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SOME ABORT TECHNIQUES AND PROCEDURES FOR MANNED SPACECRAFT

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Introduction

This paper is concerned with our present state of knowledge on the procedures and effectiveness of aborts from manned orbital and lunar missions. By abort is meant a deliberate or unintentional termination of the primary mission followed by an expedient return to earth of the payload (spacecraft and crew). A deliberate abort may be initiated if a critical component failed and thus impaired the probability of mission success or the crew's safety. Examples of this kind of abort are failure of the launch guidance system or the oxygen and cabin pressure system. Also, an unintentional abort may occur during the launch phase due to the complete or partial failure of one of the launch-vehicle stages to operate. For each mission every conceivable set of circumstances must be considered and evaluated to see if an abort of the mission is called for and to determine what sequence of events should occur to bring the spacecraft and crew safely back to earth with the highest probability of success. Factors which must be considered are: how to specify and sense an abort condition, how much fuel should be put onboard the vehicle for abort purposes, what size and kind of rocket engines are best suited for each mission, and what compromise is incurred due to other mission requirements? If an abort occurs, the optimum time of application as well as the magnitude and direction of the velocity changes must be determined beforehand and the crew must be clearly informed of what action is required or what the situation is so that they can decide the proper action.

It follows then, that mission abort considerations are not simply of academic interest, nor do they play a minor role in the overall requirements of the vehicle design. For example, in the Mercury program the mission analysis group spends four times as much effort on planning for abnormal missions as they do for the normal missions. From their experience we know that in the atmosphere the most critical factor is the lateral and negative accelerations that may occur if the spacecraft tumbles during abort at high dynamic pressure. Once outside the atmosphere, the reentry decelerations may be high but usually are not critical, and the more important consideration will be in landing the vehicle in a specified recovery area. Regardless of what happens, both the spacecraft and pilot must be conditioned to survive a wide spectrum of emergency conditions. While the people who obtain funding for space projects must optimistically point out what can be achieved if the mission succeeds, a group of pessimists must consider what to do to save the crew and the spacecraft if the mission fails.

In this paper some of the abort considerations applicable to manned missions such as Apollo and Gemini will be discussed with occasional reference made to the techniques used in the present Mercury program. It should be clearly understood, however, that the information to be presented represents research data obtained by many groups and does not necessarily represent any final decision on the abort techniques that may be used in these programs. A number of published reports on the subject of extra-atmospheric aborts are listed at the end of the paper[1 to 11]. Many other papers not suitable for general publication (such as Mercury and Apollo Working Papers) have provided background material. While only a review of the subject is given herein, most of the details can be obtained from the referenced papers.

Atmospheric Abort

The atmospheric phase of launch will be treated only briefly since a great deal has already been written and said about procedures for the abort of a manned spacecraft at the two most critical conditions within the atmosphere: Off-the-pad and at high dynamic pressures. For the purpose of abort at these conditions, the Mercury spacecraft uses a set of tower-mounted solid rockets. The tower and rockets went through an extensive series of tests early in the development program and are now considered one of the most reliable systems connected with the Mercury spacecraft. However, two important changes have been considered for improving this system.

The question often arises as to whether an escape tower with rockets that pull the spacecraft off the launch-vehicle is the best way to do the job, or whether pusher-type rockets should be used. After investigating the various alternatives to the escape tower, the conclusion to date is that no better alternative exists for the contemplated Apollo configuration. However, the Gemini spacecraft will probably not use the escape tower but will use ejection seats for the crew in the event of an abort at altitudes below 20,000 to 40,000 feet.

One aspect of the present abort tower, however, will probably be improved. An illustration of this aspect is the case in which the launch-vehicle and spacecraft have reached a launch velocity of 2,000 or 3,000 ft/sec and an abort is required. The fuel flow to the launch-vehicle is shut off and by using the tower-mounted rockets, the spacecraft is separated from the launch-vehicle with an initial separation velocity of say 400 ft/sec. Immediately following separation then, the spacecraft is subjected to a dynamic pressure somewhat greater than that of the launch-vehicle while the weight or inertia of the launch-vehicle is about 100 times greater than that of the spacecraft. Even with a tumbling launch-vehicle, this difference in inertia and dynamic pressure would be enough to cause the launch-vehicle to overtake the spacecraft in 5 to 10 seconds. For this reason the "thrust line" of the escape rockets does not pass through the center of gravity (eccentricity), and thus a pitching moment is produced and the spacecraft is given a component of velocity away from the trajectory of the launch-vehicle.

The present shortcoming with this arrangement is illustrated in figure 1. If the launch-vehicle is tumbling, then the pitching moment applied by the separation rocket ($\dot{\theta}_c$ of fig. 1) may be completely or partially cancelled by the pitching moment of the launch-vehicle—spacecraft combination $\dot{\theta}_b$ and the result will be a subsequent collision. On the other hand, if the spacecraft were able to sense the direction of rotation (pitching and yawing velocity) at the time of abort, and the rotational velocity imparted by the separation rocket were applied in approximately the same direction $\dot{\theta}_b + \dot{\theta}_c$, then sufficient clearance would be provided. This type of control would be a refinement to the atmospheric abort system and studies of such a system are now being made.

Suborbital Abort (Extra-Atmospheric)

Decelerations during entry.- One aspect of the Mercury launch program which may not be too well known is the fact that if an abort occurs at a launch velocity of 14,000 to 16,000 ft/sec, the spacecraft, without lift capability, will undergo as much as a 16g reentry. Although 16g is not critical for reentry from a launch abort, it may be of interest to examine how even lower values can be obtained.

An indication of the maximum entry decelerations following an extra-atmospheric abort is shown in figure 2. This figure, taken from[1] with some details deleted, shows the maximum deceleration that will be reached for a nonlifting entry when the trajectory conditions are measured at an altitude of 394,000 feet (120 km). Although the figure does not cover a sufficiently wide range of flight-path angles and applies only to nonlifting entries, it does illustrate the sensitivity of deceleration to flight-path angle over a wide range of velocities. (Similar curves are also given in[2] for flight-path angles up to 20° based on a reference altitude of 300,000 feet and for values of L/D from 0 to 2.) For most orbital or lunar launch trajectories, the critical area of this figure lies between velocities of 14,000 and 18,000 ft/sec for it is here that the combination of state variables (altitude, velocity, and flight-path angle) usually leads to the highest decelerations on entry following an abort. Exact values of deceleration will vary with different launch profiles, but for manned missions this is not a desirable region to undergo launch-vehicle staging if it can be avoided. Some considerations of problems of this type are discussed in[3,4].

Optimum time and direction for corrective thrust.- If an abort does occur outside of the atmosphere at suborbital velocities and the spacecraft is equipped with a rocket engine and fuel, then the decelerations at entry may be reduced by proper use of the fuel available. The proper time and direction for the thrusting maneuver can be illustrated with the aid of figures 3 to 5 taken from[5].

In figure 3 is shown a typical variation of altitude with time for a spacecraft following a suborbital abort. For this example, the spacecraft can produce a lift-to-drag ratio L/D of 0.5 once inside the atmosphere and, in addition, is equipped with a rocket engine capable of thrusting for 11.05 seconds and producing a velocity increment of 3,000 ft/sec. This ΔV may be applied in any direction, and at any time along the coast trajectory up until the time of entry. Seven possible positions are indicated on the figure where the rocket could be ignited. The last position coincides with the initial buildup in deceleration, which occurs at a relative low altitude due to the large flight-path angle (-20°) and low velocity (14,000 ft/sec) at entry.

The effect on the entry decelerations of applying the thrust at these various positions is shown on figure 4. On the left of figure 4 is the maximum deceleration measured during the entry plotted against the direction of thrust application for four of the seven positions considered in figure 3. The curves for positions 4, 5, and 7 lie in between those shown for positions 3 and 6 and have been left out for simplicity. The data show that the decelerations are minimized if the thrust is started at position 6 and applied at an angle of about 100° to the initial velocity vector. The results also show that applying the velocity change at the time of abort (position 1) is the least effective of the cases considered.

On the right of figure 4 a formula is given for the impulsive velocity change ΔV producing the maximum change in flight-path angle $\Delta\gamma$. The values of ζ obtained by this formula are shown with ticks on the curves of figure 4 and, as may be seen, give a good representation for the minimums of these curves.

The data of figure 4 indicate that the optimum time to apply this thrust is just prior to the buildup of dynamic pressure during the entry. However, sufficient time must be allowed for (1) the rocket to finish thrusting, (2) separating the rocket from the spacecraft (if required), and (3) to reorient the spacecraft for entry prior to an excessive buildup of dynamic pressure with the associated high rates of heating and deceleration. Therefore, it is necessary to know, among other things, how sensitive is this "optimum" condition and how do these results change with various sizes of rocket engines. In order to answer these questions, the data from figure 4 have been plotted in figure 5 showing maximum entry deceleration as a function of dynamic pressure at the time of rocket ignition. Figure 5 also shows similar data for several other thrust levels. In all cases the total velocity change was 3,000 ft/sec, the initial vehicle weight was 7,000 pounds, and an L/D of 0.5 (or 0) was used during the atmospheric pullout. Since the lower thrust levels require the longer burning times, the data indicate that low-thrust engines must be started at a higher altitude than the higher-thrust engine. The exact values will, of course, change somewhat with different entry conditions.

Vehicle attitude control.- If a set of assumptions are made as to the shape, etc., of the manned spacecraft, then a sequence or procedure for reorientation can be postulated for aborts at suborbital velocities, and a study can be made of some of the factors involved in the use of a rocket for maneuvering prior to entry.

Such a study was recently performed at the NASA Langley Research Center and will be reported in more detail in[6]. For this study it was assumed that:

1. The manned vehicle was a blunt-faced spacecraft.

2. The rocket engine discussed in the preceding section was located on the face of the heat shield of the spacecraft similar to the retrorocket now used on the Mercury spacecraft.

3. The spacecraft center of gravity was offset so that, in the atmosphere, the spacecraft would trim at an angle α of 33° with respect to the relative wind. The resulting aerodynamic force would produce a ratio of lift-to-drag L/D of 0.5. The vehicle would be rolled to direct this lifting force in the proper direction for both side-force and for "negative" lift.

An instrument used in this study to describe the motions of the spacecraft to the pilot is shown in figure 6. The instrument is simply a two-needle synchro with an image of the spacecraft on one needle and a "velocity vector" on the other needle. This instrument gives information on the angle of attack α, the flight-path angle γ, and the angle of the spacecraft with respect to the local horizon θ and covers a range of 360° with a read-out accuracy of about 1° or 2°. Such an instrument supplements the normal flight instruments and is generally referred to as a "situation display." It is indicative of the feeling of many researchers in this field that new situations often require new instruments to describe adequately the flight conditions to the pilot.

