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FIXED-BASE-SIMULATOR STUDY OF THE ABILITY OF A PILOT TO PERFORM SOFT LUNAR LANDINGS

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SUMMARY

A fixed-base-simulator study has been made of the ability of pilots to perform soft lunar landings. The study permitted all six rigid-body degrees of freedom of the landing vehicle. The pilot was given control over the vehicle thrust level and over the moments about each of the three body axes. It was assumed that the vehicle had no automatic damping or control. The pilot's task was to acquire and fly a particular trajectory and to perform a soft landing in a specified area. The task was performed either by one pilot in complete control of vehicle or by two pilots with one controlling vehicle attitude and the other controlling the thrust level.

Results of this study have shown that the pilots could consistently make good lunar landings and generally landed with touchdown-velocity components (radial and tangential) of less than 4 feet per second and within a range of about 2,000 feet of the desired landing site. This range variation was associated primarily with the readout resolution of the indicators shown to the pilot. There was no appreciable difference in touchdown conditions or fuel consumption between the one-pilot and the two-pilot flights. However, one-pilot lunar landings required close concentration on the part of the pilot in order to attain acceptable touchdown-velocity components, acceptable vehicle attitude, and the desired landing site. The landing task was made much easier by using two pilots: one operating the throttle and the other operating the attitude control.

It was possible to effectively uncouple the range and the altitude control for the landing maneuver. During most of the maneuver, the throttle is used for range-rate and range control, and the attitude controller is used to adjust rate of descent and altitude. During the vertical descent, the throttle is used for rate of descent and altitude control, and the attitude controller is used to adjust range rate and range.

The characteristic velocity for piloted lunar landings varied from about 0 to 10 percent above that computed for a perfectly flown trajectory.

INTRODUCTION

It is anticipated that man will participate in various lunar missions and that he will perform many functions in order to minimize the amount of automatic equipment (and hence weight) required on board the space vehicle and to improve systems reliability. It is important therefore to examine the various duties which man might perform and the equipment, displays, and guidance required for these duties. A number of investigations concerning a pilot's ability to perform certain tasks applicable to lunar missions have been reported and have demonstrated man's ability to establish lunar orbits (ref. 1) and to perform rendezvous by using instrument displays (ref. 2) or by using visual cues (ref. 3).

One of the more critical tasks involved in lunar missions is that of performing landings on the lunar surface. This mission is critical because of the stringent limitations placed on the touchdown-velocity components, vehicle attitude, and fuel expenditure permitted in the task. Several schemes for performing soft lunar landings which utilize automatic control have been proposed (refs. 4 to 6, for example) and a few studies of manned lunar landings have been reported (see ref. 7, for example). The degrees of freedom permitted in the manned lunar landings generally have been restricted to motion in a plane, and in some cases have been further restricted to particles, so that vehicle dynamics were neglected. The present investigation which permitted all six degrees of freedom of the vehicle was made to determine the ability of a pilot to perform soft lunar landings.

Equations of motion were solved on an electronic analog computer operating in real time. The pilot closed the control loop and had direct input into the force and moment equations. It was assumed that there was no automatic damping or automatic control in the vehicle. Most of the simulated flights were initiated in an approximately circular orbit which had an altitude of 50 statute miles. The pilot's task was to perform a soft landing in a designated area.

SYMBOLS

The British system of units is used in this study. In case conversion is desired, the following relationships may be used:

1 international foot = 0.3048 meter

1 international statute mile = 1.609344 kilometers

g_e	gravity at surface of earth, 32.2 ft/sec^2
g_m	gravity at surface of moon, 5.32 ft/sec^2
h	altitude above surface of moon, ft
I_{sp}	specific impulse of rocket, 420 sec
I_X, I_Y, I_Z	moments of inertia about body X-, Y-, and Z-axes, respectively (note $I_X = I_Y$), slug-ft^2
J_X, J_Y	control forces exerted along body X- and Y-axes, respectively, lb
l	distance from reaction control jets to vehicle center of gravity, ft
M_X, M_Y, M_Z	control moments exerted about body X-, Y-, and Z-axes, respectively, ft-lb
m	vehicle mass, slugs
$\dot{m}$	time rate of fuel consumption, slugs/sec
p,q,r	rates of rotation about body X-, Y-, and Z-axes, respectively, radians/sec
R	radial distance from center of moon, ft
ΔR_a	range error at touchdown, ft
R_m	radius of moon, 5.702×10^6 ft
$\dot{R}$	vehicle velocity component in radial direction, ft/sec
$R\dot{\theta}$	vehicle velocity component in circumferential direction, ft/sec
t	time, sec
T	rocket thrust along body Z-axis, positive in Z-direction, lb
T_{max}	maximum rocket thrust, lb
ΔV	characteristic velocity, $g_e I_{sp} \log_e \frac{m_o}{m}$, ft/sec
W_o	weight in orbit, $m_o g_e$, lb

X,Y,Z vehicle body-axis system with origin located at vehicle instantaneous center of gravity and with the Z-axis alined with vehicle axis of symmetry (fig. 1)

x_i,y_i,z_i inertial reference axes with origin located at center of moon (fig. 2)

x,y,z distances along the x_i-, y_i-, and z_i-axes, respectively, ft

x',y',z' reference axes parallel to inertial axes with origin located at vehicle center of gravity

Δz vehicle lateral displacement with respect to initial orbit plane, ft

α angular orientation of vehicle in pitch, defined as approximate angle between local horizontal and vehicle Z-axis, radians or deg

δ_T rocket throttle control displacement

δ_X,δ_Y control displacements which produce control forces along X- and Y-axes, respectively

δ_Z control displacement which produces control moment about Z-axis

ψ,θ,ϕ Euler angles of rotation, radians or deg

$\bar{\psi}$ angular orientation of vehicle referred to as yaw angle and defined as the angle between the vehicle XZ-plane and the trajectory plane, radians or deg

$\bar{v}$ sideslip velocity, ft/sec

$\bar{\phi}$ bank angle, rad

$\bar{\theta}$ angular travel over surface of moon, radians

Subscripts:

f fuel

H hover condition

o initial conditions

A dot over a symbol indicates a derivative with respect to time.

EQUATIONS OF MOTION

The six equations of motion used in this study permitted all six rigid-body degrees of freedom of the vehicle. The three force equations were written with respect to an inertial-axis system; the three moment equations were written about the vehicle body axes (appendix A). The purpose of writing the force equations about the inertial-axis system rather than about the body axes was to avoid coupling terms which can introduce inaccuracies resulting from computer drift. The pilot closed the control loop and had direct input into the force and moment equations.

The inertial-axis system for the force equations was a fixed-axis system with its origin at the center of the moon (fig. 2). The moon was assumed to be a nonrotating homogeneous sphere with a radius of 5.702×10^6 feet and a surface gravity of 5.32 feet per second per second. Vehicle mass and moments of inertia were varied as thrust was applied to account for mass reduction during thrusting. Mass changes due to the use of moment controls were negligibly small in comparison with the mass change associated with thrust application and were therefore neglected.

COMPATIBILITY OF ANALOG ACCURACY WITH SIMULATION REQUIREMENTS

The accuracy available in computations made by using electronic analog equipment is limited by the fact that variations in parameters of the similation are generally specified as changes in voltage levels. The voltage level available is a range of ±100 volts. At best, therefore, 200 volts is available for the full range of a particular variable. This basic limitation and some drift in analog component characteristics are such that the absolute accuracy of a good computer, for a specific output quantity, is generally quoted to be about 1 percent of the scaled range of that quantity. For example, if the altitude range of a problem is expected to be 100,000 feet, then 200 volts is used to represent 100,000 feet, and the expected altitude accuracy would be ±500 feet.

In the present study, it was desired to make soft landings on the lunar surface starting from altitudes of about 264,000 feet. It is apparent, therefore, that the computed altitude at any point in the trajectory could be incorrect by as much as ±1,320 feet. However, in studying a pilot's ability to perform certain functions, the primary interest is to see how well he reacts to presented information, regardless of whether this information is correct or in error. In other words, the pilot assumes that the information displayed to him is correct, and he tries to fly his vehicle accordingly. Based on this philosophy, the

analog outputs are useful beyond their absolute accuracy. There is, of course, a limit to which this philosophy can be used, and this occurs when the output is reduced to the noise level of the computer. At this point the outputs vary randomly and are of little use.

VEHICLE DESCRIPTION

The vehicle assumed for this study was a relatively squat body of revolution, appearing somewhat as indicated in figure 1. The vehicle had a large fixed engine which thrusts along the axis of symmetry with a maximum available thrust capable of accelerating the vehicle weight-in-orbit at $\frac{1}{2}g_e$; thus $\frac{T_{max}}{W_o} = \frac{1}{2}$. Thrust was assumed to be variable from full thrust to zero thrust. This range of throttleability is not attainable in practice; however, it was recognized that some range of throttleability would be required in landing, and it was of interest to investigate the range used by the pilots.

Moment control about the axis of symmetry was assumed to be available from reaction jets producing a pure couple. Moment control about the other two axes was produced by reaction jets located at the top of the vehicle.

The variations of the vehicle moments of inertia with vehicle mass are shown in figure 3.

Cockpit and Controls

The general layout of the cockpit used in the study is given in figure 4 and shows the relative positions of the pilot's chair, throttle, controls, and displays. The axis system selected for the vehicle was such that the controls and attitude displays were consistent with those common to aircraft. Vehicle thrust was commanded by the use of the displacement controller at the pilot's left. Thrust varied linearly with controller displacement. The pitch and roll attitudes were commanded through the two-axis hand controller located to the pilot's right. Yaw control was obtained by use of treadle-type foot pedals. This type of control system (foot pedals and two-axis controller) was selected because it was anticipated that precise combined control would be required in performing soft landings. In such situations pilots had indicated some preference for this type of control system over other types (see refs. 8 and 9, for example). Control torques commanded by the pilot for attitude control were proportional to control deflection except for a small dead band around zero deflection (fig. 5).

Instrument Display

Selection of a suitable instrument display involves a number of important factors, such as determination of what quantities should be displayed, type of display required for rapid interpretation, and proper arrangement to minimize scanning. In performing landings, it is apparent that altitude, velocity, and vehicle attitude are of primary importance, particularly near touchdown. These quantities were displayed to the pilot on the display panel. Because rather good resolution was required in altitude, rate of descent, circumferential velocity, and vehicle attitude, and because each of these quantities varied over a wide range, some special treatment of displays was required and is discussed in the following sections.

Altitude.- At high altitudes, good resoltuion is not particularly important and could be obtained crudely from the guidance plots (see section entitled Guidance Scheme). Near touchdown, however, altitude is very important. The three-hand altimeter shown in figure 6 became activated at an altitude of about 10,000 feet and could be read to within about 2 feet.

Rate of descent.- The rate of descent was shown on a three-hand meter which was scaled for 1,000 feet per second. It therefore was possible to read the meter to a rate of descent of 0.2 foot per second. However, below about 2 feet per second, the analog noise level in the output to this meter and the gear backlash reduced the resolution to about 1 foot per second.

Circumferential velocity.- The circumferential velocity was displayed on a galvanometer-type meter in combination with automatic switching so that there were three scales available: 0 to 6,000 feet per second, 0 to 600 feet per second, and 0 to 60 feet per second. A system of lights was used to indicate the scale in use at any particular time and also the direction of the circumferential velocity. This meter could be read to within about 1 foot per second on the lowest scale.

Vehicle attitude.- The vehicle attitude is important during the entire trajectory inasmuch as it is a primary factor in determining the efficiency of the landing maneuver. Near touchdown the attitude assumes added importance because it determines to a large extent the magnitude of the circumferential velocity $R\dot{\theta}$. In the vertical descent, a small tilt of the vehicle can build up $R\dot{\theta}$ rather rapidly. In order to permit the pilot to monitor pitch attitude very closely, a vernier pitch-attitude indicator was supplied. This was scaled to read from 85° to 95°, and could be read accurately to a fraction of a degree.

Although the display and guidance scheme used in the investigation was not optimized, it was evolved after trying various other schemes and was found to be adequate for this investigation.

LUNAR LANDING TRAJECTORIES

There are any number of trajectories which could be used in performing lunar landings. For the purpose of this study, the actual trajectory is not of particular significance but should be practical with regard to engine capabilities and fuel economy. Several trajectories were used in this investigation - a nominal gravity-turn trajectory in which thrust was always directed against the velocity vector and several "off-nominal" trajectories. In addition one constant-altitude deceleration maneuver was studied briefly.

Nominal Gravity-Turn Trajectory

The nominal gravity-turn trajectory was selected from a group presented in reference 10. This trajectory is illustrated in figure 7, and can be described as follows: The landing vehicle is assumed to be in a circular orbit at an altitude of 50 statute miles. When the vehicle passes over some predetermined landmark, the pilot initiates thrust at a level which results in a value of T/W_o of 0.143 (i.e., deceleration of $0.143g_e$ or 4.61 feet per second per second). This thrust level is maintained for a surface range of about 390 miles, after which the thrust is increased to $T/W_o = 0.430$. The thrust level is maintained until the vehicle velocity components are reduced to almost zero and the altitude is reduced to a few thousand feet. The pilot then varies thrust and attitude as might be required to perform a soft vertical landing.

Off-Nominal Trajectories

Off-nominal trajectories had initial conditions which were different from those of the nominal trajectory. The assumed variations were ±50 feet per second in the initial velocity components ($\dot{R} = \pm 50$ feet per second, $R\dot{\theta} = 5{,}385 \pm 50$ feet per second) and ±20,000 feet in altitude ($h = 264{,}000 \pm 20{,}000$ feet).

Constant-Altitude Deceleration Maneuver

The constant-altitude deceleration maneuver used in this study was initiated at an altitude of about 7,600 feet. At this altitude, the pilot initiates retrothrust and controls the vehicle to maintain constant altitude until the velocity is reduced to about 2,000 feet per second. The pilot then permits the vehicle to lose altitude so that the vehicle velocity components are reduced to zero when the altitude is zero.

The following table lists the initial conditions for the various types of trajectories flown during the investigation:

Radial velocity, $\dot{R}$, ft/sec	Circumferential velocity, $R\dot{\bar{\theta}}$, ft/sec	Altitude, h, ft	Remarks
0	5,385	264,000	Nominal trajectory
			Off-nominal trajectories
0	5,385	284,000	High in altitude
0	5,385	244,000	Low in altitude
-50	5,385	264,000	High in $-\dot{R}$
-50	5,335	244,000	High in $-\dot{R}$, low in $R\dot{\bar{\theta}}$ and h
50	5,435	284,000	High in $\dot{R}$, $R\dot{\bar{\theta}}$, and h
50	5,335	264,000	High in $\dot{R}$, low in $R\dot{\bar{\theta}}$
-50	5,435	264,000	High in $-\dot{R}$, high in $R\dot{\bar{\theta}}$
50	5,435	264,000	High in $\dot{R}$ and $R\dot{\bar{\theta}}$
-50	5,335	264,000	High in $-\dot{R}$, low in $R\dot{\bar{\theta}}$
0	5,570	7,598	Constant-altitude deceleration

GUIDANCE SCHEME

The lunar landings made in the present study were "instrument" landings, in that instruments provided all of the information which the pilot used for vehicle control. In order for the pilot to fly the

"nominal trajectory," it was necessary to provide some guidance to permit him to compare his situation (velocities and position) with those of the nominal trajectory. There are various methods of presenting such information, for example, null-meters, cathode-ray-tube displays of the nominal trajectory characteristics, and prediction schemes. During the present study, two 2-variable plotters were used; one showed rate of descent as a function of altitude (fig. 8(a)) and the second showed the variation of altitude with range-to-go (fig. 8(b)). Prior to each flight, traces of these curves were put on the plotters. Each plotter was activated at the start of a flight, and it traced the corresponding parameters of the vehicle as the pilot flew each mission. The pilot attempted to fly the simulator so that the flight traces would follow the guide curves. The multiple curves shown in figures 8(a) and 8(b) were required to obtain the desired degree of resolution.

DEVELOPMENT OF LANDING PROCEDURE

There are three major conditions to be attained in making soft lunar landings:

(a) Low touchdown velocities (radial and tangential)

(b) Proper landing site

(c) Proper vehicle attitude

Prior to making any flights, it was believed that the primary task would be that of attaining low touchdown velocities and that the problem of reaching the proper landing point would be rather simple. This latter belief was based on the premise that by closely following the plot of rate of descent against altitude, and by using the proper thrust level, the pilot would fly the nominal trajectory and thereby land near the desired landing site. The pilots therefore concentrated primarily on flying the plot of $\dot{R}$ against h, and generally neglected the plot of h against range-to-go.

Preliminary Flights

Several interesting points were observed during initial flights. First, it was relatively easy to follow the guide plot of rate of descent against altitude over most of the flight. Near touchdown, however, the pilot was kept extremely busy in scanning and interpreting the display and in manipulating thrust and attitude controls. There was a tendency to be very cautious in this vicinity. This often resulted

in prolonged periods of hovering and even some climbing before actually landing. Such a case is illustrated in figure 9. This figure shows the rate of descent as a function of altitude. The solid line is the guide curve which was traced on the plotter before the flight was initiated. The dashed line shows the variation of $\dot{R}$ with h as the pilot flew the mission. As shown, the pilot followed the guide curve very well until he descended to an altitude of about 7,000 feet. At this point, he was concentrating on manipulating attitude controls to reduce $R\dot{\bar{\theta}}$ to zero and failed to note that his rate of descent was higher than the nominal value. At an altitude of about 550 feet, he noted the excessive rate of descent and applied increased thrust. The thrust level was somewhat excessive, and after descending to about 170 feet, the vehicle began climbing. The pilot noted this, and reduced thrust so that the climb stopped at an altitude of about 550 feet. The pilot then made a good landing. This type of maneuver, involving prolonged periods of hovering, can be costly in fuel consumption. The "characteristic velocity" associated with hovering can be computed very simply by the following:

For hovering near the lunar surface,

$$T = mg_m \tag{1}$$

or

$$\dot{m}g_eI_{sp} = mg_m$$

from which

$$\log_e \frac{m_o}{m} = \frac{g_m}{g_eI_{sp}}t \tag{2}$$

The characteristic velocity is defined by

$$\Delta V = g_eI_{sp} \cdot \log_e \frac{m_o}{m} \tag{3}$$

From equations (2) and (3),

$$\Delta V = g_m t \tag{4}$$

This shows a characteristic velocity ΔV of 319.2 feet per second for each minute of hovering.

In general, the early flights terminated with impact velocities from 0 to 30 feet per second in rate of descent, and from 0 to 10 feet per second in circumferential velocity.

In concentrating on obtaining a soft landing, the pilots often neglected the range plot, and landed anywhere from "on target" to distances up to 6 miles from the desired landing site. It appeared that it would be difficult for one pilot to monitor and interpret the displays and to control the vehicle properly at this stage of development. In order to relieve the pilot of some of his duties, the throttle was moved to the outside of the cockpit and was controlled by a second pilot. With two pilots, the problem of obtaining soft landings was relatively easy. During the vertical descent, the throttle operator permitted the vehicle to descend slowly until the attitude controller had reduced $R\dot{\bar{\theta}}$ to a low value. The vehicle was then permitted to land. The two-pilot landings removed the occurrence of prolonged hovering periods; however, the difficulty of attaining the proper landing site remained.

The reasons for missing the landing site by such large distances were not immediately apparent. Since the pilots were flying the plot of $\dot{R}$ against h quite closely, it was thought that the range error might be associated with a combination of errors in vehicle attitude and thrust level, and that these errors were integrating over the entire flight time (about 14 minutes). A few flights therefore were made in which the thrust level was very carefully set at the proper value by checking the input to the computer. But even when this was done, the range varied about 6 miles at touchdown. In making some of the two-pilot flights, it was observed that the value of circumferential velocity $R\dot{\bar{\theta}}$ departed from the nominal-trajectory values shortly after initiation of a flight. It became obvious that if the vehicle $R\dot{\bar{\theta}}$ could be matched to the nominal values throughout the flight, the range would of necessity be correct provided the vehicle maintained altitude to the landing site. It was clear that, in order to match the nominal $R\dot{\bar{\theta}}$, it would be necessary to vary the thrust level during the flight. The landing task can be separated into two major portions - one from thrust initiation until the start of the vertical descent, and the other from start of the vertical descent until touchdown. The pilot's tasks then become clear. During the first portion of the flight the throttle operator varies thrust in whatever manner is required to match the vehicle $R\dot{\bar{\theta}}$ values with those of the nominal trajectory. His pertinent displays consist of the $R\dot{\bar{\theta}}$ meter in the cockpit and the plot of altitude against range which indicates the nominal values of $R\dot{\bar{\theta}}$ at various points along the trajectory. The attitude operator varies the vehicle attitude in whatever manner is required to maintain the nominal altitude and rate of descent. During the vertical descent, the throttle operator applies thrust to control the rate of

descent, and the attitude operator varies the vehicle attitude to adjust $R\dot{\bar{\theta}}$ and range.

