximately 55° depression, or about 10° above the lower limit of the window. For reference purposes a 3000-foot circle has been drawn about the landing position and the landing footprint associated with a ΔV of 100 ft/sec is shown.

Landing point designator.- The pilot will know where to look to find the intended landing area, or the area toward which the guidance system is taking him, by information coming from the guidance system display and keyboard (DSKY). This information will be in the form of a digital readout that allows him to locate the correct grid number on the window, commonly called the landing point designator (LPD). After proper alinement of the grid, the pilot merely has to look beyond the number corresponding to the DSKY readout to find the point on the lunar surface toward which the automatic system is guiding the spacecraft. The proposed grid configuration for the LPD is shown in figure 30.

The process of landing point designation and redesignation is illustrated in figure 31. The guidance system always "believes" that it is following the correct path to the landing site. It has the capability at any time to determine the proper look angle or line-of-sight to the intended landing site. Because of orbital navigation errors and also drifts of the inertial system during the powered descent, the actual position of the spacecraft will not be the correct position. Thus, if the pilot looks along the calculated line-of-sight he would see an area different from that of the desired landing area. Should the desired landing area appear in another portion of the window, the pilot, by taking a measurement of the angle formed by the line-of-sight readout from the guidance system and the new line-of-sight (to the desired point), can enter the change in line-of-sight into the guidance computer.

Next the guidance system will recompute the location of the desired landing area. The guidance system, in effect, begins a period of relative navigation, where the new landing point is calculated in the present reference frame and is not significantly affected by whatever inertial system or other navigational errors that may have occurred. The accuracy with which the landing point designation or the redesignation process can be made is a function of how accurately the line-of-sight can be interpreted, or correctly displayed to the pilot.

There are several sources of redesignation errors. These include the variations in terrain along the approach to the landing site, the guidance dispersion effect upon altitude (provided the landing radar updating is not complete), boresight installation, the inertial measuring unit reference misalinement, and the errors of application by the spacecraft crew. The effect of altitude errors, whether from the terrain, or from the guidance system altitude uncertainties, is shown in figure 32. In this case, the guidance system assumes the landing site is at the same elevation as the terrain over which the spacecraft is flying, and, therefore, determines the line-of-sight through that point. However, when the crew views this line-of-sight, the point of intercept with the lunar surface is at an entirely different point than the intended landing position. For flight path angles of about 14°, this ratio of downrange error to altitude error is approximately 4 to 1. Altitude errors do not affect the lateral dispersions. It is obvious that although the landing radar performs a very vital function in reducing the altitude dispersions of the guidance system, there is a probability that the same landing radar function will update the inertial system with a false indication of the landing position altitude.

The errors other than the altitude type errors (the installation IMU and the pilot application errors) all tend to be biases. Preliminary testing indicates that these errors could be of the order of 0.5°. Again, for typical flight path angles of about 14°, this 0.5° of application boresight error will lead to downrange redesignation errors on the order of 800 feet for redesignations occurring in the altitude range of 5000 to 8000 feet. These downrange errors will reduce to the order of 100 feet when the redesignations are made at altitudes of 1000 feet or less. Thus, there is an interchange with regard to the probable magnitude of the errors that vary with altitude, particularly if the approach terrain is likely to have large variations of altitudes.

The process of redesignation will be a task cooperative between the pilot and the copilot. The copilot will read the DSKY and call out the numbers corresponding to the LPD. The pilot will then orient his line-of-sight so that he can look beyond the proper number on the LPD and see where the guidance system is taking him. If he is not satisfied with this position, then he can instruct changes in the guidance system by incrementing his attitude hand controller. During this portion of the approach, the guidance system is flying the spacecraft automatically so that the pilot's attitude hand controller is not effective. With each increment that the pilot makes in moving the hand controller in a pitching motion, there is a signal sent to the guidance system to change the landing point by the equivalent of 0.5° of elevation viewing angle. Lateral changes in the landing position can be made by incrementing the hand controller to the side in a motion that would normally create rolling motion of the spacecraft. Each increment of a hand controller in this direction causes a 2° line-of-sight change laterally to the landing area. When the guidance system receives these discrete instructions it recalculates the position of the desired landing area and commands the pitch or roll attitude in combination with a throttle command required to reach the desired position. This results in a transient response from the spacecraft until the new attitude and throttle setting commands are implemented. After the transient has settled, the copilot would normally read the DSKY again and inform the pilot what new number to look for to find the desired landing area. The pilot would then orient himself to look at this number and check to see if his instructions to the guidance system had been fully correct. If not, some refinement in landing site selection would then be made.

The response of the spacecraft to redesignations of landing position is important. For example, if the new site selected is further downrange, the spacecraft will pitch closer to the vertical and reduction in throttle will be made so that the new position will be more closely centered in the pilot's window. If, however, the site chosen is short of the original landing site, the spacecraft would have to pitch back and increase throttle in order to slow down and obtain the new desired position. These attitude motions affect the line-of-sight and become important because of the danger of losing sight of the target. Some typical responses to changes in the landing point are shown in figure 33. The variation of the line-of-sight to the landing site (look angle) with time from Hi-gate is shown for the nominal case, a redesignation downrange, and a redesignation uprange. The redesignations occur at an altitude of 5000 feet. For the nominal landing site, the line-of-sight look angle is maintained between 35° and 30° throughout the phase. For the 3000-foot long redesignation the look angle is increased over the nominal case, varying between 45° and 35° (after the resulting transient response is completed). For the 3000-foot short redesignation the pitchback motion of the spacecraft causes the line-of-sight angle to the very target area to be initially decreased to approximately 20°, increasing to about 28° for a short time interval. Thus, for this case, visibility of the landing area would be lost for a portion of time since the lower window limit is 25°. For this reason, the normal procedure would be not to redesignate short by more than the equivalent of about 2000 feet at this altitude. At lower altitudes, shorter range redesignations should be limited to proportionally less magnitude. For crossrange redesignations, the effect on the look angle is slight for redesignations up to 3000 feet; however, the spacecraft will require a new bank attitude (which is nominally zero for in-plane redesignations). Thus, this figure does not present the total attitude response transients for the effect of site redesignations.

An important aspect of the redesignation process is the problem of accounting for the propellant expenditure. There is no accurate procedure to account for this fuel other than to interrogate the guidance system for the amount of fuel remaining. The guidance computer load is quite heavy at this time; therefore, it is probable that a rule of thumb approach may be utilized, which, in effect, informs the pilot that so many units of elevation and azimuth redesignation capability can be utilized. Sufficient conservatism can be placed on this number to insure that the pilot does not waste fuel to the extent that the landing could not be completed. This would also allow the pilot a rough assessment of whether or not the new landing area would be within the fuel budget.

<u>Delta V budget</u>.- The fuel expenditure during the nominal final approach phase will be an equivalent to 889 ft/sec characteristic velocity. To this number is added, for budget purposes, a bias allowance of 45 ft/sec for the landing point redesignation capability, and a 3σ random allowance of 15 ft/sec for refinements in the landing site designation.

<u>Summary of final approach phase</u>.- The final approach phase covers about 5-1/2 nautical miles during which the altitude is decreased from 9000 feet to 500 feet, and the velocity from 600 ft/sec to 50 ft/sec. Normally the time required will be about 105 seconds during which time the pilot will have a continuous view of the landing area. It is during this time that assessments of the landing area will be made, and required redesignations of the landing position to more favorable landing terrain will be accomplished.

The Landing Phase

Objectives and constraints.- The basic purpose of the landing phase is to provide a portion of flight at low velocities and at pitch attitudes close to the vertical so that the pilot can provide vernier control of the touchdown maneuver, and also to have the opportunity for detailed assessment of the area prior to the touchdown. In order to accomplish this, the trajectory is further shaped after the final approach phase. The guidance system is targeted so that the design constraints of the Lo-gate position are met, but the actual target point will be at or near the position where the vertical descent begins. The final approach phase and the landing phase are then combined with regard to the manner in which the guidance system is targeted. The targeting design would satisfy the constraints of both the terminal portion of the final approach phase and the landing phase by proper selection of the targeting parameters. There will be a smooth transition from the extreme pitchback attitude associated with the final approach phase and the near vertical attitude of the landing phase.

In the final approach phase, the trajectory was shaped in order to pitch the attitude more toward the vertical, so that approach conditions would allow the pilot to view the landing site. The resulting pitch attitude, approximately 40° back from the vertical, is however, still quite extreme for approaching the lunar surface at low altitudes; hence, it is necessary to provide additional shaping in order to effect a more nearly vertical attitude at the termination of the total descent. The first objective is to allow the crew to make the detailed assessment, and a final selection, of the exact landing point. In order to accomplish this, there will be some flexibility in the propellant budget to allow something other than a rigid following of the design trajectory. This leads to the second objective, in which it is desired to allow some maneuvering capability and adjustment of the landing point. The constraints are familiar ones, including fuel utilization, physical limitations of the window, and in turn, the lighting and associated visibility of the surface, the visibility associated with the lighting, the actual terrain, and the possibility of blowing dust maneuvering with the desired attitude limits in order to retain the advantages of a fairly stable platform, and last, what is termed the staged abort limiting boundary. This boundary defines the circumstances under which an abort maneuver cannot be performed without the ascent stage hitting the surface. This curve is based upon a combination of vertical velocities, altitudes, and the pilot-abort-staging system reaction time.

Nominal trajectory.- The variables that are available to try to satisfy all of these constraints and objectives include variations in the approach flight path and the involved velocities, the spacecraft attitude, and the actual touchdown control procedures. The desired landing phase profile, which has resulted from almost 2-1/2 years of simulating the maneuver, is illustrated in figure 34. The Lo-gate point is at an altitude of approximately 500 feet, at a position about 1200 feet back from the intended landing spot. The landing phase flight path is a continuation of the final approach phase flight path so that there is no discontinuity at the Lo-gate position. At the start of this phase, the horizontal velocity is approximately 50 ft/sec and the vertical velocity is 15 ft/sec. The pitch attitude is nominally 10° to 11° throughout this phase, but rigid adherence to this pitch attitude is not a requirement. The effect of the pitch attitude is to gradually reduce the velocities as the flight path is followed in order to reach the desired position at an altitude of 100 feet from which a vertical descent can be made. Modification of this trajectory can be accomplished simply by modifying the profile of pitch attitude in order to effect a landing at slightly different points than that associated with the nominal descent path. No actual hover position is shown in the approach profile because the vertical velocity or descent rate nominally does not come to zero. The approach is a continuous maneuver in which forward and lateral velocities would be zeroed at approximately the 100-foot altitude position and the descent velocity allowed to continue at approximately 5 ft/sec. This allows a very expeditious type of landing. However, if a hover condition is desired near the 100-foot altitude mark, it is a very simple matter for the pilot to effect such a hover maneuver. The only disadvantage of the hover maneuver is the expenditure of fuel. The total maneuver from the Lo-gate position will normally take approximately 80 seconds. If flown according to the profile, the descent propellant utilized will be equivalent to about 390 ft/sec of characteristic velocity. During the landing approach, the pilot has good visibility of the landing position until just before the final vertical descent phase. A nominal sequence of pilot views of a 100-foot radius circular area around the landing point is shown in figure 34. However, even during the vertical descent, the area immediately in front of and to the side of the exact landing position will be visible. The LM front landing pad is visible to the pilot. In addition to being able to observe the intended landing site, the pilot has ample view of much of the lunar surface around him so that if the original site is not suitable he can deviate to the other landing position, provided that the new landing position is obtainable with the fuel available. The basic system design will allow the entire maneuver to be conducted automatically. However, the LM handling qualities make it a satisfactory vehicle for the pilot to control manually. The satisfactory nature of the LM manual control handling qualities has been demonstrated by fixed base simulation and by flight simulation at the Flight Research Center using the Lunar Landing Research Vehicle and the Langley Research Center using the Lunar Landing Research Facility. Simulations have shown that there should be no problems involved if the pilot decides to take over from manual control at any time during the terminal portion of the final approach phase or the landing phase. Much concern has been generated regarding the problem of visibility during the landing approach. In the event that the pilot has some misgivings about the area on which he desires to land, the landing phase can be flexible enough to accommodate a dog-leg type maneuver that will give the pilot improved viewing perspective of the intended landing position. Manual control of this maneuver should present no problem and could be executed at the option of the pilot. At the present time, trajectory is not planned for an approach in order to maintain simplicity of trajectory design, because of the expected ease in which the maneuver could be accomplished manually should the need occur. Should, however, the dog-leg be identified as a requirement for an automatic approach, it will be incorporated.

A profile of the altitude and altitude rate of the landing phase is shown in figure 35. The altitude rate is gradually decreased to a value of about 5 ft/sec at the 100-foot altitude position for vertical descent. The descent rate of 5 ft/sec is maintained at this point in order to expedite the landing. At an altitude of approximately 50 feet, the descent rate would be decreased to the design touchdown velocity of 3-1/2 ft/sec. It is not necessary that this be done at exactly 50 feet so that uncertainties in the altitude of the order of 5 to 10 feet would not significantly affect the approach design. The value of 3-1/2 ft/sec descent rate is then maintained all the way until contact with the surface is effected and procedures initiated for cutoff of the descent engine. The curve labeled staged-abort boundary shown in figure 35 is applicable to the situation in which the descent engine has to be cut off and the vehicle staged to abort on the ascent engine. It is obvious that this boundary must be violated prior to effecting a normal landing on the surface. However, with the current design, this boundary is avoided until the pilot is ready to commit himself to a landing so that it is only in the region of below 100 feet that he is in violation of the boundary.

Delta V budget.- A summary of the landing phase fuel budget is given in table III. The budget reflects allowances for several possible contingencies. For example, the pilot may wish to proceed to the landing site and spend some time inspecting it before he finally descends to the surface. This would require that the spacecraft hesitate during the approach, and the penalty involved is the amount of fuel expended. A period of 15 seconds of hover time will cost about 80 ft/sec of fuel equivalent. There is also the possibility that the performance of the landing radar may be doubtful, in which case the spacecraft crew might want to hover in order to visually observe and null out the velocities. It has been found by means of flight tests in a helicopter, that velocities can be nulled in this manner within 1 ft/sec after less than 15 seconds of hover time (another 80 ft/sec of fuel expenditure). It would be possible to update the inertial system in this manner and allow the spacecraft to proceed and land on the surface with degraded landing radar performance during the final portion of the descent. If there are errors in the radar vertical velocity, there will be a direct effect upon the time required to complete descent and a random ±65 ft/sec of equivalent fuel has been allotted in the fuel budget. Another descent engine fuel contingency that must be accounted for is the possible variation in the pilot's control technique including the deviations from the planned flight profile the pilot might employ. Simulation experience has indicated a need for an average addition of 80 ft/sec of fuel and a random ±100 ft/sec. It is noteworthy that only 30 seconds of hover time has been budgeted and that for specifically designated purposes.

Fuel Budget Summary

A summary of the total LM descent fuel budget is given in table IV. The budget is divided into that required by the baseline trajectory requirement totaling 6624 ft/sec, and items, described as contingencies, totaling 398 ft/sec mean requirement with an additional ±183 ft/sec random requirement. This leads to a total of 7205. The inclusion of the RSS random contingencies as a fuel requirement is considered a conservative approach in that each of the random contingencies could lead to a fuel savings as well as a fuel expenditure. The tankage capacity of about 7300 ft/sec of fuel provides an additional margin of fuel for as yet unassigned contingencies.

The fuel budget summary is presented in figure 36 as a "How-Goes-It" plot of the expenditure of fuel both in equivalent characteristic velocity and pounds as a function of time and events during the descent. The solid line gives the baseline trajectory and results in a fuel remaining of 778 ft/sec at touchdown. Adding the utilization of all the budgeted contingency mean values of fuel is represented by the dashed line. When these contingencies are used, the time basis of the plot will be incorrect, particularly for the time between lo-gate and landing. The total time could extend to as much as 12.5 minutes (735 seconds) in the event that all of the contingency fuel was used for hovering over the landing site.

LUNAR LANDING TOUCHDOWN CONTROL, AUTOMATIC AND MANUAL

Perhaps the most important single operation in the lunar landing mission is the actual touchdown maneuver. It is during this maneuver that the uncertainties of the lunar surface become a real problem. A recommended procedure for controlling the approach has been developed. This procedure, developed partly through simulation, involves reaching a position at about 100 feet above the landing site and descending vertically to the lunar surface, as previously described. During the vertical descent, the lateral velocities are nulled and the vertical velocity controlled to a prescribed value until the descent engine is cut off just prior to touchdown. The procedures for effecting descent-engine shutdown will be discussed in detail.

Control Modes

There are two control modes by which the landing operation can be performed. The first is completely automatic. Although the pilot may have used the landing point designator to select the touchdown point, in this mode he is not active in the actual control loop. The second mode is manual, but is aided by automatic control loops. In the second mode the pilot takes over direct control but also has stabilization loops to provide favorable control response. In addition, the manual mode normally will be used in conjunction with a rate-of-descent command mode to further aid the pilot in control of the touchdown velocities.

Within the manual landing mode, the pilot has two options: (1) to land visually, which would require that there be no visual obscuration as might come from dust or lunar lighting constraints, or (2) because of such obscurations to control the landing through reference to flight instruments. Because of the expected good handling qualities of the LM, the manual-visual mode should be very similar to flight of a VTOL aircraft here on earth. No landing attitude or velocity control problem is anticipated and the control should be within 1 ft/sec lateral velocities.

Manual-instrument mode of control loops have sources of error that may degrade control and those that have been considered include control system response, landing radar velocity measurement, landing radar altitude measurement, IMU accelerometer bias, IMU misalinement, display system for manual only, the pilot (for manual only), and the center-of-gravity position.

Descent Engine Cutoff Constraints and Logic

In considering the control of the landing, emphasis has been placed on the method of timing for shutoff of the descent engine. Because of possible unsymmetrical nozzle failure due to shock ingestion and a desire to limit erosion of the landing surface, an operating constraint of having the descent engine off at touchdown has been accepted. Probable errors in altitude information from either the inertial system or from the landing radar preclude the use of this information for the engine cutoff function, even though the accuracy may be of the order of 5 feet, because of the deleterious effect on touchdown vertical velocities. The need for an accurate, discrete indication of the proper altitude to shutoff the descent engine led to the adoption of probes extending beneath the landing pads.

A light in the cockpit indicates probe contact with the lunar surface. The light-on signal informs the pilot that the proper altitude has been reached for the engine cutoff. The probe length must be determined from a consideration of delay times in pilot response, descent engine shutoff valve closures, and tail-off of the nominal descent velocities. The sequence of events is shown in figure 37.

The variation of descent rate at touchdown as a function of descent rate at probe contact is shown in figure 38 and includes the effect of pilot reaction time. The curves are representative of a probe 53 inches in length, coupled with a 0.25-second total engine shutoff delay time. This engine delay time includes that time required for the electronic signal to be generated, the shutoff valves to close, and the thrust tail-off to be essentially completed. The heavy dashed line on the chart going up on a 45° angle indicates a combination of descent rate at probe contact, plus system delay and pilot reaction times, that would cause the engine to still be on at touchdown. If the desired final rate of descent has been achieved, up to 1.0-second pilot delay time can be tolerated and still have the descent engine off at touchdown.

As shown in figure 38, the actual touchdown velocity is just slightly more than the descent rate at probe contact, or about 4 ft/sec. Faster reaction time would increase the final touchdown velocity, but not beyond present landing gear impact limit. If manual control allowed a slightly higher final descent rate than desired, and radar errors at the time of final update also allowed a slightly higher descent rate, these compounded increases might yield descent rates on the order of 5 to 6 ft/sec. These increased rates coupled with the 0.6-second reaction time would mean the criterion of having the descent engine off at touchdown would not be met. One solution for this situation would be to extend the probes to allow more range in pilot reaction time. However, the advantages of longer probes must be traded off against a probable decrease in reliability and an increased probability of touching down with greater than acceptable vertical velocity. A simulation study of this maneuver with the pilot initiating shutoff of the descent engine established pilot reaction times that average about 0.3 second (fig. 39).

Simulation Results

Pilot-in-the-loop and automatic-control-simulation studies have been conducted on the landing control maneuver. The pilot-in-the-loop studies were made using a simulated LM cockpit including all the control actuators (attitude, throttle, and descent engine cutoff). The simulation included the major sources of system errors, such as platform misalinement, accelerometer bias, instrument display resolution, center-of-gravity offsets, and landing radar errors. The landing radar errors are a prime factor in the touchdown control process and the models assumed for the analysis are shown in table V. The specification performance of the landing radar calls for each of the three components of velocity to be measured within 1.5 ft/sec on a 3σ basis. Current predictions are that this specification will be met in lateral and forward directions and bettered by 0.75 ft/sec vertically. For a conservative analysis, the predicted performance has been degraded by a factor of two.

The simulation results of landing velocity using manual control with specification performance by the landing radar are shown in figure 40. The dashed lines indicated the present design criteria for the landing gear. The 0.9, 0.99, and 0.999 probability contours are shown and are well within the design envelope. The effect of changing the length of the landing probes is to adjust the vertical velocity bias velocity approximately 1 ft/sec per foot change in probe length.

The effect of the landing radar performance upon the landing velocity envelope is shown in figure 41. The 0.99 probability contours are shown for the cases of no radar errors, specification performance, predicted performance, and degraded (predicted) performance. The resulting contours show the almost direct dependence of touchdown velocity error upon the landing radar velocity performance.

The comparative results between automatic and manual control of the landing touchdown velocities are shown in figure 42. The 0.99 contours show that automatic control results in lower touchdown velocities, but the difference is much less pronounced for the degraded radar performance as compared with the predicted radar performance. The figure does not reflect the advantage that manual control provides in closer selection of the actual touchdown position in the event that the terrain is not uniformly satisfactory.

Additional analysis of these same results for the control performance for attitude and attitude rates indicated that control within the present criteria of 6° and 2° per second can be expected on a 3σ probability.

ABORT AFTER TOUCHDOWN

Although analysis and simulation tests indicate a high probability that the landing touchdown maneuver will be within the landing gear design criteria, there is still an interest in the ability to abort should the landing dynamics become unstable. The ability to abort will be a function of when the need for the abort is recognized, the time required to initiate abort, the time involved in separation of the ascent stage, the thrust-buildup time of the ascent stage, the attitude and the attitude rate at separation, and the control power and control rate limitations of the ascent stage.

At staging, the control power of the ascent stage is about 35°/sec^2 for pitch and roll attitude maneuvers. Under emergency manual control where the pilot deflects his attitude hand controller hard-over, there is no attitude rate limitation. Normal manual control commands are limited to 20°/sec and automatic control limited to 10°/sec in pitch and 5°/sec in roll. These attitude rate limitations are important from the standpoint of determining how quickly the ascent stage attitude can be returned to the vertical in the event of an impending tipover.

An analysis was made of the boundary of over-turn conditions from which a successful staged abort could be made. The results of this analysis are shown in figure 43. Two boundaries are shown; one for emergency manual attitude control which requires the pilot to put his hand controller hard-over and the other for a rate limit consistent with automatic roll response (5°/sec). Both boundaries apply to the conditions under which an abort action must be recognized as being required. The boundaries allow a total of 1.4 seconds for the pilot to actuate the abort control, the staging to take place, and the ascent thrust to build up to 90 percent of rated thrust.

In addition to the boundaries, there is also a line indicating the neutral stability boundary or the sets of conditions under which the spacecraft would just reach the tipover balance point of about 40°. The curve labeled Landing Gear Design Envelope Maximum Energy applies to the improbable, if not impossible, case where the landing was made at the corner of the velocity criteria envelope 7 ft/sec vertical and 4 ft/sec horizontal, and all of the energy was converted to rotational motion. It is, therefore, highly improbable that conditions will be encountered that lie to the right of this curve.