A typical sequence of maneuvers following an abort at suborbital velocities is illustrated in figure 7. A typical trajectory profile is shown in the upper portion of the figure with the orientation of the capsule shown at several key positions on the trajectory. In the lower portion of the figure the pilot's display at these same positions is shown. If the pilot and crew are in an upright position* with respect to local horizons at the time of abort, then a positive pitch maneuver of about 100° is required in order to fire their rocket in the optimum direction. The pilot will normally have several minutes in which to make this maneuver while coasting well above the earth's atmosphere. During this time, the vehicle should also be rolled 180° (the sequence of pitch + roll or roll + pitch is optional) so that, following burnout of the maneuver rocket, only a pitching maneuver will be required to orient the spacecraft in the proper attitude for entry. This final pitching maneuver requires an angular change of about 67° (from an angle of attack of 100° to an angle of 33°) and may have to be done very rapidly. Since the available thrust is most effectively used just prior to entering the atmosphere, the elapsed time between the end of the thrust and the beginning of the high atmospheric heating rates may have to be kept necessarily small.

*During the boost phase and under "zero-gravity" conditions, there is nothing unique about an "upright" position. On the other hand, there does not appear to be any overpowering reason to change the position of the crew to any other position either.

The minimum time required to make these maneuvers is primarily a function of the control power of the spacecraft's attitude control system. In the simulation program of[6], it was found that when pilots were presented with the aforementioned reorientation problem, they used the maximum pitch rate capability of the spacecraft even when the spacecraft's automatic damping system was inoperative. A typical time history of one of these maneuvers is shown in figure 8. In the case shown, the spacecraft had a rate command system with a maximum pitching rate of 6.2 deg/sec. Similar results were also obtained with maximum rates of 12.4 and 18.6 deg/sec. In the example shown in figure 8, the heating rates would have already become extremely high by the time the maneuver was completed. However, in simulation programs, it is usually necessary to give the pilot an incentive for high performance. For this reason, the pilot was forced (in the simulation) to perform the maneuver very close to the atmosphere. If he failed to complete the pitch maneuver before the buildup in dynamic pressure, the spacecraft became aerodynamically unstable and uncontrollable. In actual flights the high heating rates would precede this condition of instability and would undoubtedly provide the same incentive.

Range after suborbital abort.- Exact values for the range traveled by the vehicle following the abort will depend on the launch trajectory and characteristics of the spacecraft L/D and W/C_DS. However, some results on a blunt-faced spacecraft with L/D of 0.5 are indicative of the general trends that can be expected. (See[6,7].) Figure 9 shows a typical launch trajectory with the trajectories that would be followed if aborts had occurred when the launch velocity was 12, 15, 18, 21, and 24 thousand feet per second. Associated with each abort trajectory, there is a footprint of available landing sites which can be reached by varying the L/D and the corrective thrust ΔV prior to entry. Only one such footprint is illustrated in figure 9. These footprints can be characterized by shape, width, length, and the down-range distance from the launch site to some reference point on the footprint. Taking these characteristics in their given order, some range and cross-range data are given in figures 10, 11, and 12. In figure 10 is shown the general shape of the available footprint obtained by a pilot using an $L/D = 0.5$ and restrained to not exceed entry decelerations of 10g. The difference between the maximum and minimum available longitudinal range L determines the length, the maximum lateral range B determines the width, and the longitudinal distance traveled while achieving maximum lateral range A determines the reference center of the footprint.

In the cases studied, corrective velocity changes of 0, 1,500, and 3,000 ft/sec were applied in the direction giving a maximum reduction in entry angles (see fig. 4) just prior to entry. It was found that the ratios of A/L and B/L were not affected by the thrust maneuver. Furthermore, the ratio of A/L remained constant at 0.67 ± 0.02 regardless of abort velocity and the ratio of B/L varied as shown in figure 10.

In figure 11, the variation of L with the velocity at abort and with ΔV is shown. Since each degree represents about 60 nautical miles, it can be seen that at 18,000 ft/sec the maximum footprint is 240 nautical miles long L and ± 48 nautical miles wide B if a ΔV of 3,000 ft/sec is used and only 1/2 that size if no ΔV is used.

In figure 12, the maximum down-range distance obtained is plotted on a semilog scale against the velocity at abort. The curves with symbols are distances measured from the position at abort while the dashed curve gives the distance from the launch site to the position at abort. It can be seen that up to 24,000 ft/sec, we are concerned with recovery sites on the order of 80° to 90° down range from the launch site.

One useful application of these data could be made by plotting these footprints along the nominal ground track of the vehicle during launch to determine the desirable location of recovery sites for a lifting vehicle. A further study is also necessary to determine how much ΔV should be used by a pilot from range considerations since, as the figures show, both the entry decelerations and the range will be affected by the onboard propulsion. Of the two, range considerations and the simplicity of procedures will probably predominate over considerations of reducing the decelerations on entry.

It may be of interest to note that in the operational procedures for the Mercury spacecraft the retrorocket normally used for deorbit can also be used at suborbital abort to shorten the range. However, this rocket produces a velocity change of only 450 ft/sec and procedures call for its use in a retrograde direction from 30 to 250 seconds after abort. The proposed larger velocity capability of Gemini and Apollo offer considerable more choice in the field of range control. Complexity is, however, the price to be paid for this freedom of choice.

Superorbital Abort

General.- Probably the most difficult region in which to establish an ironclad abort procedure is the region between orbital and escape velocity. Depending upon the amount of fuel carried and the time required to use that fuel, an immediate return to earth can be executed only up to a certain point during the launch; beyond that point, an immediate return cannot be made and the procedure for using the available fuel becomes entirely different. Once superorbital velocities have been reached, the centrifugal acceleration due to velocity will exceed the acceleration due to gravity and the vehicle will move away from earth on a Keplarian ellipse even though the launch-vehicle is shut down. Because of this fact, any increase in velocity, altitude, flight-path angle, or time will also increase the velocity change required to deflect the vehicle's trajectory so that it intersects the atmosphere before apogee is reached. If sufficient fuel to accomplish this velocity change is not available, then the vehicle must continue to apogee, using the available fuel along the way to change the orbit so that perigee lies within the earth's atmosphere. These considerations are necessary both for direct launches and launches from orbit and we will not distinguish between the two in this paper. Some of these factors have been covered in[8] for impulse thrust applications. Also, a very comprehensive parametric study giving the impulsive thrust requirements for immediate return to earth from superorbital aborts is given in[2]. However, to the impulsive velocity requirements of[2,8] must be added the additional velocity cost due to the time required to reorient the vehicle and to apply the thrust. We will consider only these factors here.

Vehicle attitude control.- In figure 13, a typical reorientation procedure designed from an immediate atmospheric entry following an abort at superorbital velocities is illustrated. If the pilot is in an upright position at the time of abort, the sequence of figure 13 can be described as essentially that of a forward somersault. At the signal for an abort, the pilot (or automatic control sequencer) has several choices. If the abort is not caused by the launch-vehicle and the pilot can override the launch-vehicle control system, then he may use the launch-vehicle to change his trajectory back toward the atmosphere before separation. If the abort is caused by a failure of the launch-vehicle control system, a launch-vehicle thrust failure, or by an impending explosion, immediate separation must be made using an abort or separation rocket. If sufficient fuel is available for an immediate return to earth, the sequences shown in figure 13 are applicable. Upon separation, the pilot should immediately pitch down about 96° $\left(\text{according to the equation } \zeta = 180^{\circ} - \cos^{-1}\frac{\Delta V}{V_o}\right)$ and fire the maneuver rocket. This procedure will produce a maximum negative change in flight-path angle for the given available ΔV. Following the firing of the maneuver rocket the pilot will probably continue the pitch-down maneuver to an angle of attack of -33° in preparation for the atmospheric entry. This attitude will produce a negative L/D of 0.5 and should hold the spacecraft in the atmosphere until near orbital velocities are reached, at which time a roll-out of 90° to 180° can be made. Since the ΔV requirement for immediate earth return changes vary rapidly during the superorbital phase of the boost (see[8]), an excess of velocity may be available to the pilot and if this is the case then positive lift may be required at entry. This situation is similar to the undershoot and overshoot boundaries as described by Chapman[12] and others, and care must be taken at abort to stay within the accepted corridor for supercircular entries.

Pilot response times.- The maneuvers just described have been examined in[6] from a pilot response viewpoint, and a typical time history is shown in figure 14. A critical case was chosen in which atmospheric capture depended upon the pilot's response time following the abort signal. Allowing for the 3-second burning time of the separation rocket, the time τ required to hit the separation switch, reorient, and fire the maneuver rocket was evaluated for a number of test pilots. The results are shown in figure 15 along with results obtained from the suborbital aborts of figure 8. The data are presented as the probability of the pilot exceeding a given time τ for the maneuver where τ is given on the abscissa. The critical values of τ for the two flight conditions considered are given on the figure.

An indication of the additional velocity requirement imposed by the time to reorient and fire the maneuver rocket is given by figure 16. Taken from[5], the figure gives the ratio of propellant weight W_p to total initial weight W_o of the spacecraft as a function of delay time τ at two different abort conditions and for three different thrust levels. The reason for haste in making these maneuvers and using maximum available thrust levels is apparent in this figure.

The orientation time between the firing of the maneuver rocket and entry does not appear to be critical even for launches from low-altitude orbits. This fact can be illustrated by considering a case in which the spacecraft is in a capture condition, say $V = 30{,}000$ ft/sec, $\gamma = 0^{\circ}$, $h = 250{,}000$ ft, and $L/D = -0.5$. If we then compute backward in time, the trajectory leading up to this capture condition is obtained. Each position on this trajectory represents a possible condition of state following an abort and the firing of the maneuver rocket. The time required to go from each of these state conditions to the final capture condition is shown in figure 17. It may be seen, for instance, that it takes 150 seconds to go from 300,000 feet and 233 seconds to go from 400,000 feet to the final condition at 250,000 feet. When considering a maneuver requiring 10 to 20 seconds, sufficient time appears to be available for reorientation prior to entry.

Range available for energy management.- Due to the wide range of entry conditions following aborts between orbital and escape velocities, it would be a formidable task to establish the range capabilities of the spacecraft throughout this entire region. No such analysis is known to exist and may not be merited until more information on the launch trajectory, the available fuel, and the abort entry conditions are established. On the other hand, a less extensive study does appear to be warranted. Although it may be difficult to guide the space vehicle from a superorbital abort to a direct return to the atmosphere with a prespecified set of entry conditions, it should be possible to achieve these desired entry conditions accurately if the spacecraft is allowed to coast to apogee first. For this reason, it may be desirable at this time to investigate what the range envelope for the "middle-of-the-corridor" conditions at superorbital velocities might be.