Two-Pilot Flights

The general procedure followed in making flights utilizing two pilots was as follows: The flight was initiated in a circular orbit at an altitude of 50 statute miles. When the range plotter passed over landmark "A" (fig. 8(b)), the thrust operator applied thrust at the nominal level. From then on he carefully compared the value of $R\dot{\bar{\theta}}$ shown on the instrument display with that shown on the plot for the nominal trajectory. If required, he modulated thrust to match the nominal value of $R\dot{\bar{\theta}}$. Meanwhile, the attitude operator controlled the vehicle attitude in a manner to acquire the proper altitude and rate of descent as given in figure 8(b). Once having acquired the proper altitude and rate of descent, the attitude operator could concentrate on flying the plot of $\dot{R}$ against h (fig. 8(a)) which was more responsive to attitude control than was the range plot. (During the vertical-descent phase, the controls were used as indicated in the previous section.) Close coordination is required between the two pilots, but this is acquired quickly with practice. This landing procedure basically results in uncoupling the range and altitude controls and makes the entire landing procedure rather easy for a two-pilot operation.

It was believed that after some practice in making two-pilot flights and after becoming thoroughly familiar with the landing procedure, it would be possible for individual pilots to perform the entire landing mission. The plan therefore was to make numerous two-pilot flights and then to make numerous one-pilot flights.

RESULTS AND DISCUSSION

The results of this study following the preliminary development flights are separated into several sections for convenience. The sections will be discussed in the order of occurrence in the investigation.

Two-Pilot Flights

Nominal trajectory.- The nominal trajectory was used in the early phases of this study to develop the landing technique and was then used primarily for pilot training and familiarization with the indicated vehicle dynamics, controls, and displays. The flights were initiated in a 50-mile-altitude circular orbit with the vehicle thrust axis alined

with the velocity vector and with a range of about 40 miles to go before thrust was to be applied. This allowed the pilots about 40 seconds to orient the vehicle (if desired) before thrust was to be applied. Results of a typical piloted nominal trajectory are shown in figure 10. The solid line is the nominal or guide trajectory which was traced on the plotter before the flight was initiated. The dashed line is the result of the piloted flight. The results show that the pilots were able to follow the guide traces very closely throughout most of the flight. As pointed out previously, by matching $R\dot{\bar{\theta}}$ very carefully, the pilots should land directly on target. The difficulty was actually one of obtaining high enough resolution on the plot of h against range to give the pilots better guidance near touchdown. Even with the range-to-go scale expanded by a factor of 10, it was impossible to insert sufficient values of $R\dot{\bar{\theta}}$ for guidance because of space limitation on the plotters. Increasing the range scale by another factor of 10 was tried, but this was fruitless because the region of interest in range-to-go fell into the analog output noise level for this parameter. Some range adjustment near the target was possible and was made in several flights. The available resolution on the range-to-go scale permitted reading the range to within about ±500 feet.

A summary of touchdown conditions obtained with two pilots flying the nominal trajectory is given in figure 11. Figure 11 shows that the pilots generally made landings with velocity components $\dot{R}$ and $R\dot{\bar{\theta}}$ of less than 4 feet per second and within a range of about 2,000 feet of the desired landing site. A perfectly flown nominal trajectory requires a characteristic velocity of about 6,300 feet per second. The results of the piloted flights show characteristic velocities from 6,340 feet per second to 7,000 feet per second (0 to 10 percent in excess of that required for a perfectly flown trajectory). A study of the flight records showed that this large spread in characteristic velocity is closely associated with the manner in which the pilots permitted the rate of descent to vary with altitude close to the landing site. When the pilots reduced the rate of descent more rapidly than the nominal variation (see fig. 12, for example), the characteristic velocity tended to be high. The reason for this is that a low rate of descent for an appreciable period of time requires that thrust be applied against the gravity vector for prolonged periods. This is comparable to hovering and is expensive in terms of fuel consumption as indicated previously.

Off-nominal trajectories.- A total of nine off-nominal combinations of initial conditions were used in the investigation. The flight history of one off-nominal flight is shown in figure 13. In this particular flight the initial conditions were a rate of climb of 50 feet per second, a circumferential velocity of 5,435 feet per second, and an altitude of 284,000 feet. At flight initiation the thrust operator applied thrust

slightly above the nominal value to attain the proper variation of $R\dot{\bar{\theta}}$. Meanwhile the attitude operator oriented the vehicle to reduce altitude, and managed to acquire the nominal trajectory at an altitude of 250,000 feet. However, at this point the rate of descent was excessive (fig. 13(a)); therefore, the vehicle descended below the nominal trajectory (fig. 13(b)). The attitude operator then slowed the rate of descent and managed to acquire the nominal trajectory with the proper rate of descent at an altitude of about 180,000 feet. From this point on, the pilots flew the nominal trajectory to the landing site.

A summary of conditions at touchdown for some of the off-nominal flights is shown in figure 14. These results are of the same order of magnitude as those for the nominal flights (fig. 11) and show touchdown-velocity components of less than 5 feet per second and landings generally within 2,000 feet of the desired landing site. The rather large spread in characteristic velocity in flights having the same initial conditions is again associated with the flight characteristics near touchdown. Low rates of descent are essentially hovering maneuvers and increase the characteristic velocity appreciably.

One-Pilot Flights

After many two-pilot flights were made, the throttle control was installed in the cockpit, and the investigation was continued with one-pilot flights. The individual pilots first flew the nominal trajectory and then flew the off-nominal trajectories.

Nominal trajectory.- The flight history of a one-pilot flight (fig. 15) shows that the pilot followed the guide traces closely over most of the trajectory. The general comment of the pilots was that more concentration was required in making one-pilot flights than was required in the two-pilot flights. This applied primarily at low altitudes near landing where close monitoring of the $\dot{R}$, $R\dot{\bar{\theta}}$, h, and α displays was required, in addition to control manipulation.

During the entire investigation, up to the one-pilot flights, the range parameter $\left(\text{Range} = \int R_m \frac{d\bar{\theta}}{dt}\, dt\right)$ had been computed for comparison with the nominal range, but no attempt had been made to determine the lateral displacement of the landing point from the nominal-trajectory plane. At this stage, therefore, two additional display meters were added, one showing lateral displacement from the nominal-trajectory plane (Δz) and the other showing the rate of displacement ($\dot{z}$). The pilot then had the additional task of monitoring these meters to keep Δz close to zero. This complicated the piloting task still further;

however, the pilot was able to keep the lateral displacement very low (to within about ±200 feet).

In order to determine what the lateral displacement would be if the pilots ignored the Δz meter and simply flew the guide traces, the Δz and $\dot{z}$ meters were covered, but time histories of the parameters were recorded. It is of interest to note that in covering a range of about 400 miles, the pilots rarely landed with lateral displacements in excess of 2,000 feet. It was therefore believed that the additional complications of the Δz and $\dot{z}$ meters were unwarranted; therefore, the meters were removed. However, time histories of Δz and $\dot{z}$ were recorded.

A summary of touchdown conditions for the one-pilot nominal flights is shown in figure 16. The results show that the terminal conditions for the one-pilot flights were of the same order of magnitude as for the two-pilot flights. (Compare figs. 16 and 11.)

Off-nominal trajectories.- Complete records of two of the one-pilot off-nominal flights are given in figures 17 and 18. In addition to the traces obtained from the guidance charts, time histories of a number of the more interesting variables are given. A summary of the touchdown conditions for many of the off-nominal trajectories is given in figure 19. The results shown in figures 17 to 19 are about the same as those for the preceding flights. The touchdown velocities are generally less than 4 feet per second radially and tangentially; the lateral and downrange displacement from the landing site is less than 2,000 feet. The characteristic velocities are, on the average, slightly less than those for the two-pilot flights. This was probably a result of the additional practice and of learning what factors were important in effecting the characteristic velocity.

Constant-altitude deceleration maneuver.- This trajectory was initiated at an altitude of about 7,600 feet with $\dot{R} = 0$ and $R\dot{\bar{\theta}} = 5,570$ feet per second. The pilot's task was simply to hold constant altitude while reducing $R\dot{\bar{\theta}}$ to about 2,000 feet per second, and then to reduce altitude and $R\dot{\bar{\theta}}$ simultaneously and to make a soft landing. The constant-altitude deceleration phase was easy to perform. The descent and landing were essentially the same as for the other types of trajectories. It was concluded that constant-altitude trajectories were of the same order of difficulty to perform as the trajectories in which thrust was applied against the velocity vector.

Thrust Levels Used in Landing

One of the points of interest in this study was the thrust variation used in making landings. The nominal trajectory specified two thrust levels, $T/W_o = 0.143$ and $T/W_o = 0.430$. As indicated previously, the actual thrust levels were varied from these values when the pilots tried

to match the vehicle circumferential velocity with the nominal values, and also near the landing site when the vehicle often was permitted to hover briefly. During the investigation, the pilots deliberately refrained from reducing thrust any more than they believed was required. Near the landing site, two conflicting interests appear if the vehicle reaches a hovering condition somewhat above the landing site. One is the desire to reduce thrust to a very low value to permit a quick descent to the surface. This would be economical, but requires a high range of throttleability. The second interest is to reduce thrust only slightly below the vehicle weight. This would prove costly because of the slow descent, but would not require a very large range of throttleability. During the present investigation the throttleability range used by the pilots varied from about 7:1 to 10:1. It was estimated that for the maximum thrust-to-weight ratio available in this study, a throttleability ratio of about 5:1 would be required for a perfectly flown trajectory. A discussion of throttleability requirements for lunar landings is given in appendix B.

CONCLUSIONS

A fixed-base-simulator study had been made of the ability of pilots to perform soft lunar landings. The study permitted all six rigid-body degrees of freedom of the landing vehicle. The pilot was given control over the vehicle thrust level and moments about each of the three body axes. It was assumed that the vehicle had no automatic damping or control. The pilot's task was to acquire and fly a particular trajectory and to perform a soft landing in a specified area. The task was performed either by one pilot in complete control of vehicle or by two pilots with one controlling vehicle attitude and the other controlling the thrust level. Results of this study have led to the following conclusions:

1. The pilots could consistently make good lunar landings, and generally landed with touchdown-velocity components (radial and tangential) of less than 4 feet per second and within a range of about 2,000 feet of the desired landing site. This range variation was associated primarily with the readout resolution of the indicators shown to the pilot.

2. There was no appreciable difference in touchdown conditions or fuel consumption between the one-pilot and the two-pilot flights. However, one-pilot lunar landings required close concentration on the part of the pilot in order to attain acceptable touchdown velocity components, vehicle attitude, and the desired landing site. The landing task was made much easier by using two pilots with one operating the throttle and the second operating the attitude control.

3. It was possible to effectively uncouple the range and altitude control by using the throttle for range control and the attitude control

to adjust altitude over most of the landing trajectory. During the vertical-descent phase, the throttle was used to control altitude and rate of descent, and the attitude control was used to adjust range and circumferential velocity.

4. The characteristic velocity for piloted lunar landings varied from 0 to 10 percent above the velocity computed for a perfectly flown trajectory.

5. A throttleability ratio of from 7:1 to 10:1 was used in making lunar landings. It was estimated that for the maximum thrust-to-weight ratio available in this study, a throttleability ratio of about 5:1 would be required for a perfectly flown trajectory.

Langley Research Center,
National Aeronautics and Space Administration,
Langley Station, Hampton, Va., July 11, 1962.

APPENDIX A

EQUATIONS OF MOTION

The six equations of motion governing the behavior of the vehicle are presented below. The force equations are written with respect to an inertial-axis system fixed at the center of a nonrotating spherical moon. The moment equations are written with respect to the vehicle body axes.

x-force equation,

$$\ddot{x} = \frac{1}{m}\Big[J_X \cos\theta \cos\psi + J_Y(\cos\psi \sin\theta \sin\phi - \sin\psi \cos\phi) + T(\cos\psi \sin\theta \cos\phi + \sin\psi \sin\phi)\Big] - g_m\left(\frac{R_m}{R}\right)^2 \frac{x}{R}$$

y-force equation,

$$\ddot{y} = \frac{1}{m}\Big[J_X \cos\theta \sin\psi + J_Y(\cos\psi \cos\phi + \sin\psi \sin\theta \sin\phi) + T(\sin\psi \sin\theta \cos\phi - \cos\psi \sin\phi)\Big] - g_m\left(\frac{R_m}{R}\right)^2 \frac{y}{R}$$

z-force equation,

$$\ddot{z} = \frac{1}{m}\left(-J_X \sin\theta + J_Y \cos\theta \sin\phi + T \cos\theta \cos\phi\right) - g_m\left(\frac{R_m}{R}\right)^2 \frac{z}{R}$$

X-moment equation (roll),

$$\dot{p} = \frac{1}{I_X}\left[M_X - p\dot{I}_X - qr\left(I_Z - I_Y\right)\right]$$

Y-moment equation (pitch),

$$\dot{q} = \frac{1}{I_Y}\left[M_Y - q\dot{I}_Y - pr\left(I_X - I_Z\right)\right]$$

Z-moment equation (yaw),

$$\dot{r} = \frac{1}{I_Z}\left[M_Z - r\dot{I}_Z - pq\left(I_Y - I_X\right)\right]$$

The M_X and M_Y terms are control moments which were generated by assumed reaction jets a distance l from the vehicle center of gravity (fig. 1). The control moment M_Z is generated by a pair of reaction jets operating together to produce a pure couple about the Z-axis. It should be noted that as mass is reduced the center of gravity moves along the Z-axis, and hence the moments of inertia and the moment arm l are variable quantities.

In addition to the equations of motion, several auxiliary equations were employed, and these were

$$\dot{\psi} = r\frac{\cos \phi}{\cos \theta} + q\frac{\sin \phi}{\cos \theta}$$

$$\dot{\theta} = q \cos \phi - r \sin \phi$$

$$\dot{\phi} = p + \dot{\psi} \sin \theta$$

$$M_X = lJ_Y$$

$$M_Y = -lJ_X$$

$$M_Z = \frac{\partial M_Z}{\partial \delta_Z}\delta_Z$$

$$J_X = \frac{\partial J_X}{\partial \delta_X}\delta_X$$

$$J_Y = \frac{\partial J_Y}{\partial \delta_Y}\delta_Y$$

$$m = m_o + \int_0^t \dot{m}\, dt$$

$$\Delta V = g_e I_{sp} \log_e \frac{m_o}{m}$$

$$\dot{m} = \frac{T}{g_e I_{sp}}$$

$$T = -\frac{\partial T}{\partial \delta_T} \delta_T$$

$$R = R_m + h = \sqrt{x^2 + y^2 + z^2}$$

$$\dot{R} = \frac{x\dot{x} + y\dot{y} + z\dot{z}}{R}$$

$$R\dot{\bar{\theta}} = \sqrt{\dot{x}^2 + \dot{y}^2 + \dot{z}^2 - \dot{R}^2}$$

APPENDIX B

THROTTLEABILITY RATIO

Practical considerations require that the available maximum thrust be sufficiently high to permit efficient landings and, in addition, that the minimum thrust be somewhat less than required for hovering over the landing site. If the maximum thrust is selected to be a certain value defined by T_{max}/W_o, then the thrust required for hovering depends on the efficiency of the landing maneuver. The throttleability ratio can therefore be determined as follows:

If hovering is to be performed when the vehicle mass is m_H, then the thrust required is

$$T_H = g_m m_H \tag{B1}$$

If the characteristic velocity is known, then the mass m_H can be determined from

$$\Delta V = g_e I_{sp} \log_e \frac{m_o}{m_H} \quad \text{or} \quad m_H = m_o e^{-\Delta V/g_e I_{sp}} \tag{B2}$$

and therefore

$$T_H = g_m m_o e^{-\Delta V/g_e I_{sp}} \tag{B3}$$

The throttleability ratio required to permit hovering after attainment of a characteristic velocity ΔV therefore is given by

$$\frac{T_{max}}{T_H} = \frac{T_{max}/m_o}{T_H/m_o} = \frac{T_{max}/m_o}{g_m} e^{\Delta V/g_e I_{sp}}$$

or

$$\frac{T_{max}}{T_H} = \frac{T_{max}}{W_o} \frac{g_e}{g_m} e^{\Delta V/g_e I_{sp}} \tag{B4}$$

Curves of the throttleability ratio are shown as functions of T_{max}/W_o and ΔV in figure 20. In the present study, I_{sp} was 420 seconds, and the characteristic velocity associated with making an ideal landing with no hovering was 6,300 feet per second. Since the maximum available T/W_o was 0.500, the chart indicates a throttleability ratio of about 4.8. If 2 minutes of hovering are permitted, then the ideal characteristic velocity would be 6,938, and the throttleability ratio would be 5. Values of from about 7 to 10 were actually used in this study. The greater range of throttleability was used by the pilots because in some cases they deviated from the nominal trajectory during the transition to a vertical descent and reduced the rate of descent too rapidly. In order to permit a rapid rate of descent, the pilots reduced thrust below the value nominally required for hovering and hence required a greater range in throttleability.

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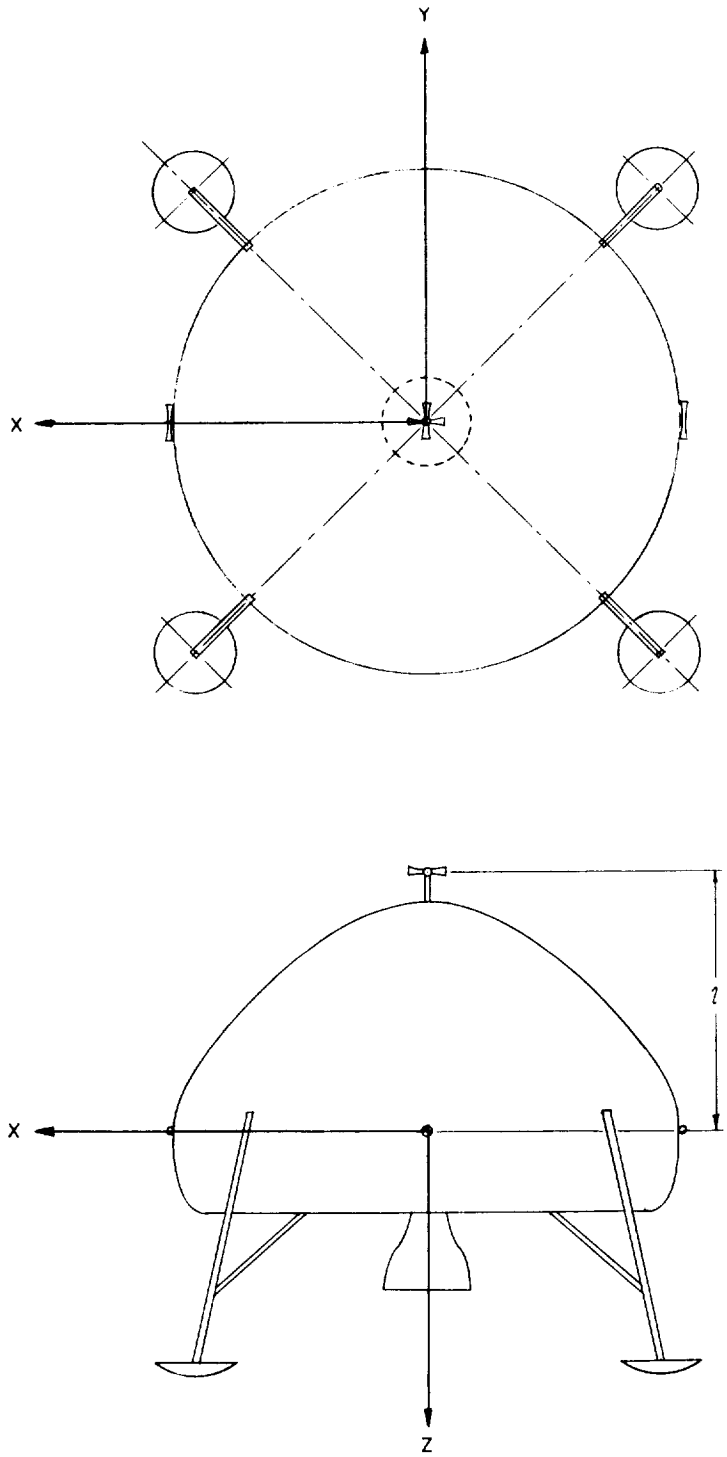


Figure 1.- Lunar landing vehicle.