For the emergency manual control, the boundary indicated an abort can be made at an attitude of about 60° if the rate is not greater than 10°/sec. This condition would take more than 4 seconds to develop after the initial contact with the lunar surface. For the other extreme of attitude rate limit (5°/sec) applicable only to the automatic roll attitude control, the boundary is reduced about 10° in attitude.

The pilot will have indication of attitude from his window view and from the attitude instrument display (FDAI). Both of these are considered adequate sources of attitude information in the event that the spacecraft passes a 40° deviation from the vertical and an abort becomes necessary.

Considering the improbability of landing contact that would result in an unstable post-landing attitude and the probability that even in such an event the pilot could initiate a safe abort, there does not appear to be a requirement for an automatic abort initiation.

SUMMARY

A lunar module descent strategy has been presented which is designed to take advantage of the lunar module systems and the lunar module crew in order that the lunar module will continually be in an advantageous position to complete the lunar landing. The three-phase trajectory is designed to maintain fuel expenditure efficiency, except in those regions of the trajectory where such factors as pilot assessment of the landing area require a judicious compromise of fuel efficiency.

The lunar landing strategy has considered all identified problems which might adversely affect the lunar landing and the resulting design calls for a fuel expenditure budget of about 7200 ft/sec of characteristic velocity. This budget is compatible with the tank capacity of the lunar module.

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TABLES

ILLUSTRATIONS

TABLE I

LM LANDING RADAR (3σ)

SPECIFICATION ACCURACY

ALTITUDE, FT	ACCURACY		
	RANGE TO SURFACE	V_{XA}	V_{YA}, V_{ZA}
5 - 200	1.5% + 5 FT	1.5% OR 1.5 FPS	2.0% OR 1.5 FPS
200 - 2000	1.5% + 5 FT	1.5% OR 1.5 FPS	3.5% OR 3.5 FPS
2000 - 25 000	1.5% + 5 FT	1.5% OR 1.5 FPS	2.0% OR 2.0 FPS
25 000 - 40 000	2%	N/A	N/A

TABLE II

DETERMINATION OF MINIMUM

HI-GATE ALTITUDE WITHOUT LR UPDATING

ORBIT NAVIGATION	3σ ALTITUDE UNCERTAINTIES,* FT			ALTITUDE BIASES, FT			MINIMUM HI-GATE ALTITUDE, FT
	PGNCS	TERRAIN PROFILE	LUNAR RADIUS	LUNAR RADIUS	TERRAIN PROFILE	STAGED ABORT	
MSFN	3700	4700	13700	9800	4300	3500	32600
MSFN & LANDING SITE UPDATE	3700	700	1700	—	700	1800	6700
PGNCS & LANDING SITE UPDATE	4500	1000	1700	—	800	1800	7500

* 3σ UNCERTAINTIES ARE ROOT-SUM-SQUARED

TABLE III

LANDING PHASE FUEL BUDGET

BASELINE TRAJECTORY ALLOWANCE 390 FT/SEC

• CONTINGENCY ALLOWANCE, FT/SEC	MEAN	RANDOM (3σ)
• MANUAL CONTROL TECHNIQUE VARIATIONS	80	100
• EFFECT OF LANDING RADAR UNCERTAINTIES	80	65
• LANDING SITE INSPECTION	80	-
• FUEL DEPLETION MARGIN	40	-
TOTAL	280	119 (RSS)

TABLE IV

SUMMARY OF LM DESCENT FUEL BUDGET

BASELINE TRAJECTORY ALLOWANCES

PHASE		ΔV, FT/SEC	
DESCENT TRANSFER		97	
POWERED DESCENT: BRAKING		5205	
FINAL APPROACH		932	
LANDING		390	
	SUBTOTAL	6624	

CONTINGENCY ALLOWANCES

		MEAN	3σ
DESCENT TRANSFER - INCREASE CSM ALTITUDE 10 N MI		13	
BRAKING: INCREASE CSM ALTITUDE 10 N MI		15	
THRUST DISPERSIONS OF + 2%			120
NAVIGATION ALT DISPERSIONS (3000 FT, 3σ)			60
FINAL APPROACH - LANDING SITE UPDATE		90	30
LANDING: MANUAL CONTROL VARIATIONS		80	100
EFFECT OF LR UNCERTAINTIES		80	65
LANDING SITE INSPECTION		80	
FUEL DELETION MARGIN		40	
	SUBTOTAL	398	183 (RSS)
	TOTAL BUDGET	7205	

TABLE V

ASSUMED LANDING RADAR ERROR MODEL

FOR LANDING CONTROL ANALYSIS

	SPECIFICATION	PREDICTED	DEGRADED
VERTICAL	1.5 FT/SEC	.75 FT/SEC	1.5 FT/SEC
LATERAL	1.5 FT/SEC	1.5 FT/SEC	3.0 FT/SEC
FORWARD	1.5 FT/SEC	1.5 FT/SEC	3.0 FT/SEC

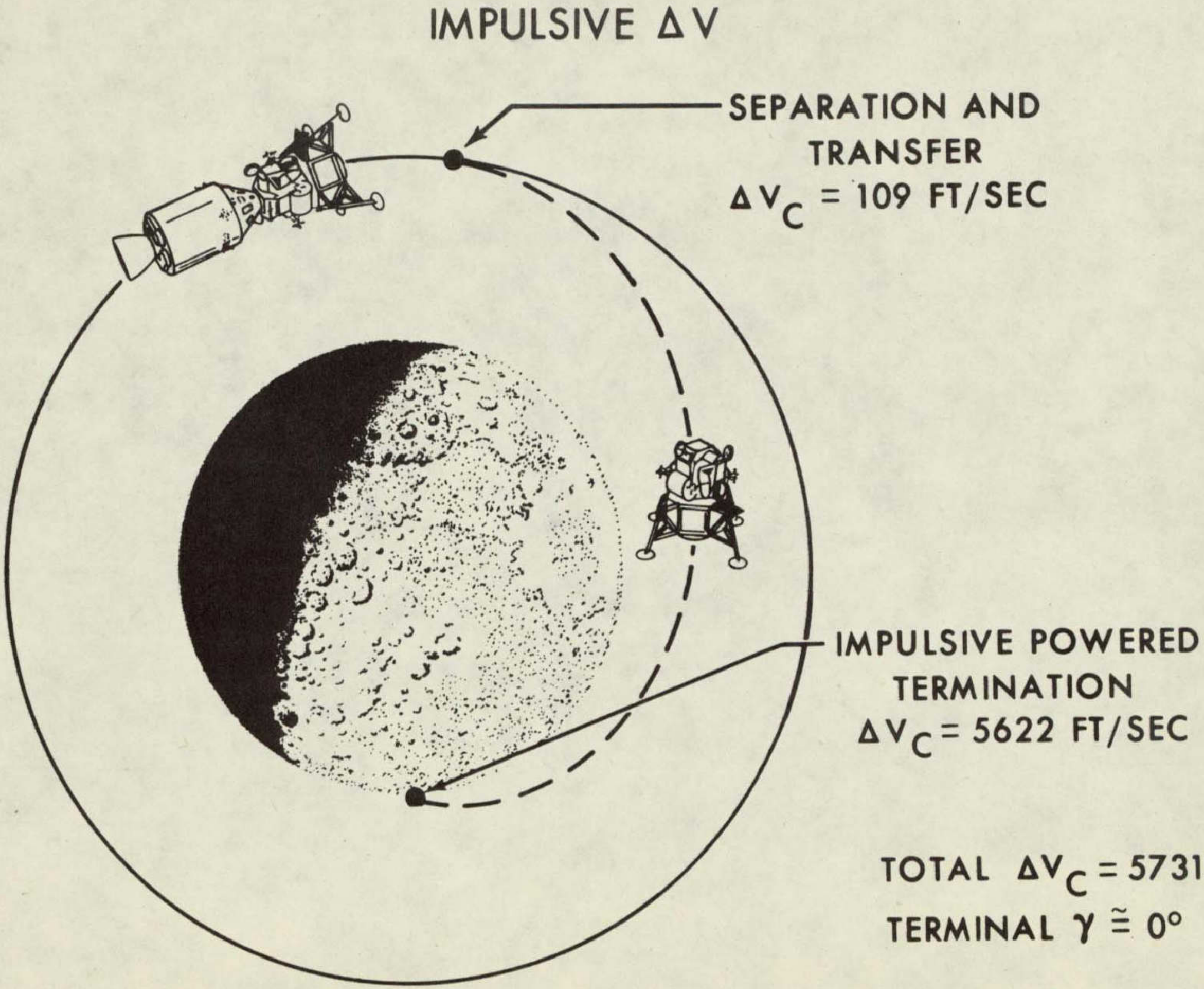


Figure 1. Theoretical LM Descent

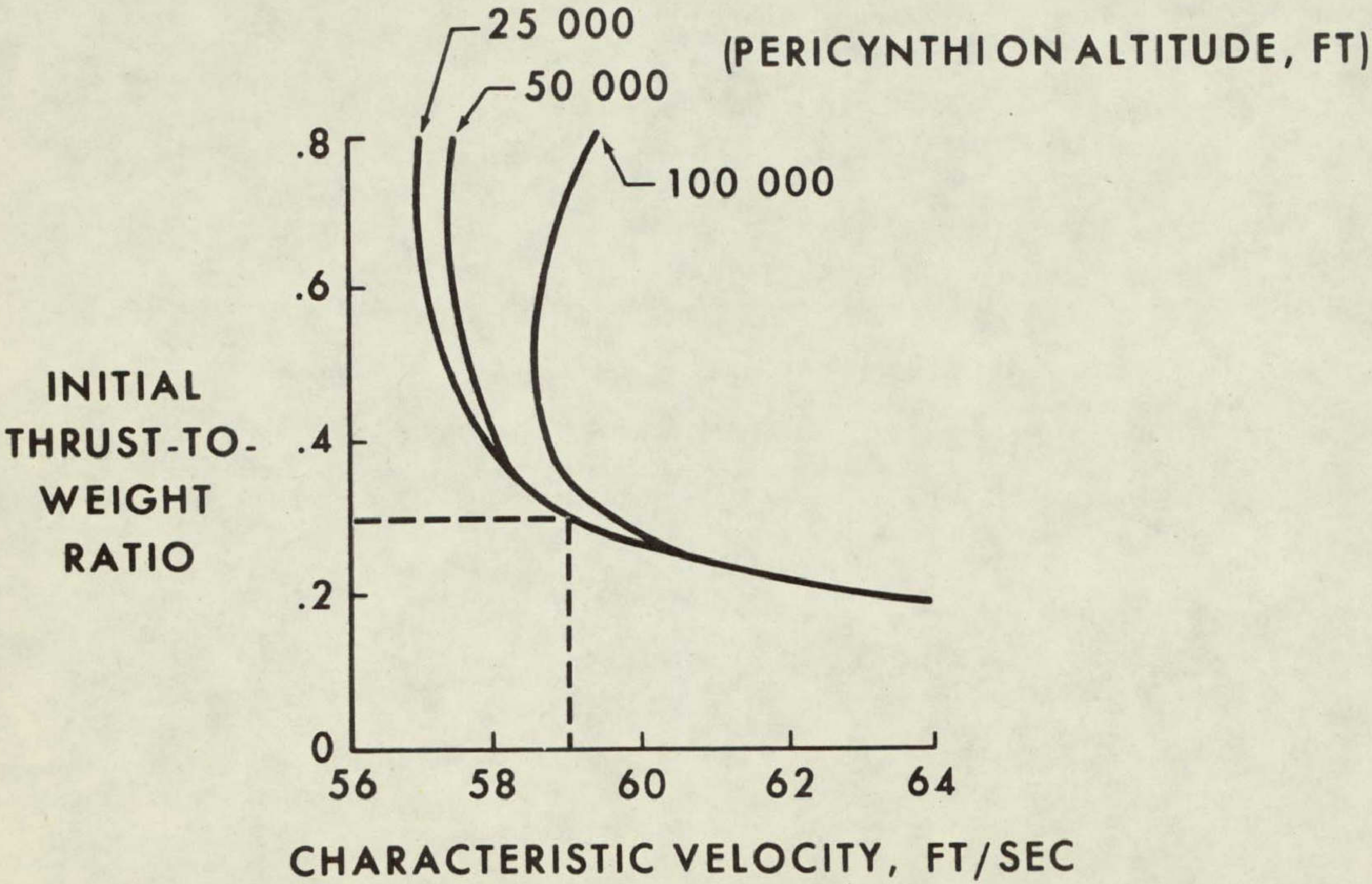
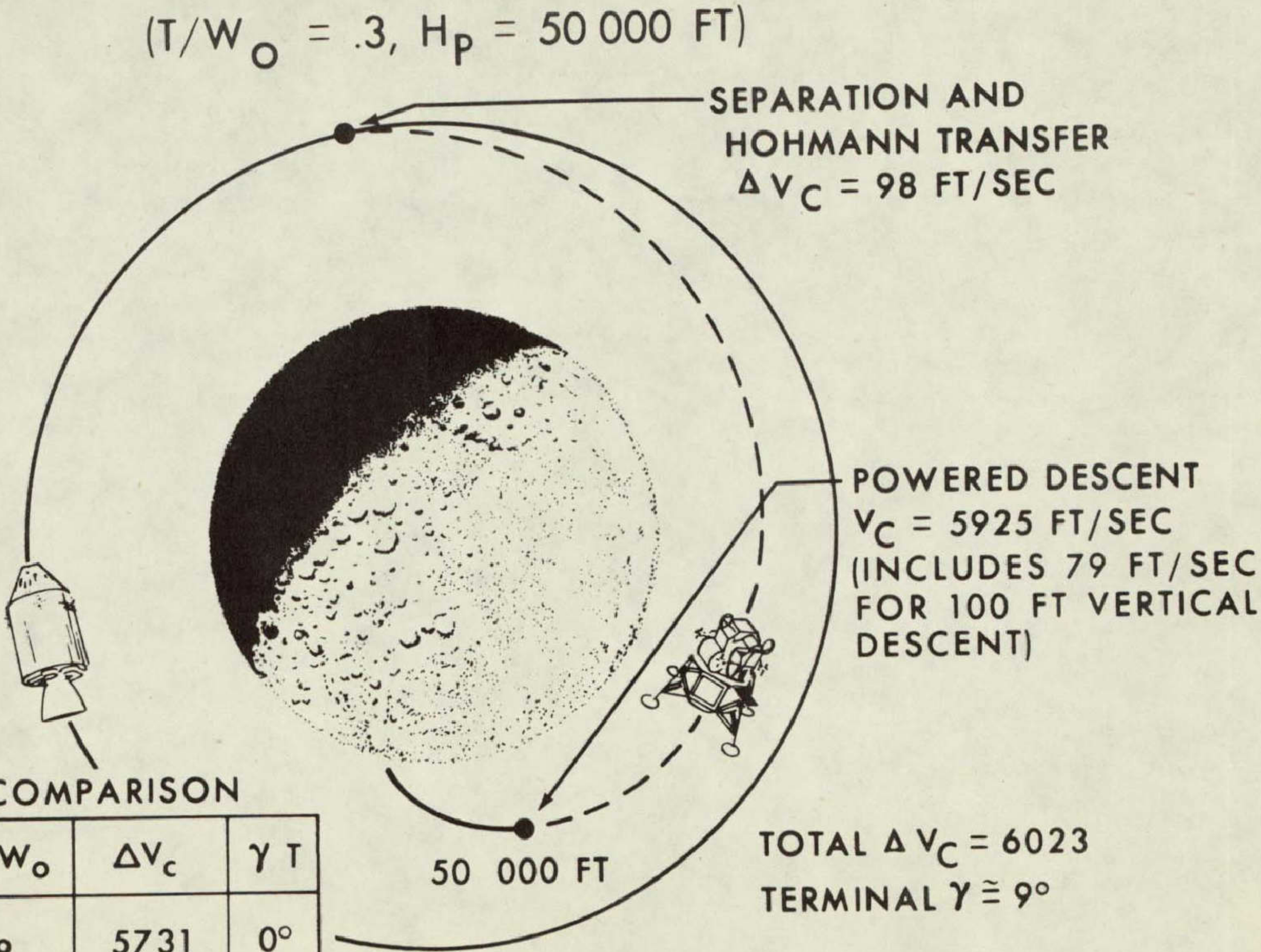


Figure 2. Variation of Powered-Descent Characteristic Velocity with Thrust-to-Weight Ratio



COMPARISON

	T/W_o	ΔV_c	γ T
IMPULSIVE	∞	5731	0°
THEORETICAL	0.3	6023	9°

Figure 3. Theoretical Optimum LM Descent

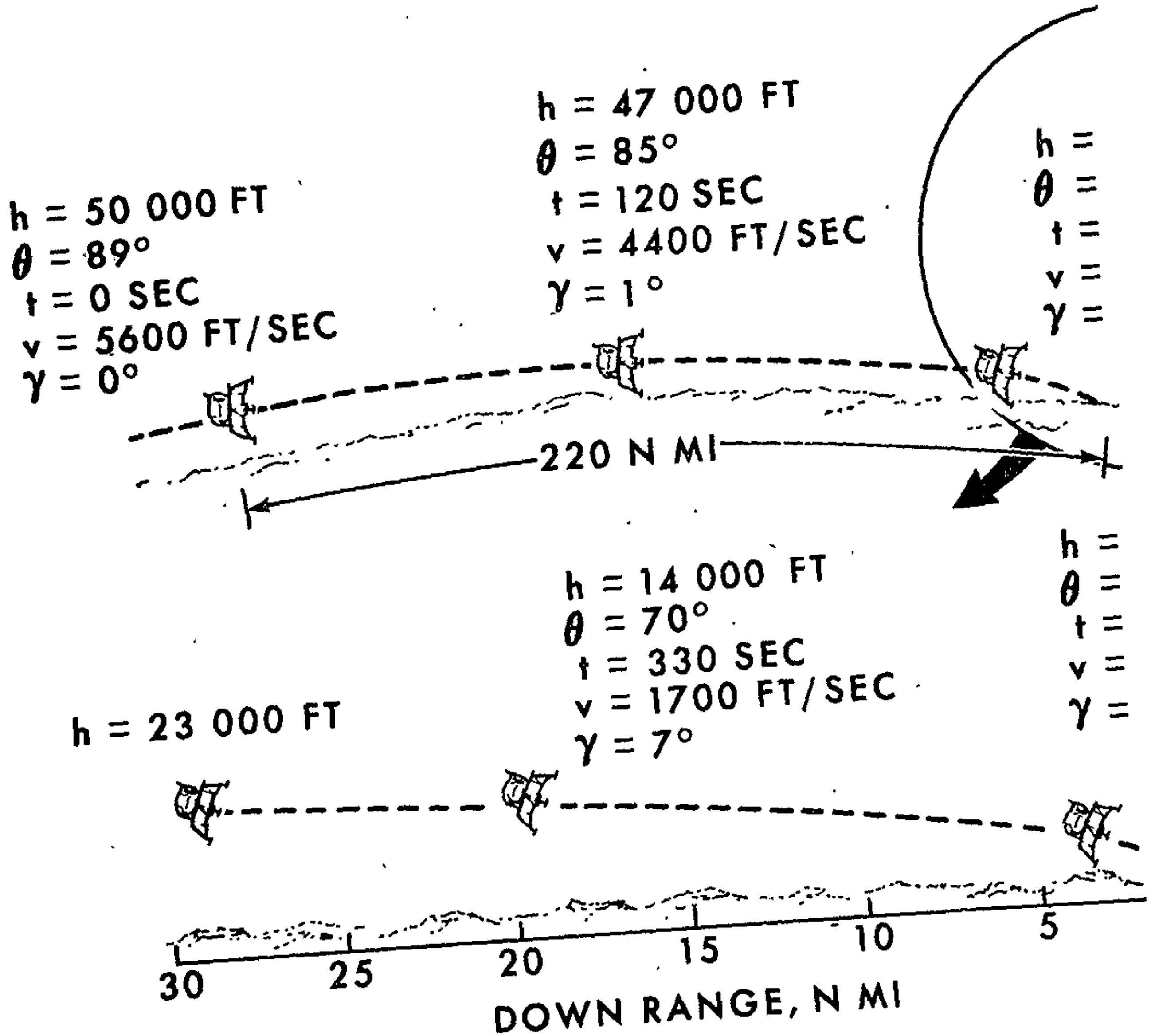


Figure 4. Optimum Powered Descent

NAVIGATION PHASE CONTRIBUTION		DOWN-RANGE σ, FT	CROSS TRACK σ, FT	CEP, FT	ALTITUDE σ, FT
LM SEPARATION AND HOHMANN DESCENT		1070	60	730	540
POWERED DESCENT		260	1410	1000	1490
RSS OF THE ABOVE TWO		1100	1410	1480	1580
LUNAR ORBIT NAVIGATION	MSFN	2320	700	1750	840
	ONBOARD	2840	540	1990	1180
TOTAL ACCURACY	MSFN	2570	1570	2410	1790
	ONBOARD	3040	1510	2630	1970

Figure 5. LM Landing Accuracy After Three Orbits

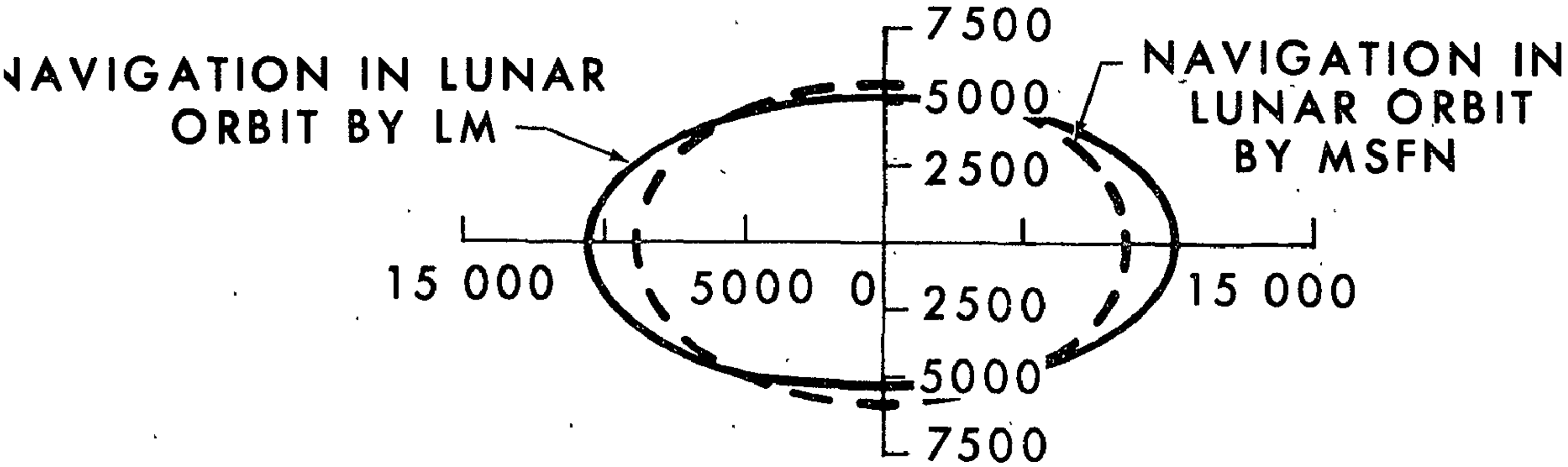
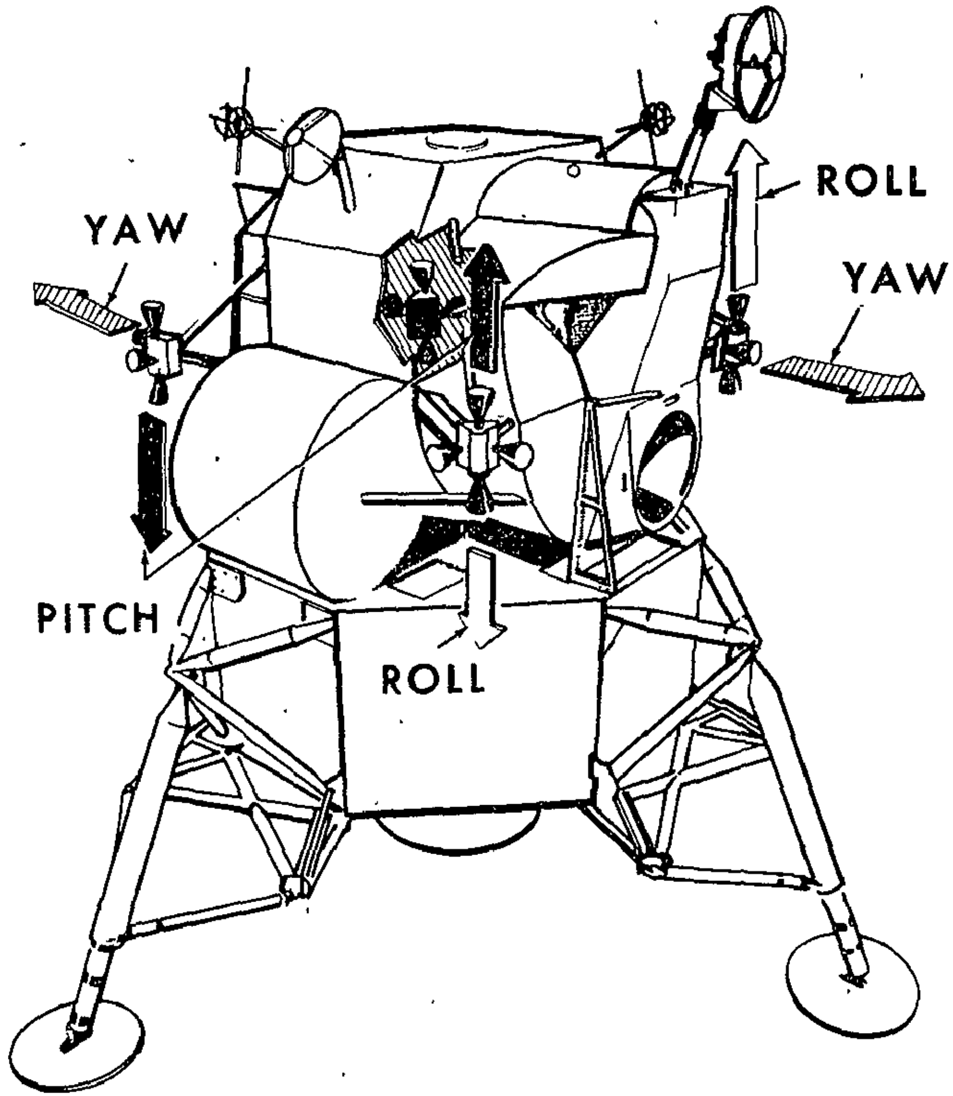