Abort From Translunar Trajectories

Although not as probable as an abort from launch, it may be necessary on occasion to abort from the translunar trajectory. Penetration by a meteoroid, solar flares, or sudden illness may justify this type of an abort. Also, closely associated to this problem is the case (referred to in the preceding section) in which the vehicle is aborted at superorbital velocities, coasts to apogee and uses the available thrust to insure a safe return to earth on the next pass. Since it will take from 2 to 24 hours (or more) to return to earth from this type of an abort, the element of time is no longer as critical as in the preceding cases considered. Furthermore, the guidance logic for making the proper velocity correction for these types of aborts should dovetail with the procedures used for midcourse corrections for the nominal return trajectory from the moon.

The only published work in this field known to the author is given in[9]. The paper essentially describes a method of calculating positions on the nominal translunar trajectory where aborts may be made which lead to a return to earth at a predetermined landing site. The concept referred to as abort "way stations" appears to be particularly suited to guidance schemes which store influence coefficients derived from perturbations on a nominal trajectory. Such a scheme is now being seriously considered for the Apollo mission.

Aborts From Lunar Landings

An entirely new field of study has recently developed with the consideration of lunar landings and lunar rendezvous. Since there is very little difference between an abort from a lunar landing and the problem of launching a rendezvous operation from the surface of the moon, these two problems are currently being studied jointly. It is too premature to discuss any detailed results now but papers on these subjects may well become quite numerous in the very near future.

Concluding Remarks

To date the United States' only manned venture into space has been made with the relatively simple and reliable system of the Mercury spacecraft. Without lift control and with only a small fixed thrust rocket, the procedures for abort and recovery have not required a complex guidance and control system. The next generation of vehicles will have both lift control and considerably more maneuver thrust capability. Just how these capabilities will be used will depend on the continued desire to keep the procedures as simple and reliable as possible. It seems possible that atmospheric and suborbital aborts can be kept relatively simple. The computations for superorbital and translunar aborts appear to be complicated by the necessity of highly accurate velocity changes. However, even these types of abort can be kept simple by using stored programs such as the abort way stations. At the moon, it now appears that onboard computations of the optimum thrust maneuver will be a virtual necessity in the case of an abort during some phases of the lunar landing. However, since onboard computations will also be necessary for the basic landing maneuver, an abort may not place an undue burden on the guidance logic system. The lack of an atmosphere at the moon should, in some ways, greatly simplify the choices and the computations of the optimum course of action.

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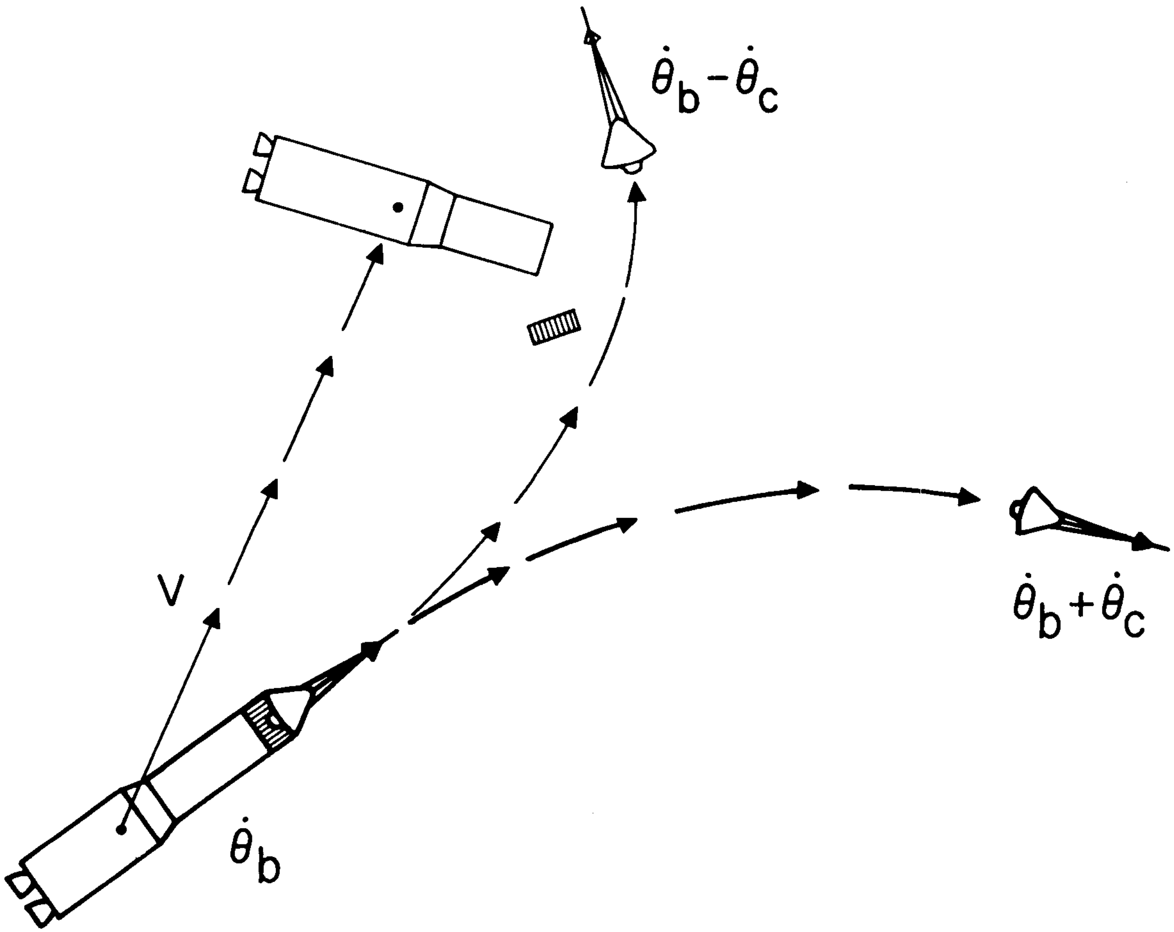
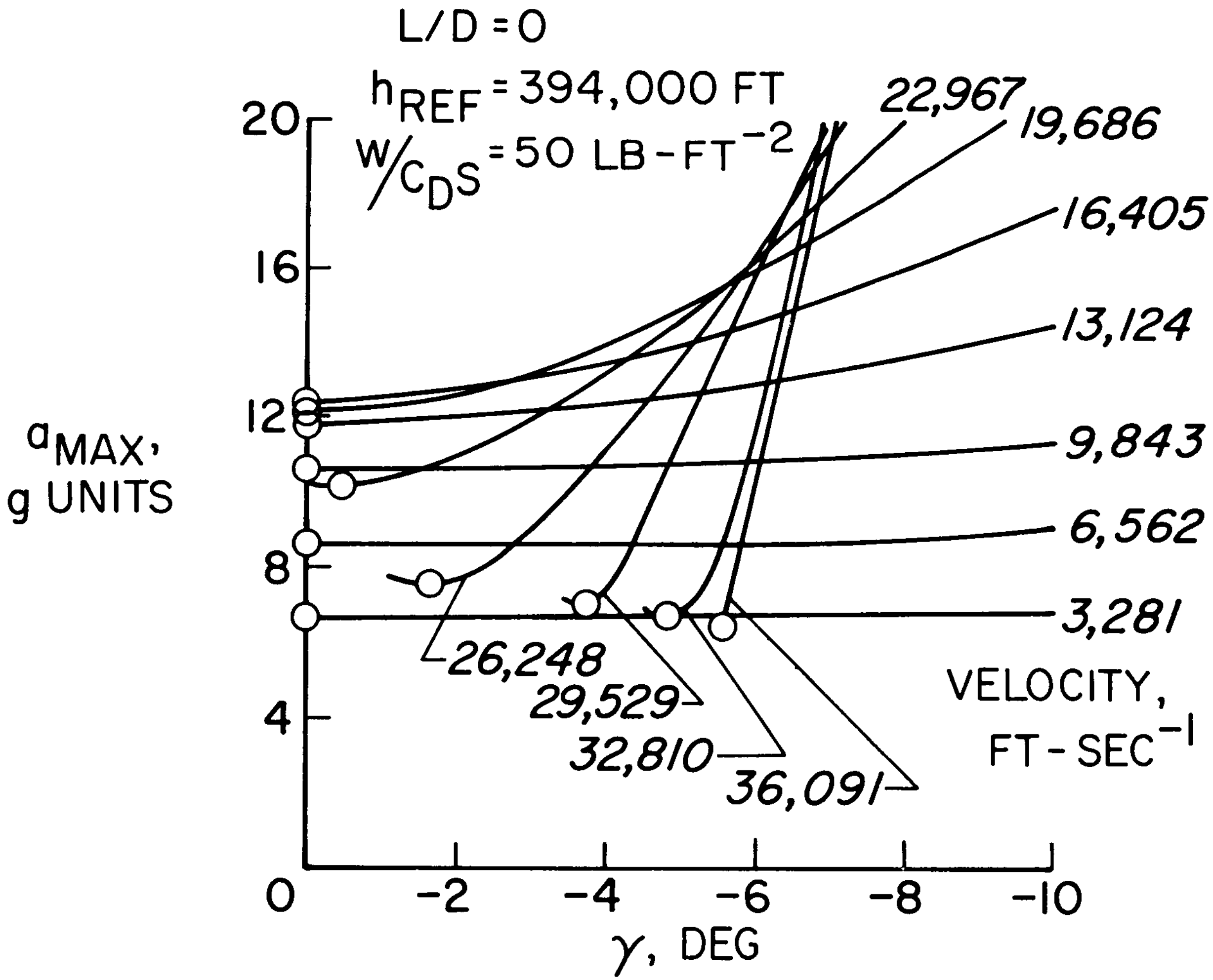