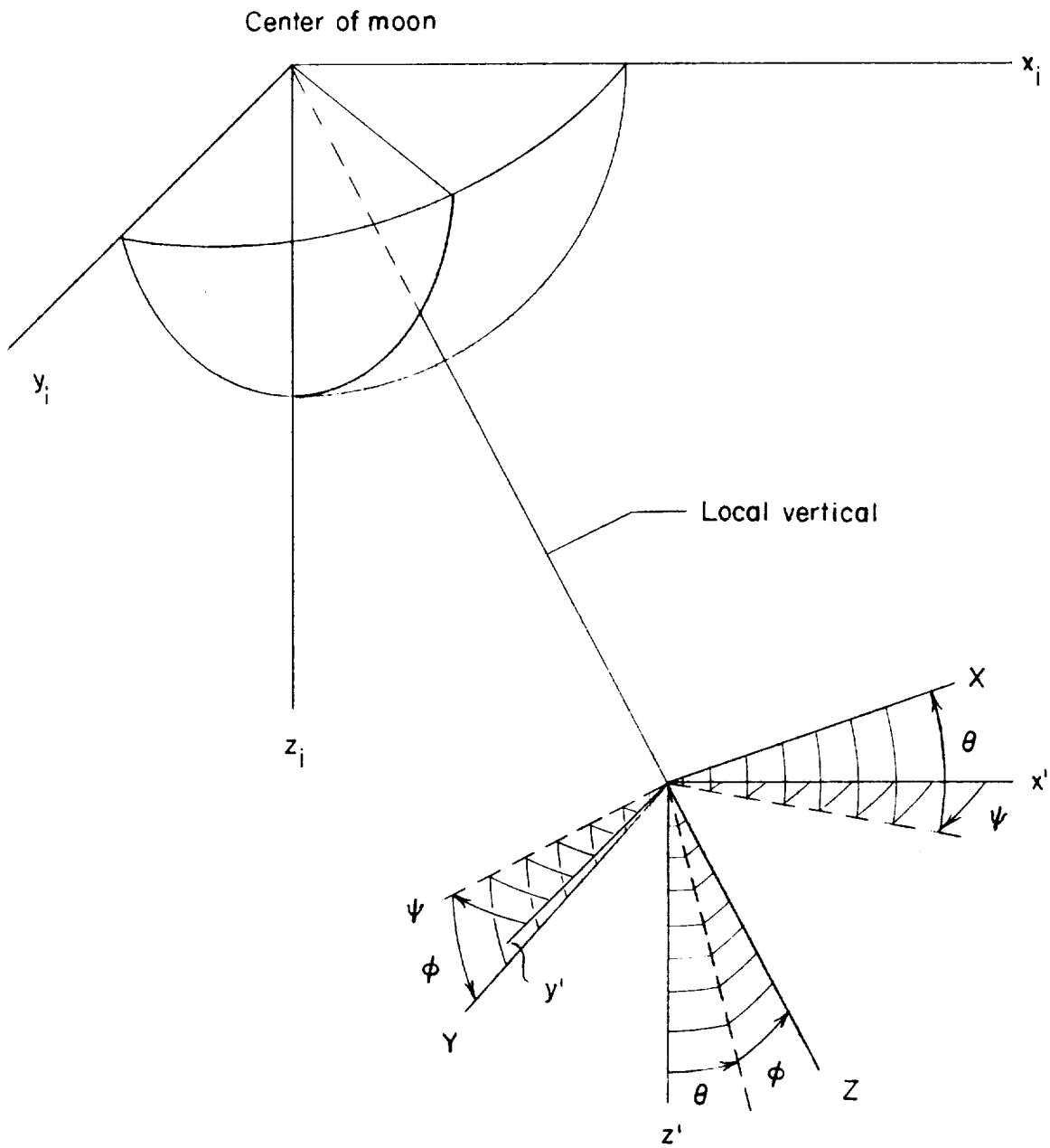


Figure 2.- Inertial-axis and vehicle-axis systems.

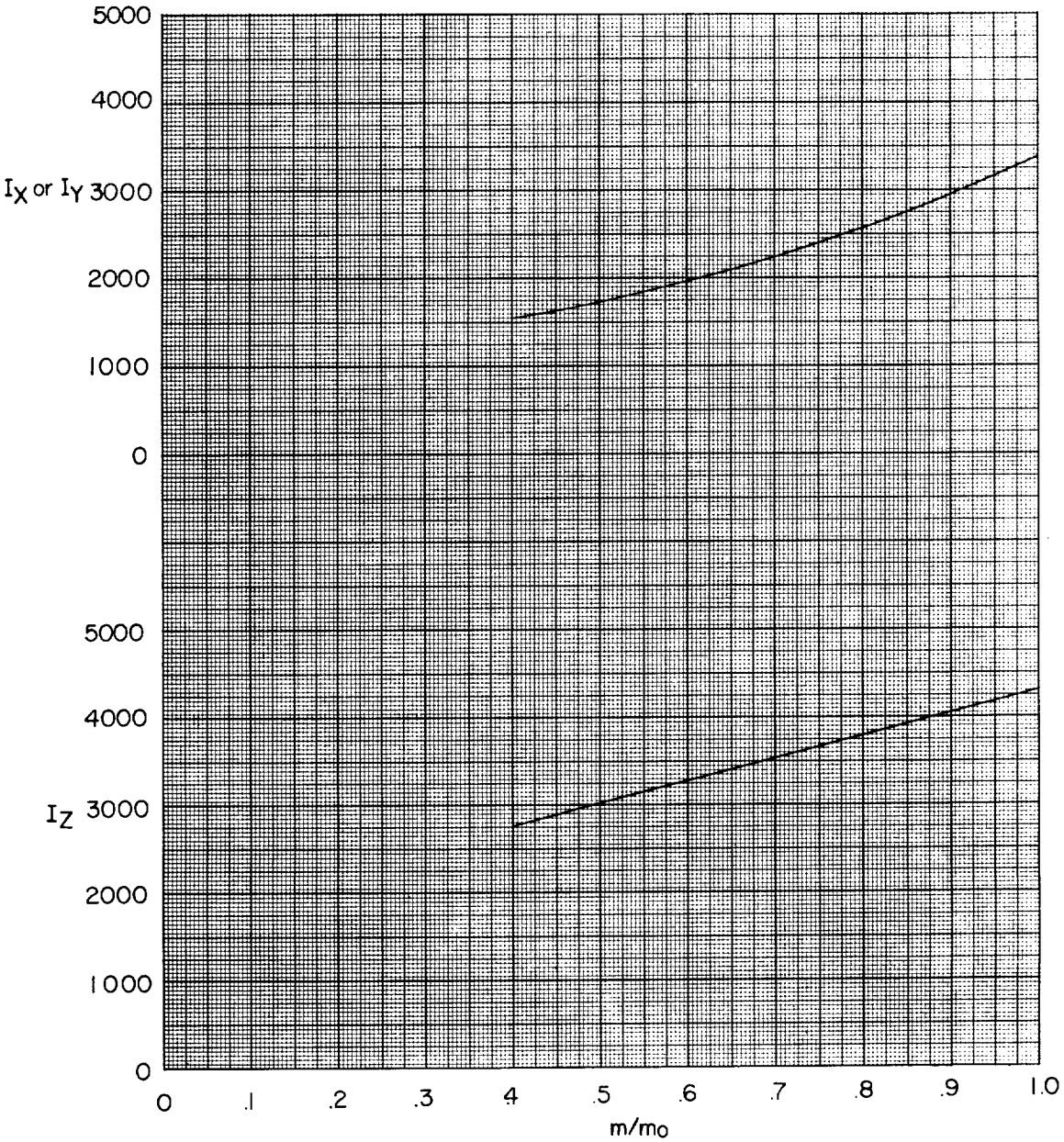


Figure 3.- Vehicle characteristics.

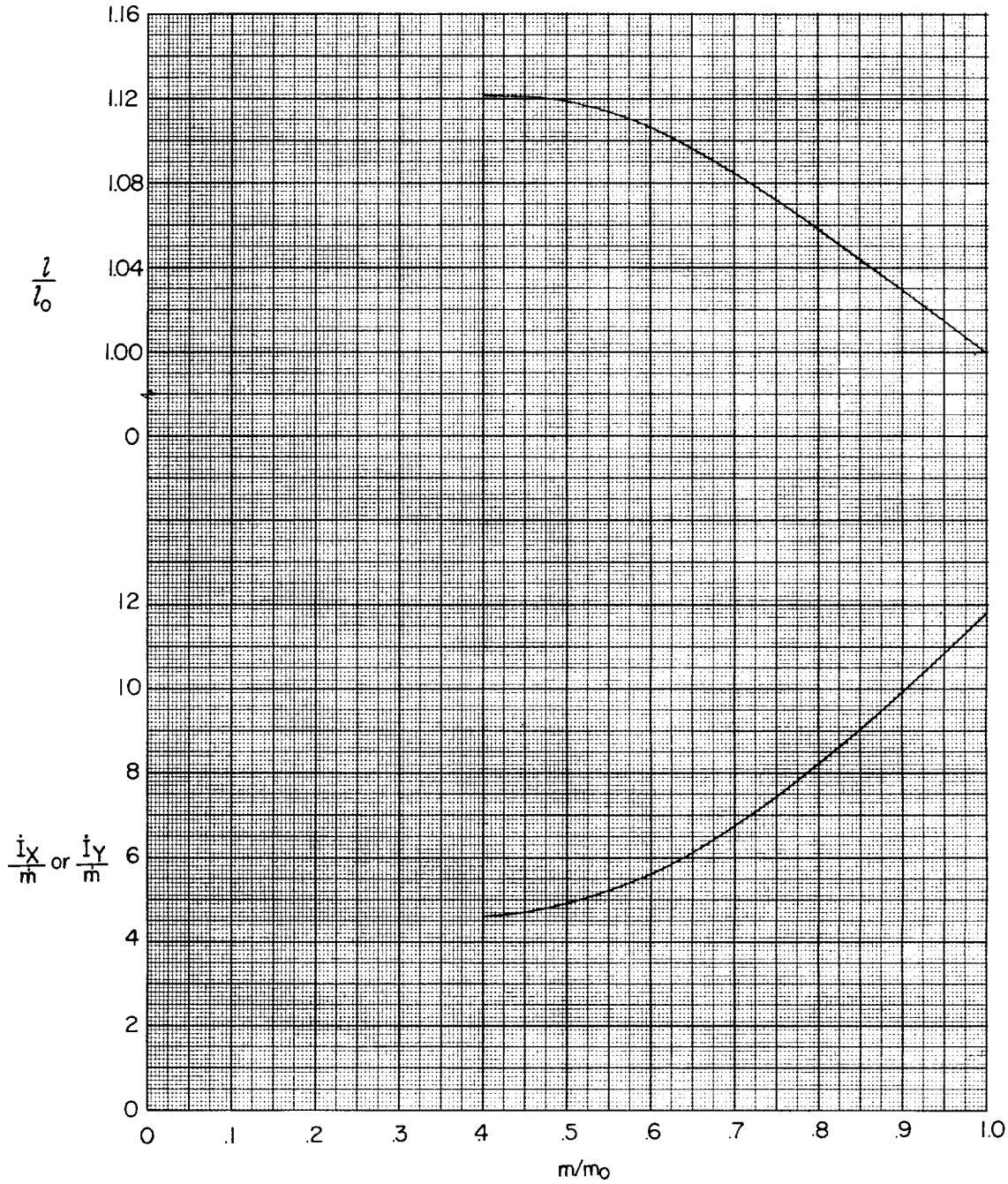


Figure 3.- Concluded.

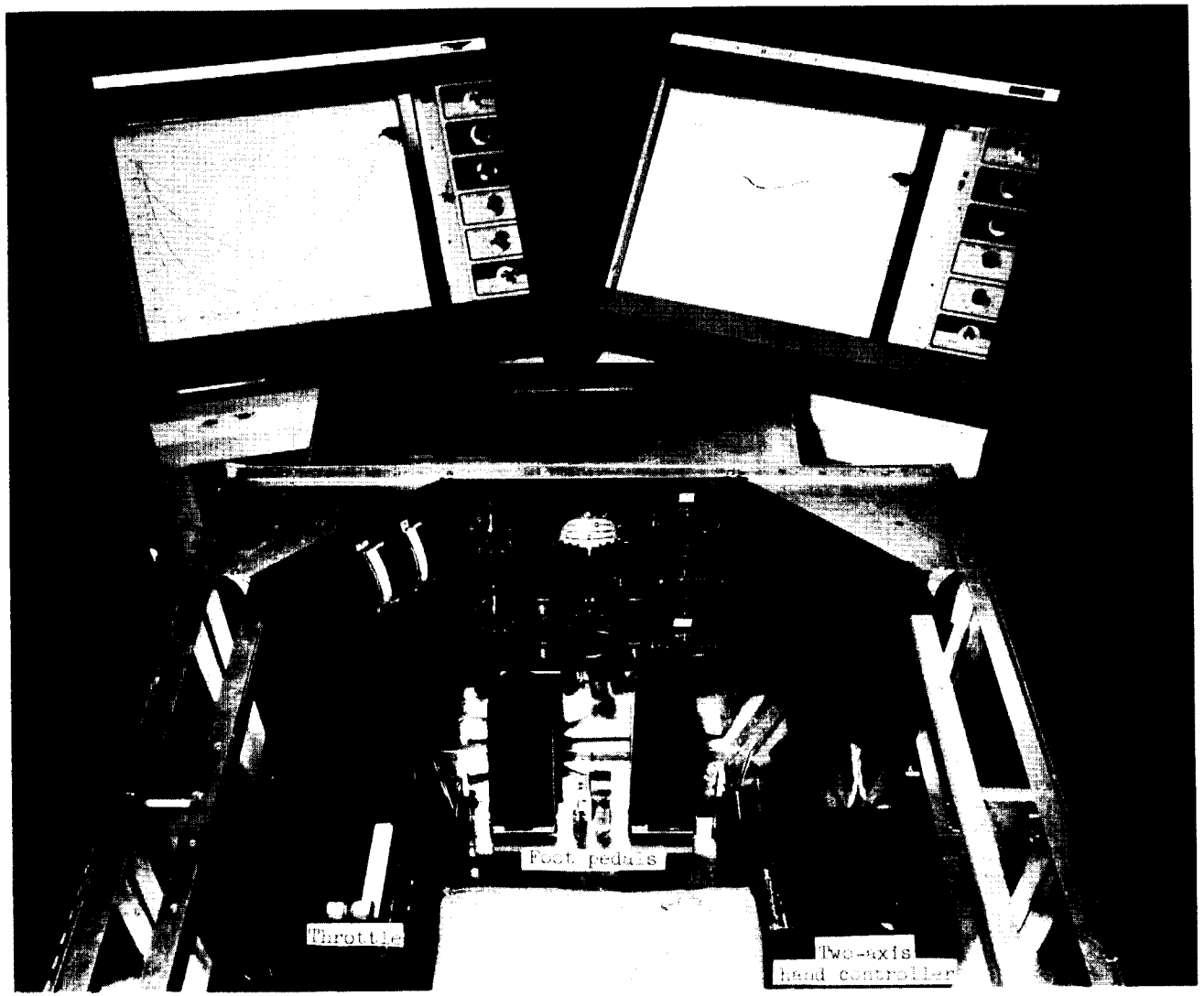


Figure 4.- General layout of cockpit. L-61-8304.1

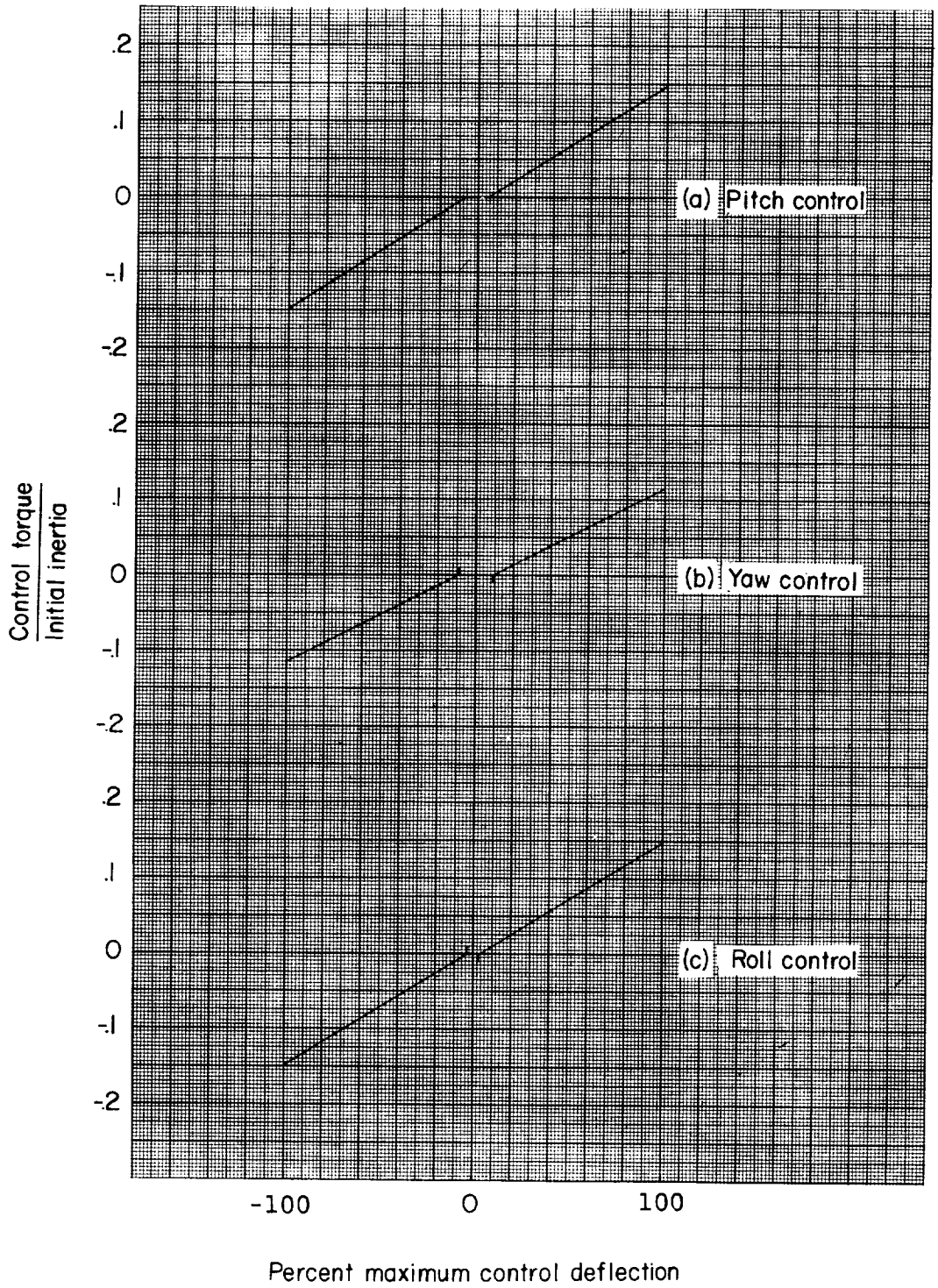


Figure 5.- Initial variations of control torque with controller deflections.

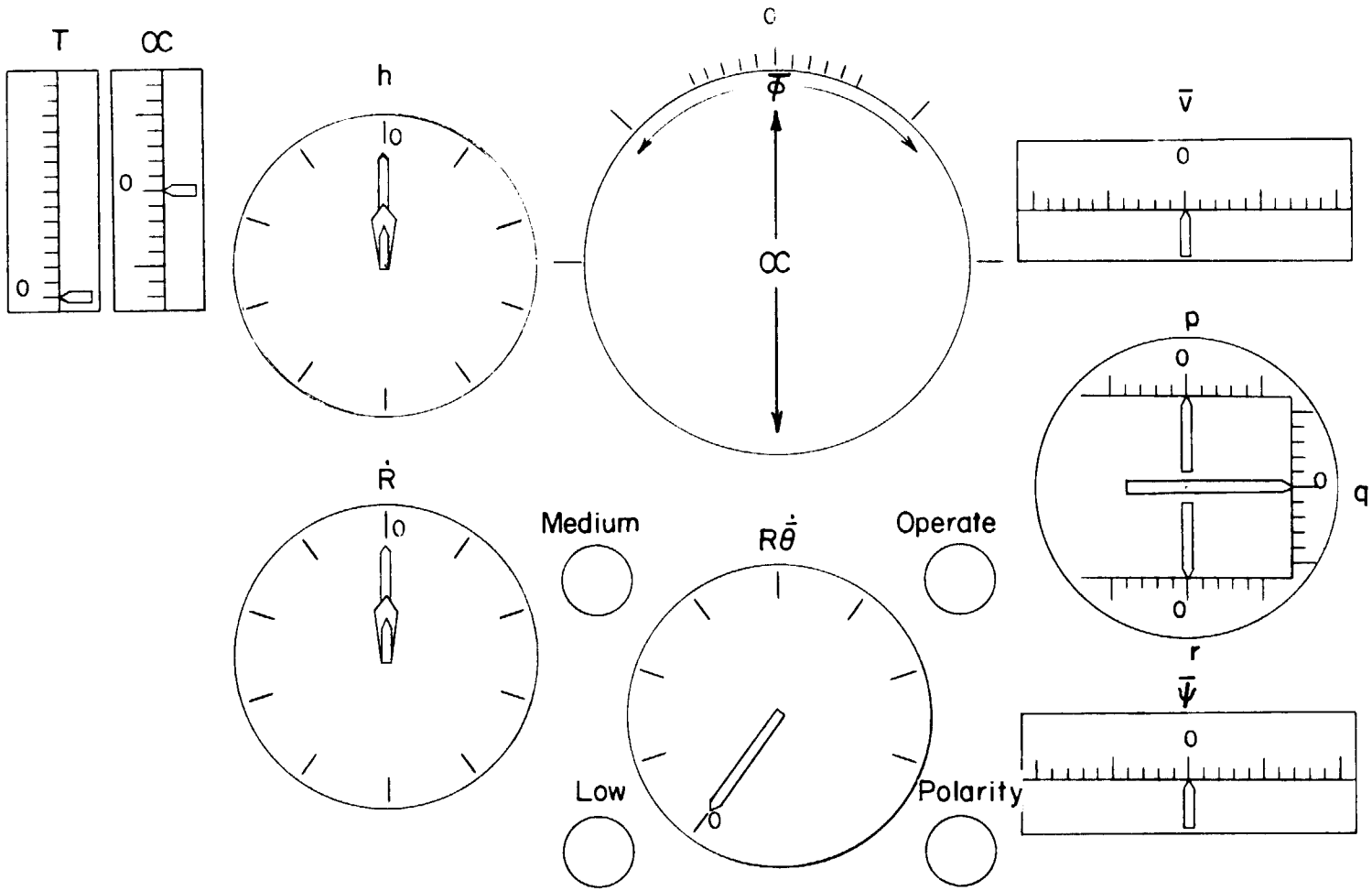


Figure 6.- Instrument panel.