Figure 6. LM Landing 3σ Uncertainty Ellipse After Three Orbits



NOTE:
IN DESCENT THRUST CONFIGURATION MAIN ENGINE GIMBAL IS EMPLOYED FOR TRIMMING THE PITCH AND YAW MOMENT DUE TO CENTER-OF-GRAVITY SHIFTS

Figure 7. Attitude Control of LM

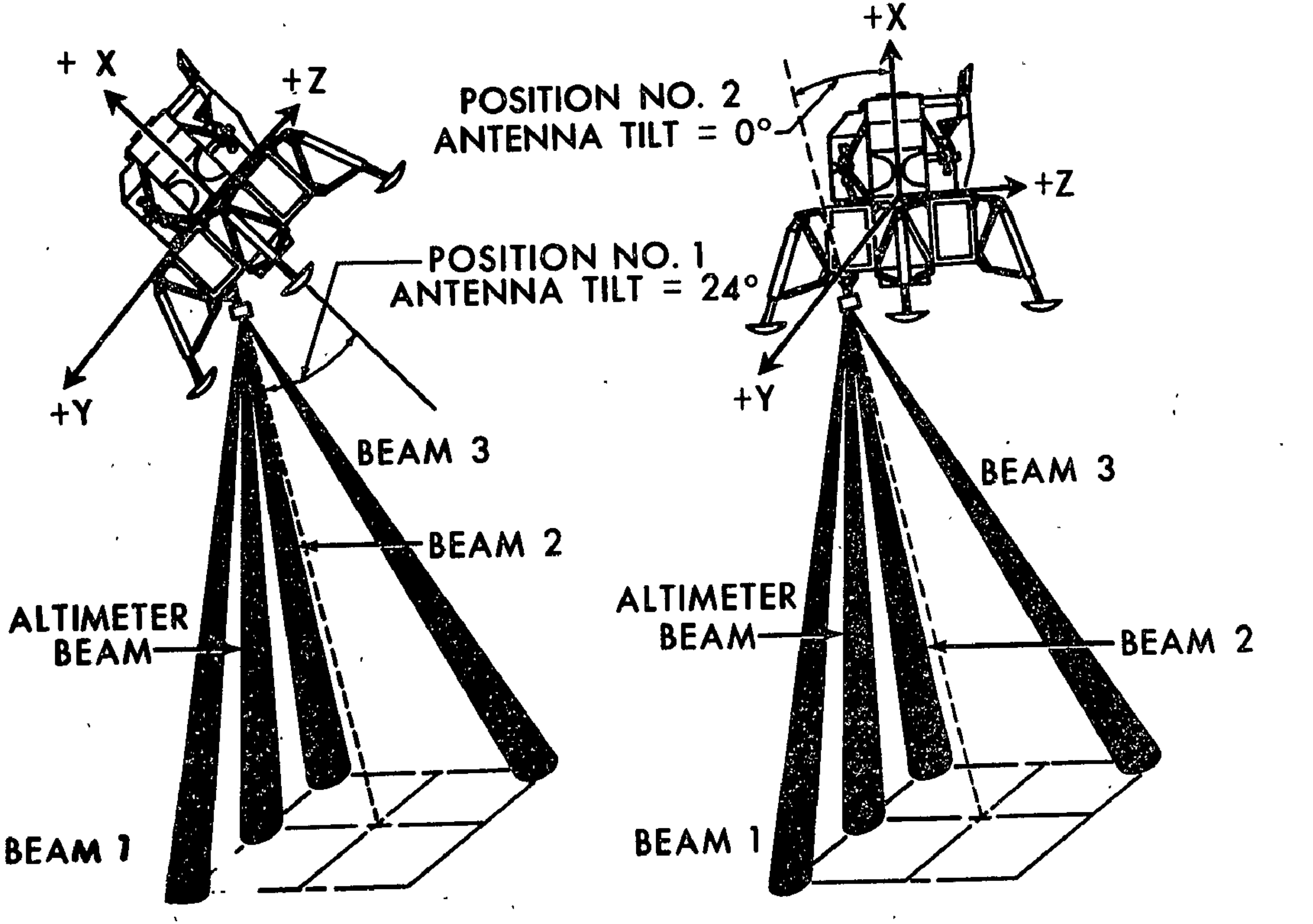


Figure 8. Landing Radar Beam Configuration and Antenna Tilt Angles

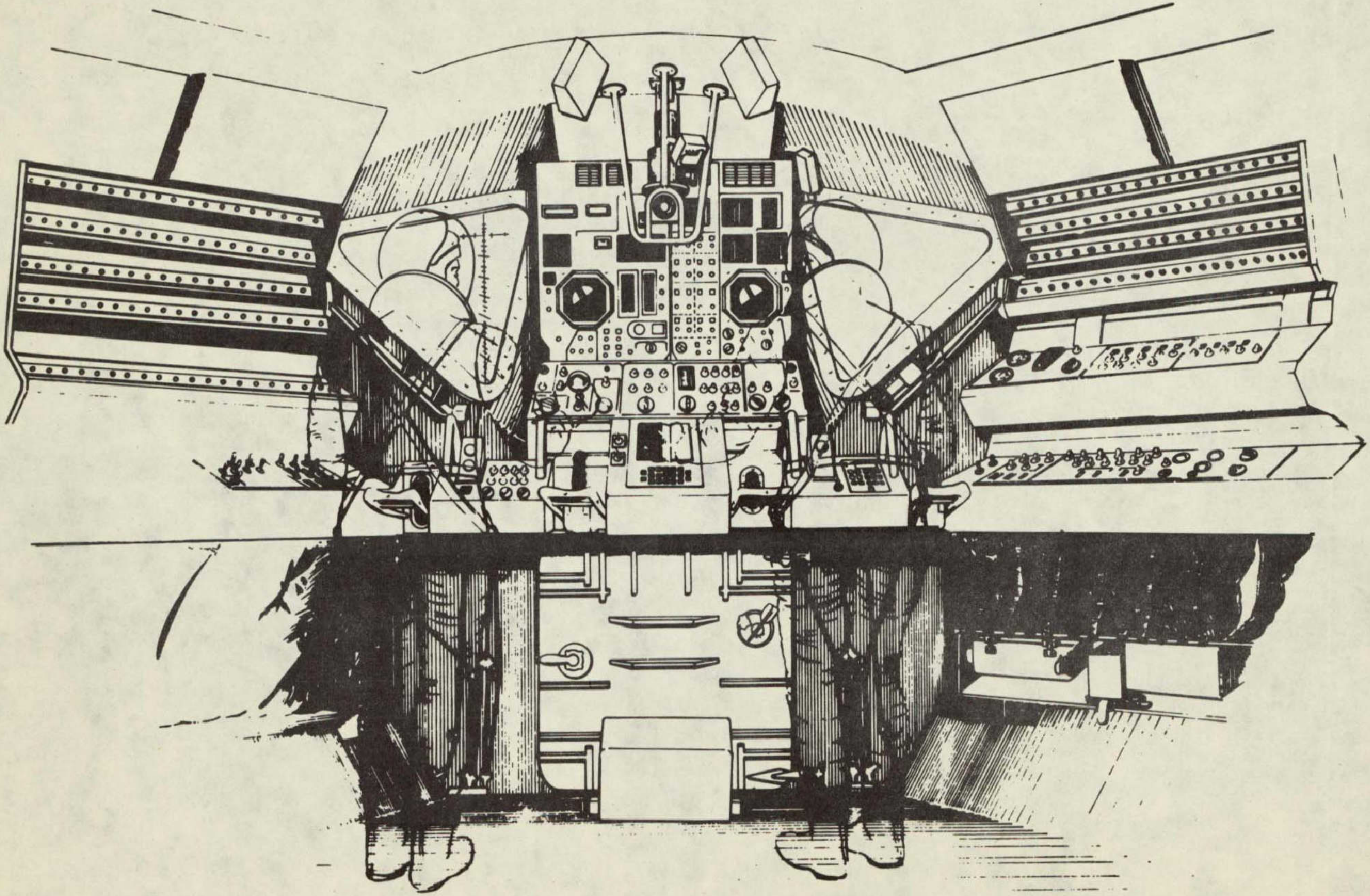


Figure 9. LM Flight Configuration

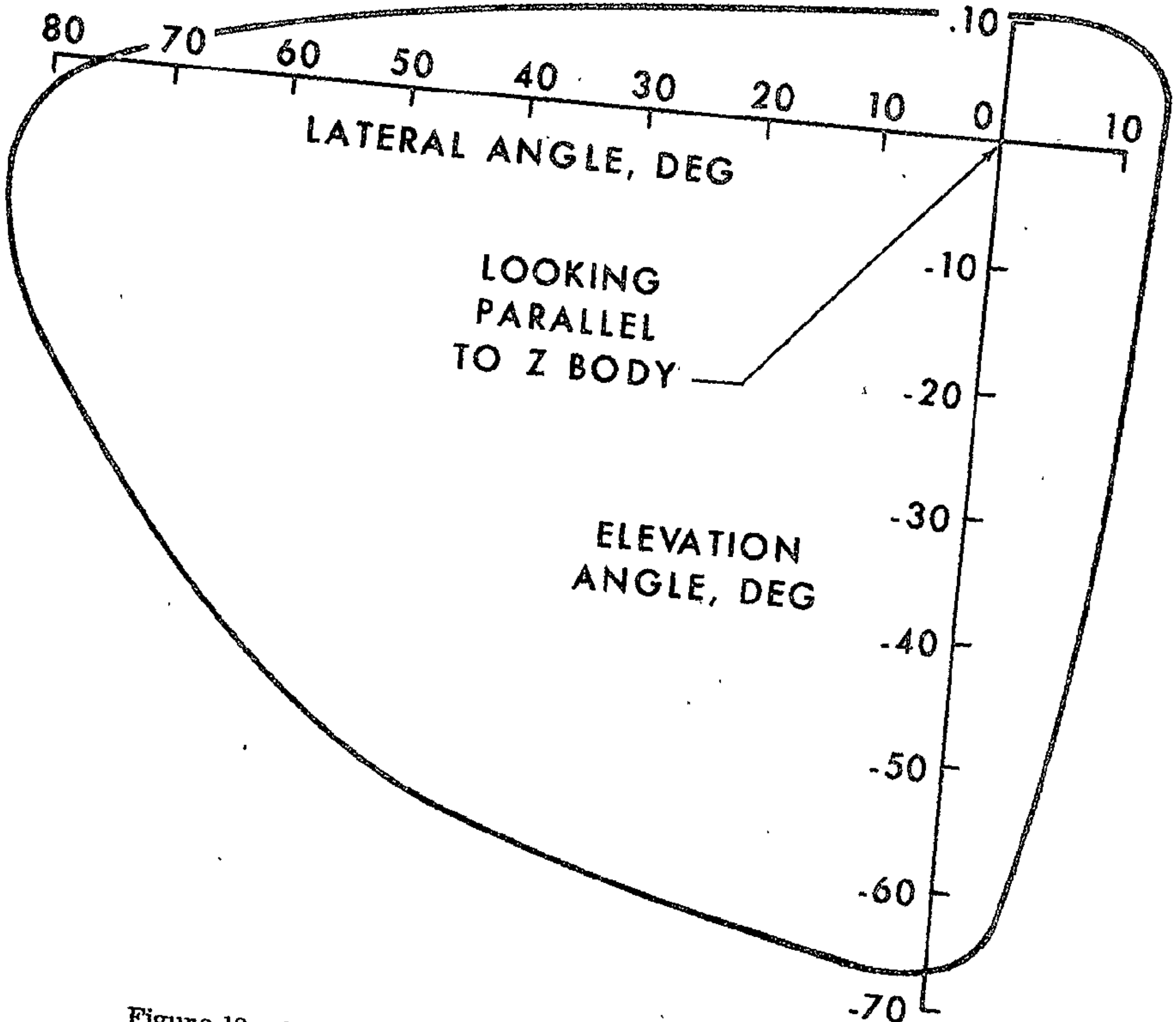


Figure 10. LM Window View Limits From Commander's Design Eye Position

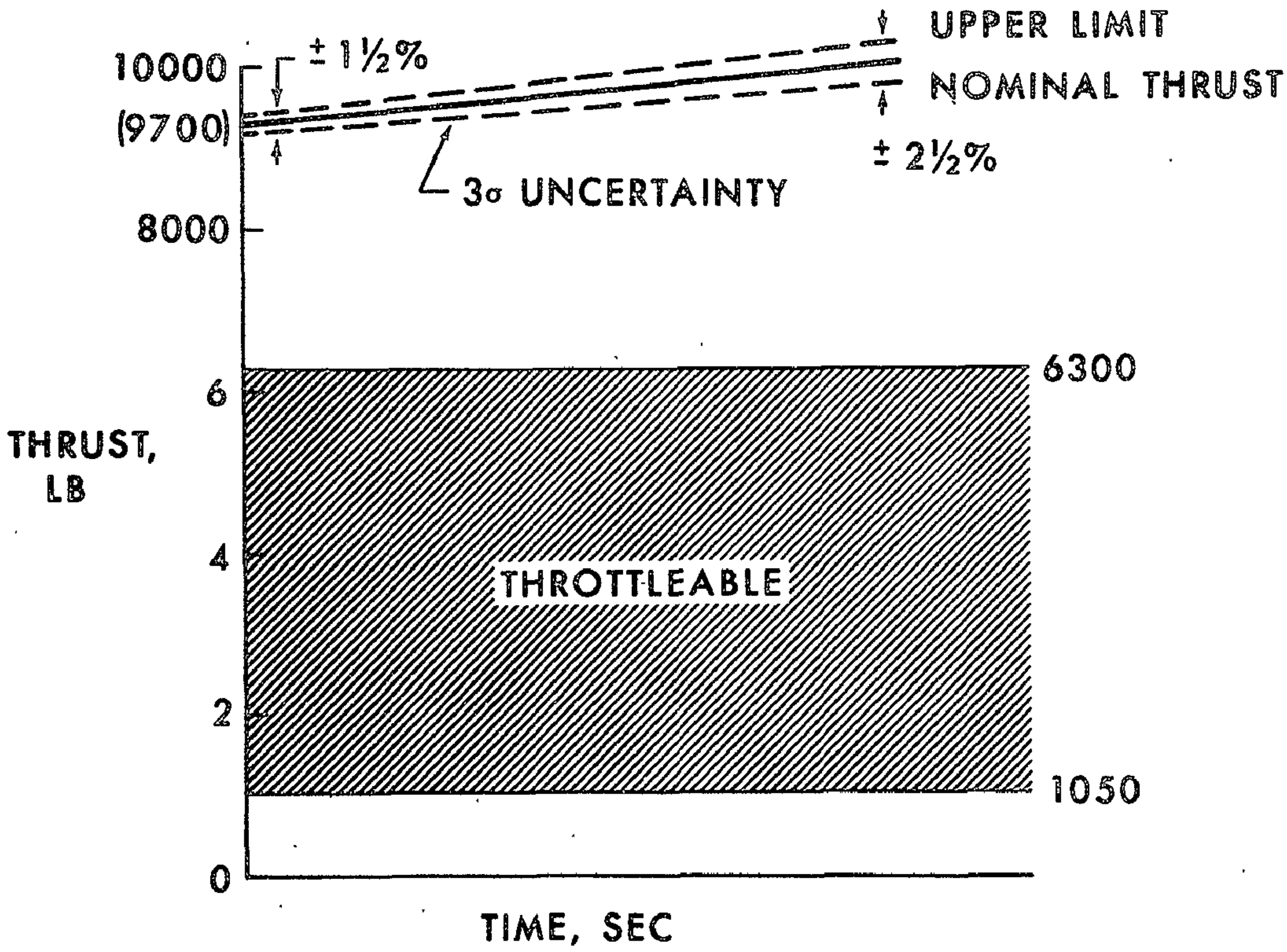


Figure 11. LM Descent Engine Thrust Characteristics