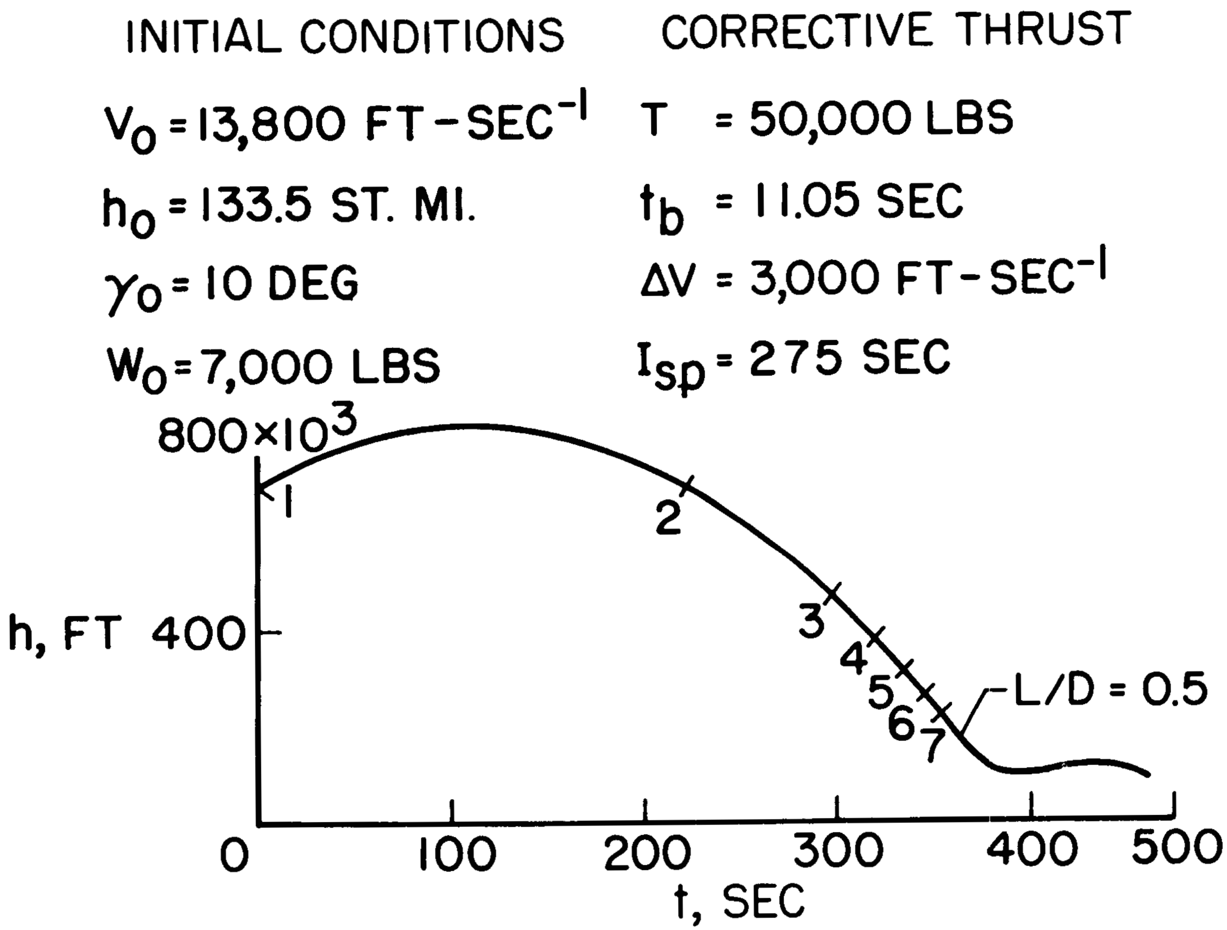
NASA

Figure 1.- Atmospheric abort anticollision control.



L/D = 0. NASA

Figure 2.- Maximum deceleration at entry.



NASA

Figure 3.- Typical trajectory after abort.

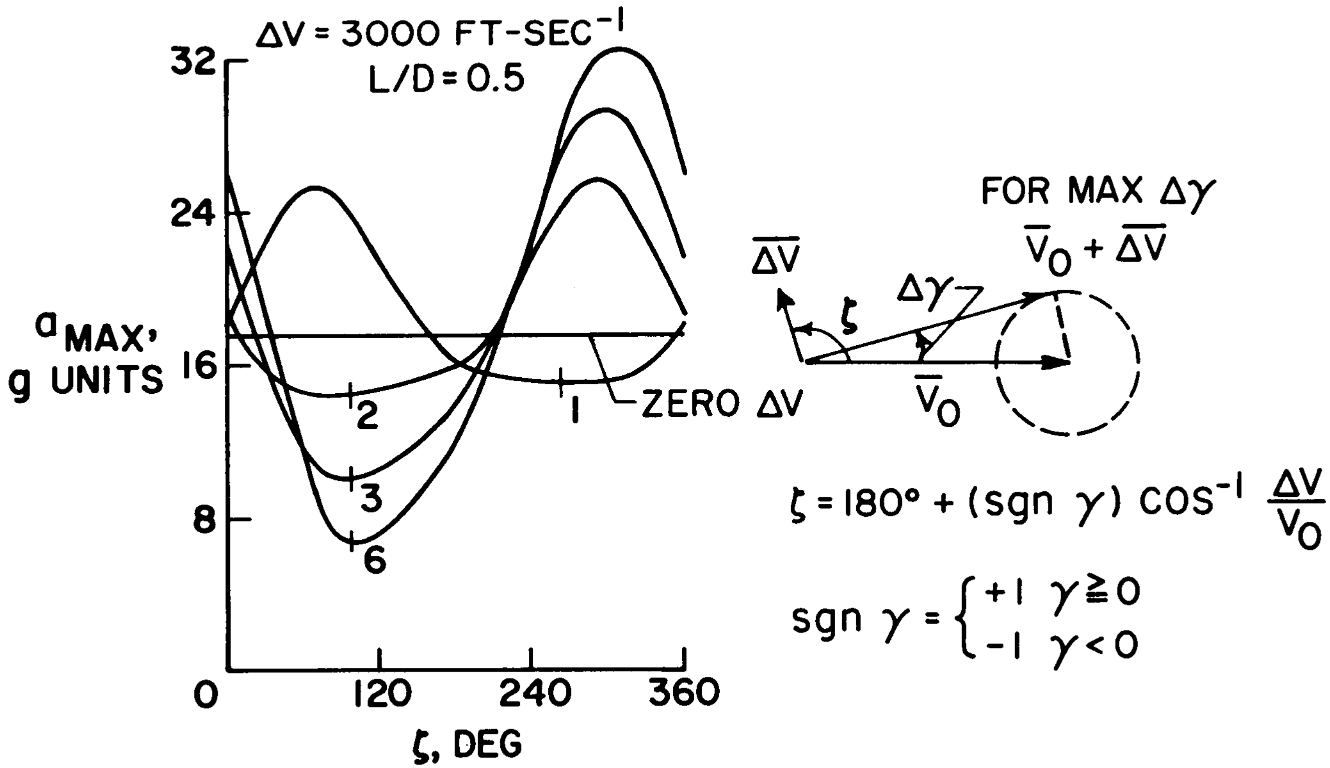


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Figure 4.- Variation of maximum deceleration with direction of thrust.

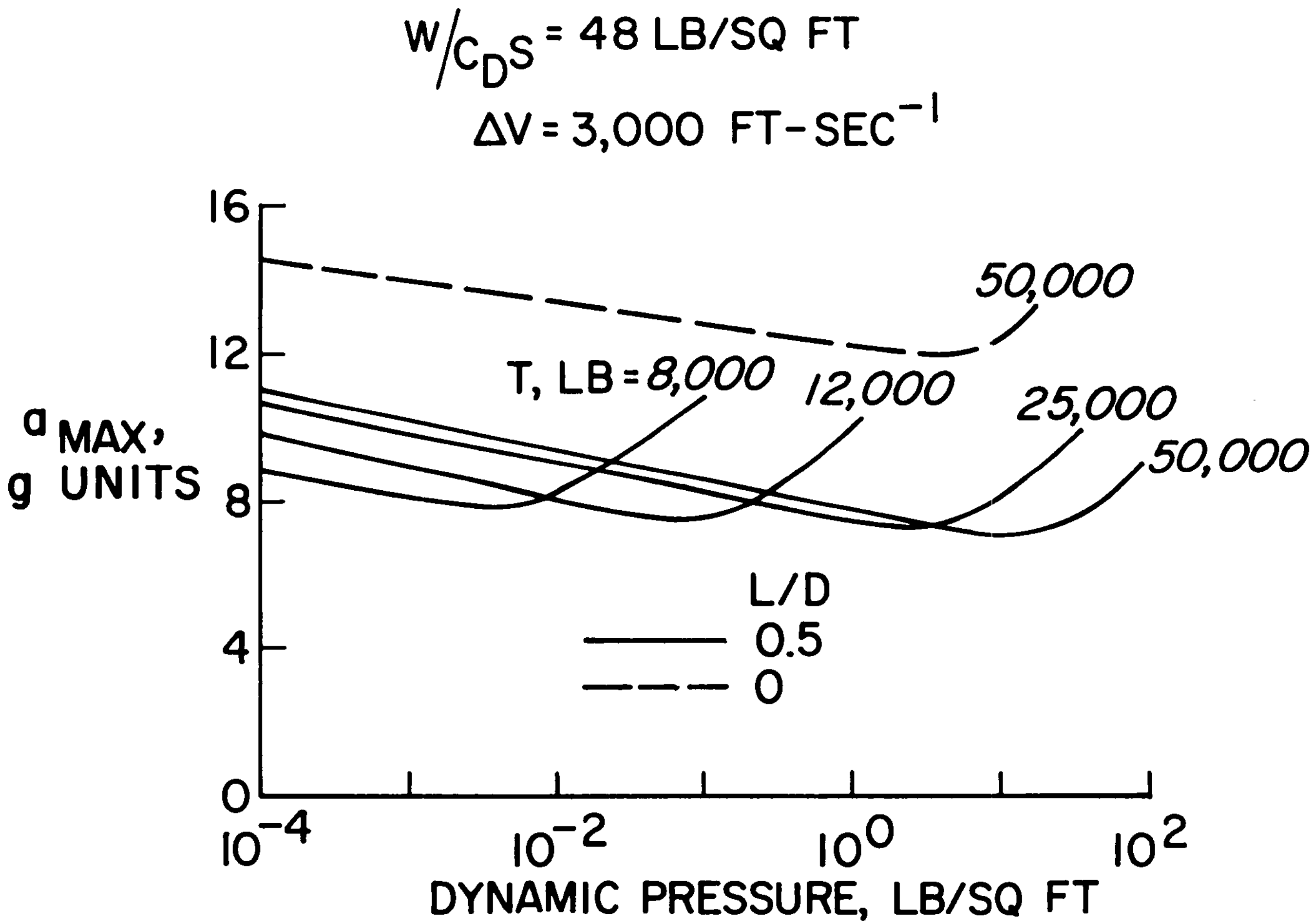

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Figure 5.- Variation of maximum deceleration with time of thrust.