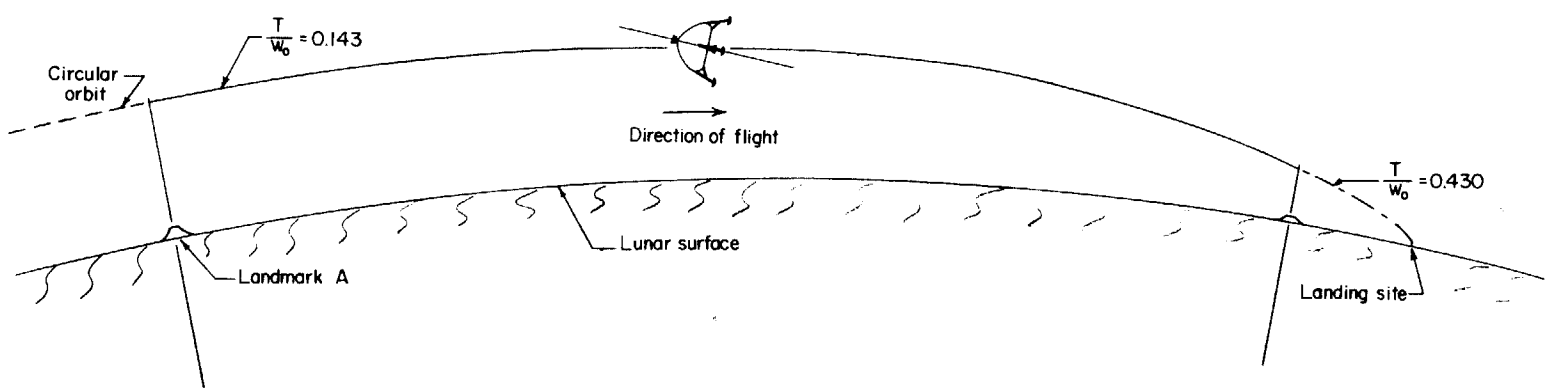
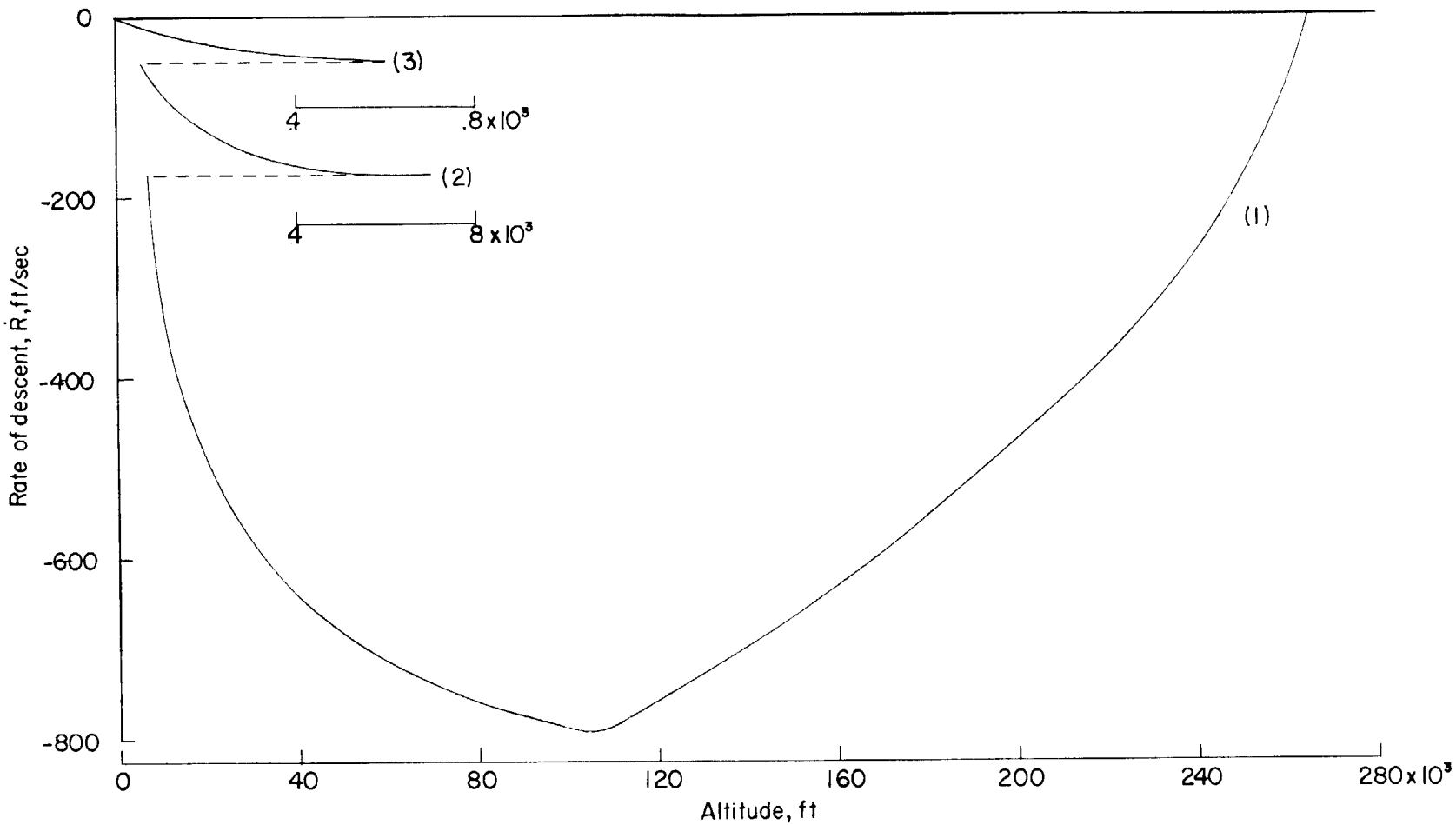
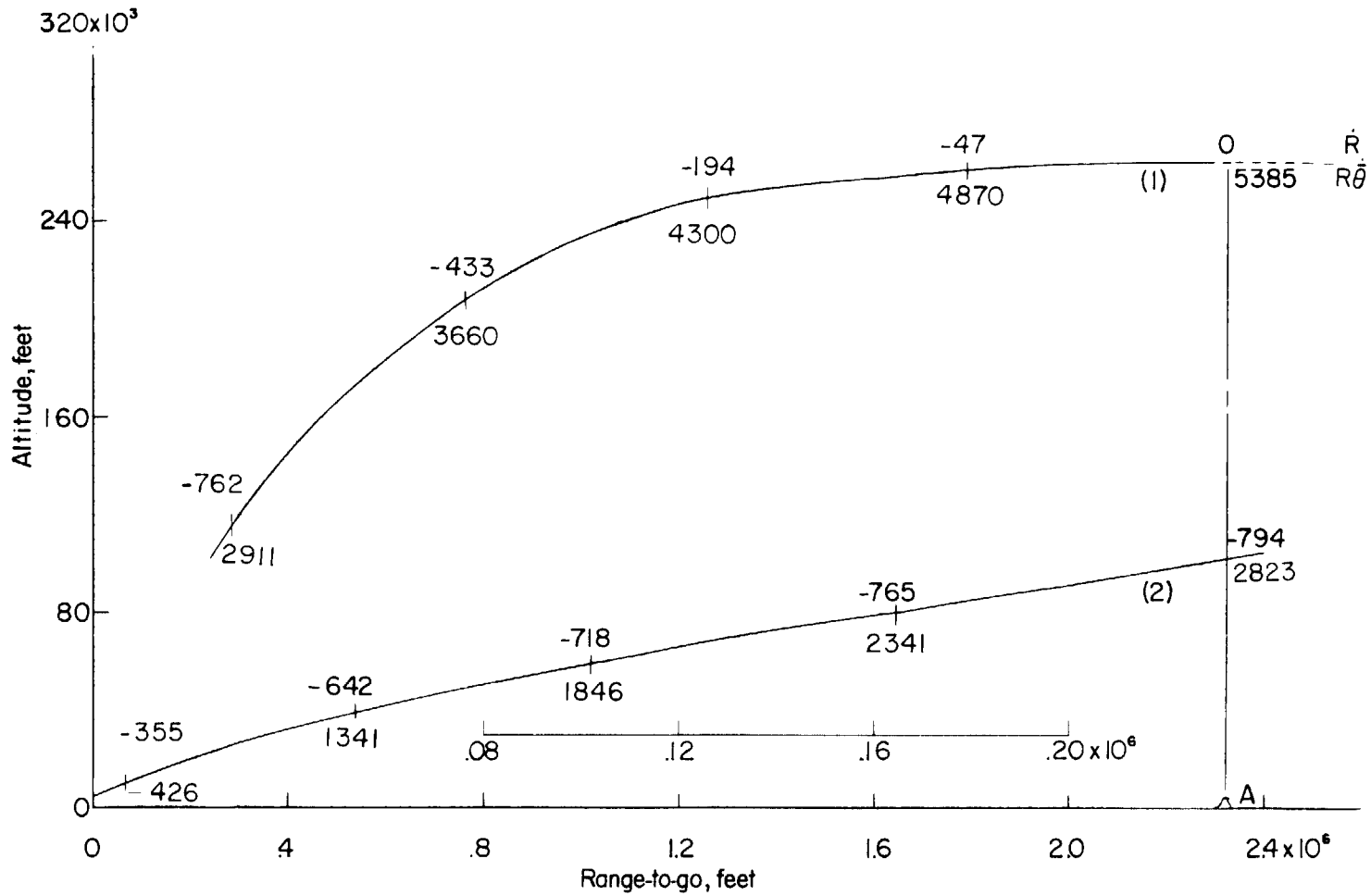


Figure 7.- Nominal gravity-turn landing trajectory.



(a) Variation of rate of descent with altitude. Curves (2) and (3) are magnifications of the final portion of curve (1).

Figure 8.- Characteristics of nominal trajectory.



(b) Variation of altitude with range-to-go. Curve (2) is a magnification of the final portion of curve (1).

Figure 8.- Concluded.

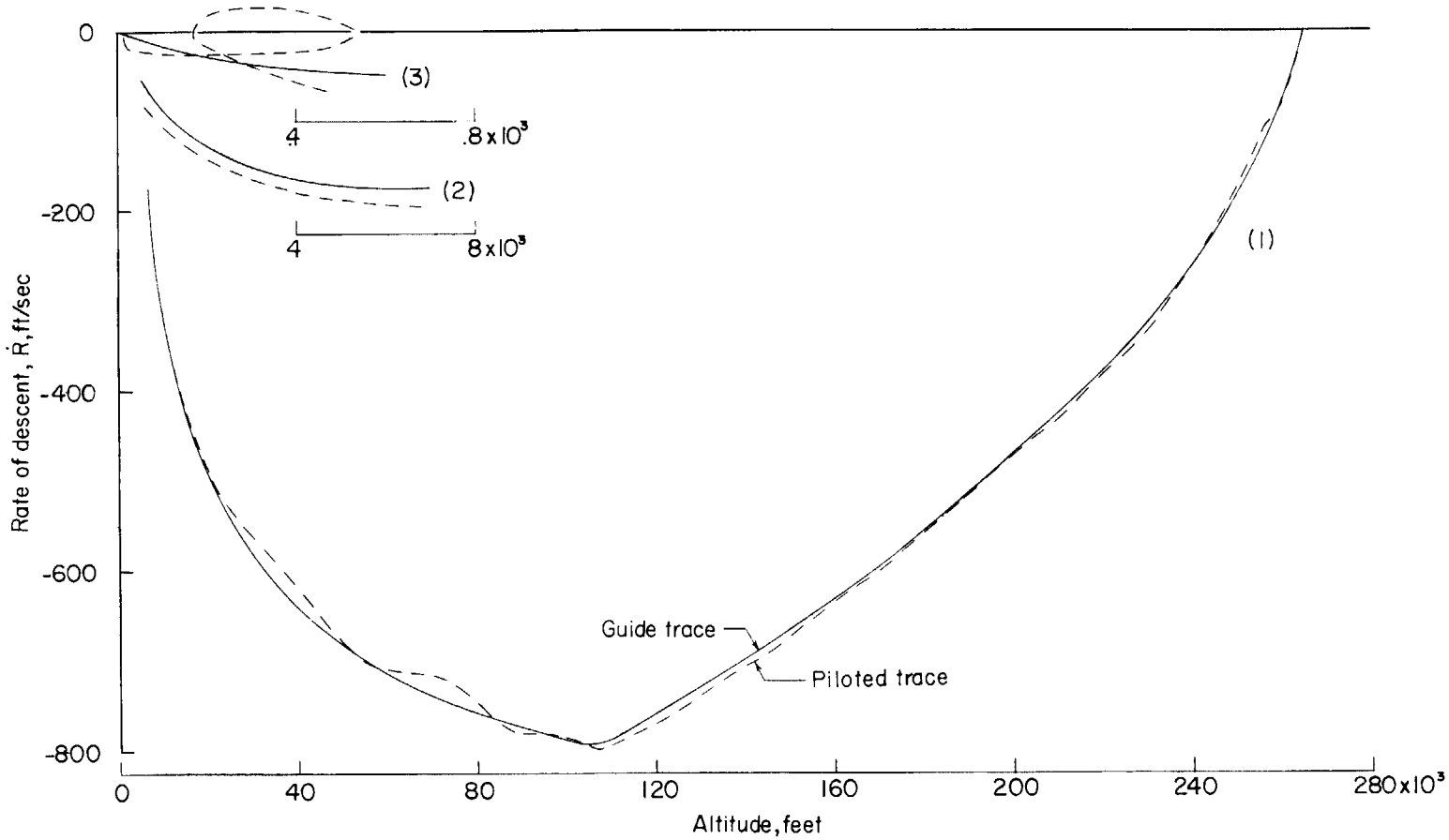
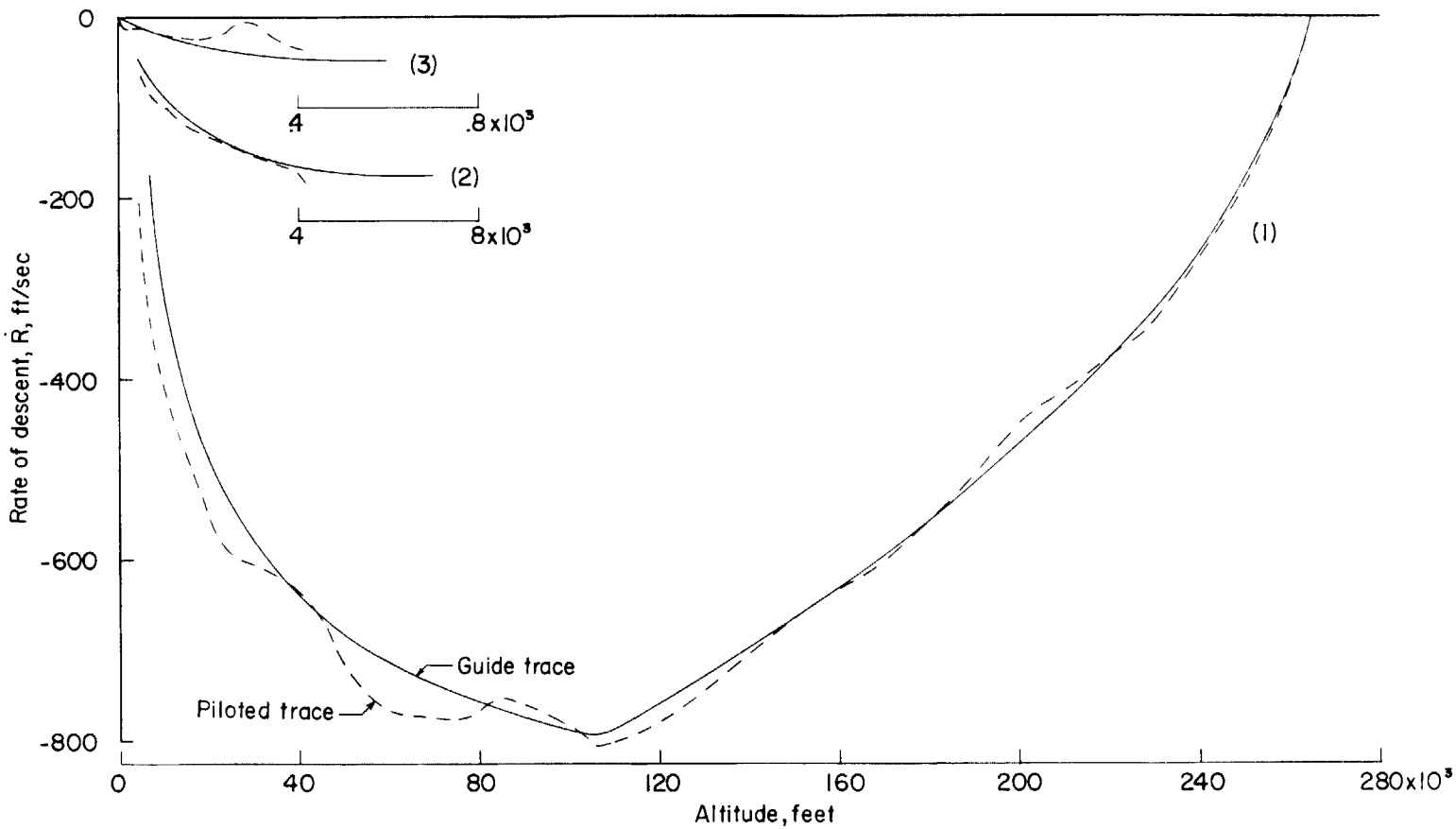
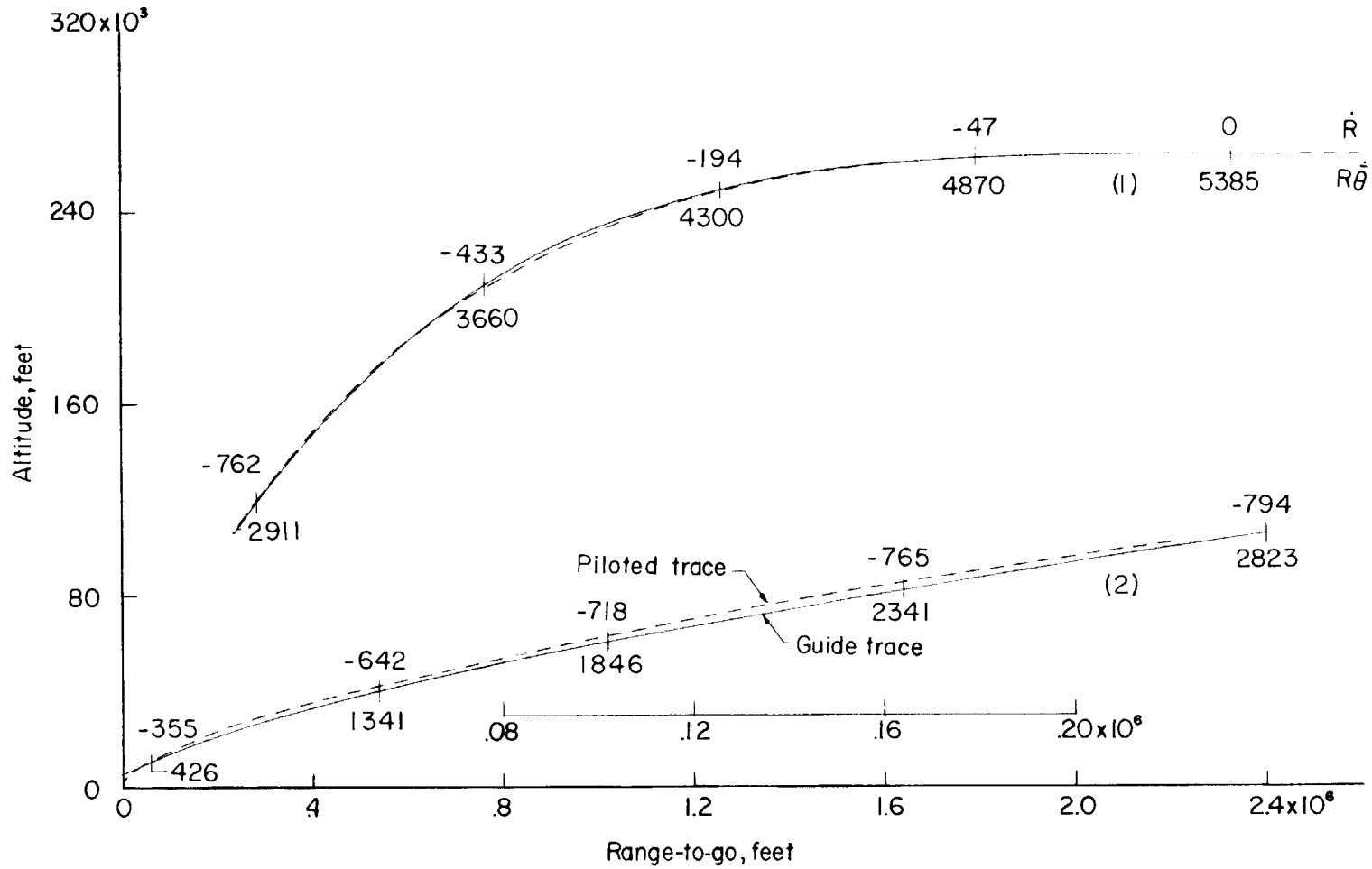


Figure 9.- Flight of nominal trajectory with one pilot. Curves (2) and (3) are magnifications of the final portion of curve (1).