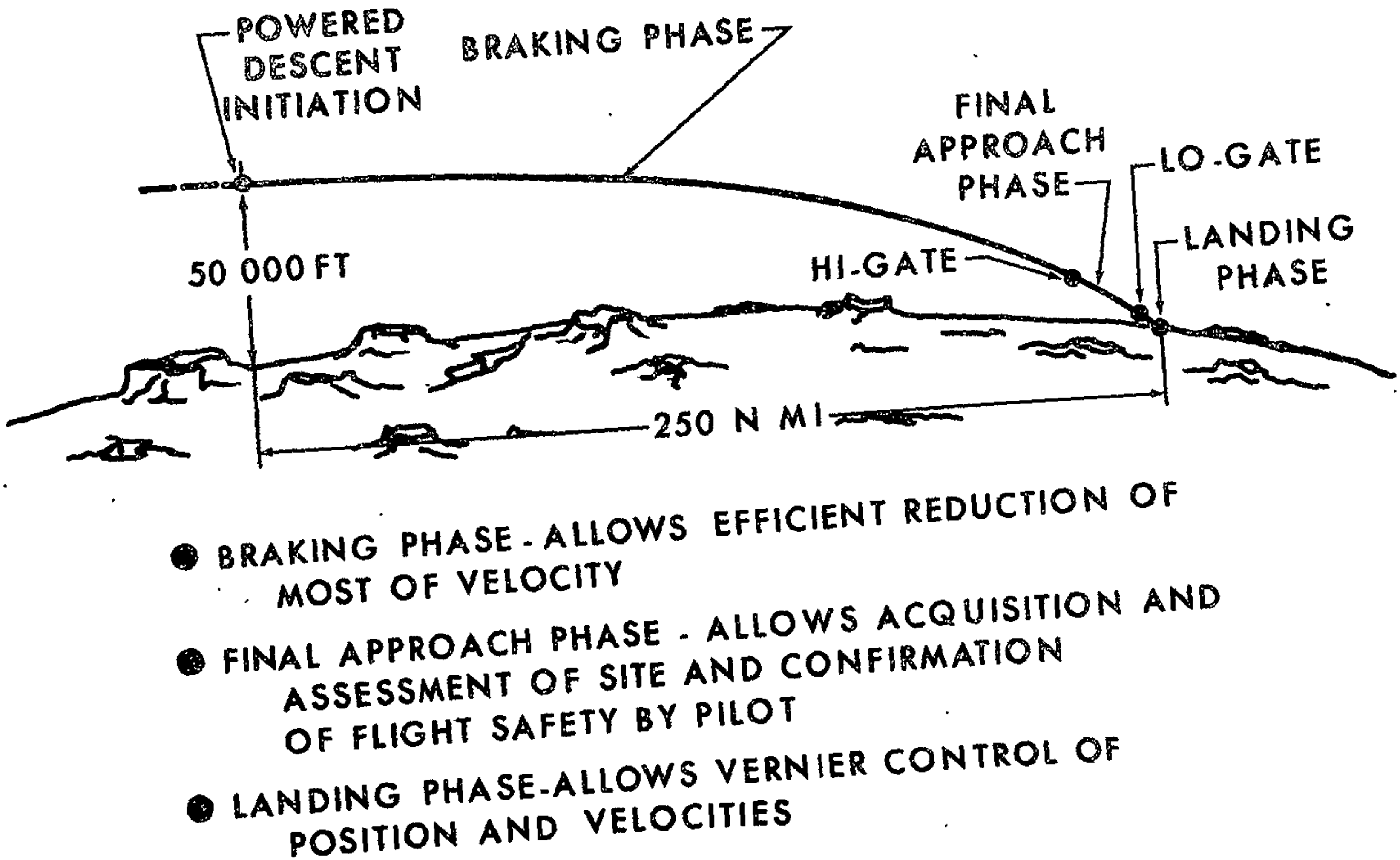


Figure 12. LM Three-Phase Powered Descent

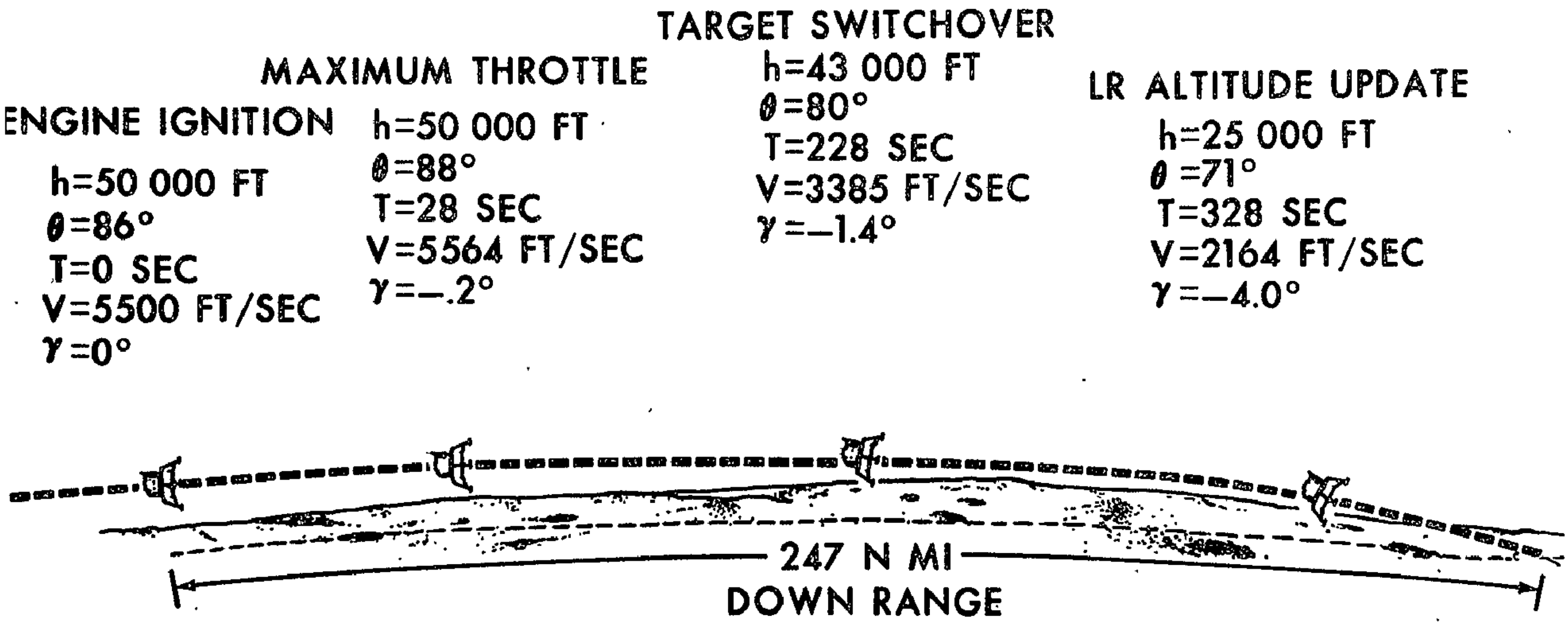


Figure 13 (a). LM Powered Descent

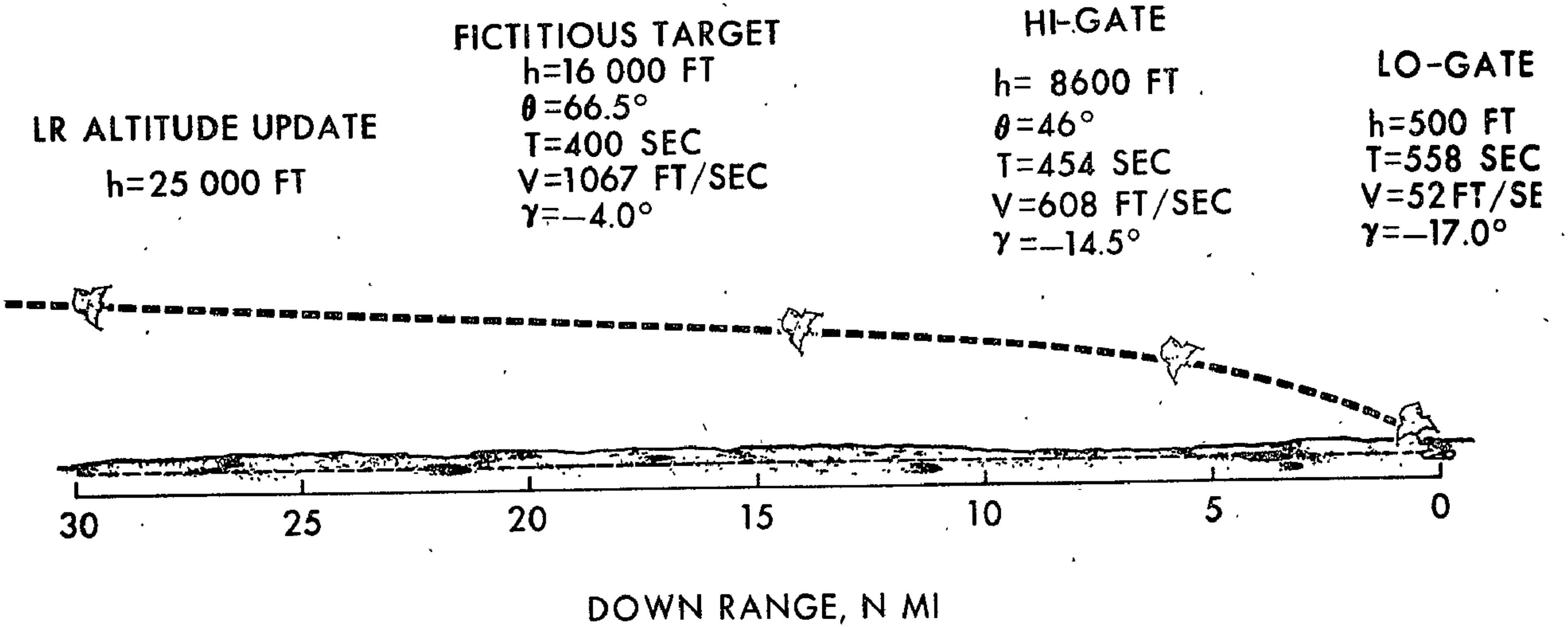


Figure 13 (b). LM Powered Descent

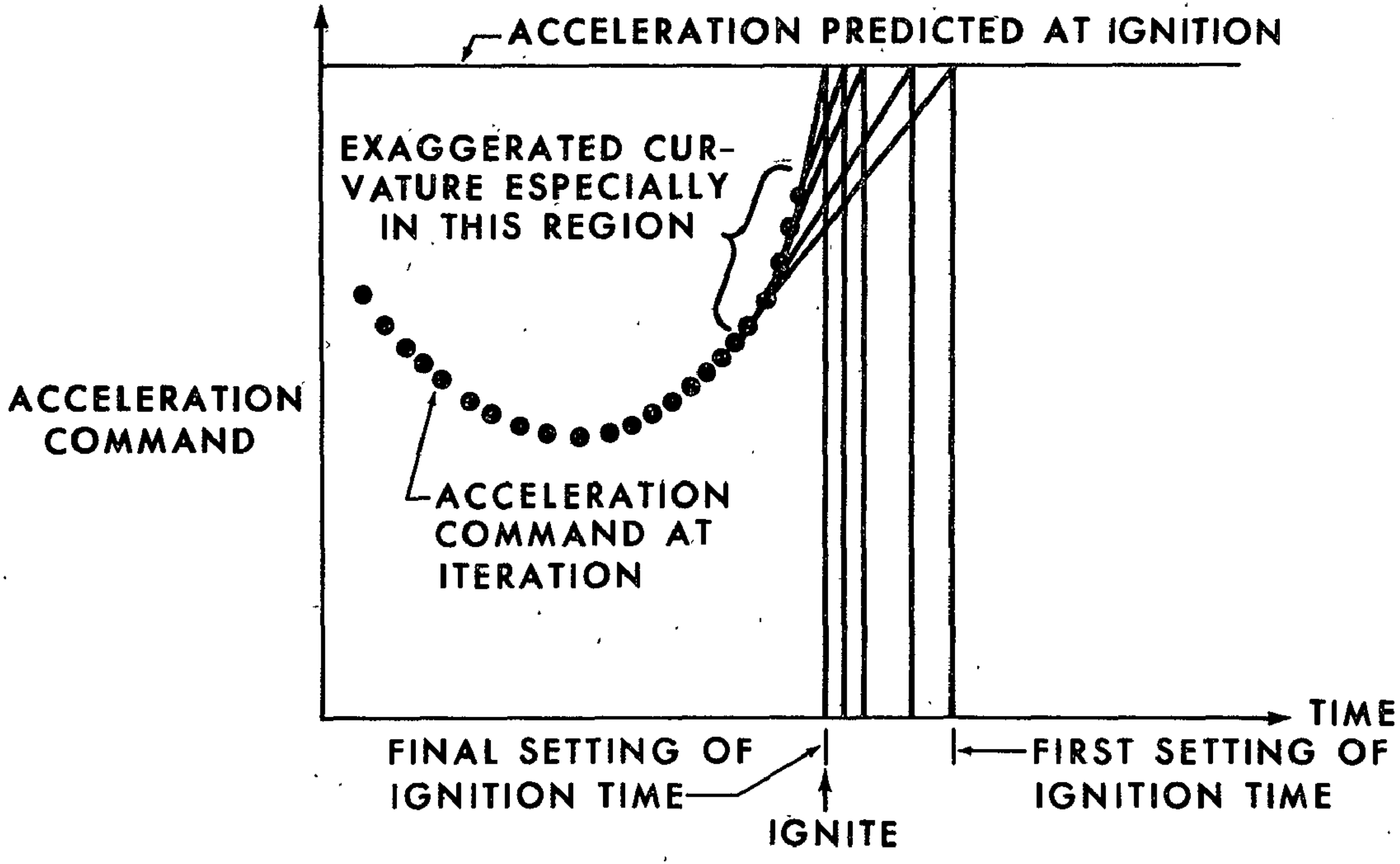


Figure 14. Powered Descent Ignition Logic

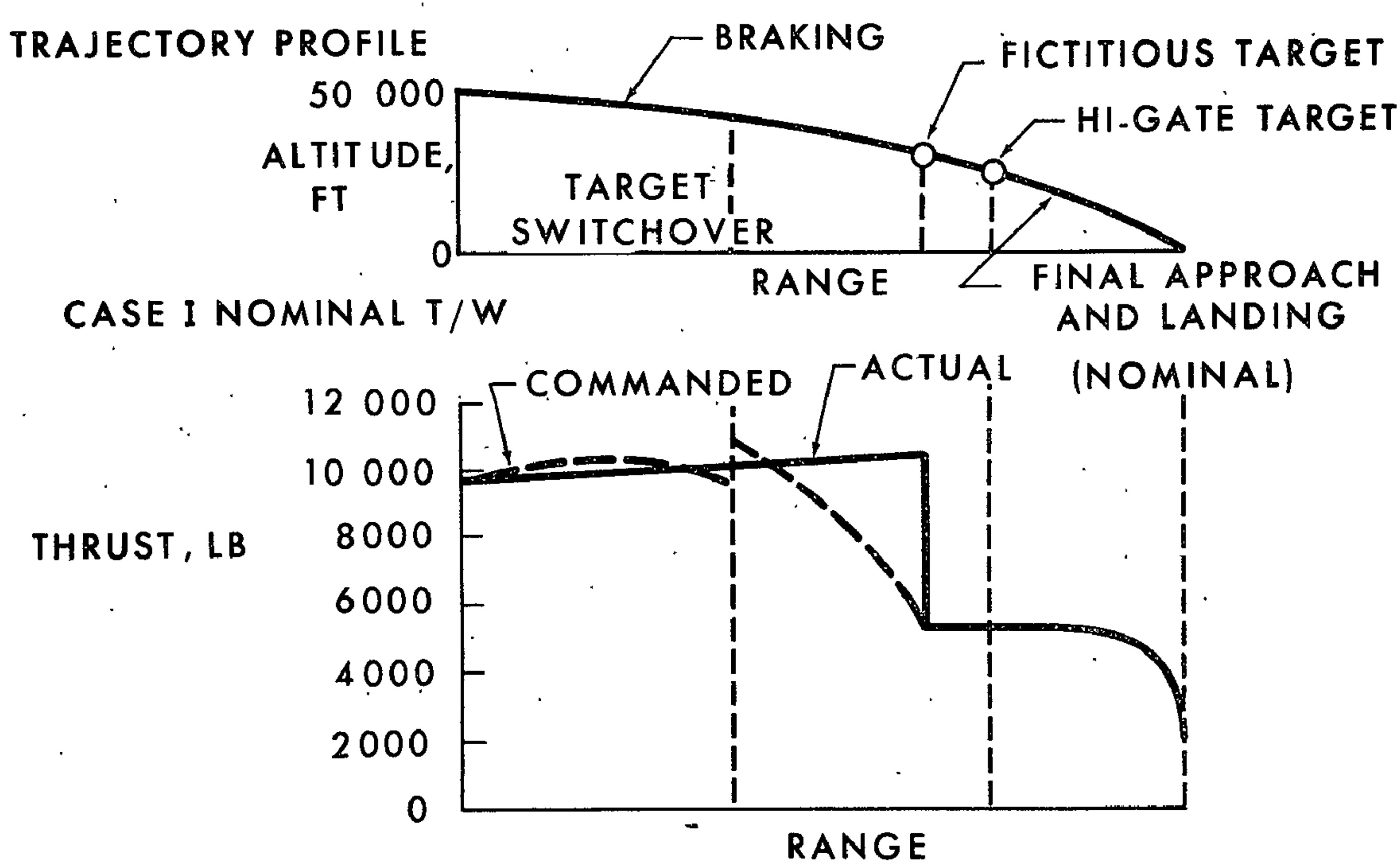


Figure 15. Thrust Behavior for Limited Throttle Guidance

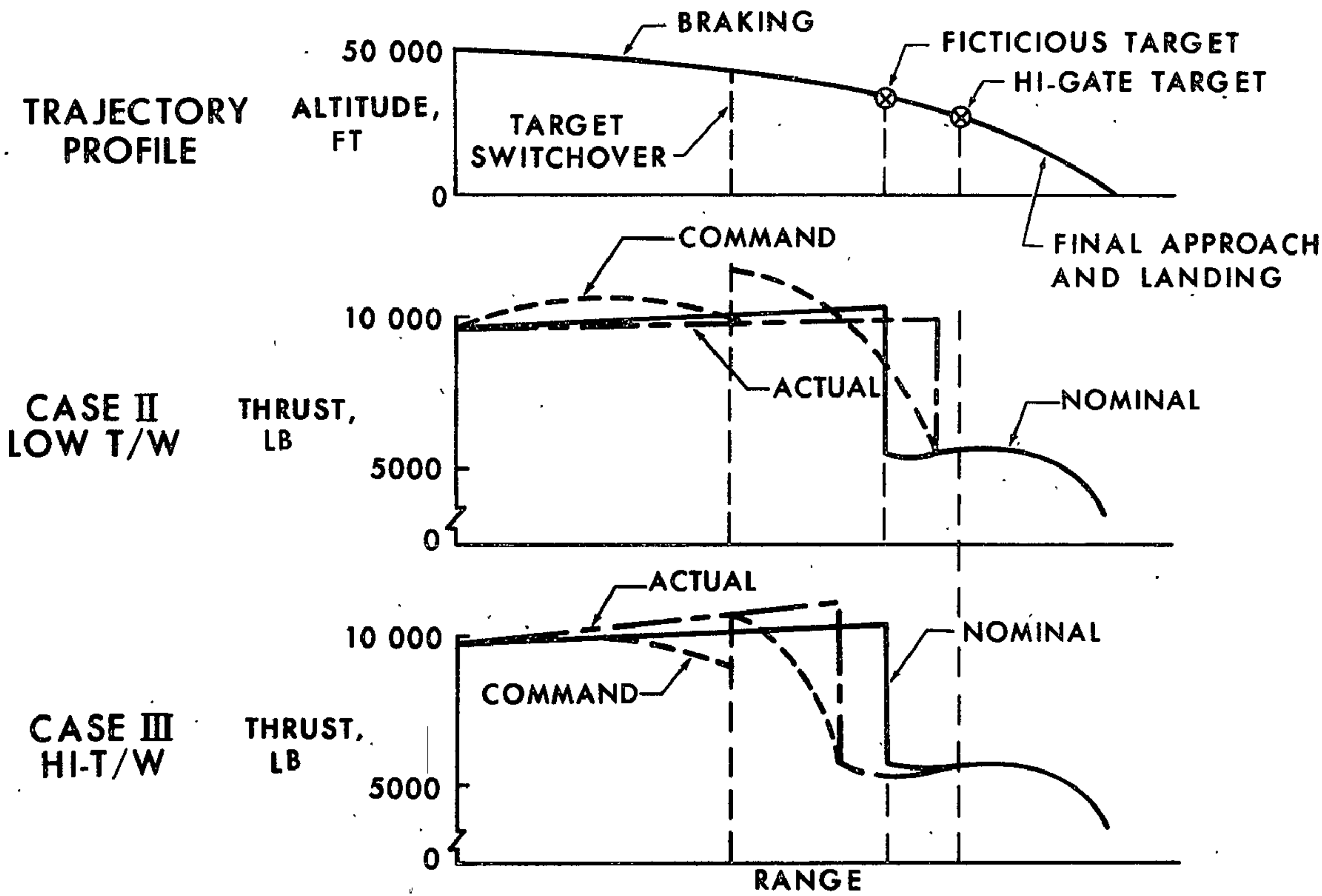


Figure 16. Thrust Behavior for Limited Throttle Guidance

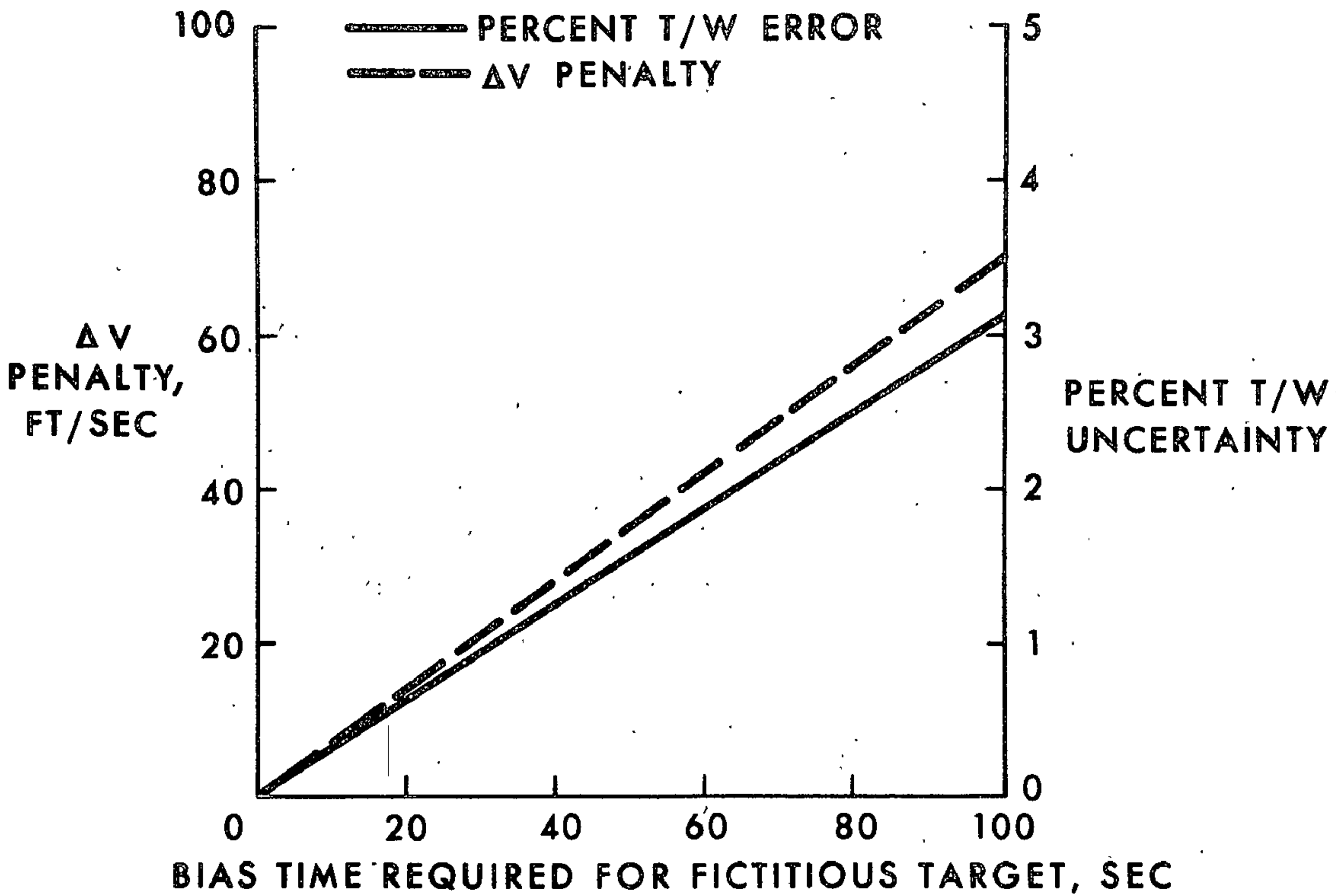
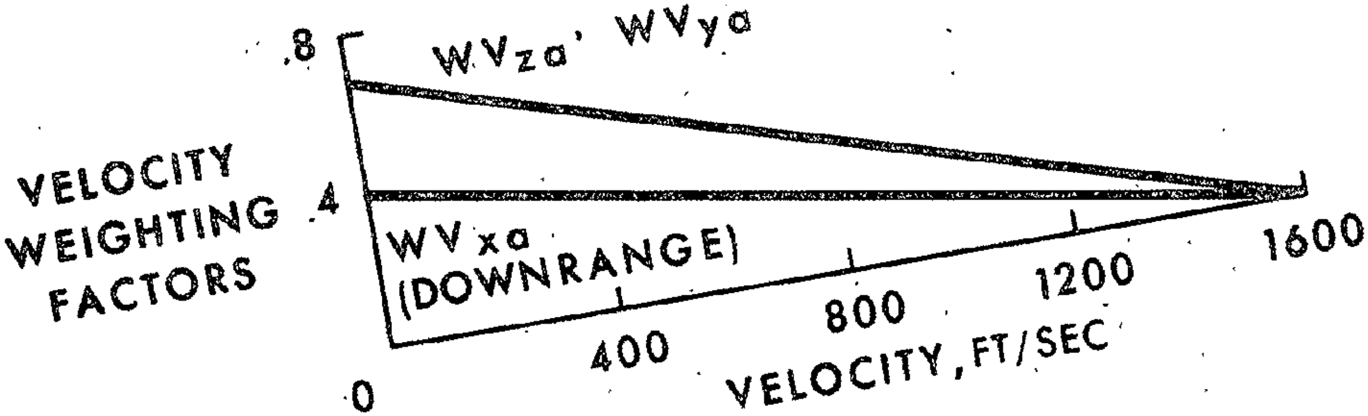