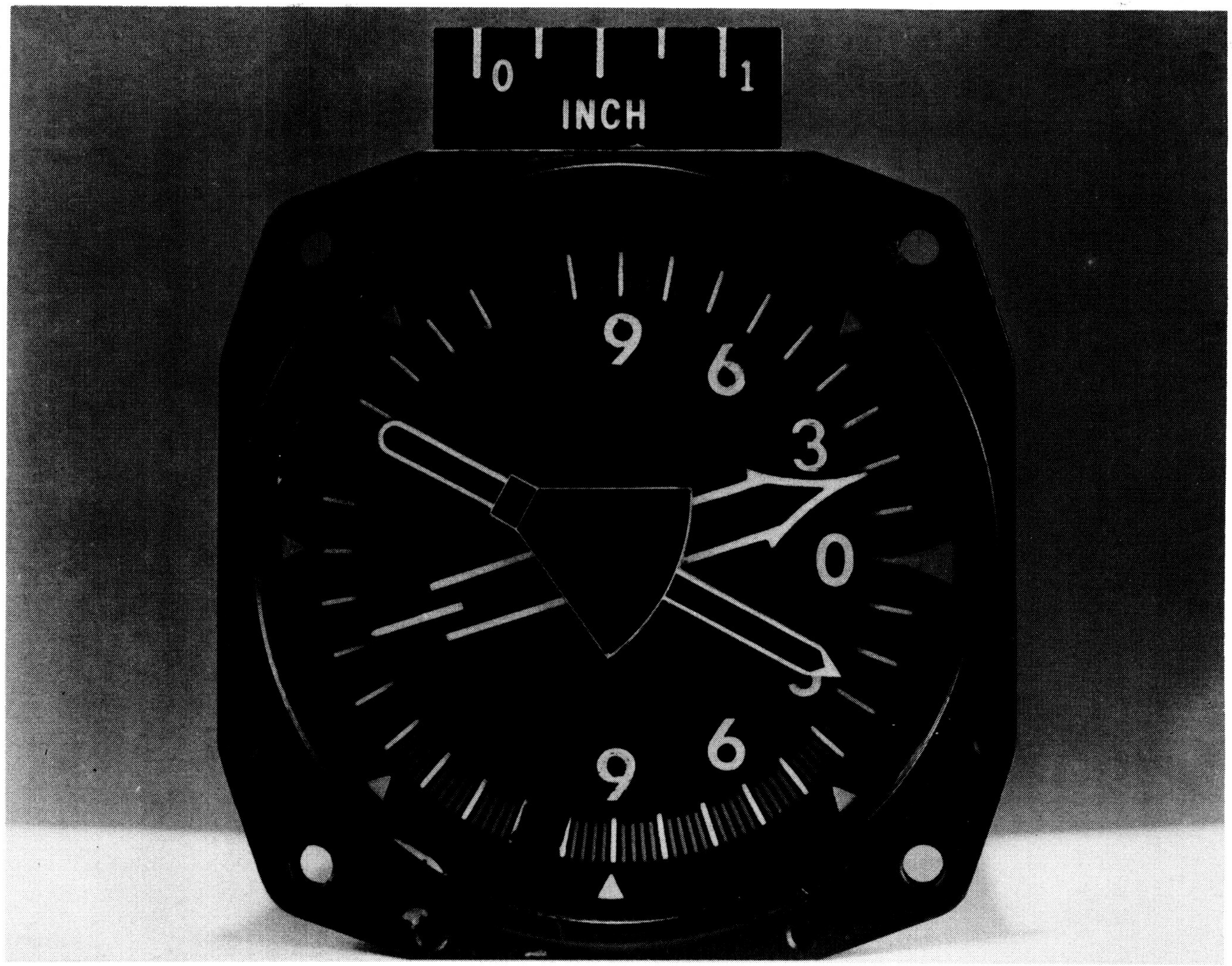

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Figure 6.- Attitude-situation display.

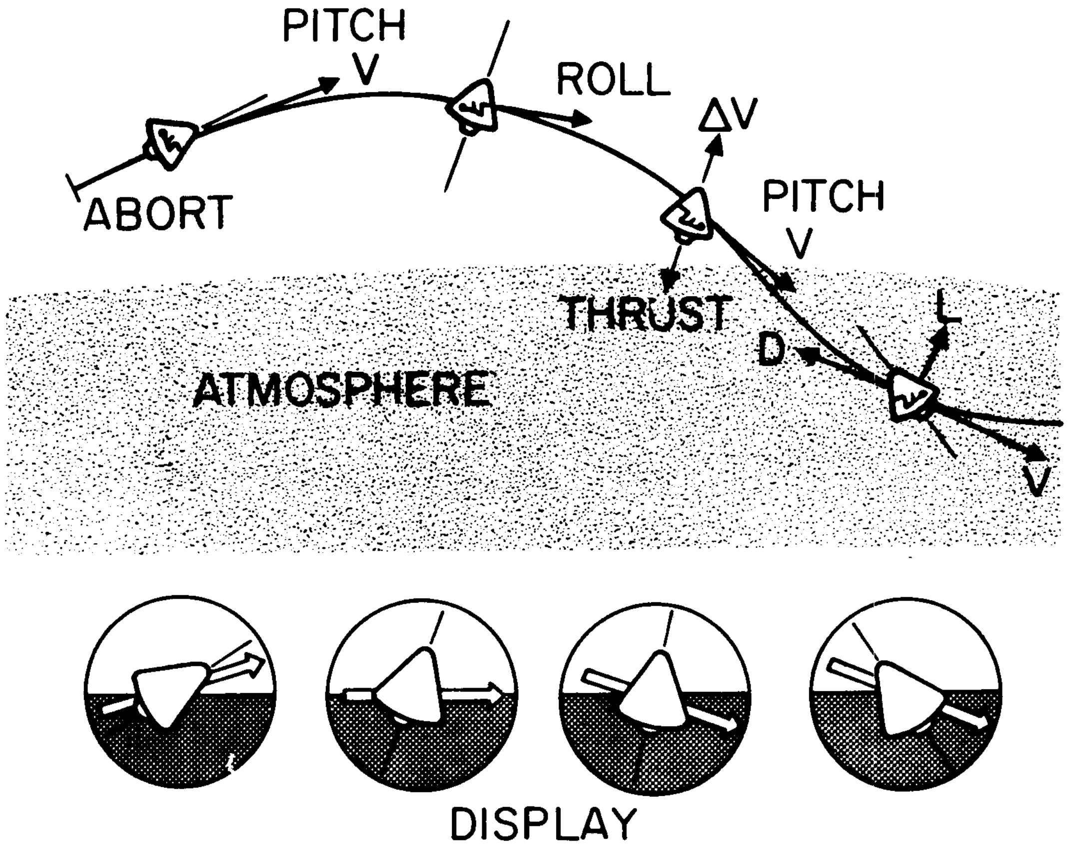

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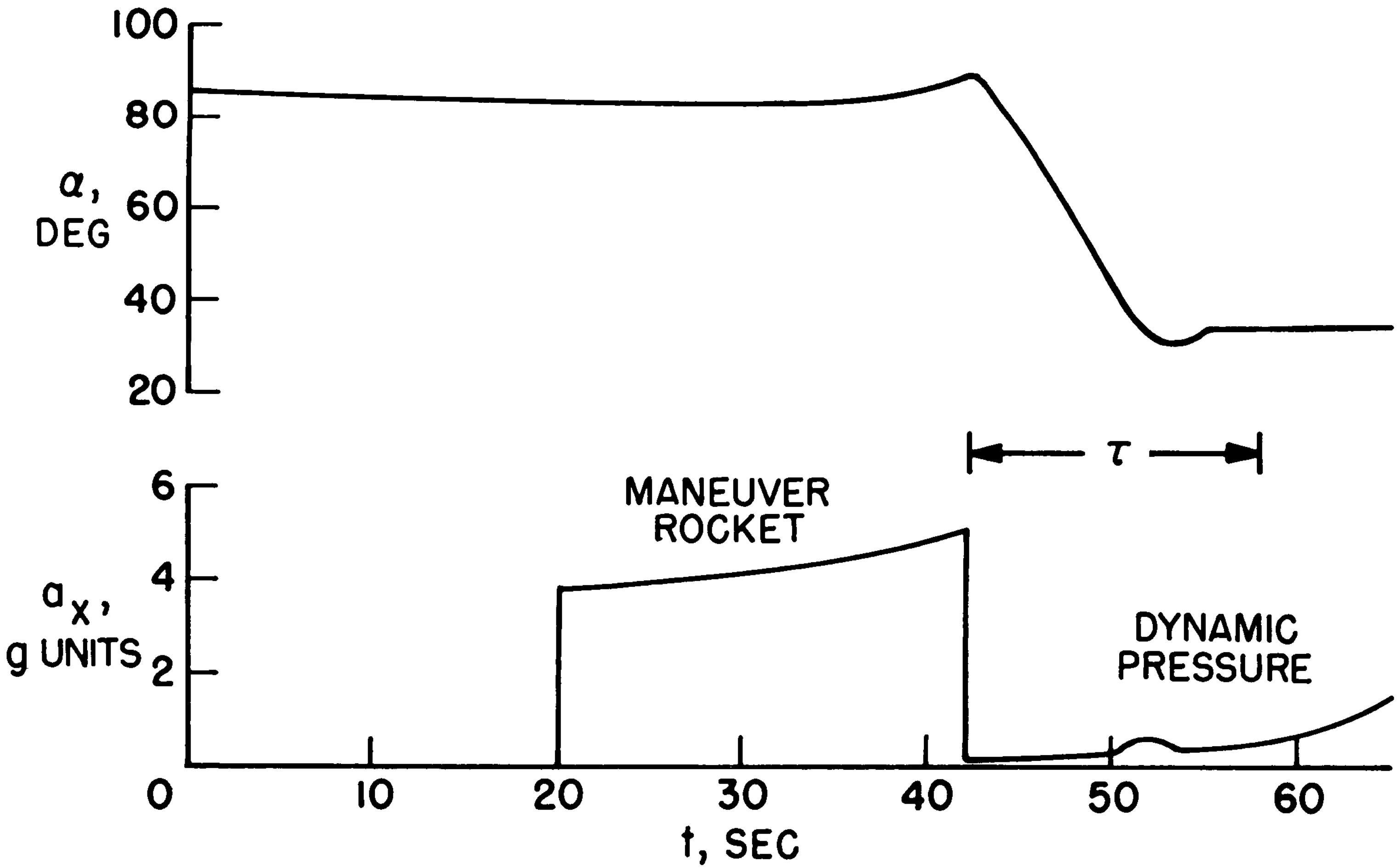
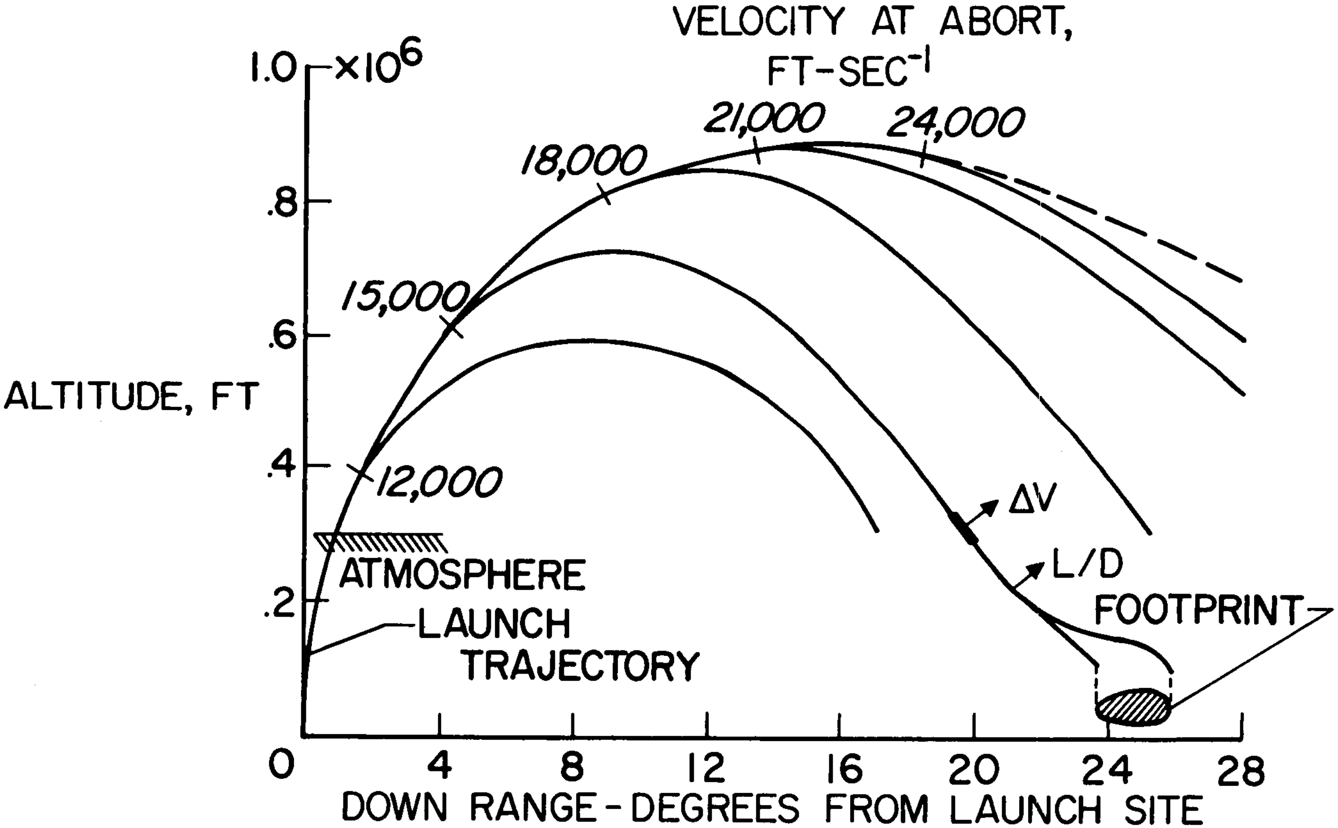
Figure 7.- Reorientation after suborbital abort.