(a) Variation of rate of descent with altitude. Curves (2) and (3) are magnifications of the final portion of curve (1).

Figure 10.- Typical nominal flight with two pilots. ΔV = 6,410 ft/sec.



(b) Variation of altitude with range-to-go. Curve (2) is a magnification of the final portion of curve (1).

Figure 10.- Concluded.

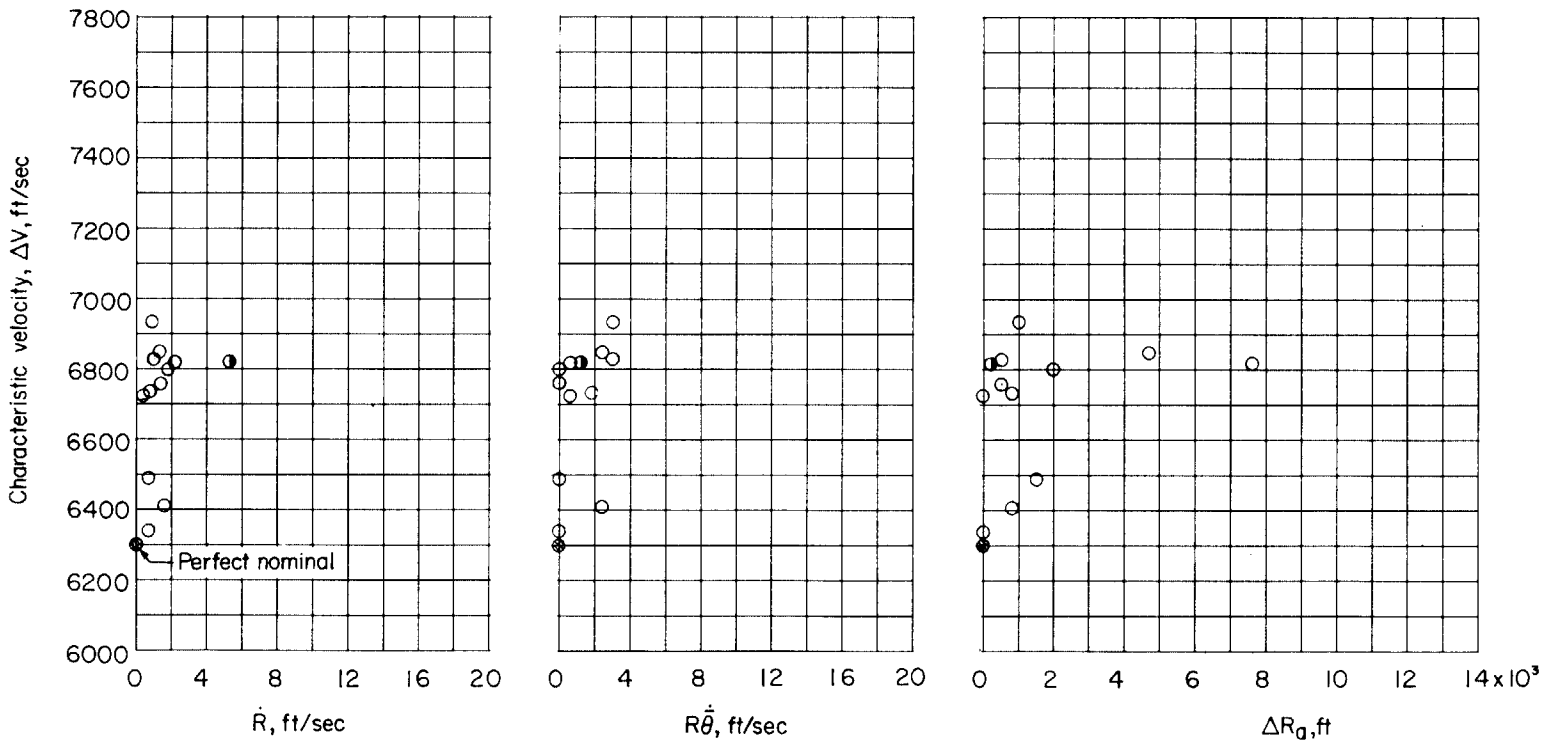
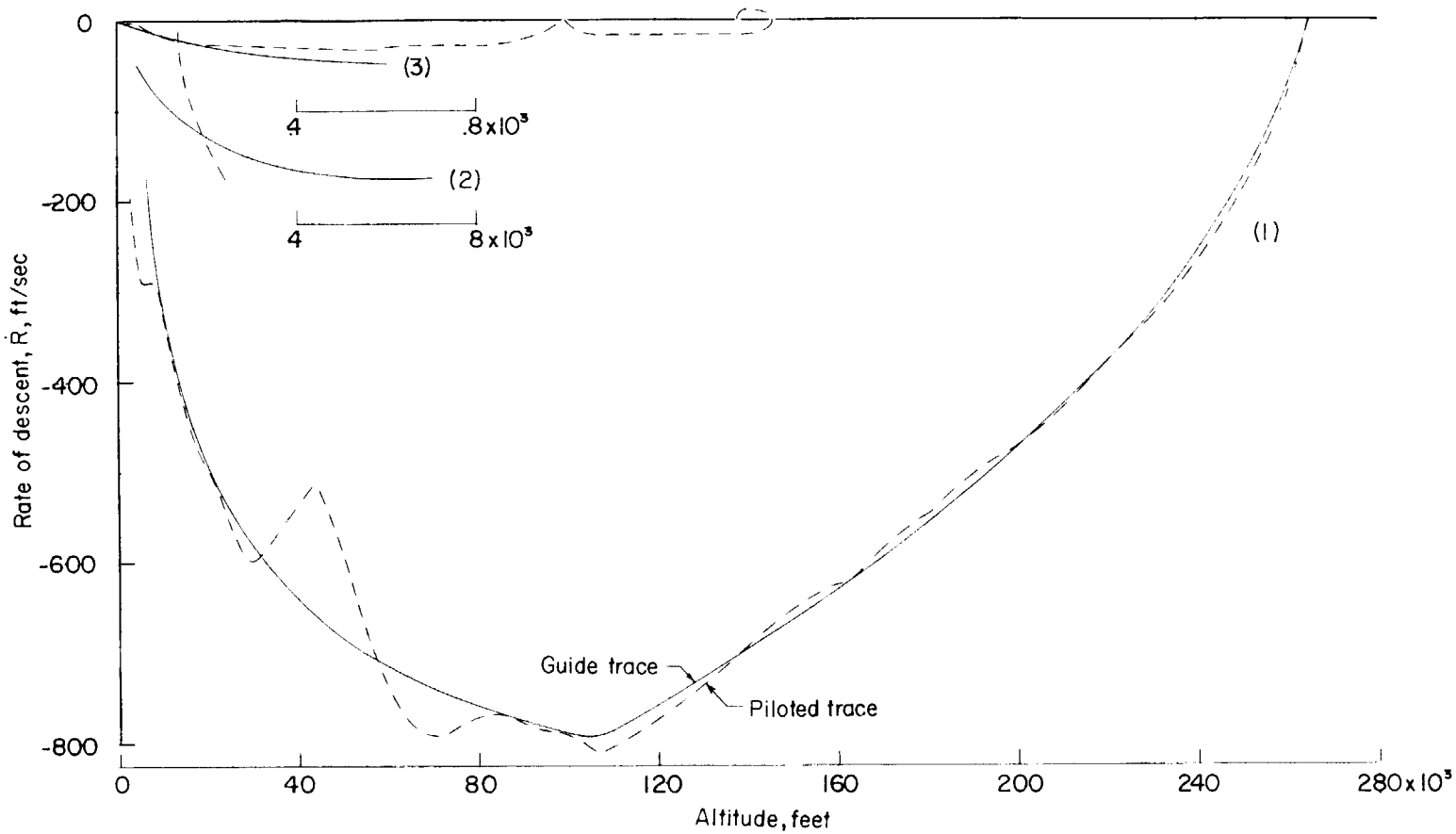
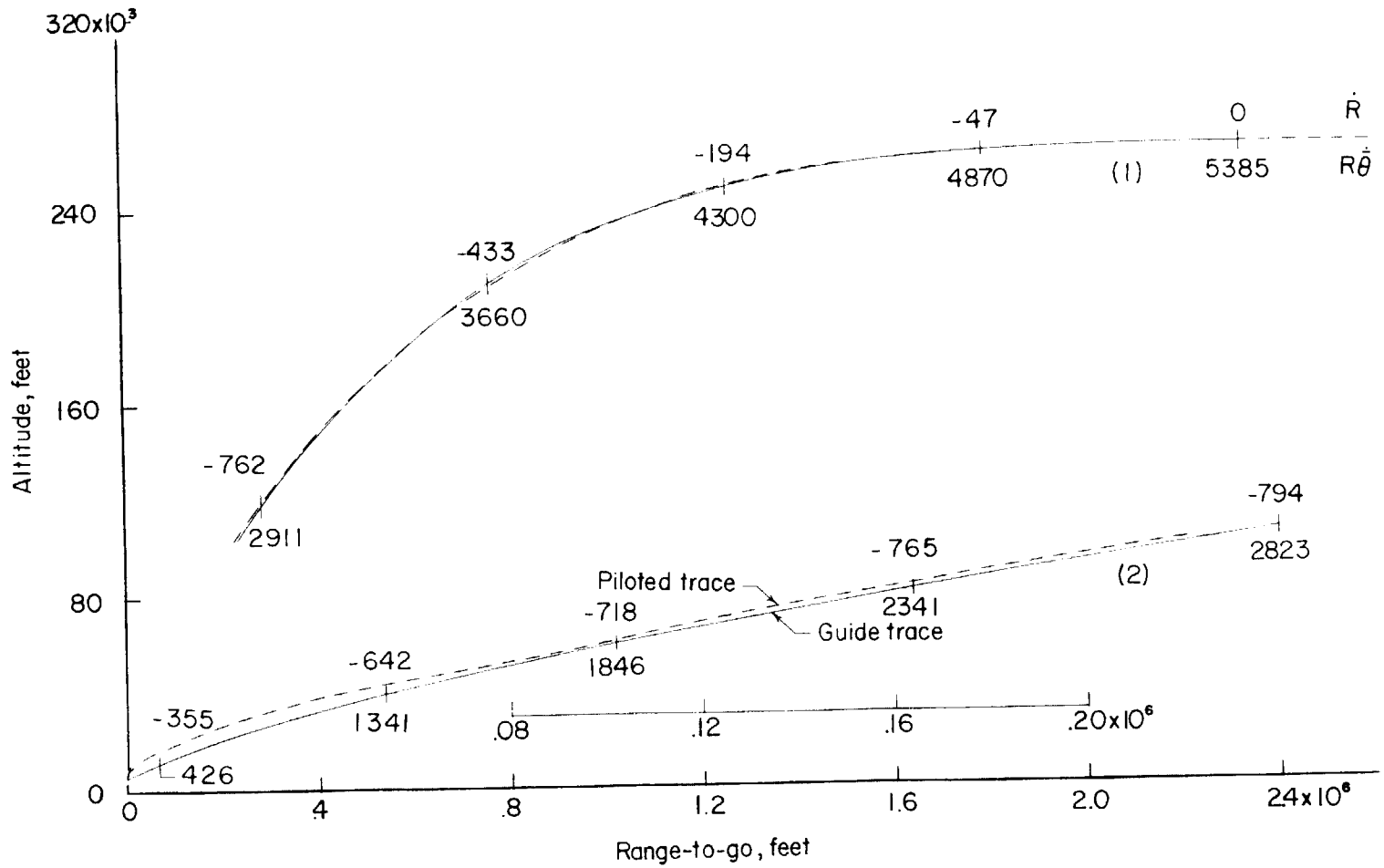


Figure 11.- Touchdown conditions of nominal flights with two pilots. Shading is used to distinguish data for flights having the same characteristic velocity.



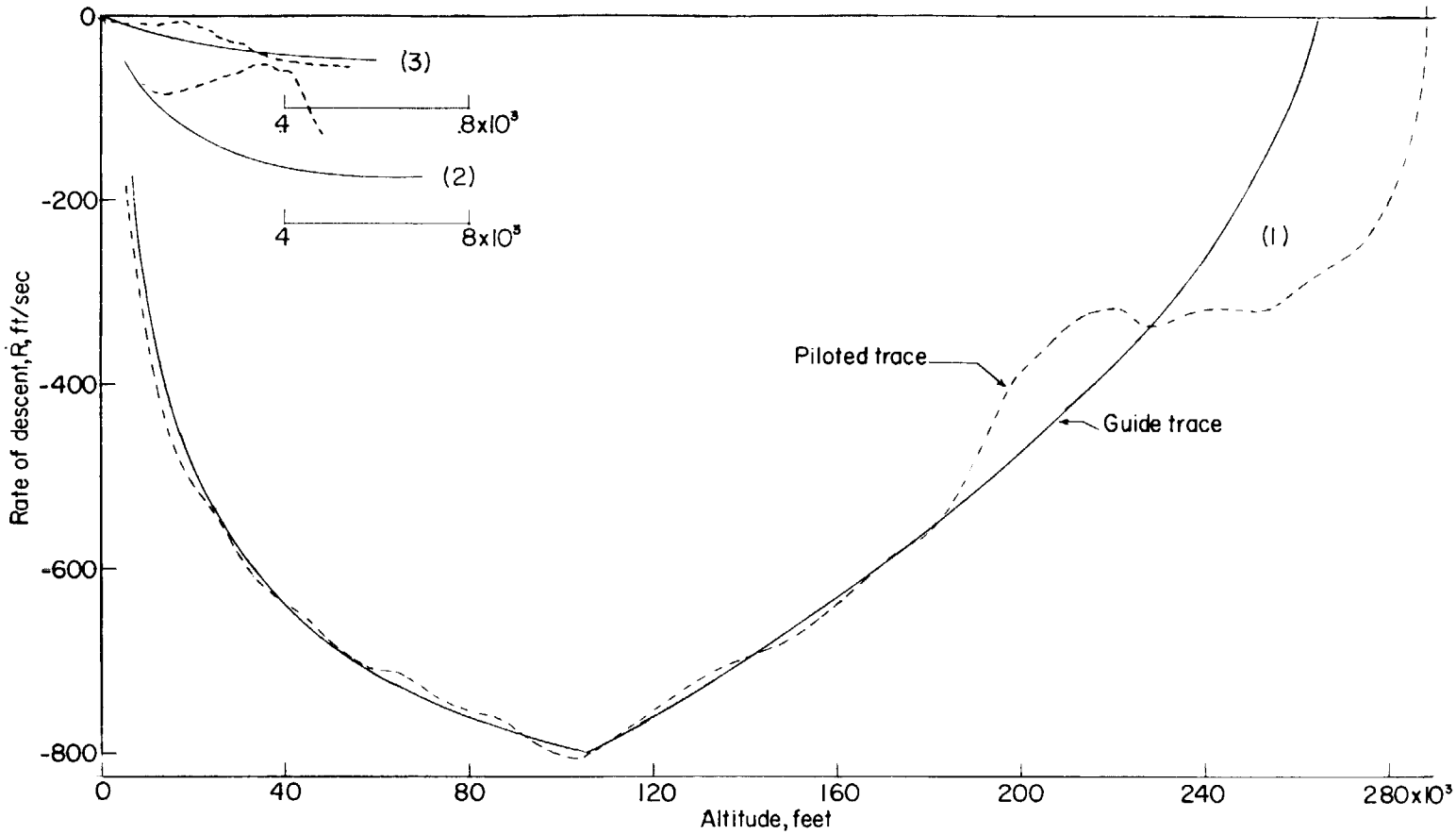
(a) Variation of rate of descent with altitude. Curves (2) and (3) are magnifications of the final portion of curve (1).

Figure 12.- Nominal flight with two pilots. ΔV = 6,935 ft/sec.



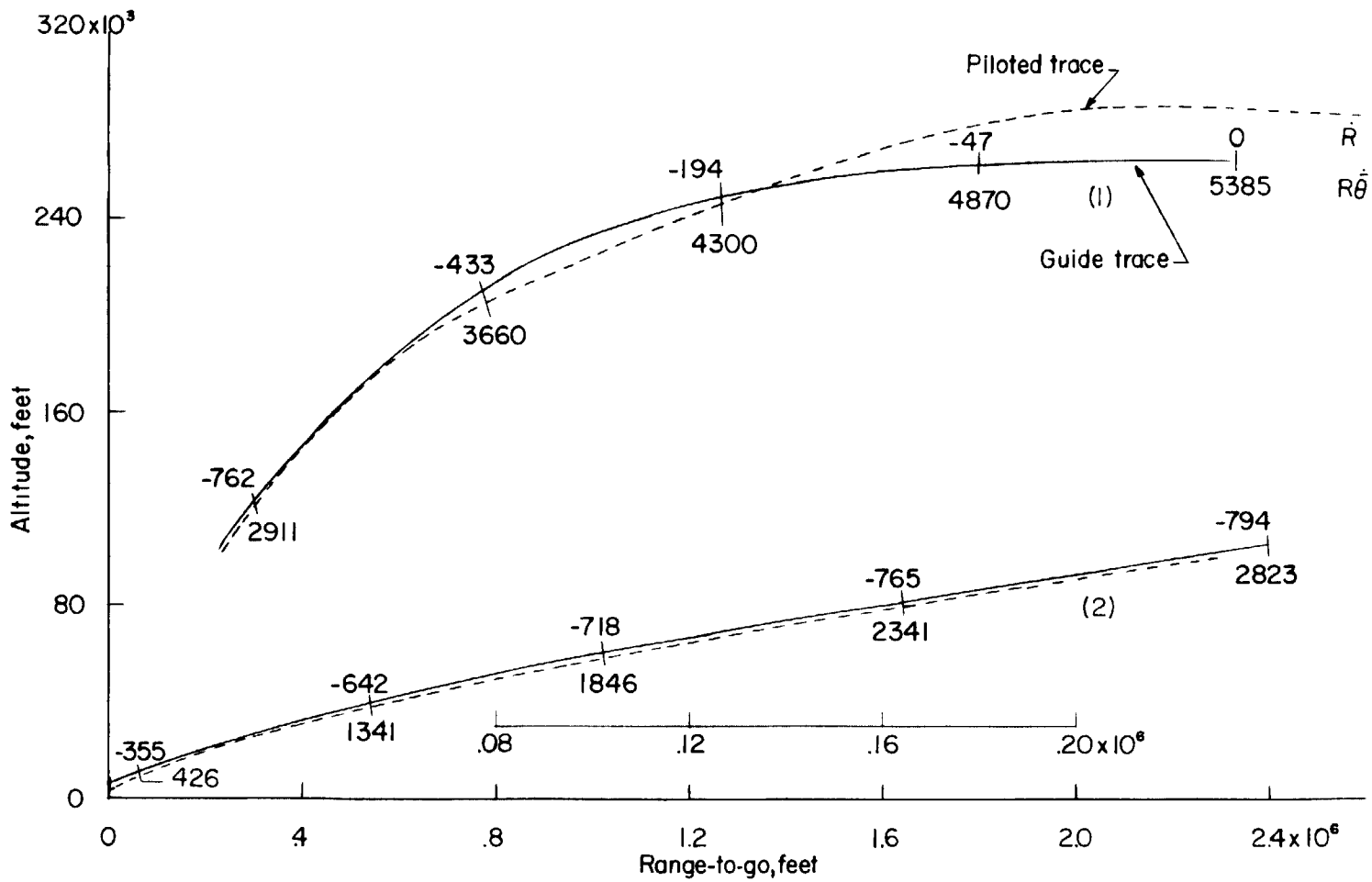
(b) Variation of altitude with range-to-go. Curve (2) is a magnification of the final portion of curve (1).

Figure 12.- Concluded.