Figure 17. V Penalty Due to Fixed Thrust Uncertainties



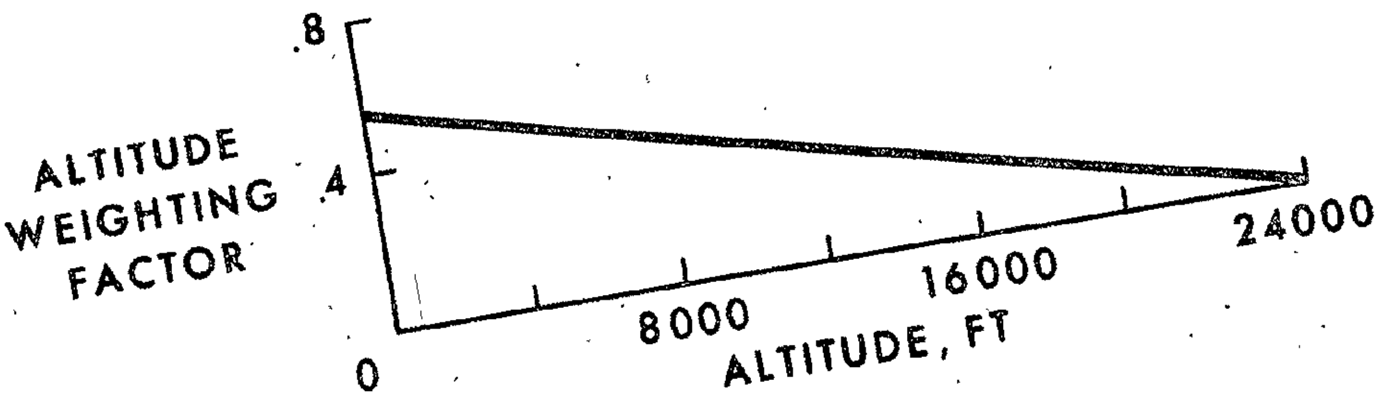


Figure 18. Landing Radar Weighting Factors for Altitude and Velocity Updates

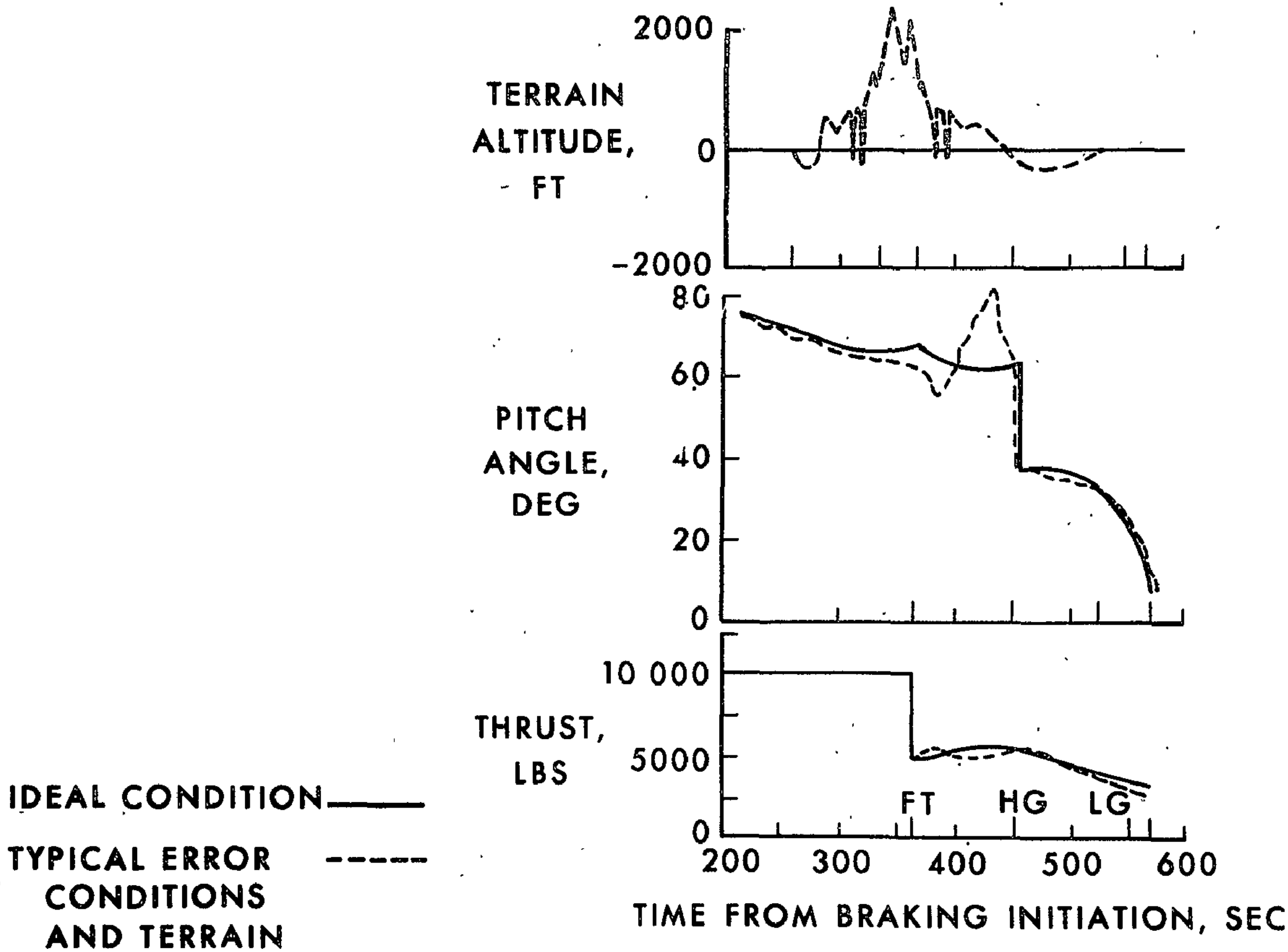
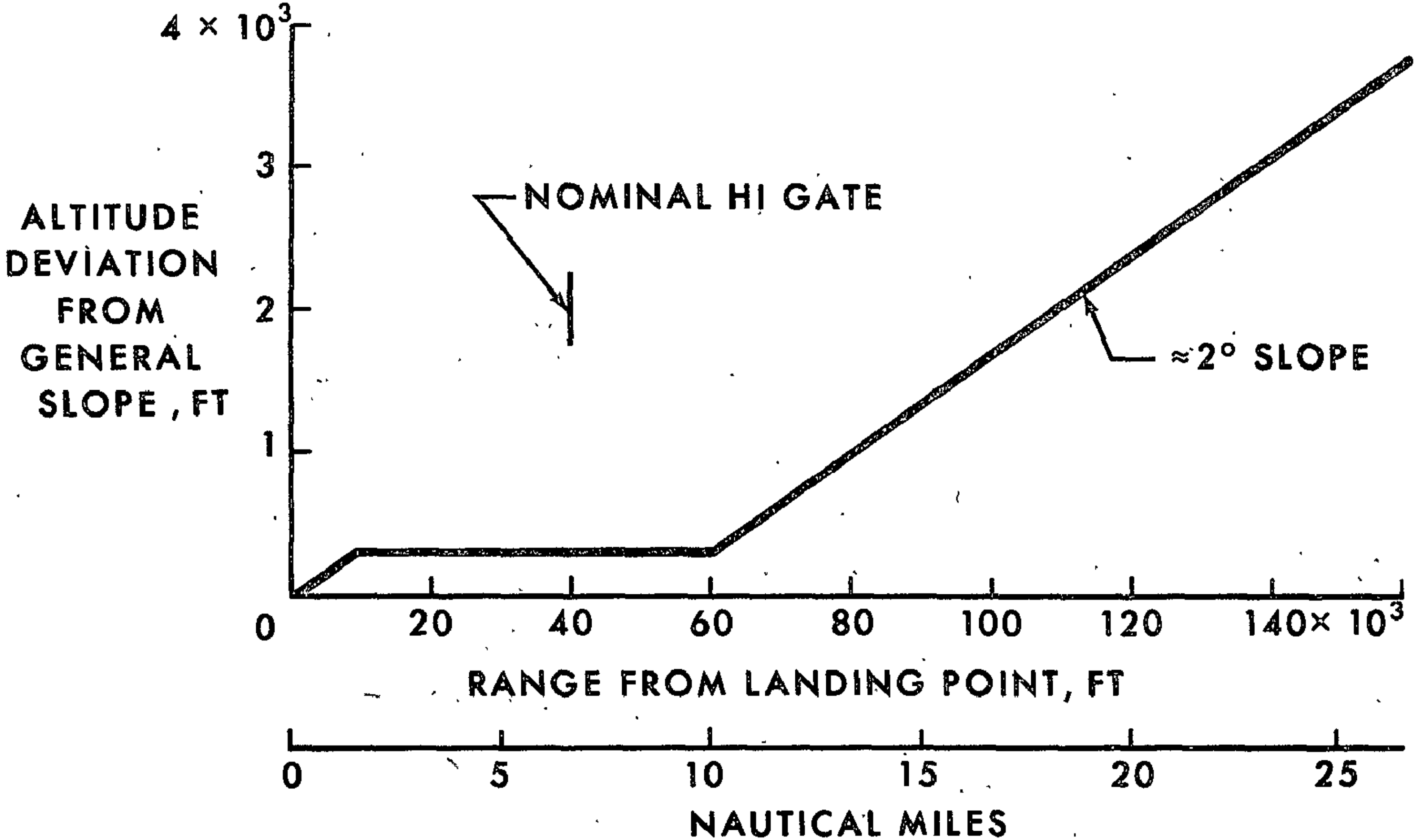
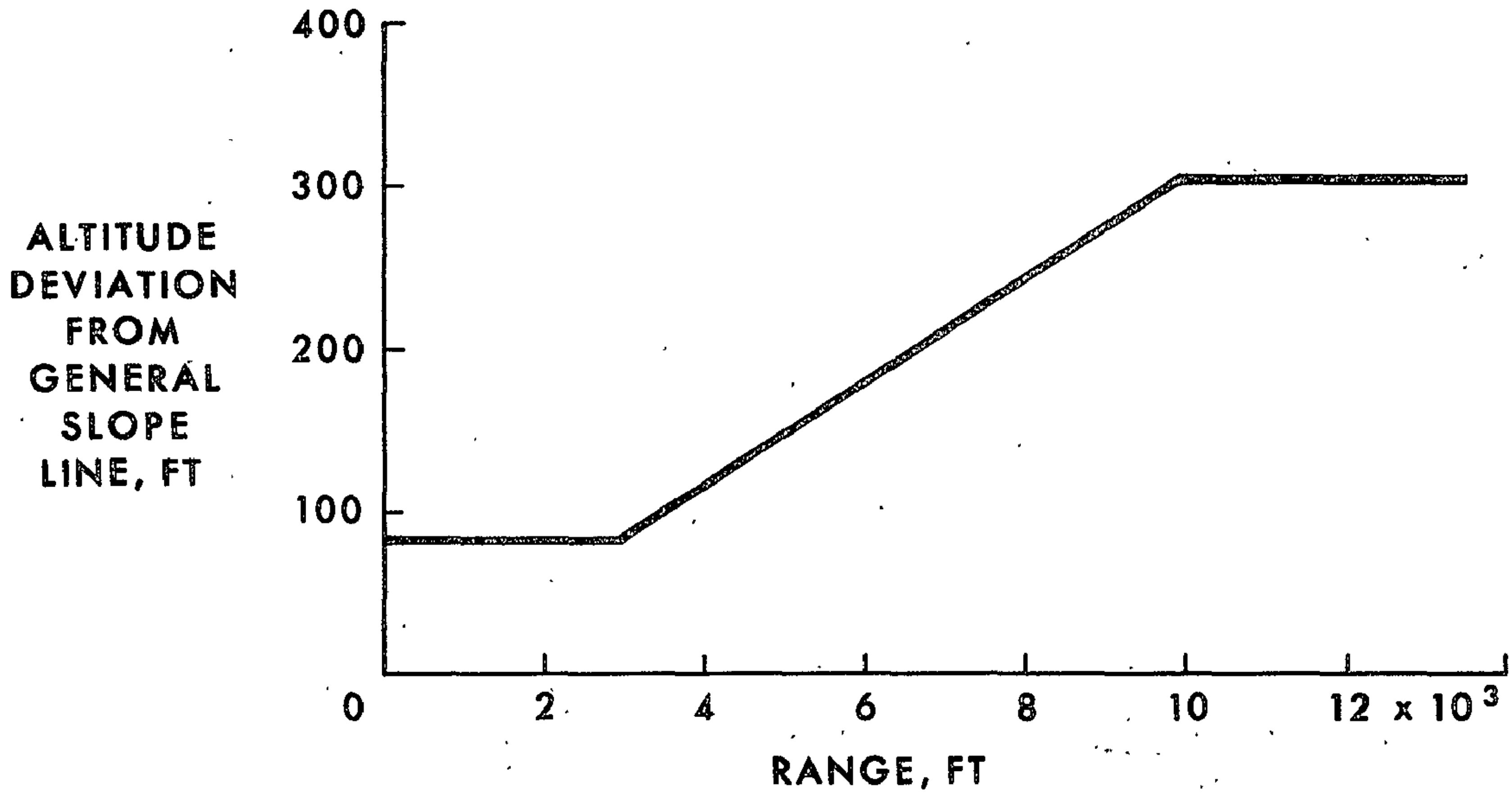


Figure 19. Guidance Commands for Power Descent



20 (a). Site Selection Terrain Criteria
(Excepts General Slope)



20 (b). Site Selection Terrain Criteria

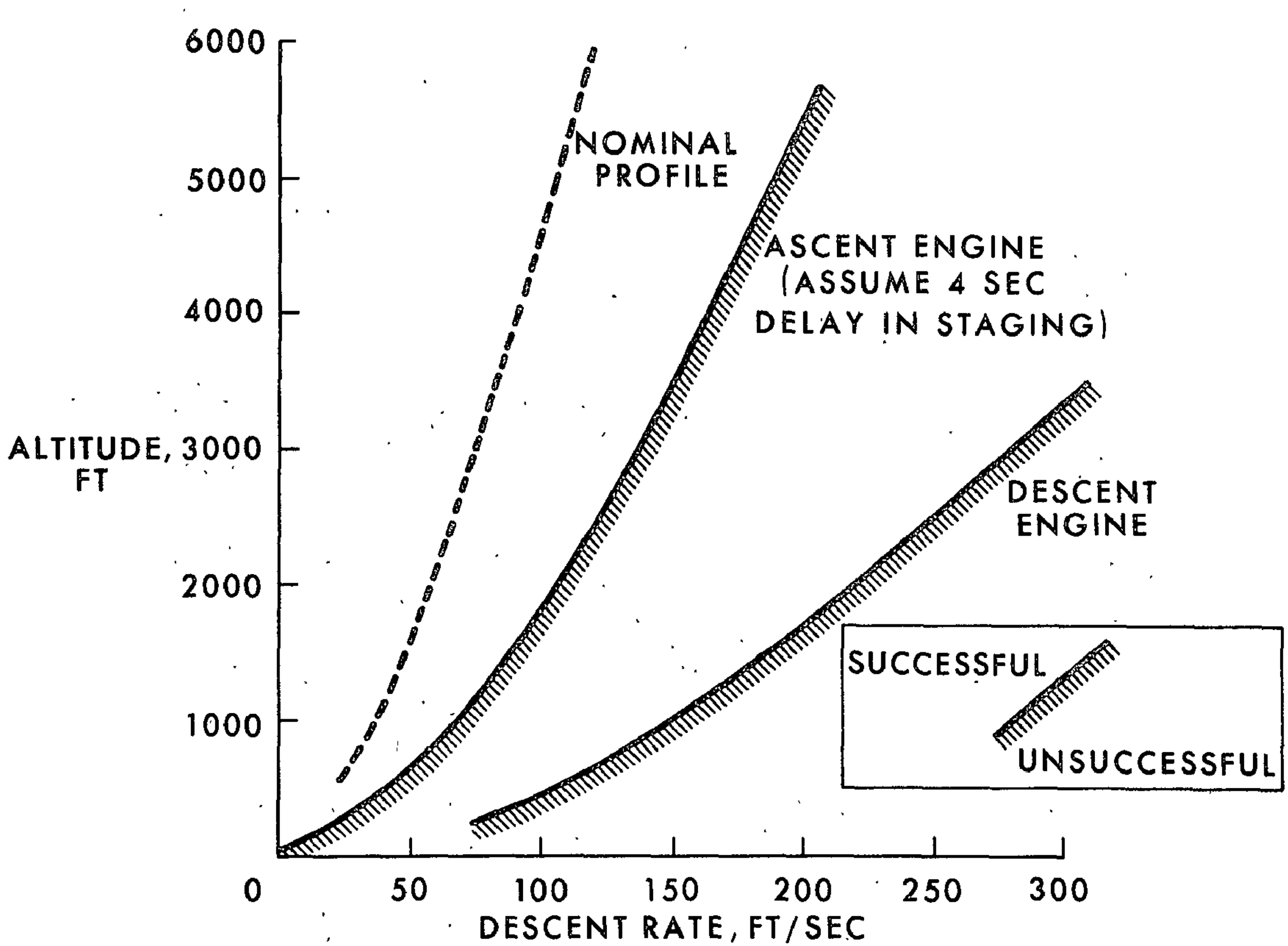


Figure 21. Abort Capability Boundaries

• GUIDANCE AND NAVIGATION UNCERTAINTIES	1500 FT ALT 1σ
• LUNAR RADIUS BIAS MAGNITUDE	3200 FT ALT
• LUNAR RADIUS RANDOM MAGNITUDE	3200 FT ALT 1σ
• PRESENT ABILITY TO DETERMINE MARIA AREA SLOPES ($\pm 3°$ 3σ)	FUNCTION OF LANDING DISPERSIONS
• ALLOWABLE TERRAIN VARIATIONS BENEATH APPROACH TRAJECTORY	FUNCTION OF LANDING DISPERSIONS

Figure 22. Factors Contributing to Uncertainties in Altitude Above Terrain

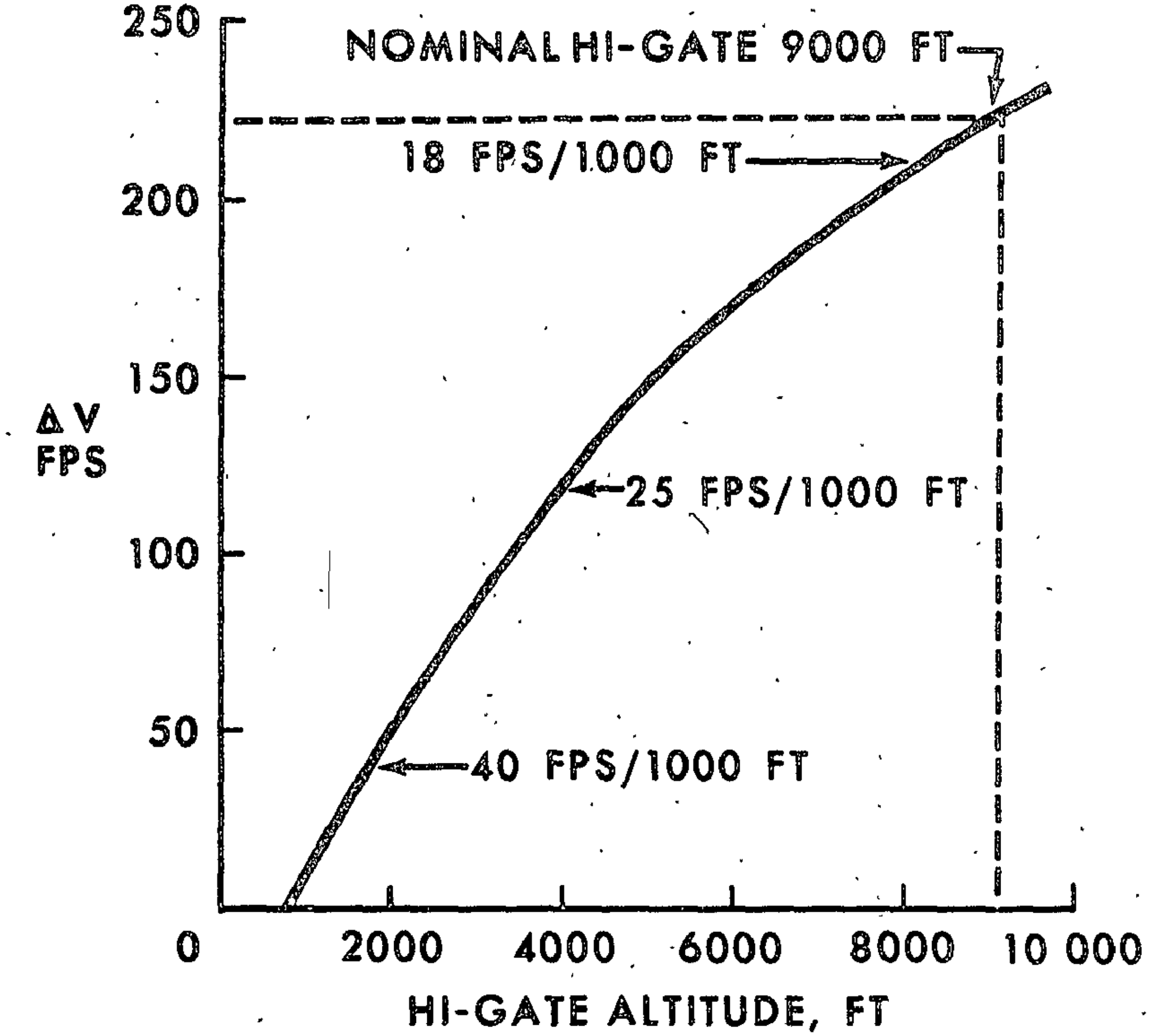


Figure 23. V Penalty for Hi-Gate Altitude Variation---Typical Flight Path Angle (Hi-Gate 9000 Ft)

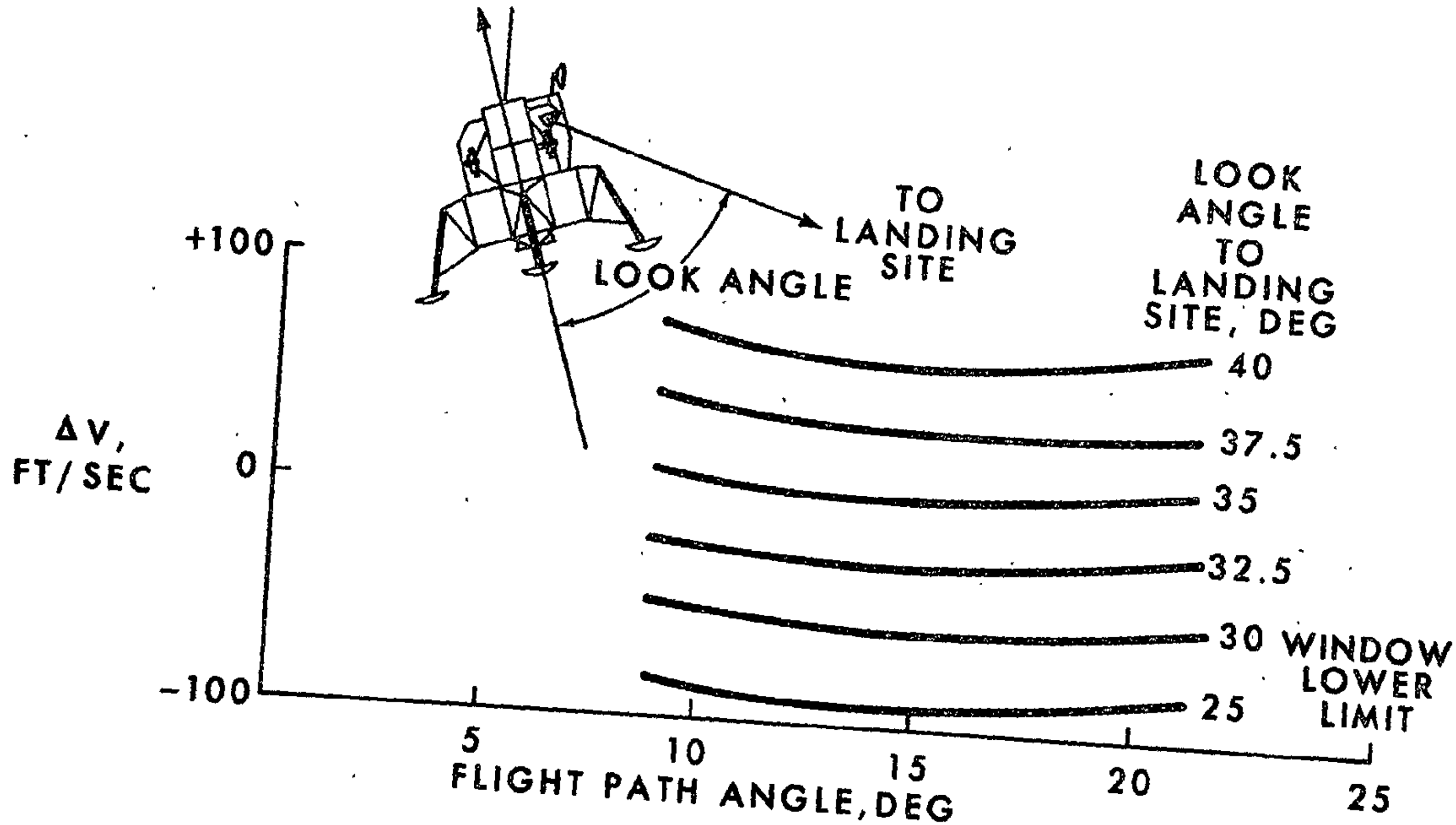


Figure 24. V Penalty for Look Angle and Flight Path Angle (Hi-Gate 9000 Ft)