q_{max} = 6.2 deg-sec^{-1}.

NASA

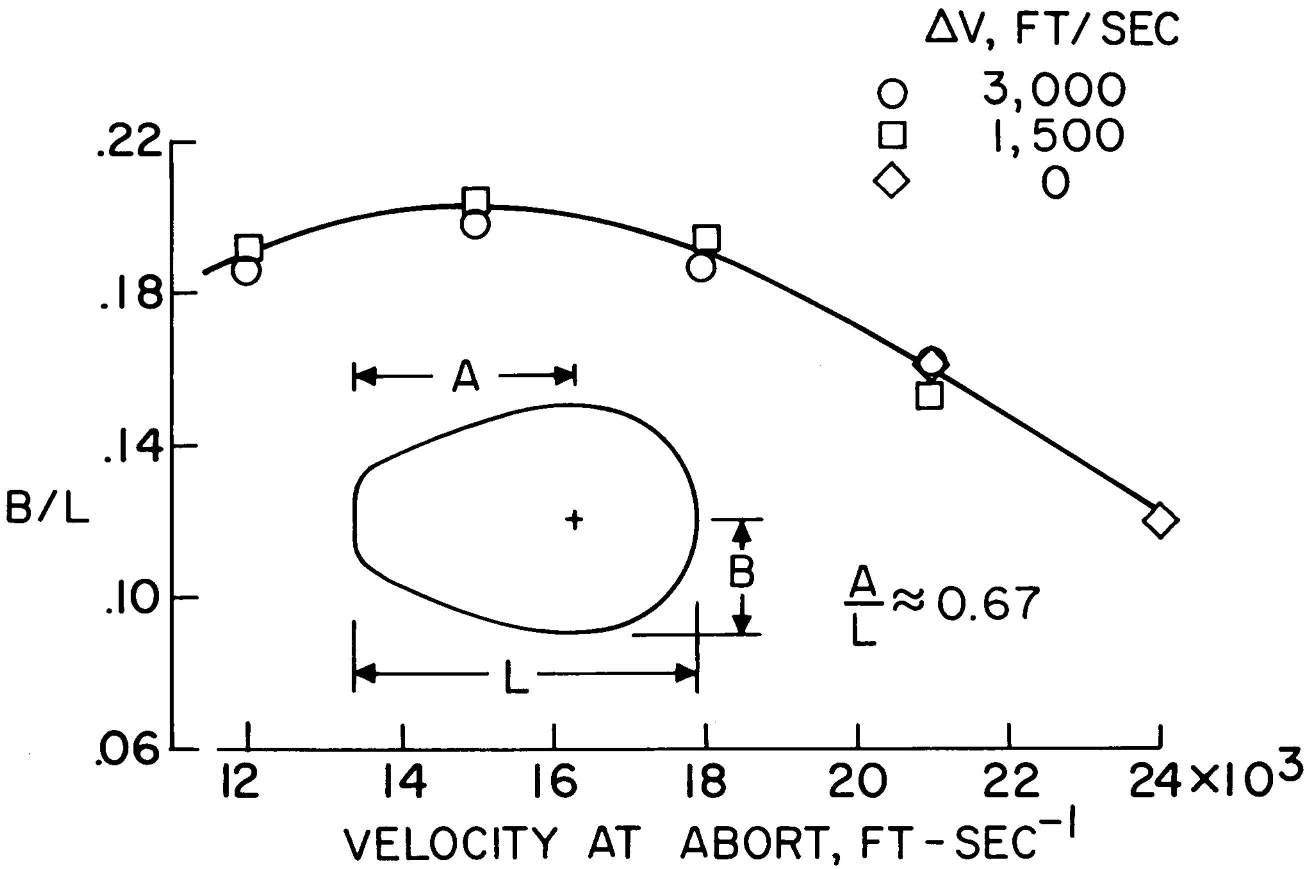
Figure 8.- Typical suborbital final pitch maneuver.



$(L/D)_{max} = 0.5.$

NASA

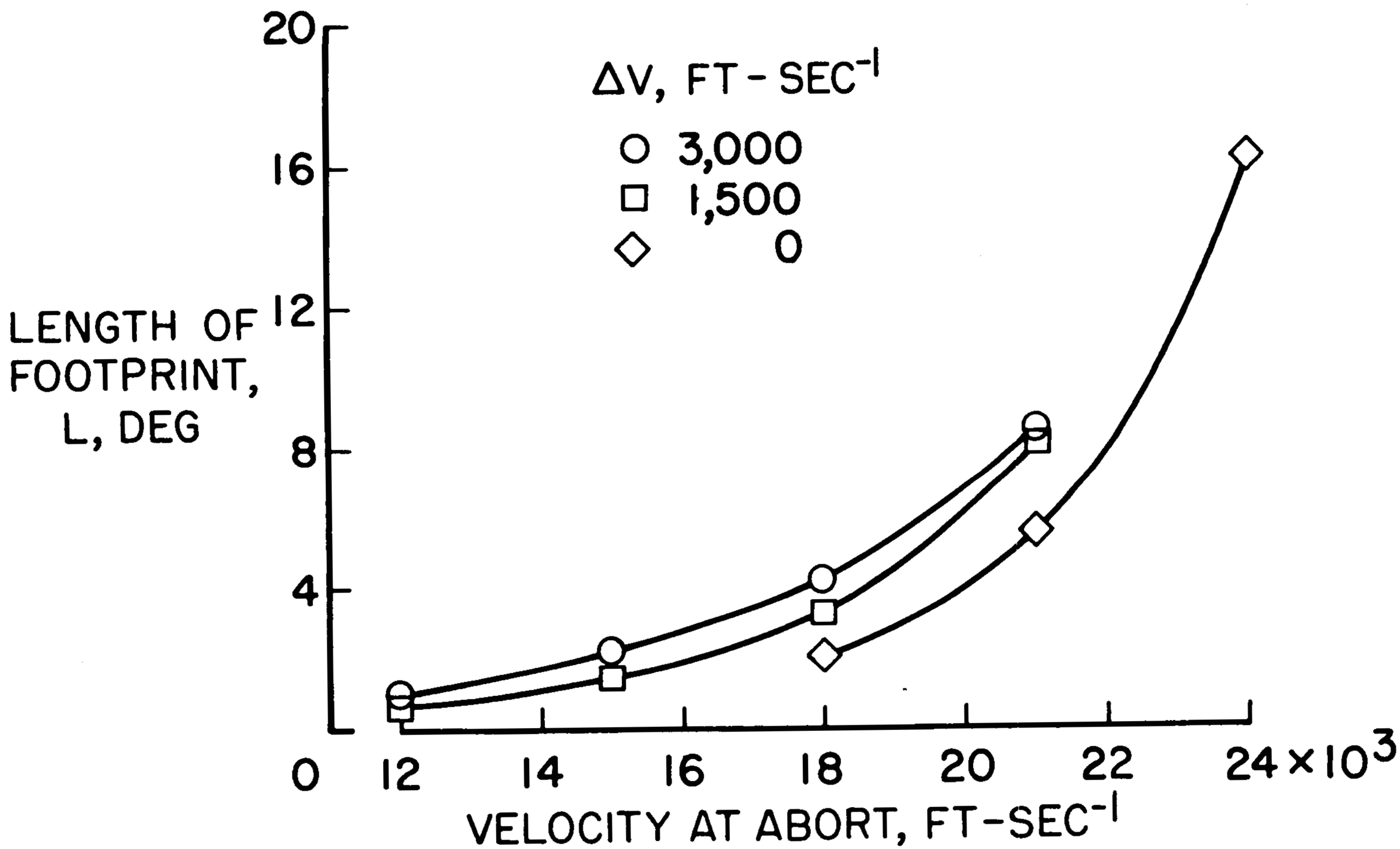
Figure 9.- Range after suborbital abort.



$(L/D)_{max} = 0.5.$

NASA

Figure 10.- Ratio of maximum cross range to length of footprint.



NASA

Figure 11.- Length of footprint.