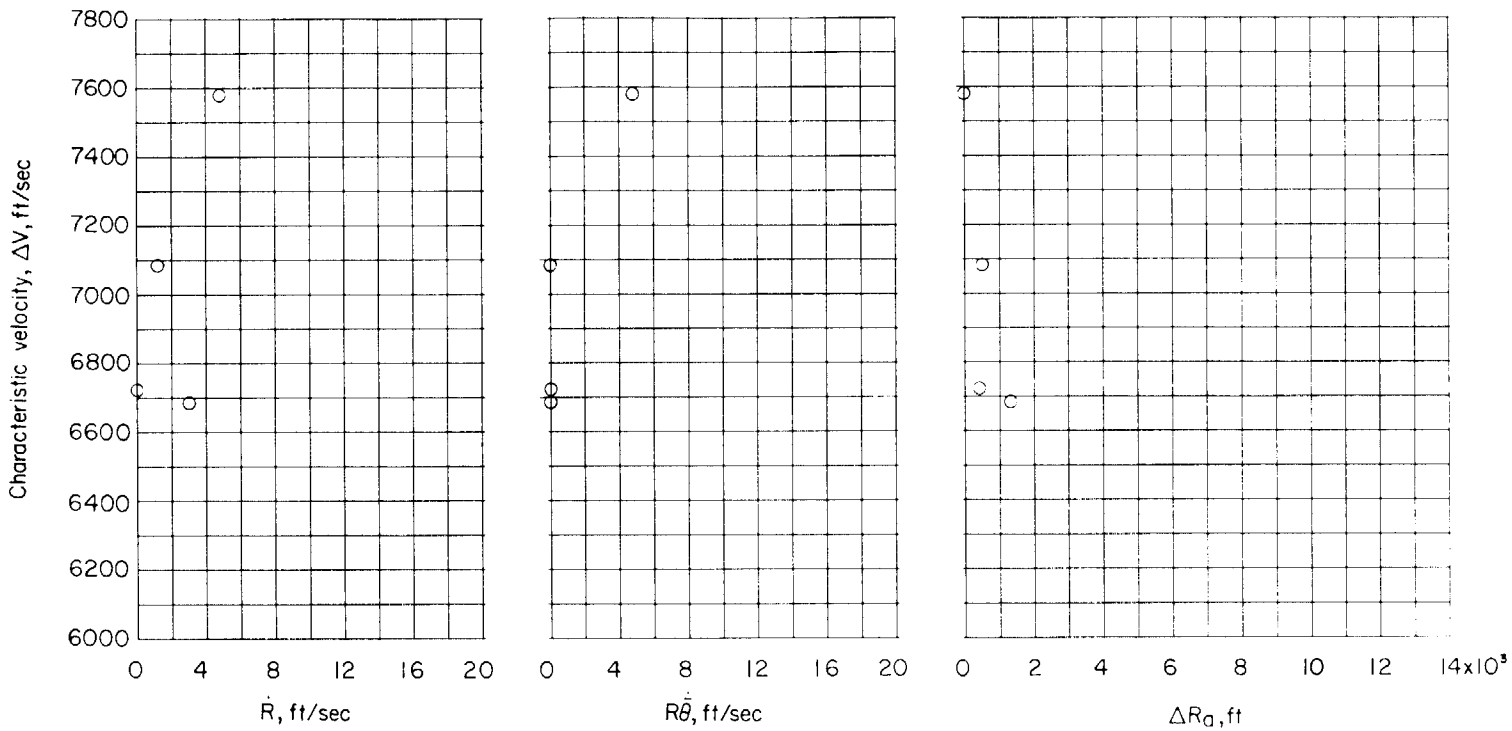
(a) Variation of rate of descent with altitude. Curves (2) and (3) are magnifications of the final portion of curve (1).

Figure 13.- Flight history of off-nominal trajectory with two pilots. $\dot{R}_o$ = 50 ft/sec; $R\dot{\bar{\theta}}$ = 5,435 ft/sec; h_o = 284,000 ft; ΔV = 6,715 ft/sec.



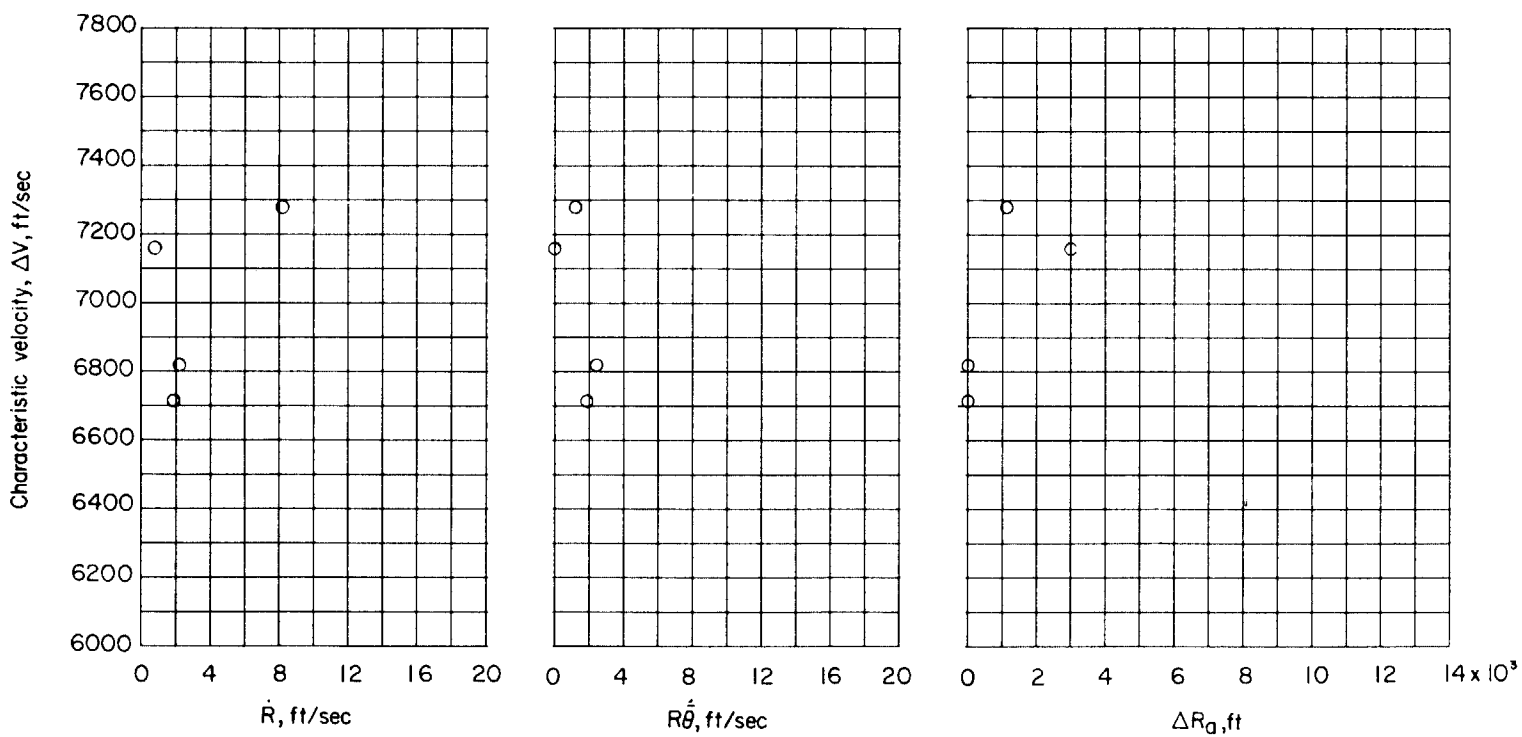
(b) Variation of altitude with range-to-go. Curve (2) is a magnification of the final portion of curve (1).

Figure 13.- Concluded.



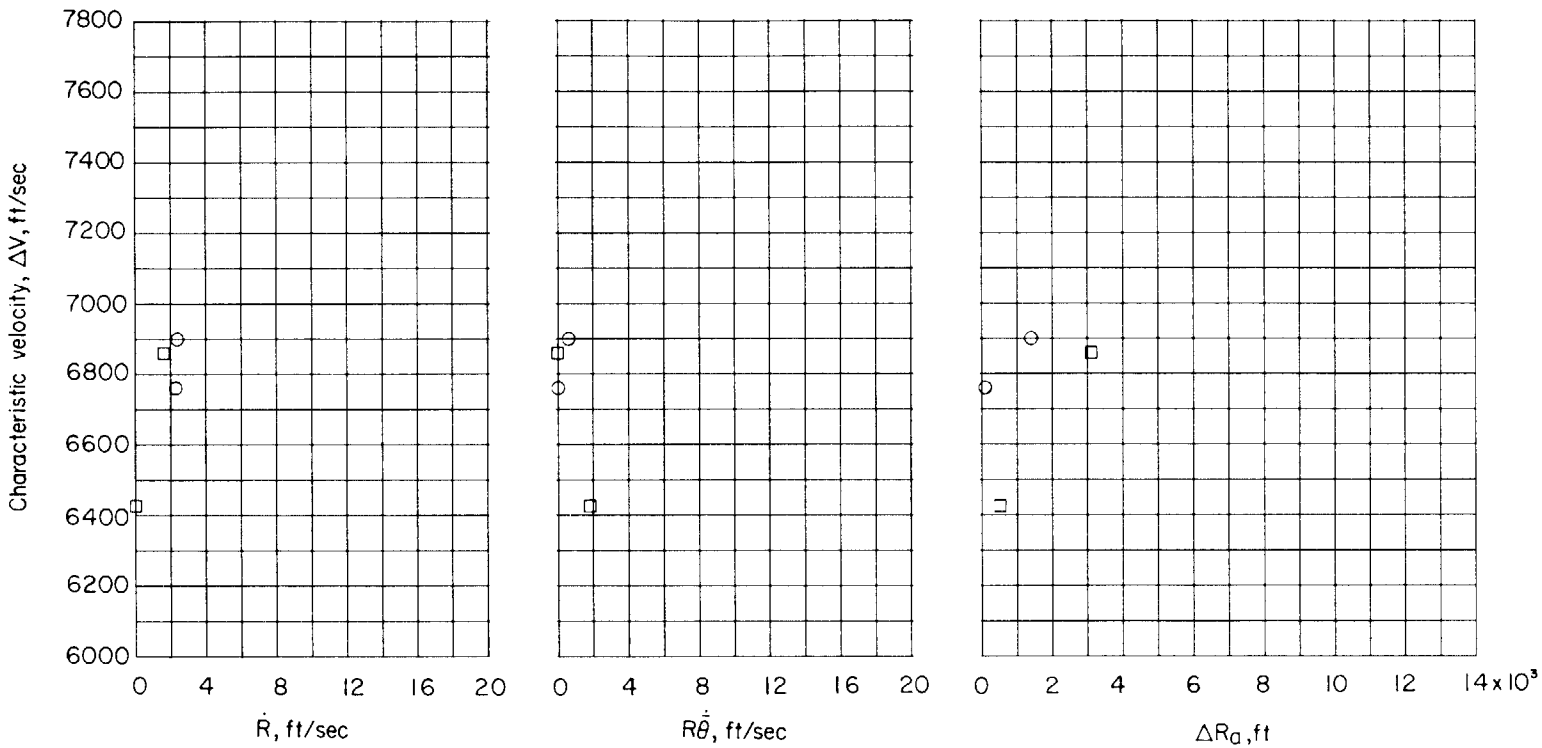
(a) $\dot{R}_o$ = 0 ft/sec; $R\dot{\bar{\theta}}_o$ = 5,385 ft/sec; h_o = 284,000 ft.

Figure 14.- Touchdown conditions of off-nominal flights with two pilots.



(b) $\dot{R}_o$ = 50 ft/sec; $R\dot{\bar{\theta}}_o$ = 5,435 ft/sec; h_o = 284,000 ft.

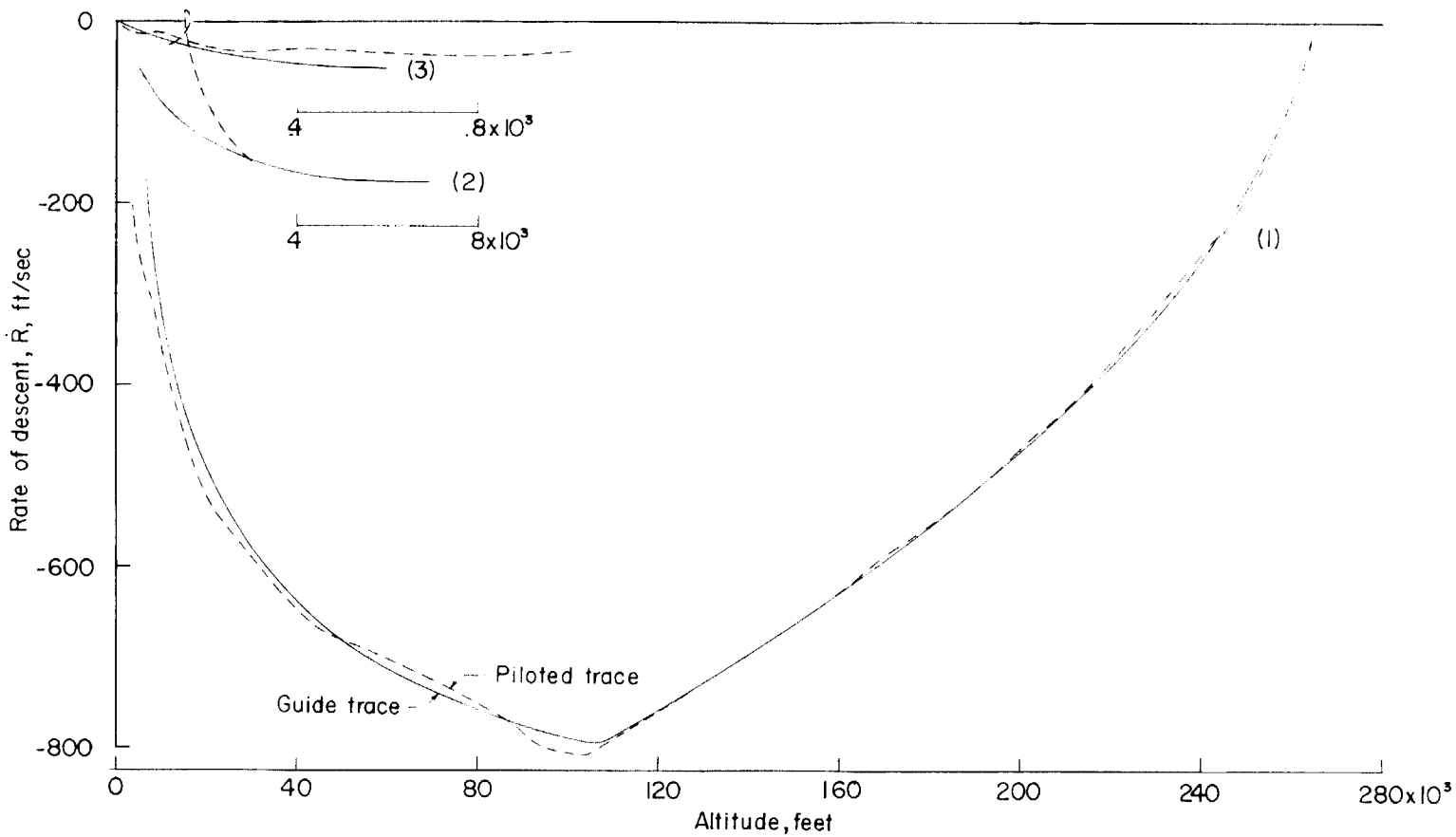
Figure 14.- Continued.



(c) ○ $\dot{R}_o$ = 0 ft/sec; $R\dot{\bar{\theta}}_o$ = 5,385 ft/sec; h_o = 244,000 ft.

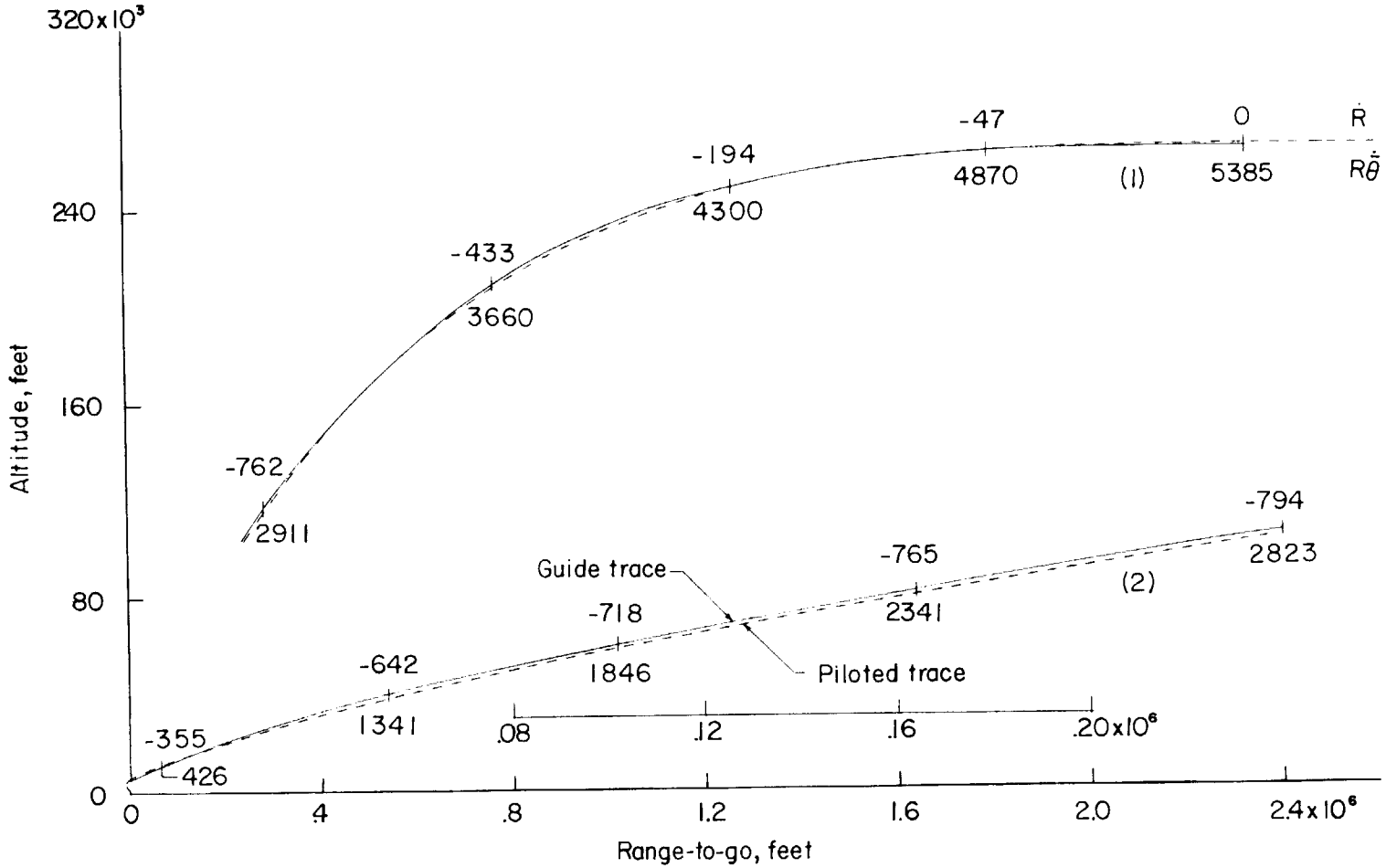
□ $\dot{R}_o$ = -50 ft/sec; $R\dot{\bar{\theta}}_o$ = 5,335 ft/sec; h_o = 244,000 ft.

Figure 14.- Concluded.



(a) Variation of rate of descent with altitude. Curves (2) and (3) are magnifications of the final portion of curve (1).

Figure 15.- Typical nominal flight with one pilot. ΔV = 6,685 ft/sec.



(b) Variation of altitude with range-to-go. Curve (2) is a magnification of the final portion of curve (1).

Figure 15.- Concluded.

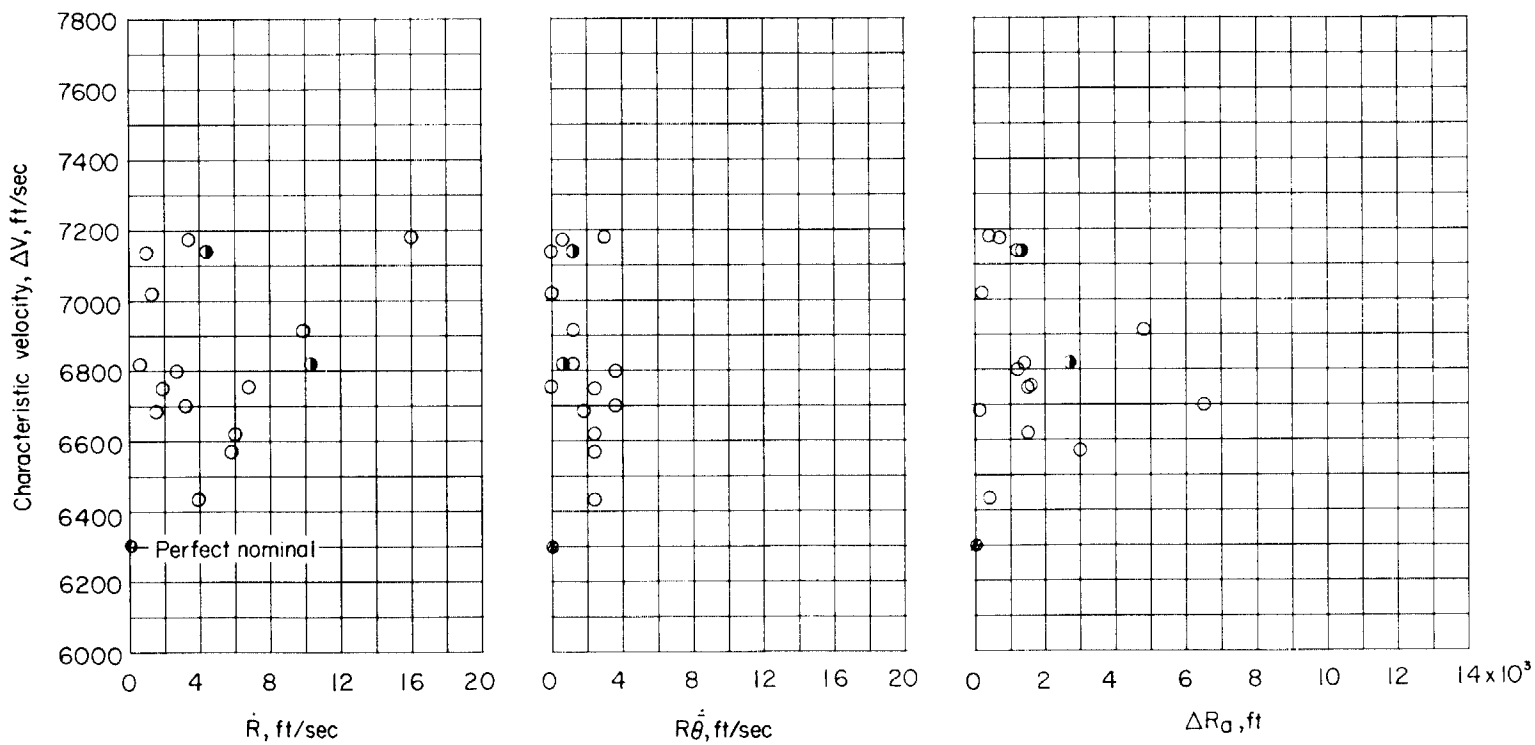


Figure 16.- Touchdown conditions of nominal flights with one pilot. Shading is used to distinguish data for flights having the same characteristic velocity.