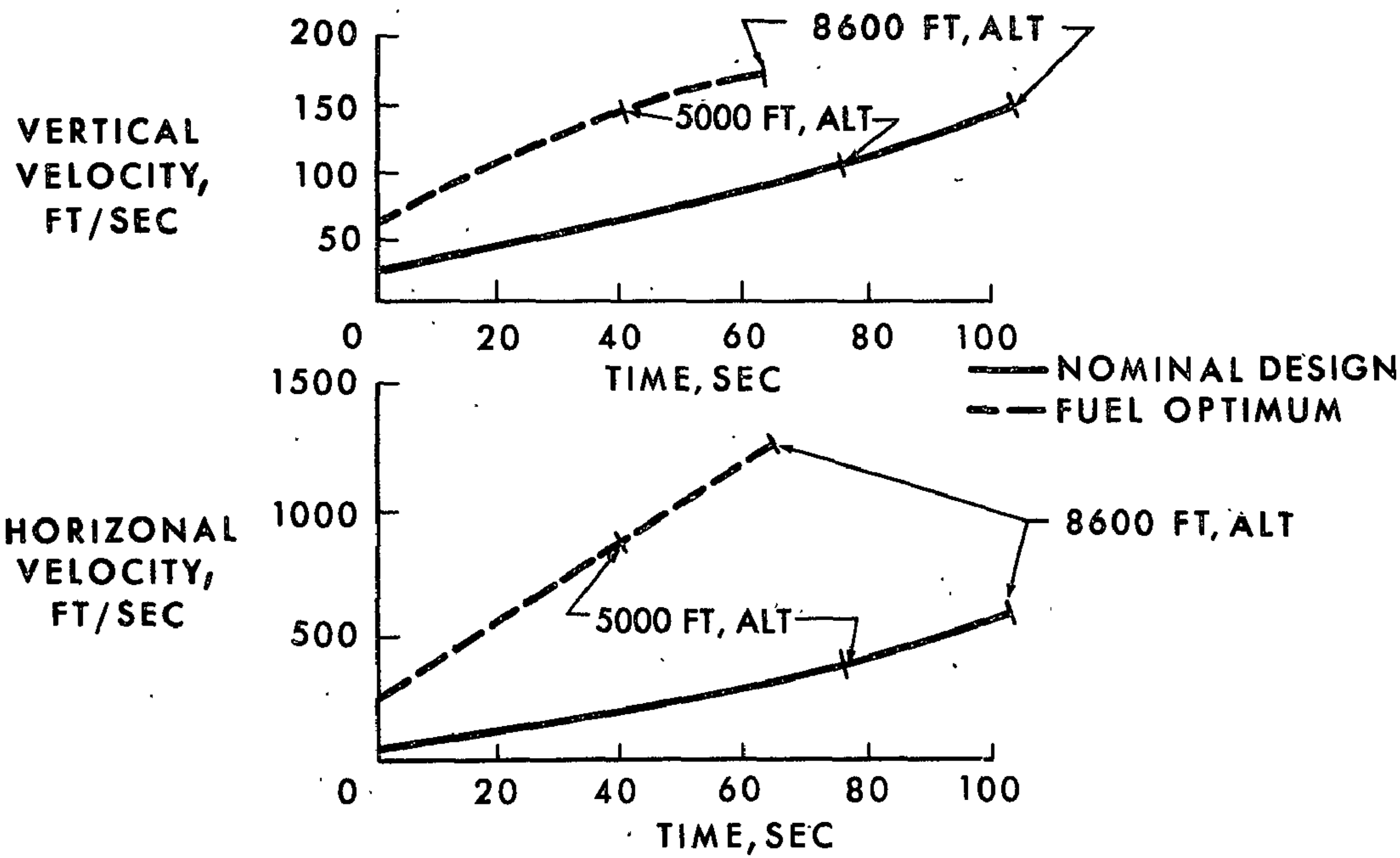


Figure 25. Comparison of Design Trajectory and Fuel Optimum

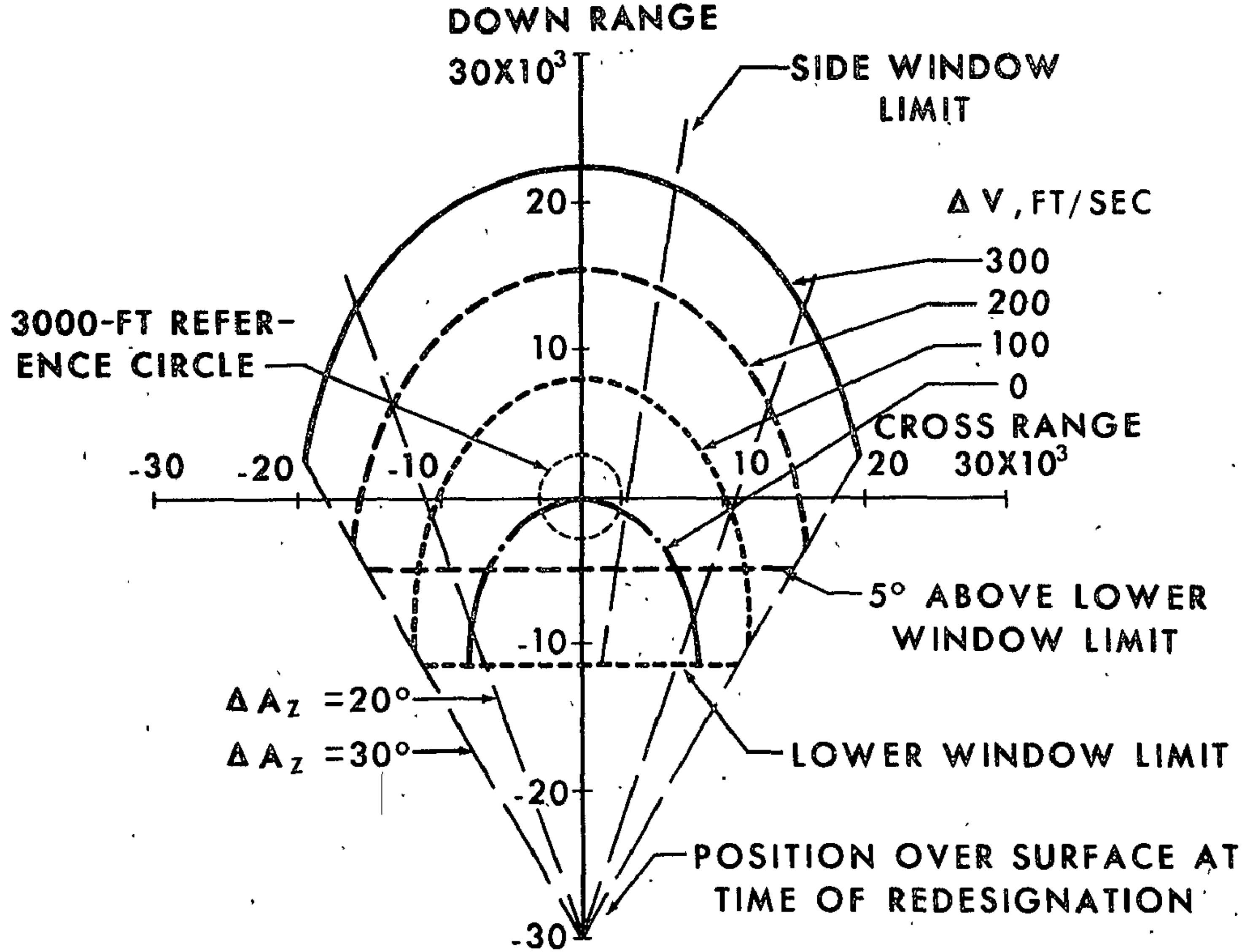


Figure 26. Overhead Profile of Footprint Capability From 8000 Ft Altitude

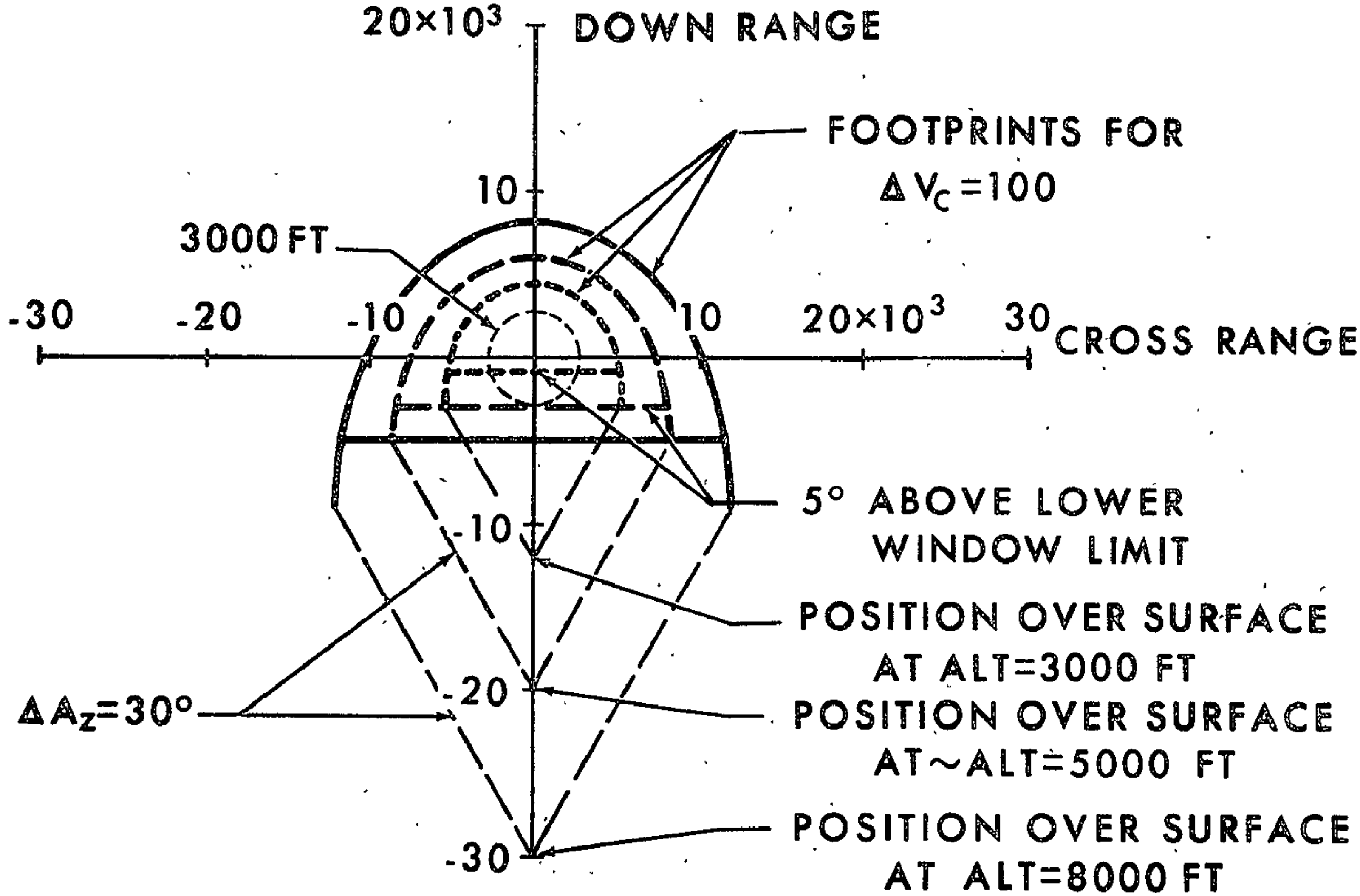


Figure 27. Variation of Footprint Capability with Altitude

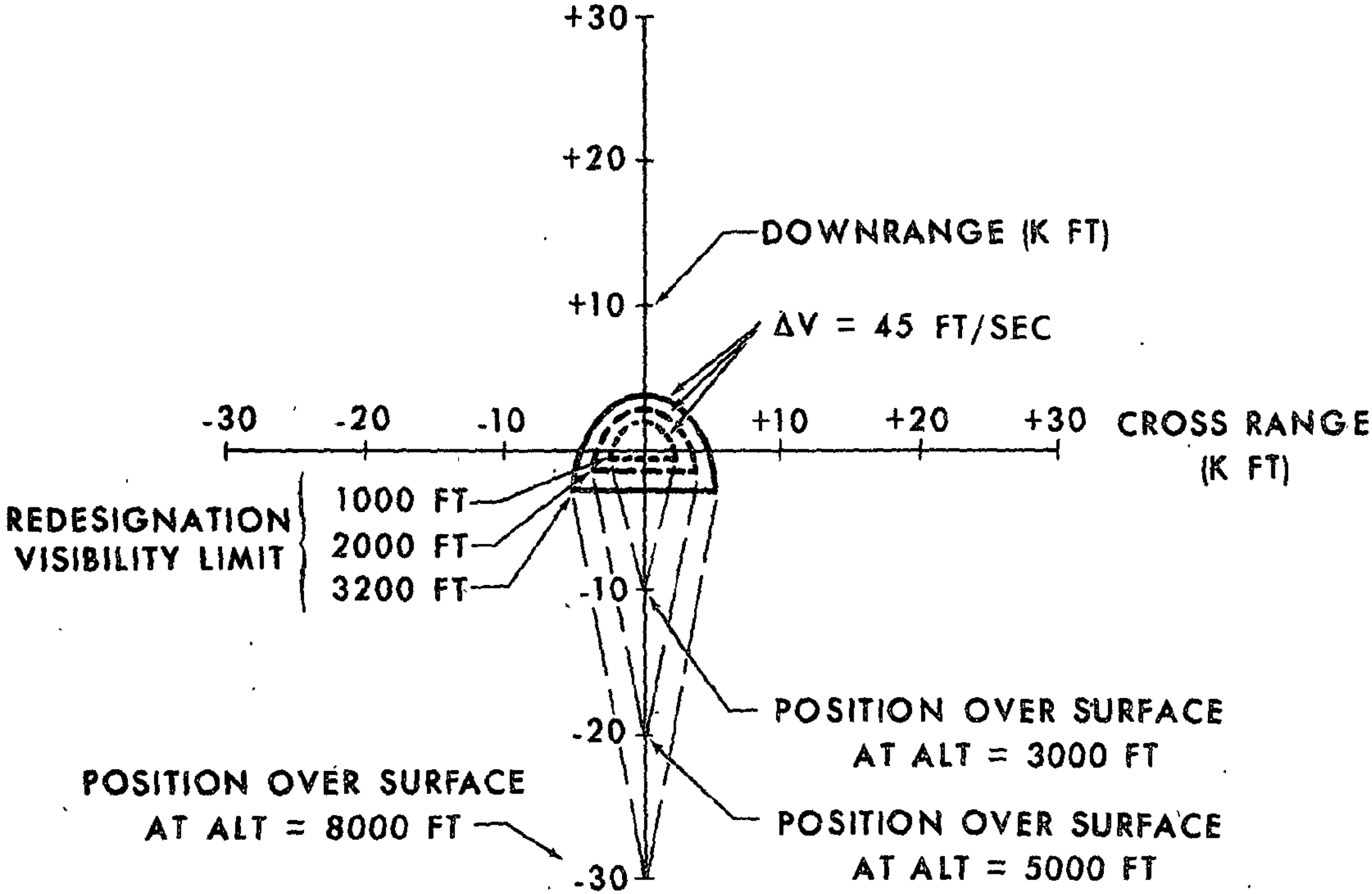


Figure 28. Variation of Footprint with Altitude During Descent

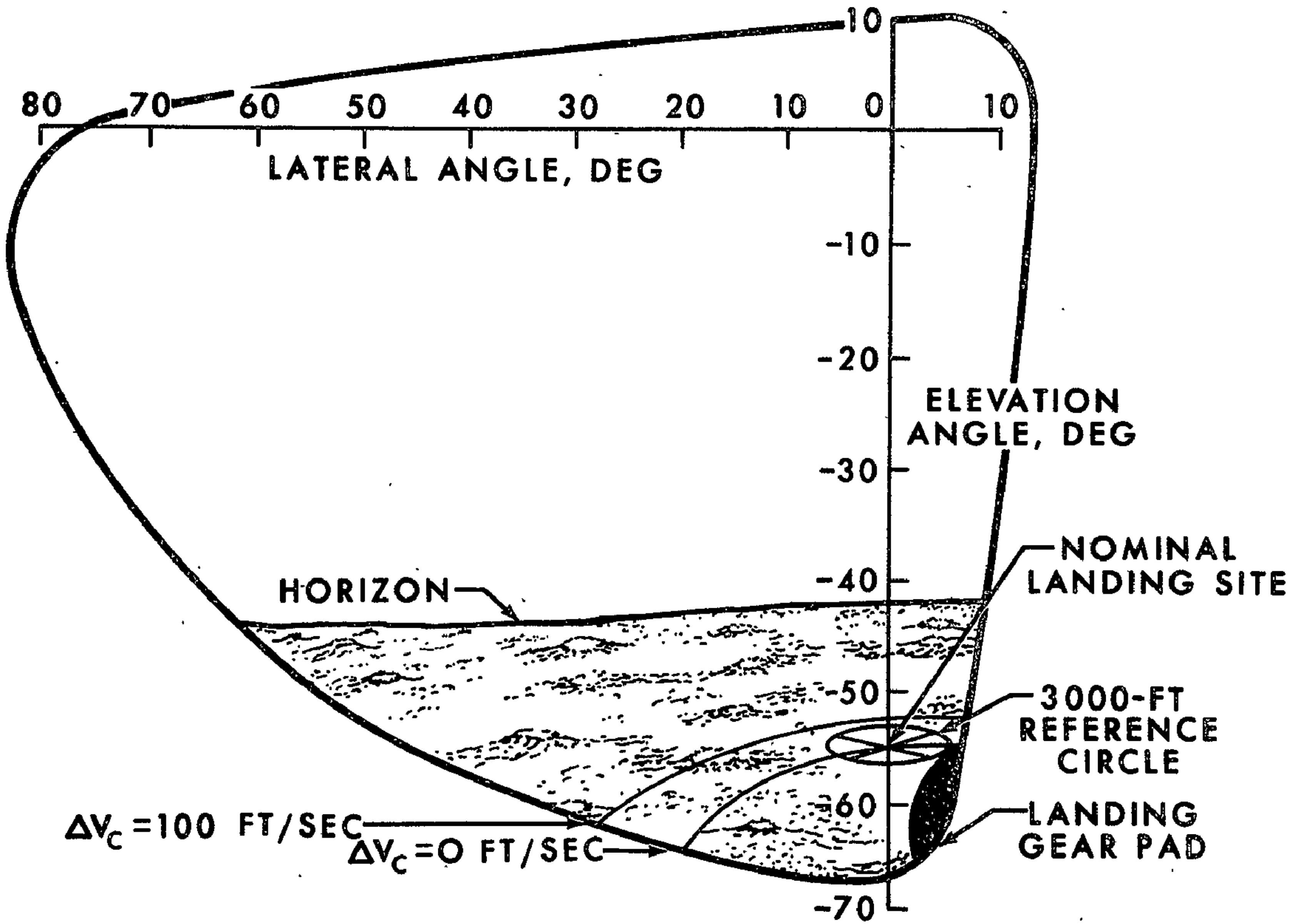


Figure 29. Landing Footprint As Seen By Pilot from 8000 Ft. Altitude

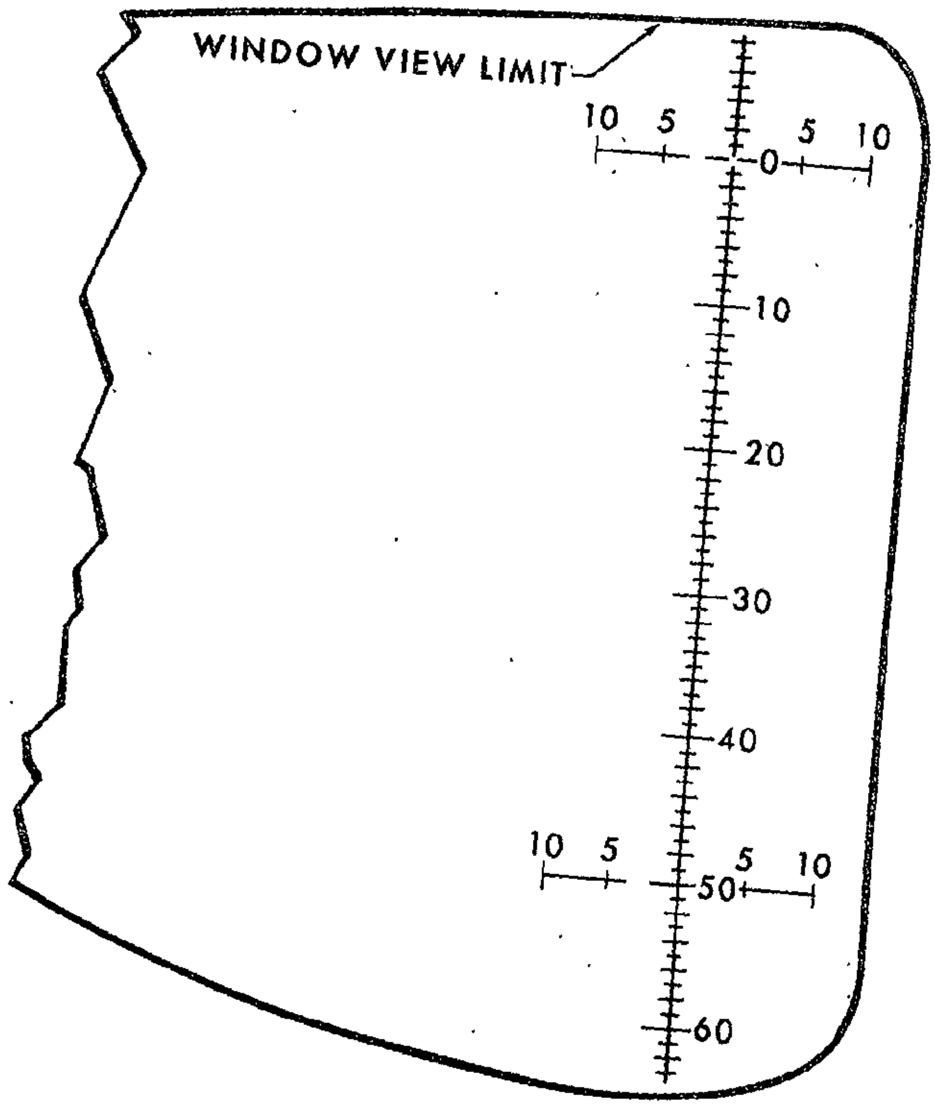


Figure 30. Marking Details of Landing Point Designator Grid