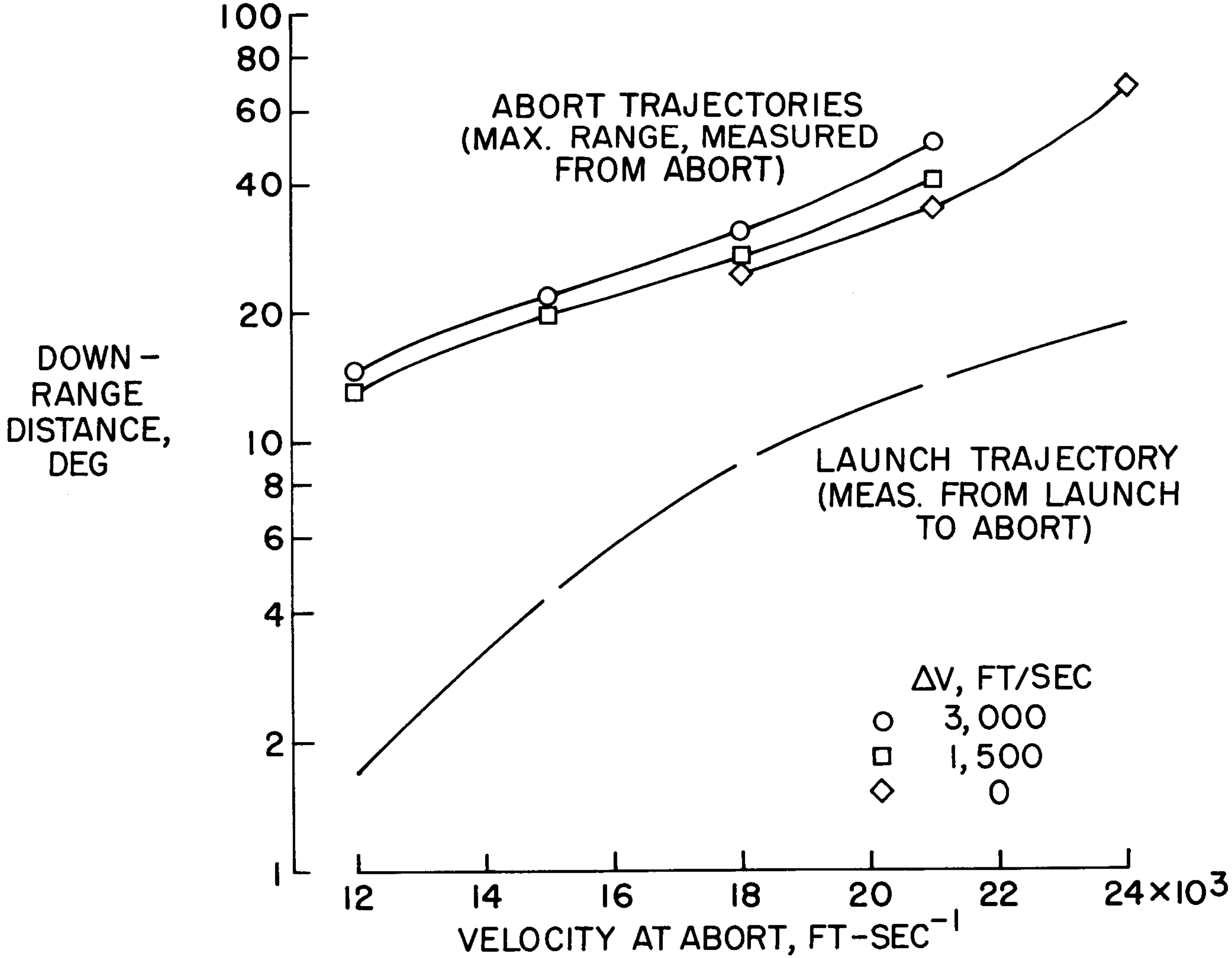


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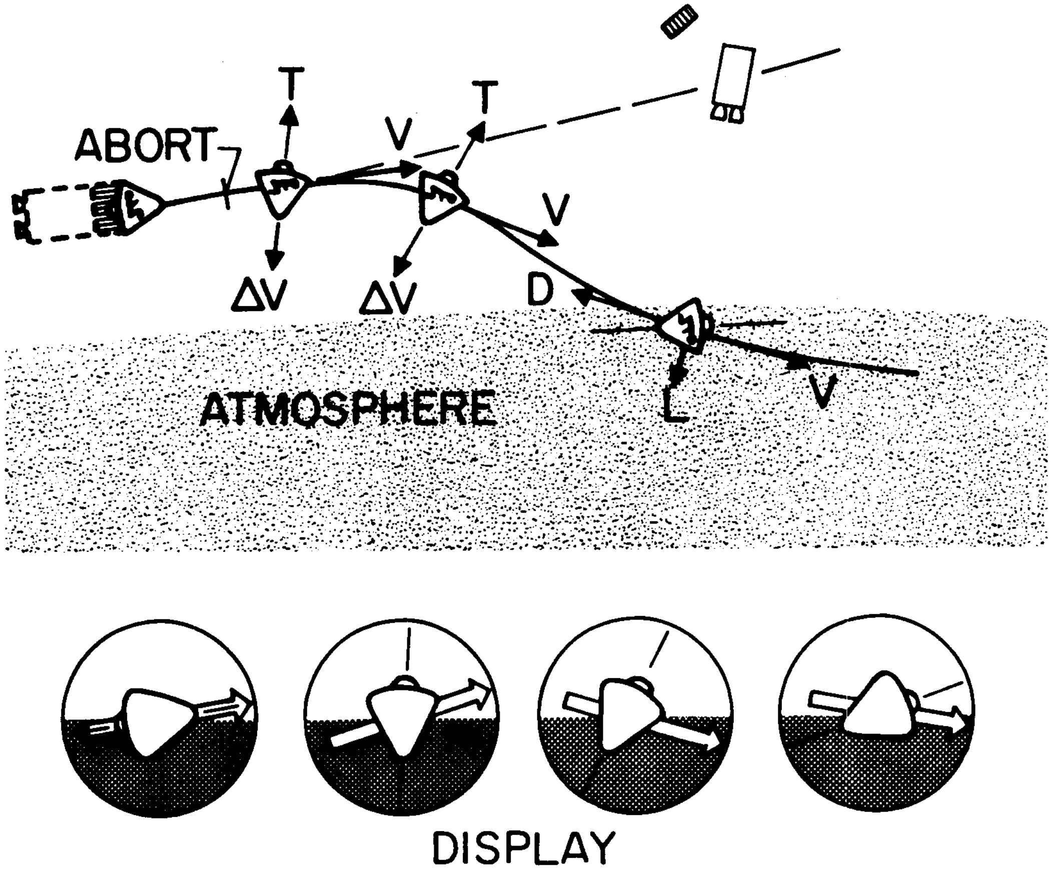
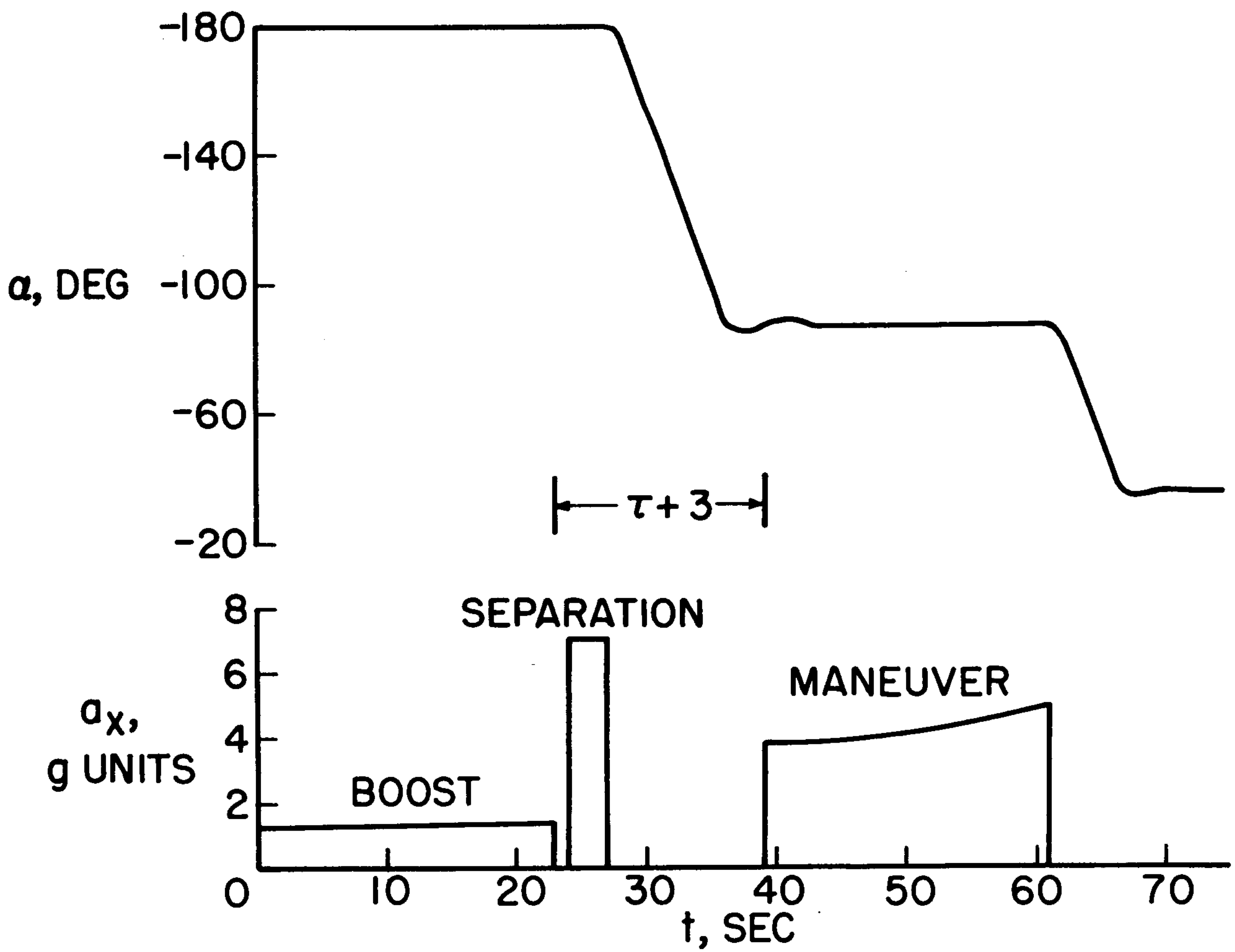
Figure 12.- Maximum down-range distance obtained following an abort.