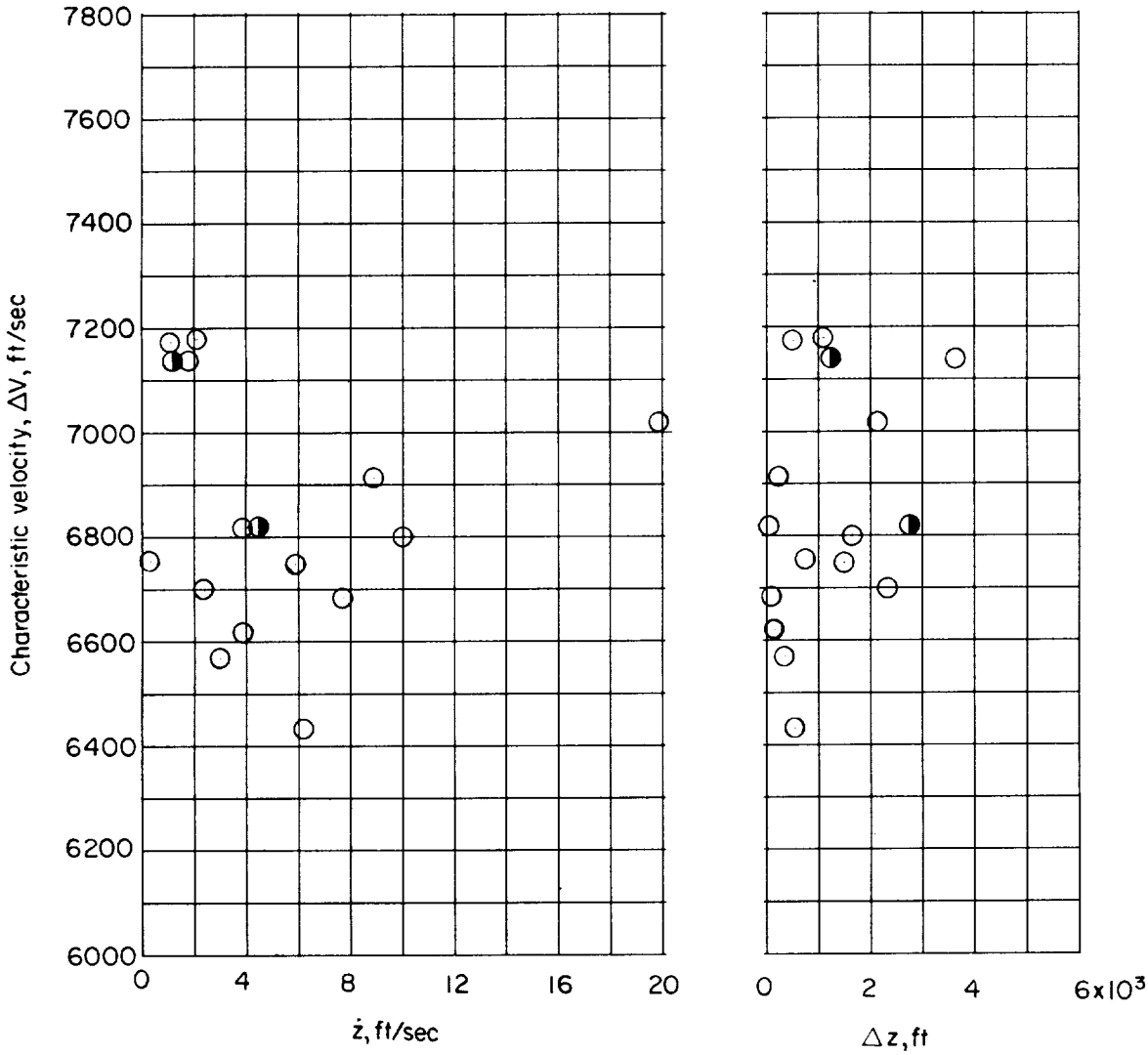
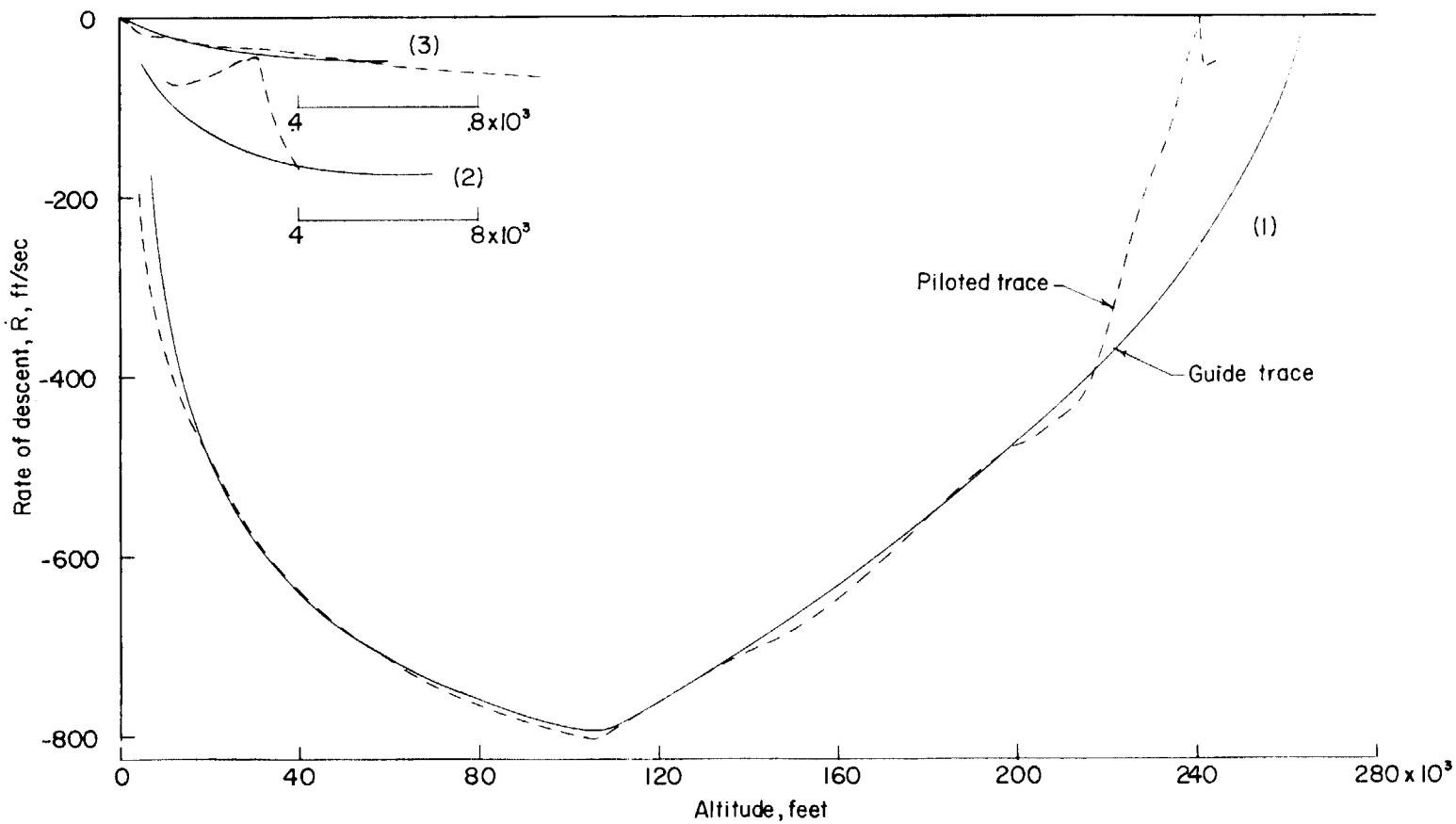
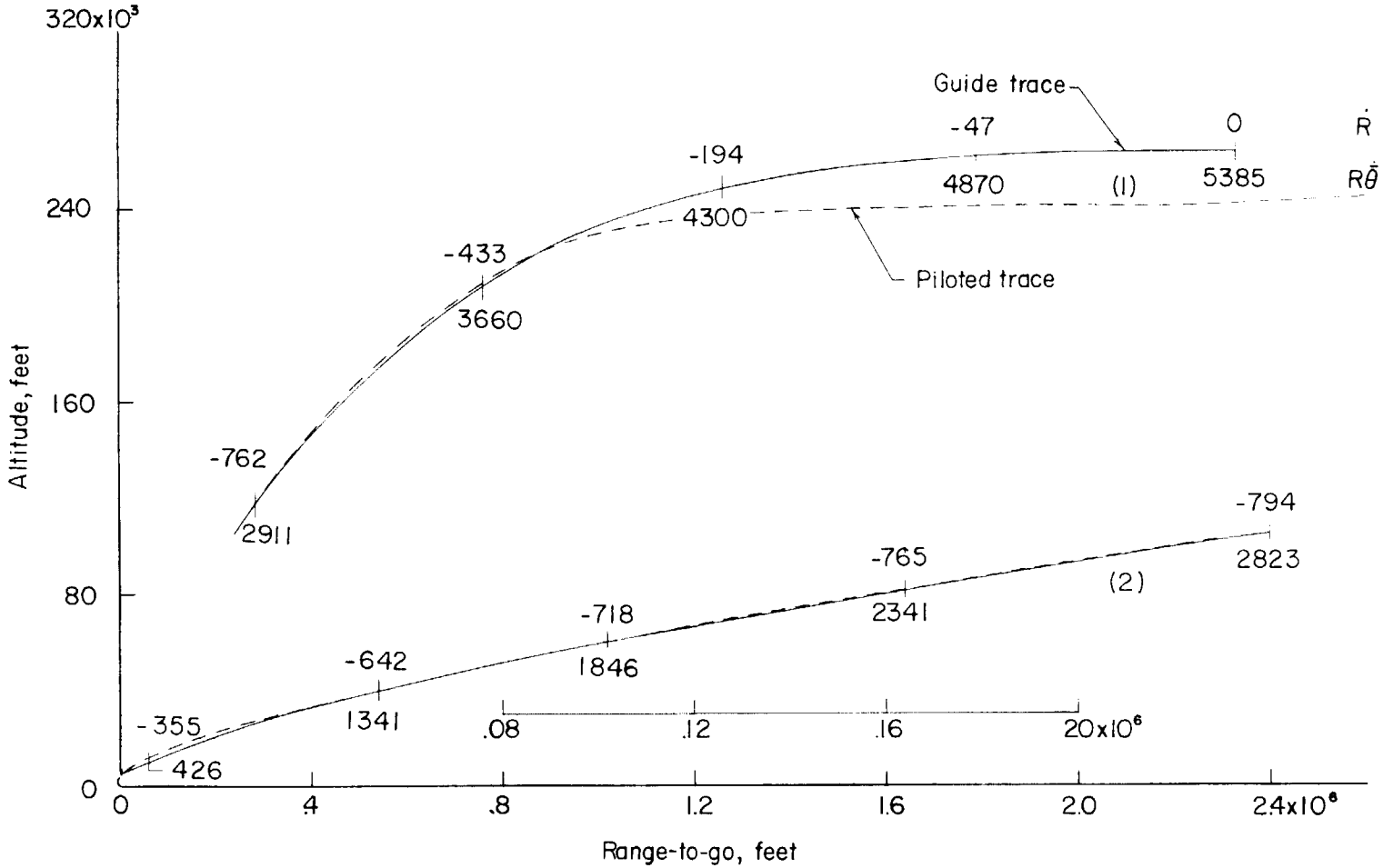


Figure 16.- Concluded.



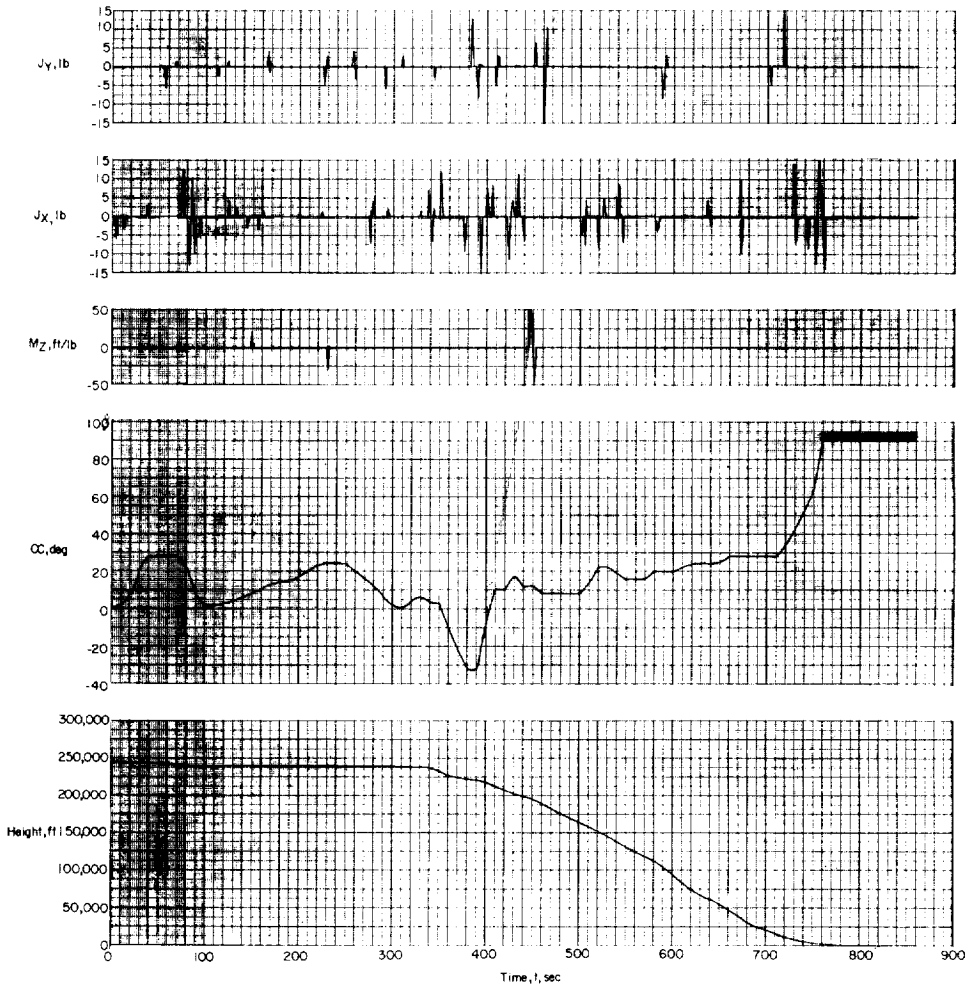
(a) Variation of rate of descent with altitude. Curves (2) and (3) are magnifications of the final portion of curve (1).

Figure 17.- Flight history of off-nominal trajectory with one pilot. $\dot{R}_o$ = -50 ft/sec; $R\dot{\theta}_o$ = 5,335 ft/sec; h_o = 244,000 ft; ΔV = 6,425 ft/sec.



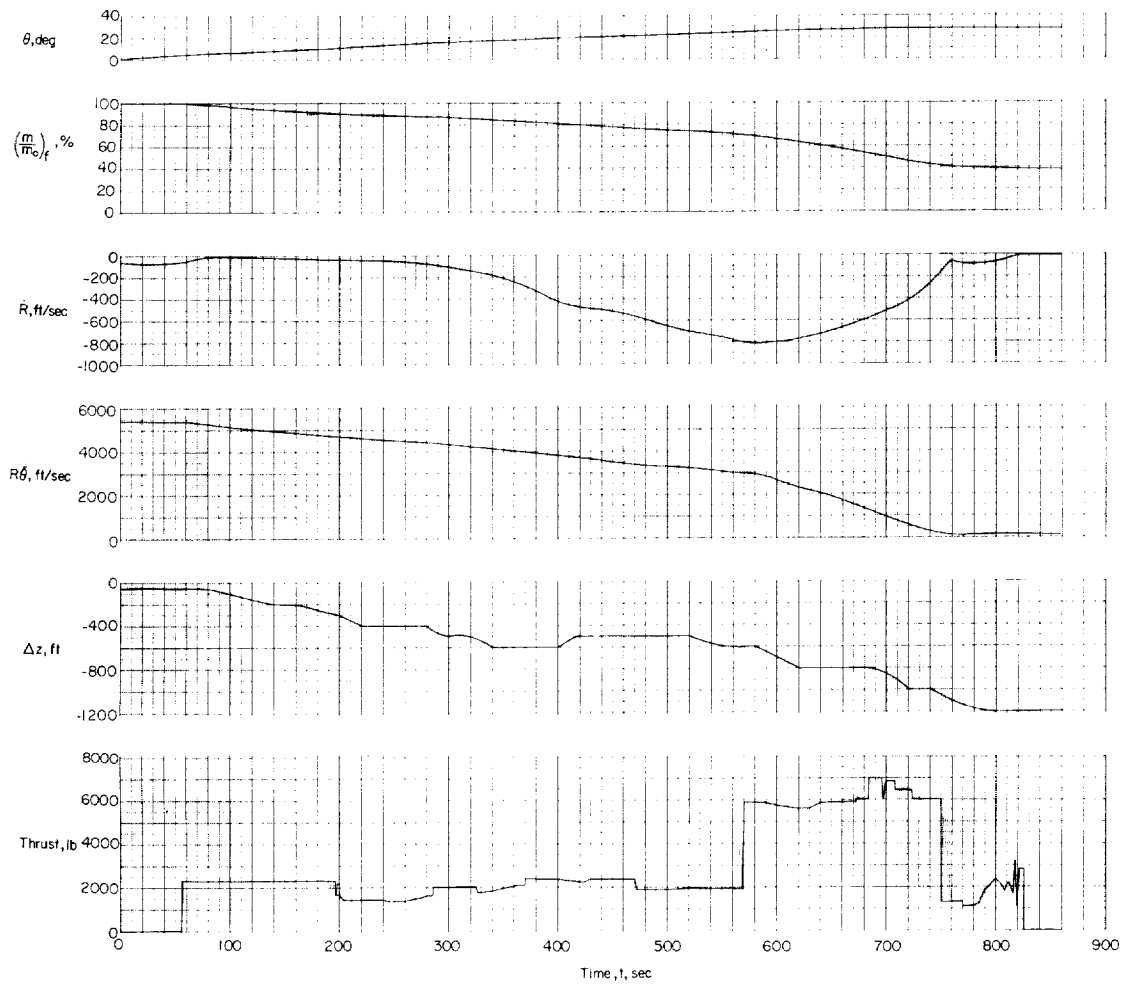
(b) Variation of altitude with range-to-go. Curve (2) is a magnification of the final portion of curve (1).

Figure 17.- Continued.