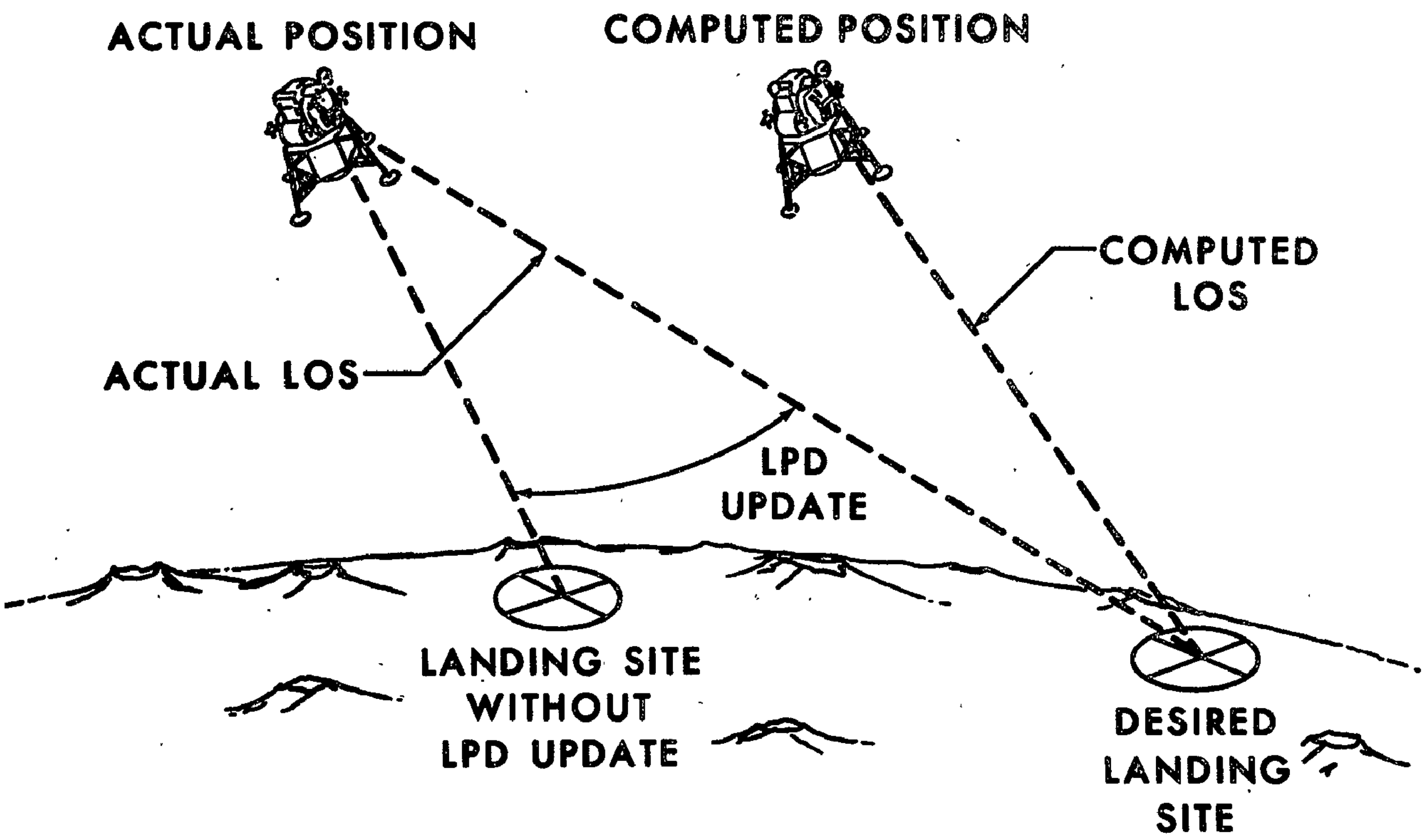


Figure 31. Landing Point Designation

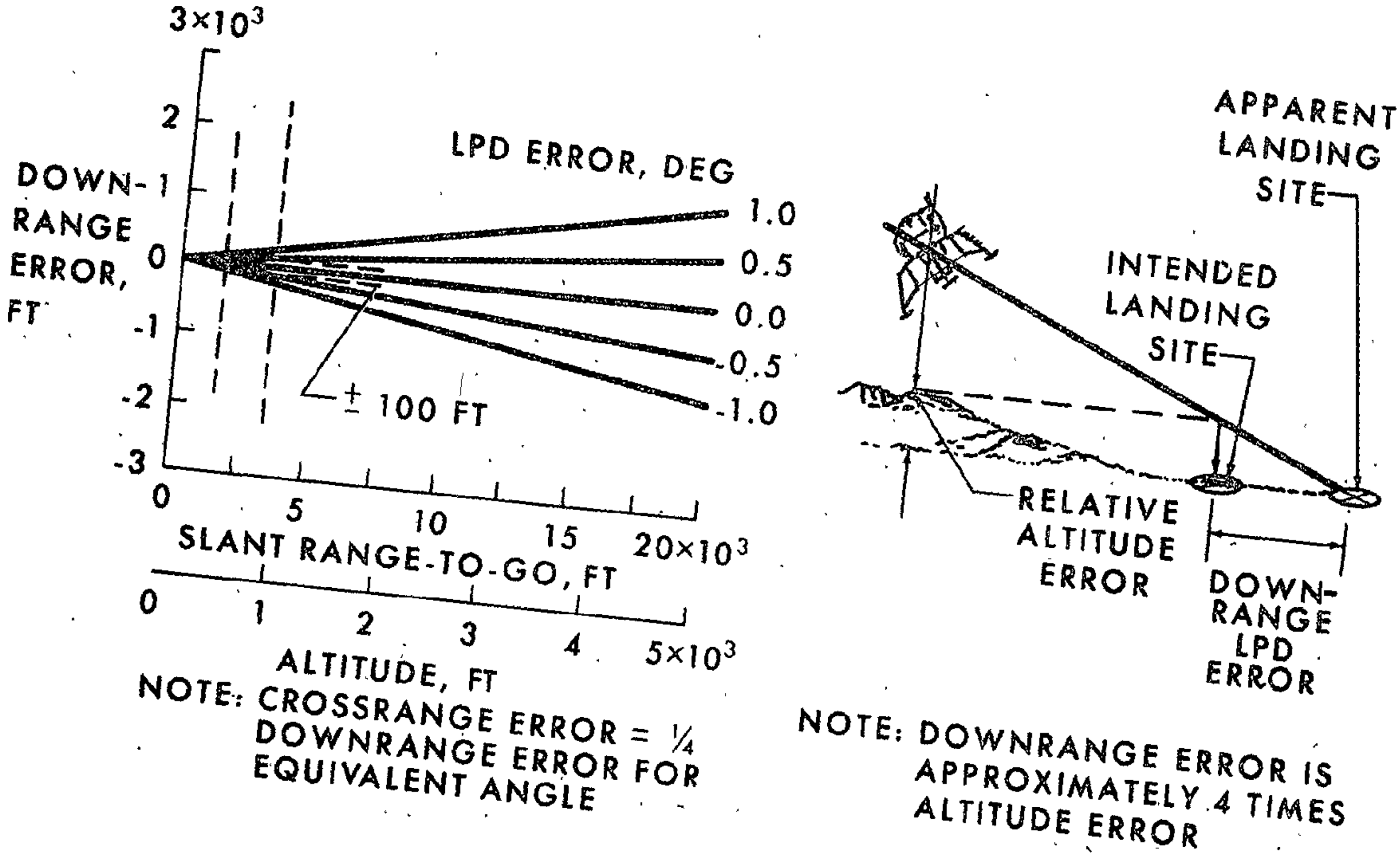


Figure 32. Landing Point Designator Error Sources
Flight Path Angle = 14°

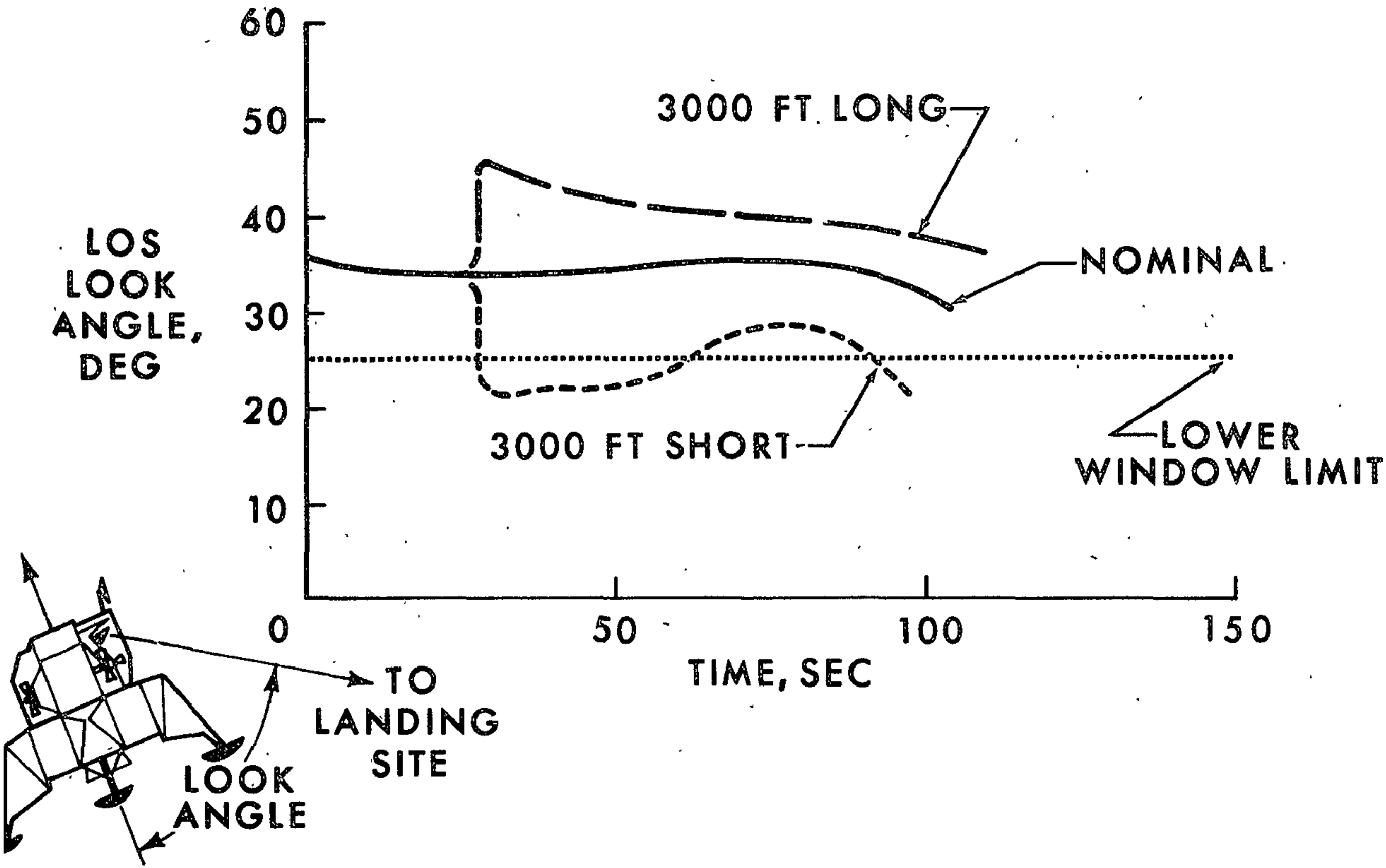


Figure 33. Time Histories of Line of Sight to Landing Point for Alternate Site Selections at 5000 Ft. Altitude

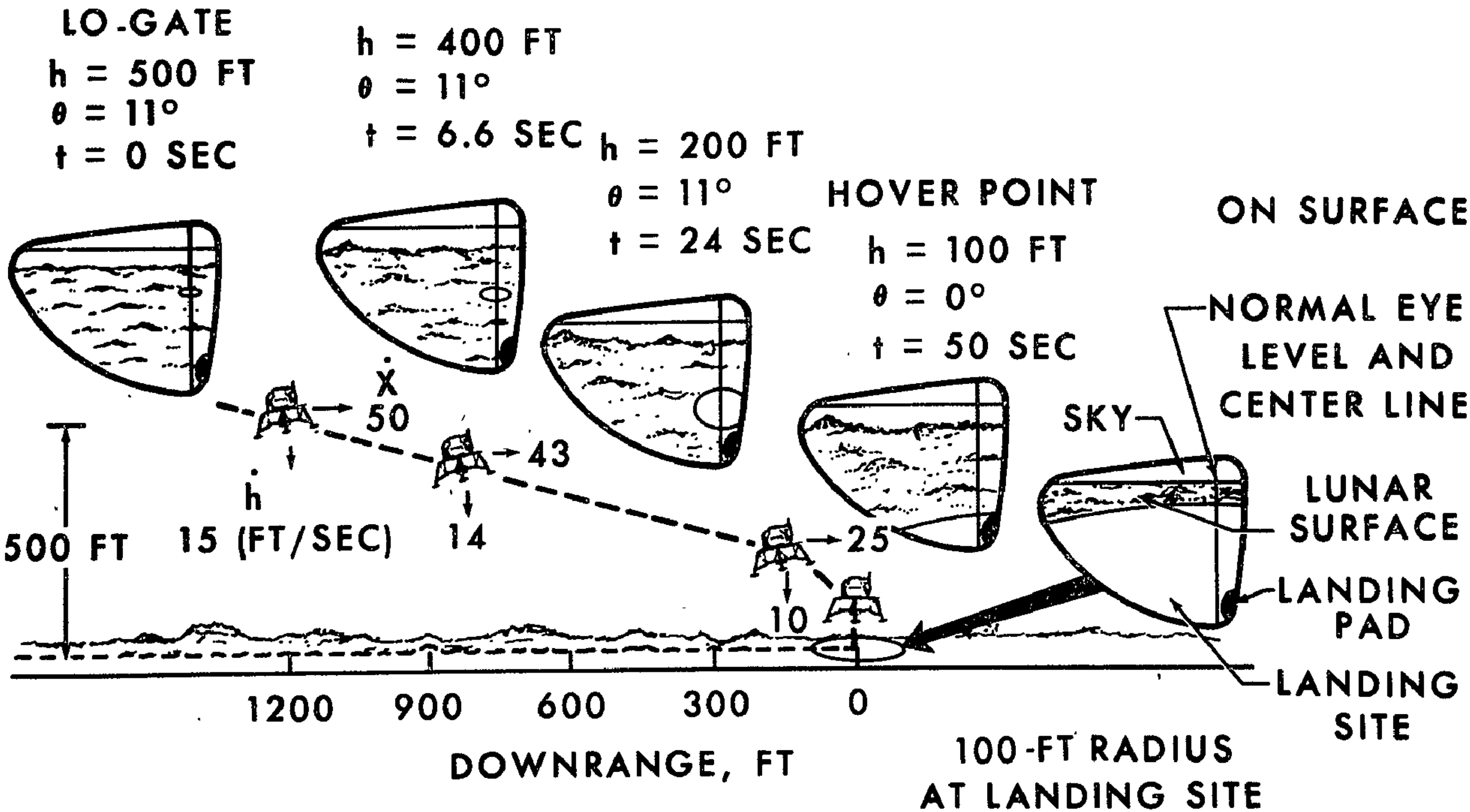


Figure 34. Pilot View During Landing Phase

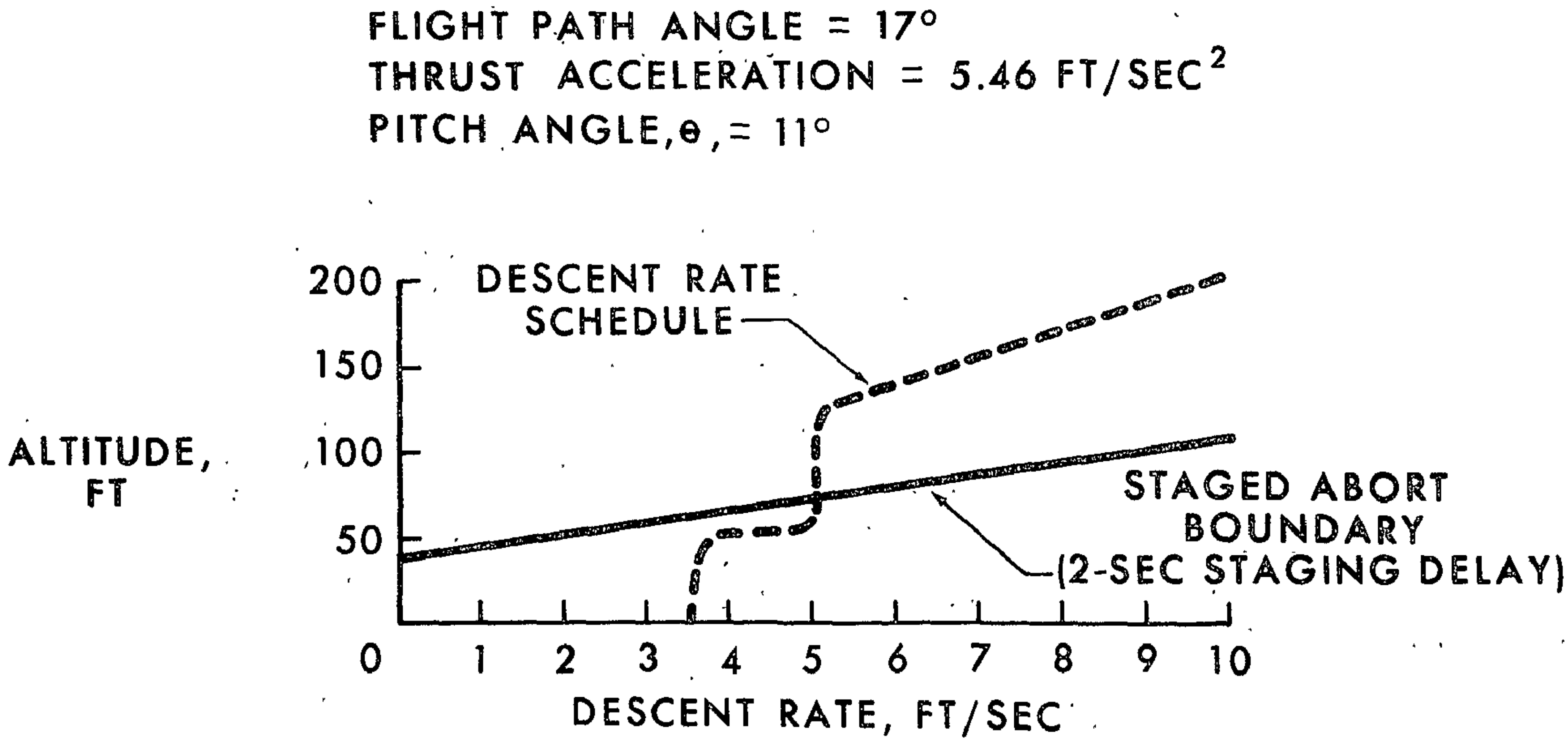


Figure 35. Trajectory Characteristics for Landing Phase

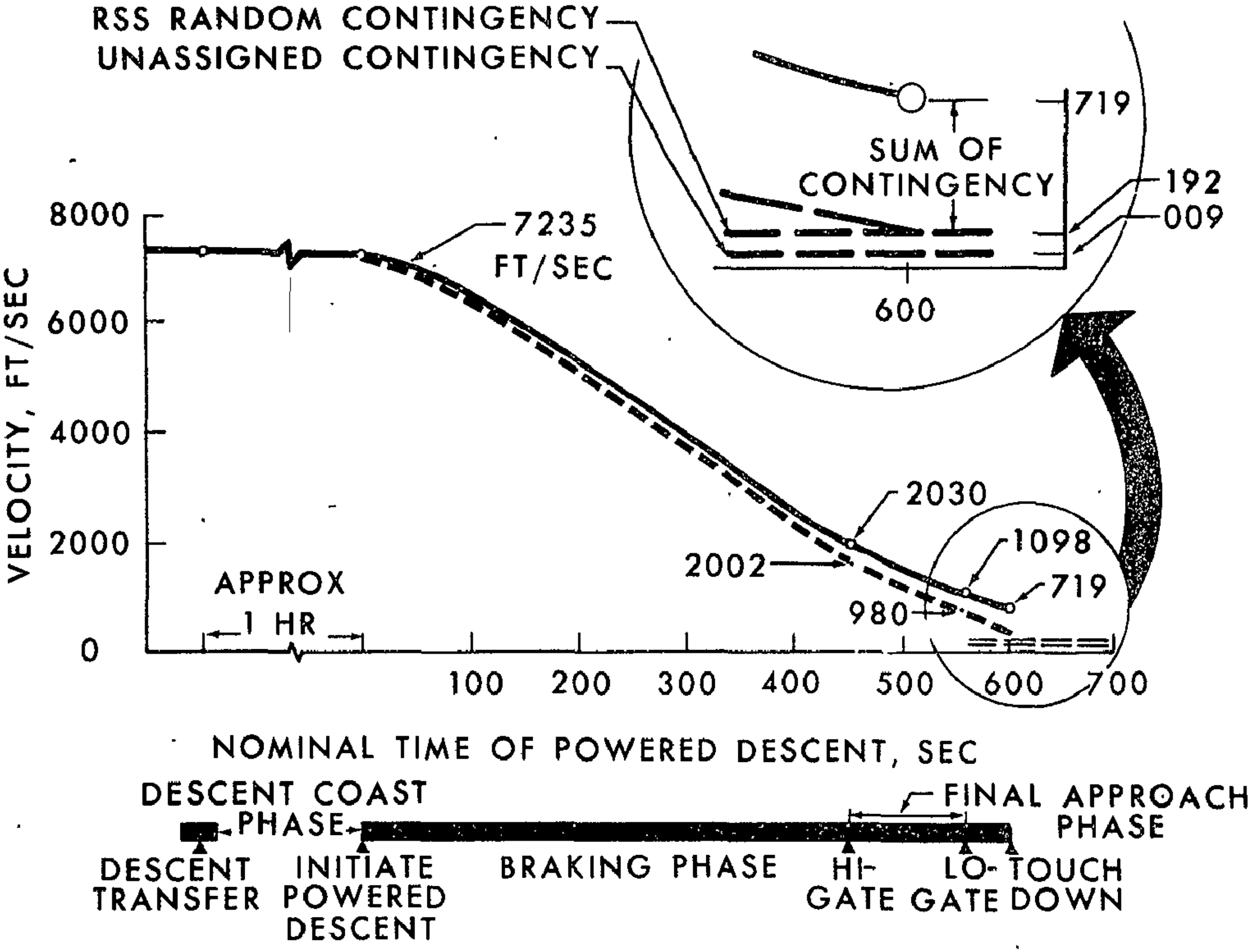
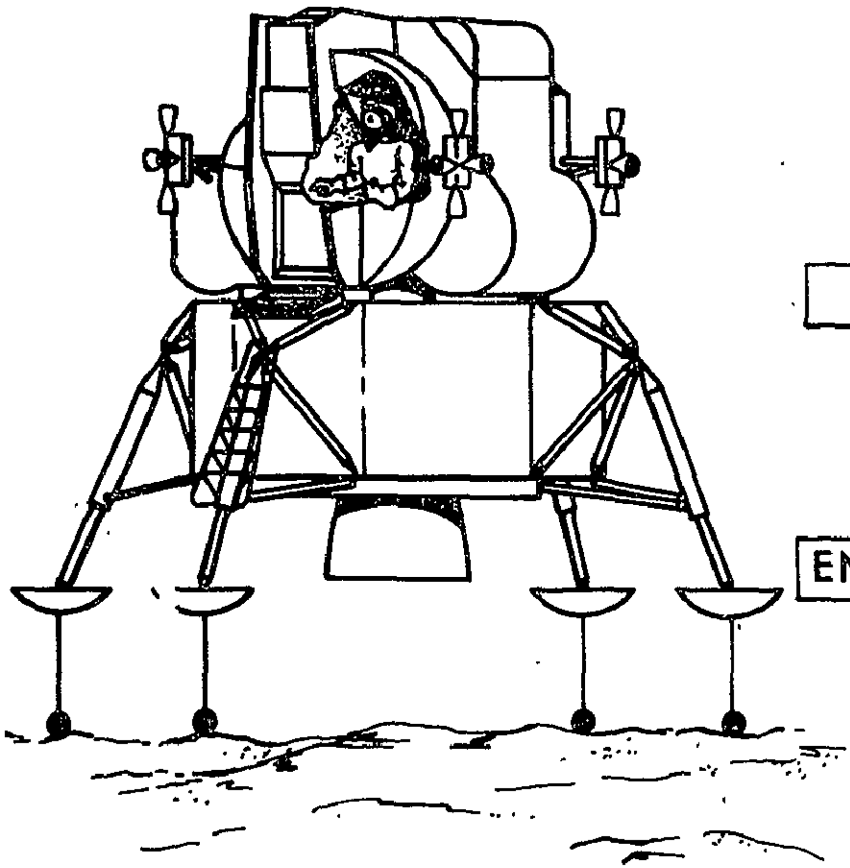