NASA

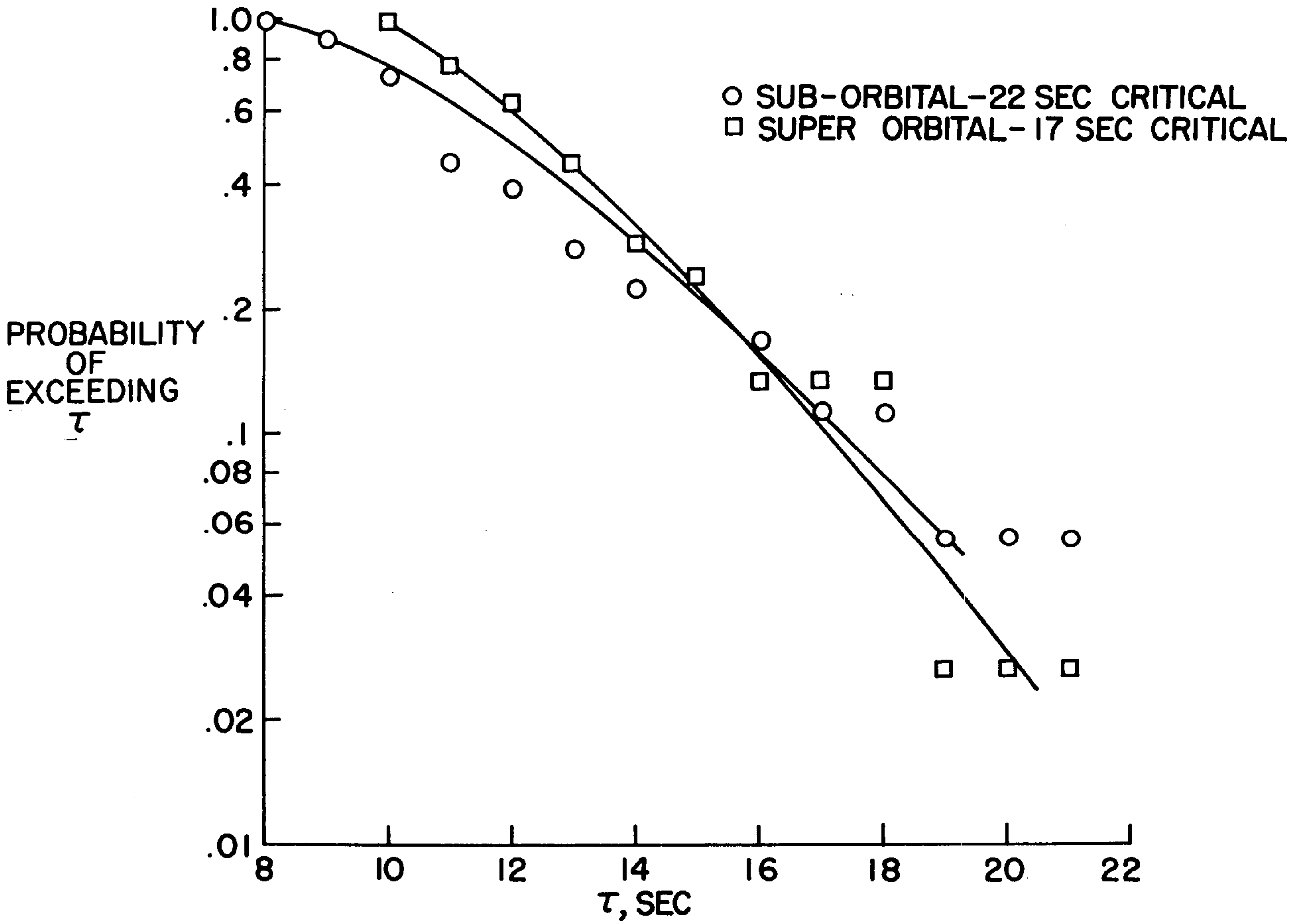
Figure 13.- Reorientation after superorbital abort.



q_{max} = 12.4 deg/sec.

NASA

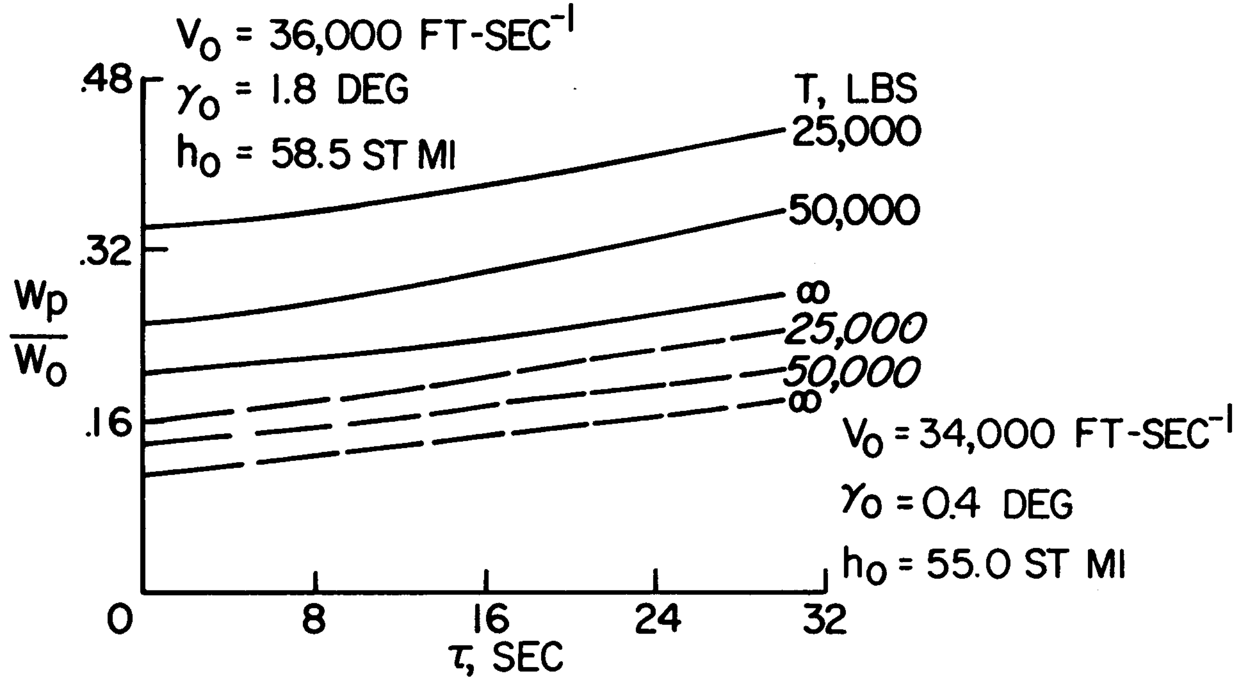
Figure 14.- Typical superorbital pitch maneuver.



$q_{max} = 12.4$ deg-sec^{-1}.

NASA

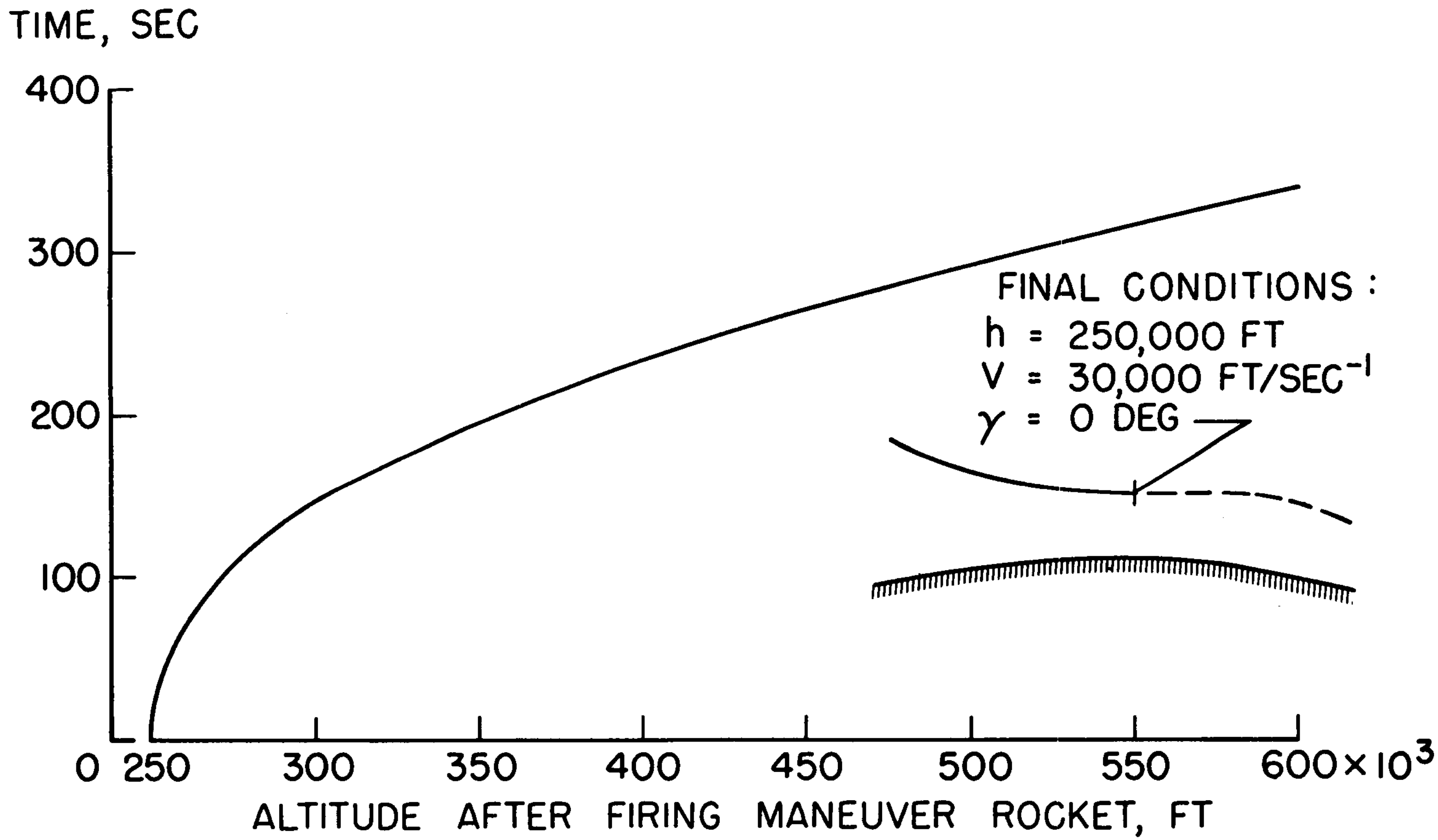
Figure 15.- Time response of pilots.



W_o = 14,200 lb; I_{sp} = 420 sec.

NASA

Figure 16.- Abort fuel requirements with time delay.



NASA

Figure 17.- Time to reach an altitude of 250,000 feet.