Figure 36. Time History of LM Descent Characteristic Velocity Expenditure



PROBE CONTACT

↓

LIGHT ON

↓

PILOT ACTIVATES DE CUT-OFF $.2 < t < .66$

↓

DECA RESPONSE 0.10 SECONDS

↓

ENGINE FUEL VALVES CLOSE 0.15 SECONDS

↓

ENGINE TAIL OFF 0.12 SECONDS

↓

ENGINE OFF

↓

TOUCHDOWN

Figure 37. Descent Engine Shutdown Sequence

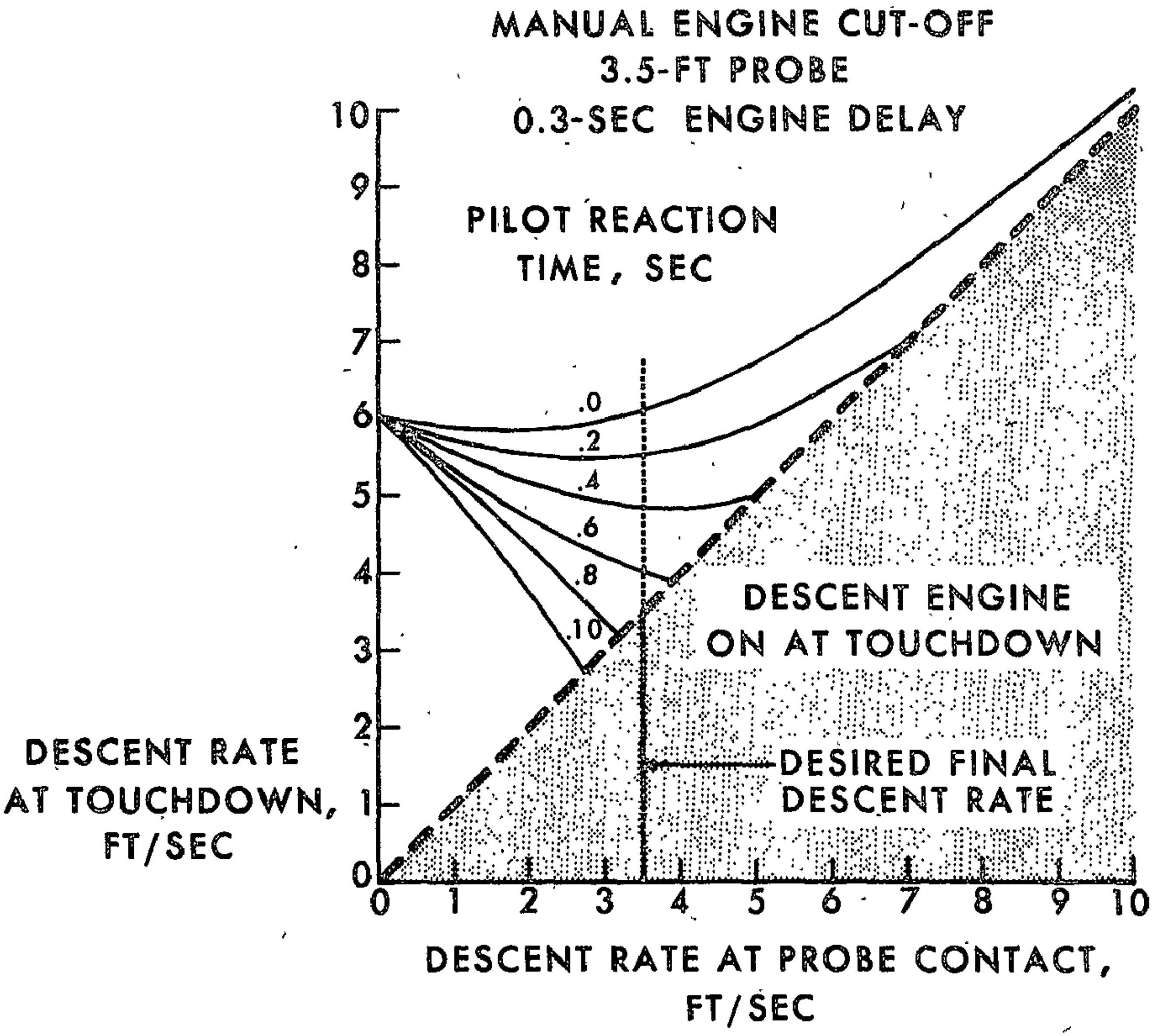


Figure 38. Landing Descent Velocity Control Using Probe for Descent Engine Cut-off Signal

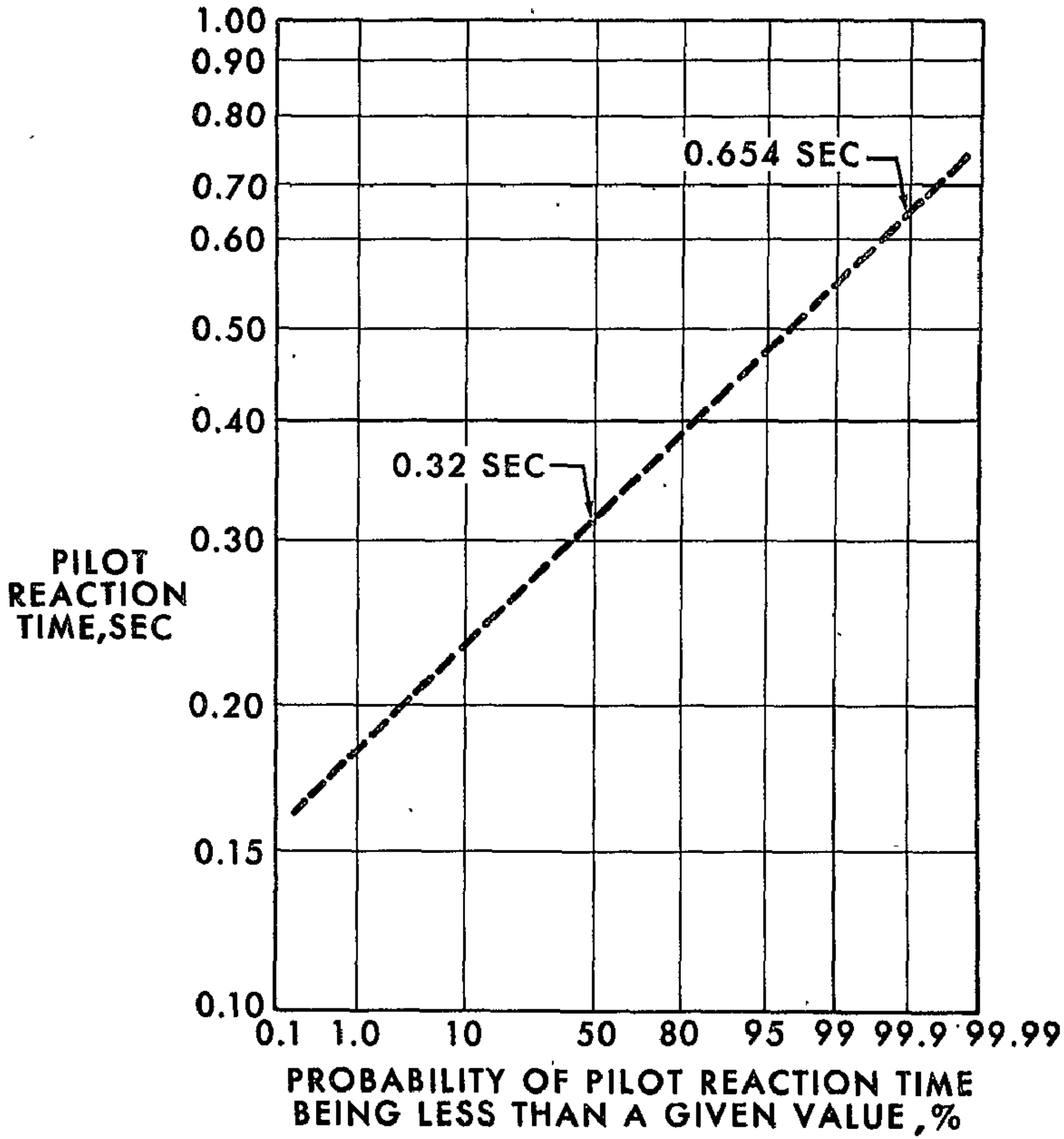


Figure 39. LM Descent Engine Cut-off Pilot Reaction Time

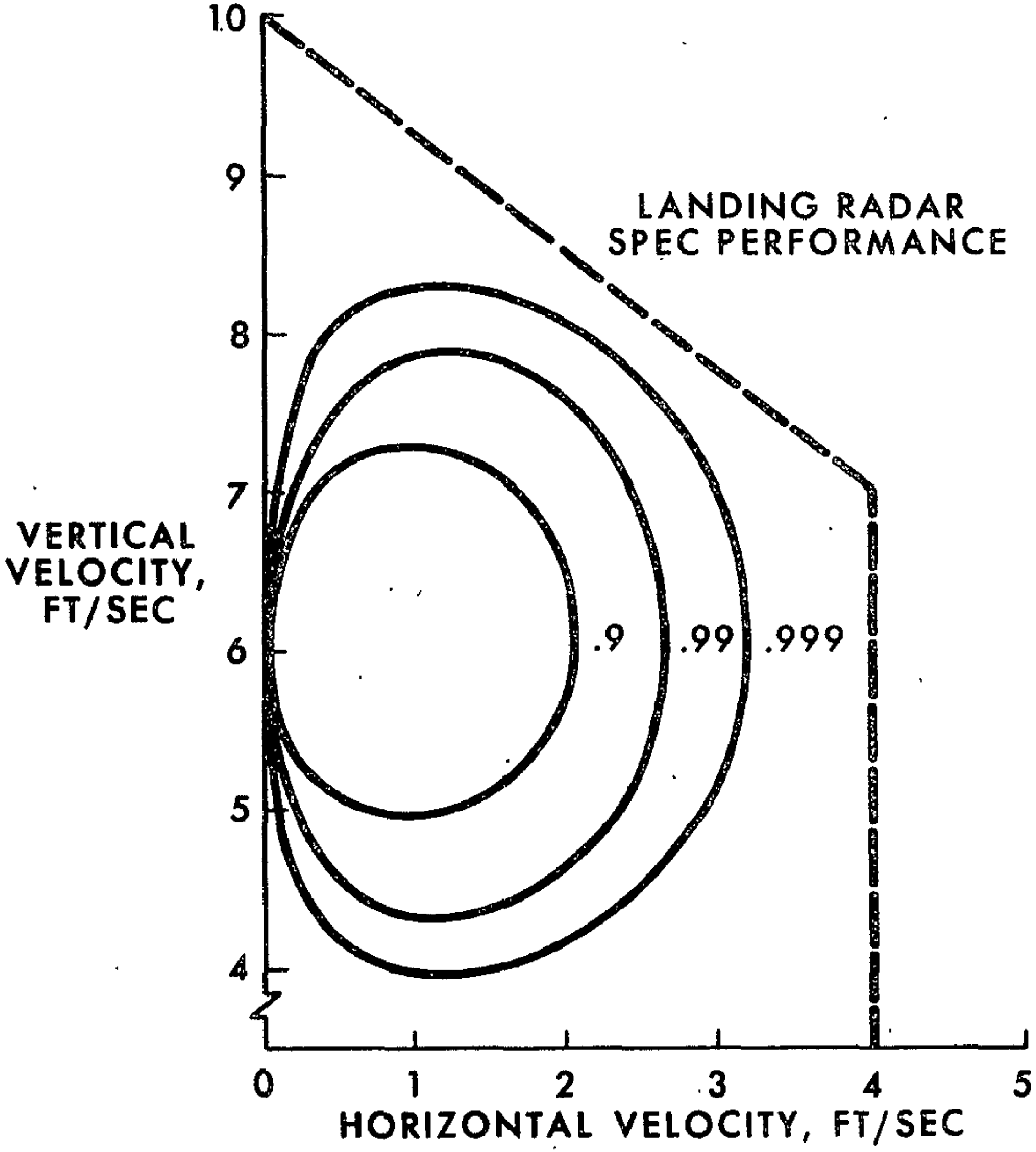


Figure 40. Manual Control of Landing Velocities With System Errors

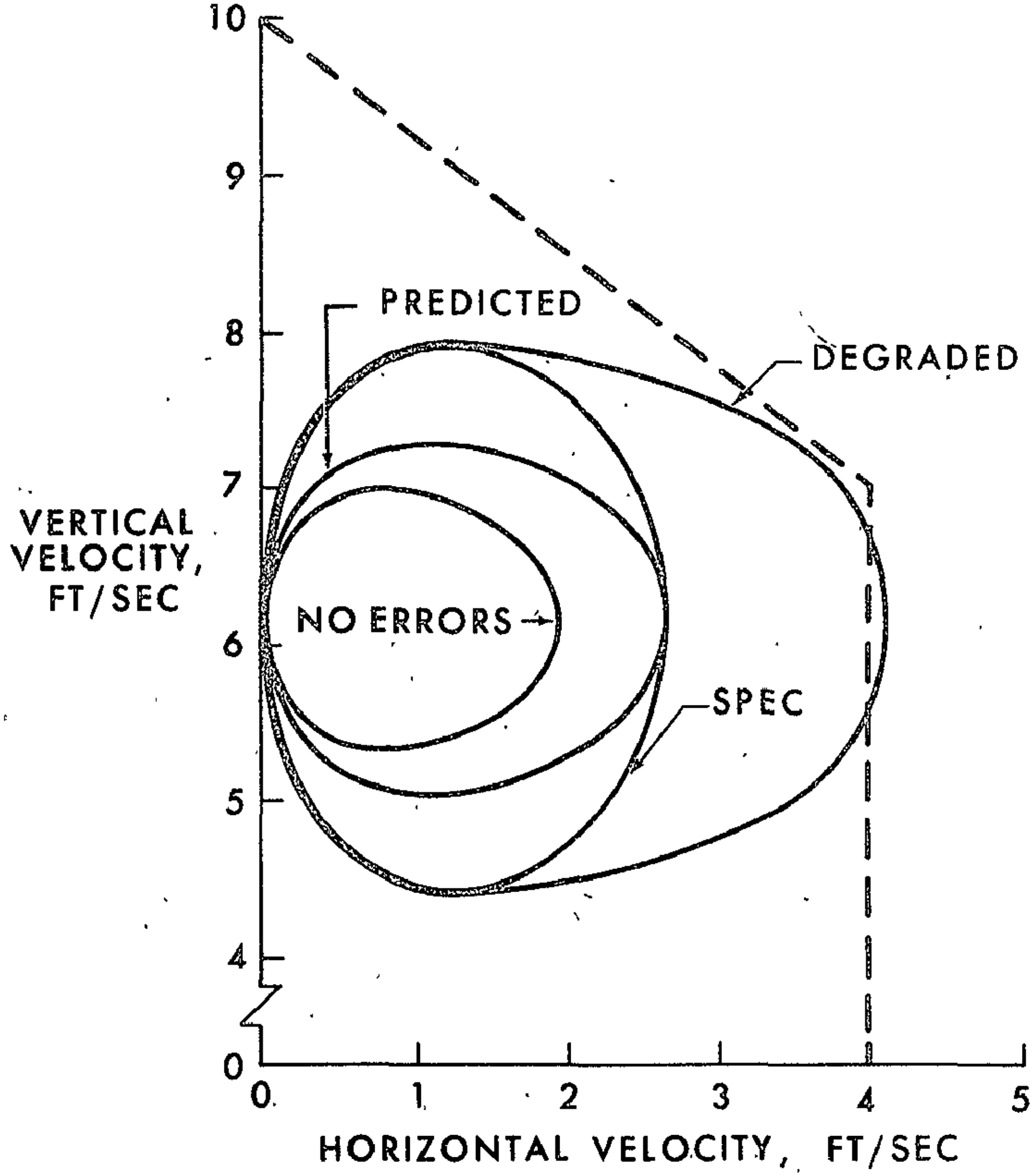


Figure 41. Effect of Landing Radar Errors On Velocities In Manual Control of Landing

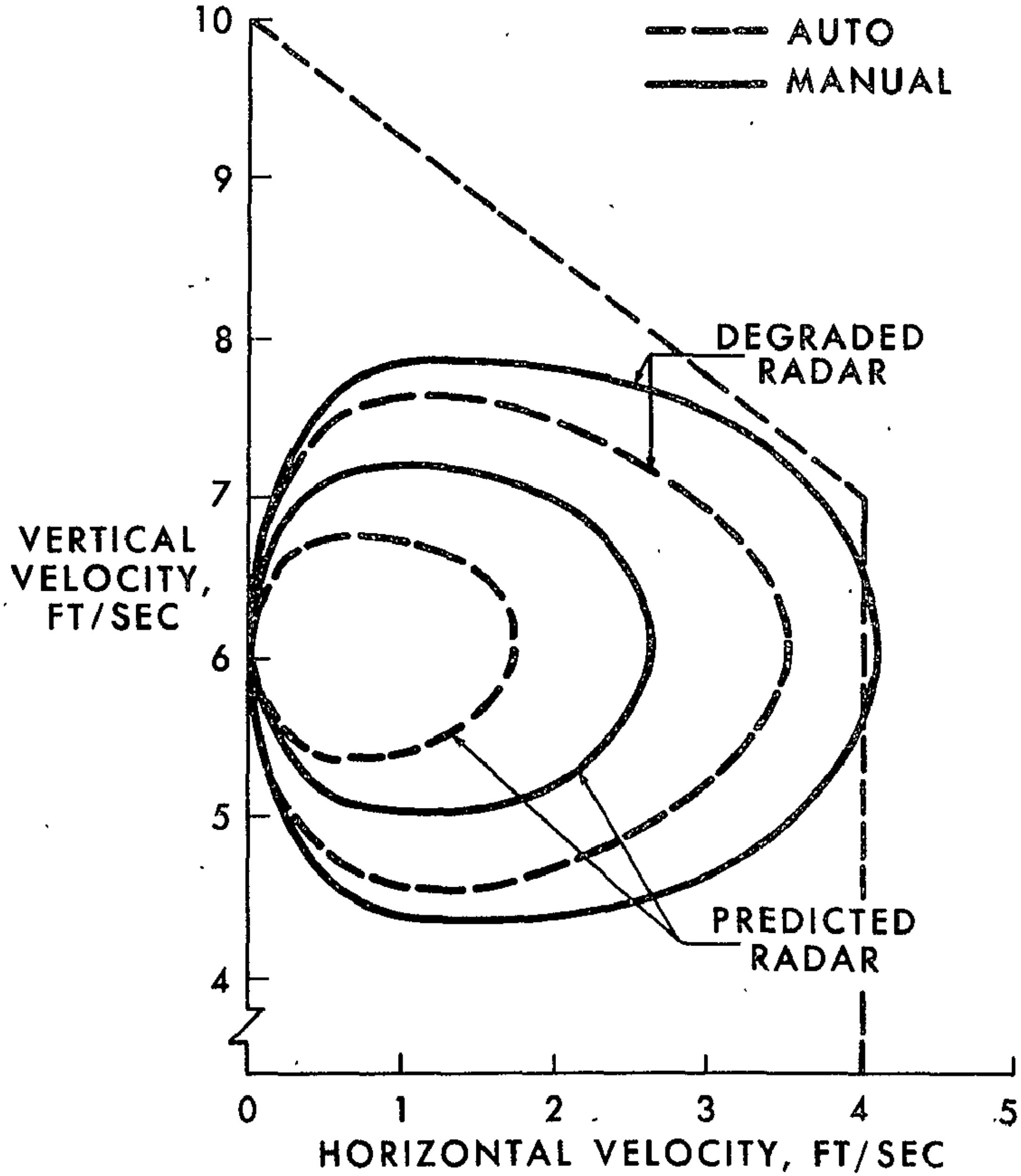


Figure 42. Comparison of Automatic and Manual Control of Velocities Landing

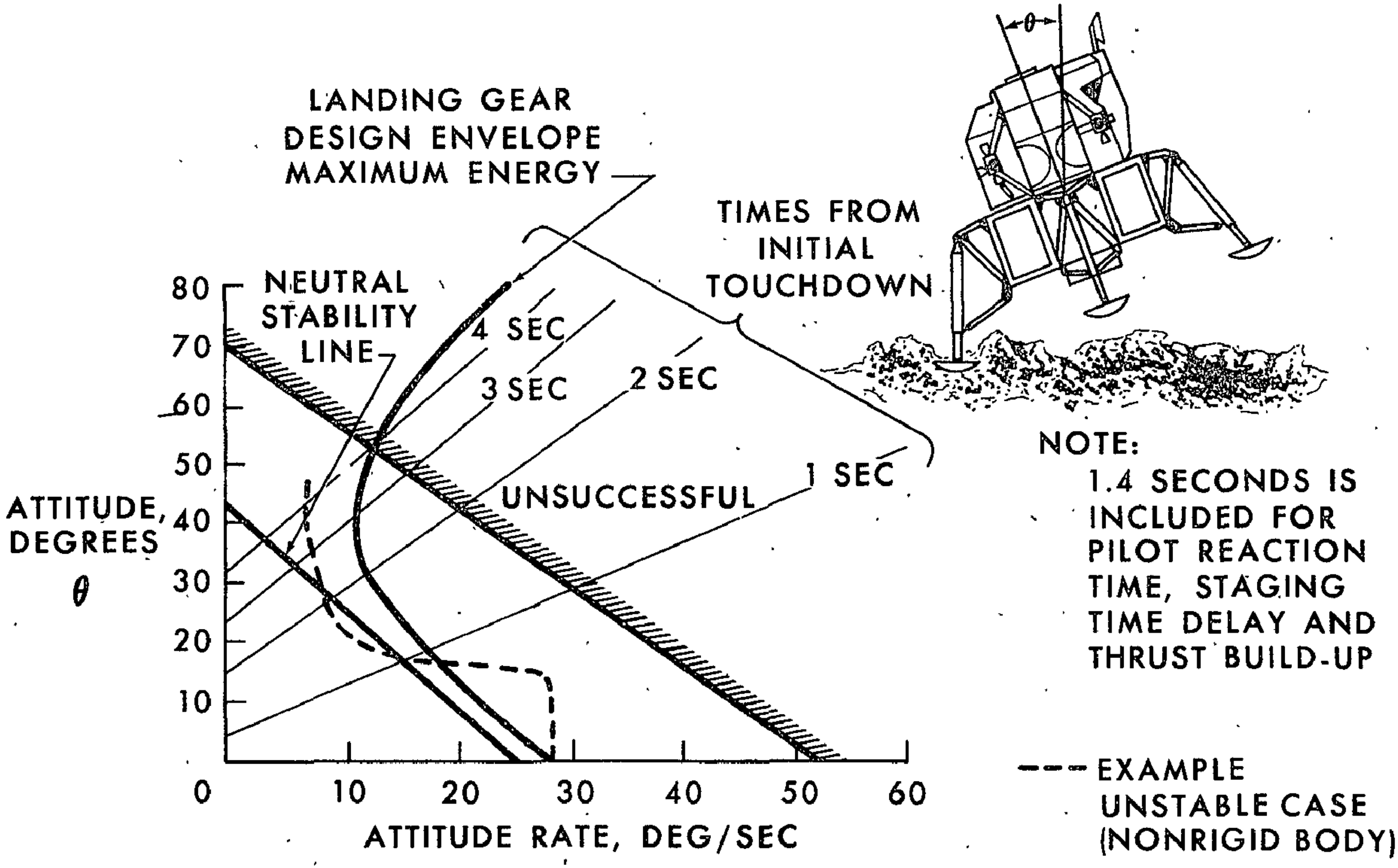


Figure 43. Boundary of Acceptable Angles and Angular Rates for Tilt-over Abort Initiate