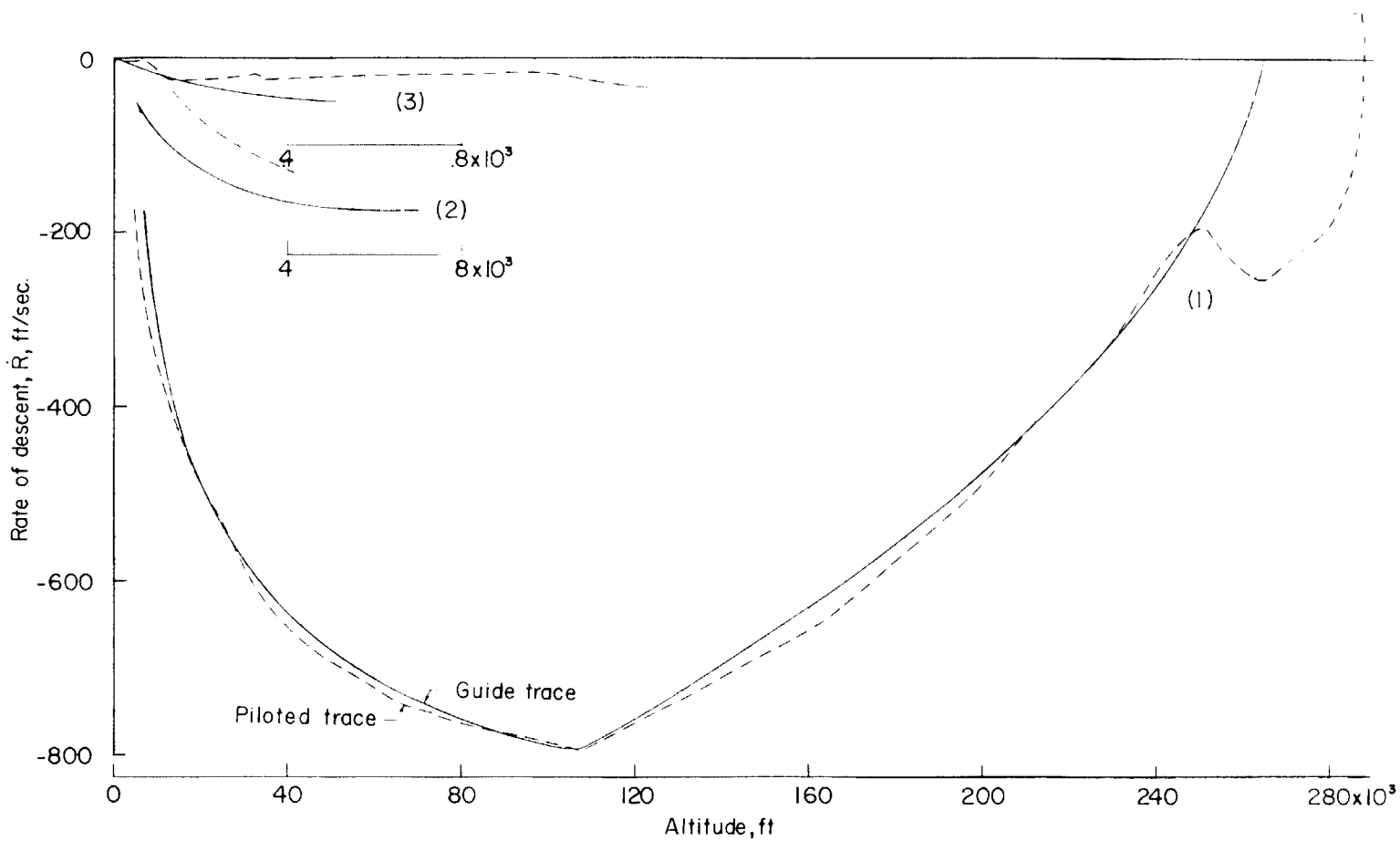
(c) Time histories.

Figure 17.- Continued.



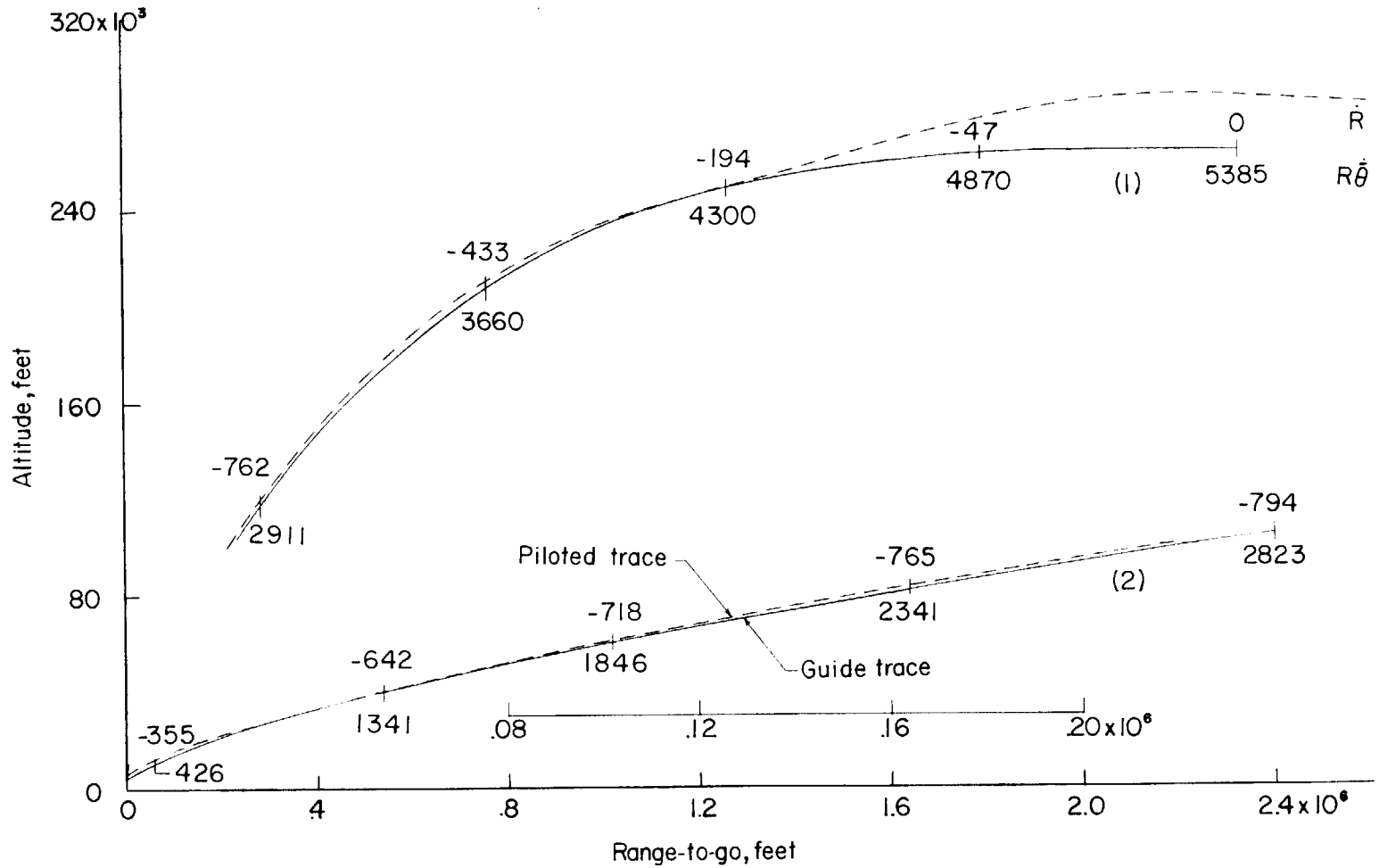
(c) Concluded.

Figure 17.- Concluded.



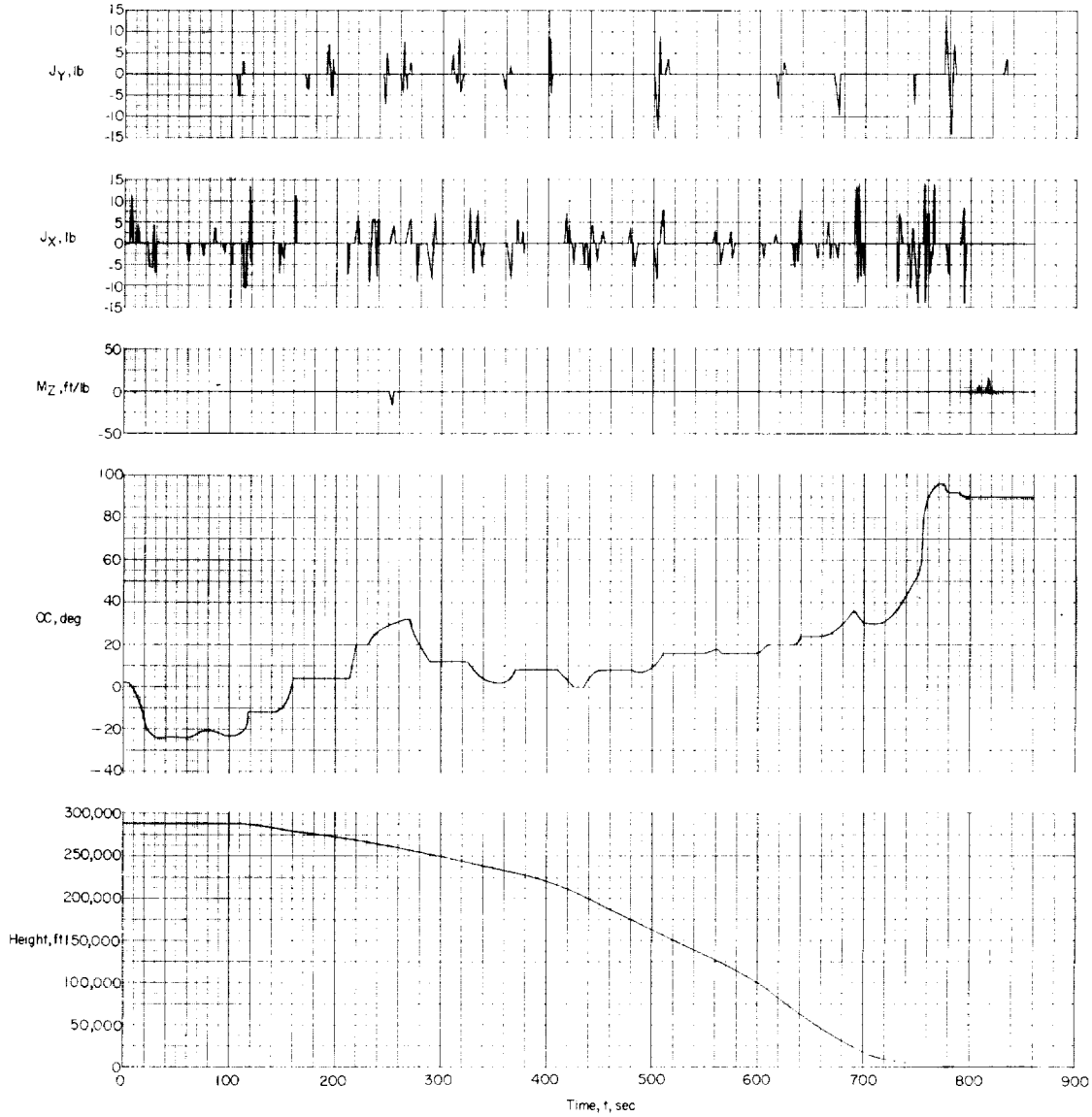
(a) Variation of rate of descent with altitude. Curves (2) and (3) are magnifications of the final portion of curve (1).

Figure 18.- Flight history of off-nominal trajectory with one pilot. $\dot{R}_o$ = 50 ft/sec; $R\dot{\bar{\theta}}_o$ = 5,435 ft/sec; h_o = 284,000 ft; ΔV = 6,795 ft/sec.



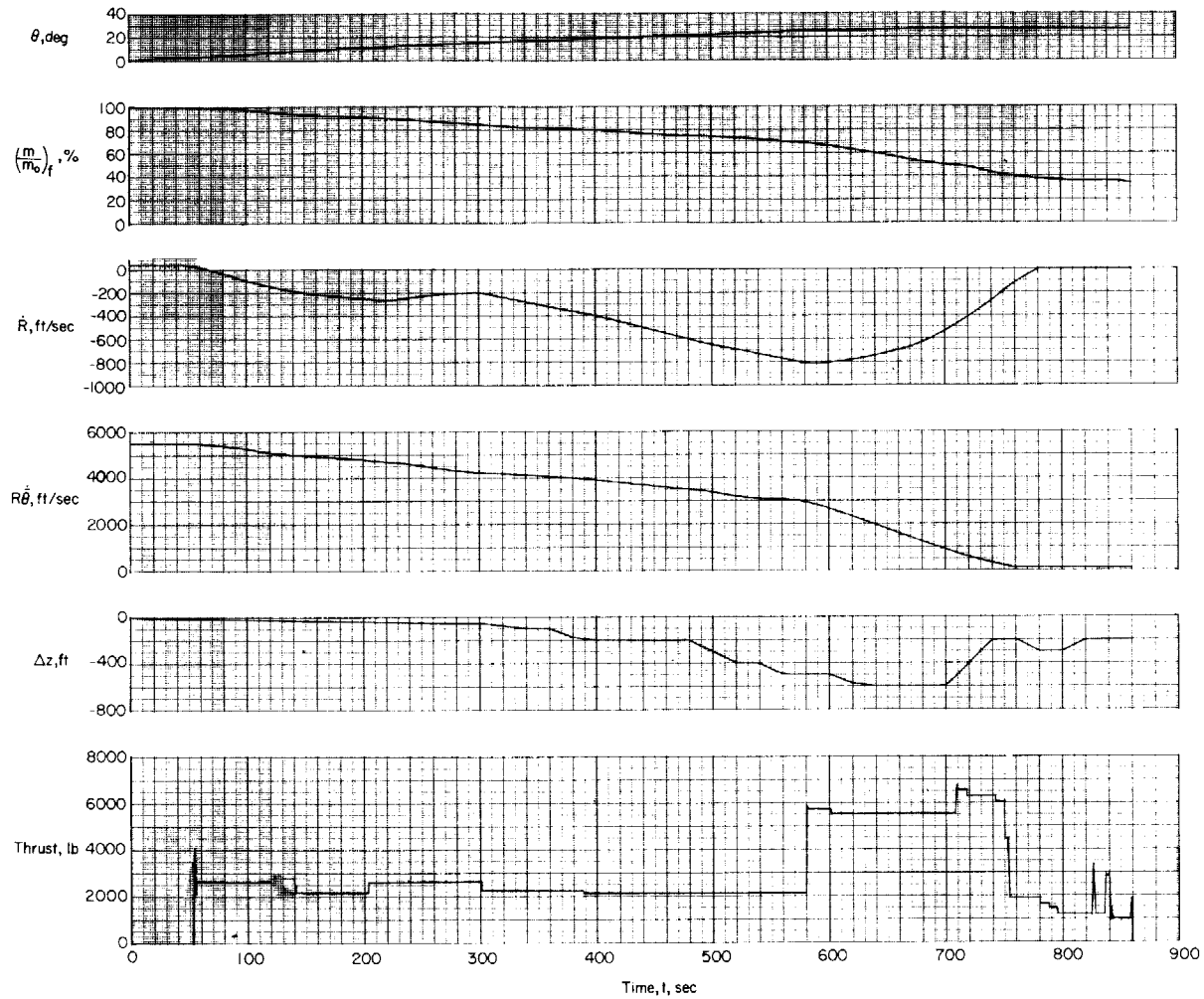
(b) Variation of altitude with range-to-go. Curve (2) is a magnification of the final portion of curve (1).

Figure 18.- Continued.



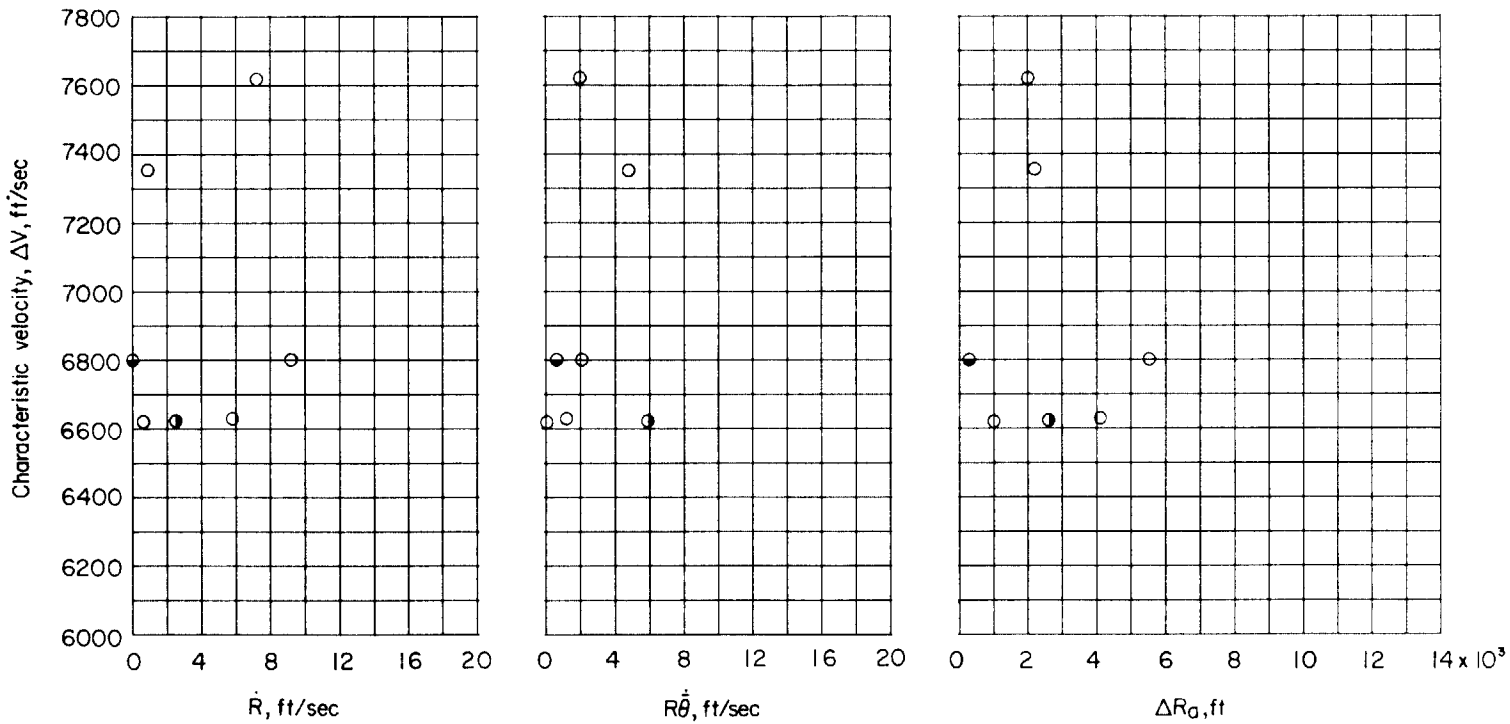
(c) Time histories.

Figure 18.- Continued.



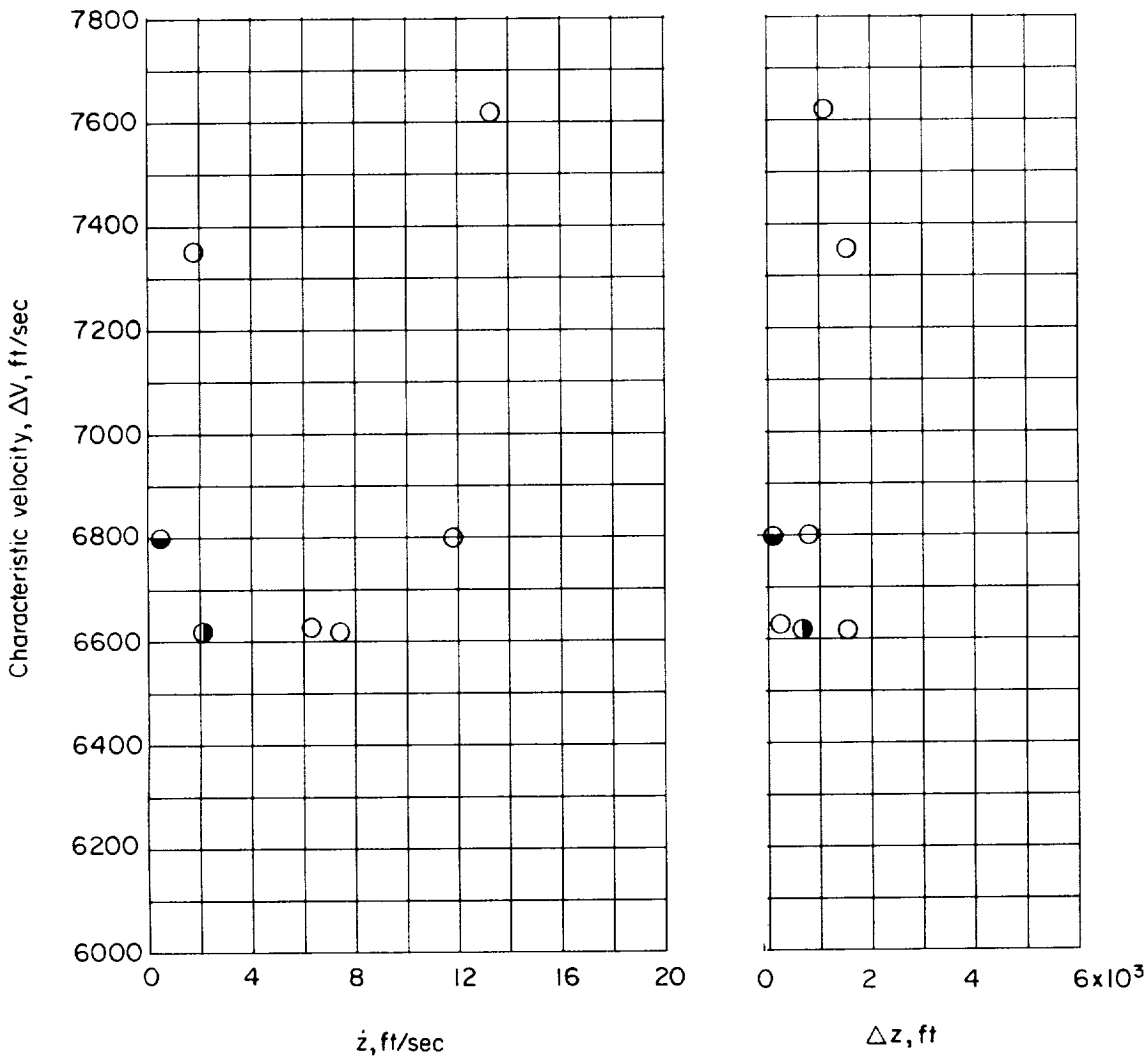
(c) Concluded.

Figure 18.- Concluded.



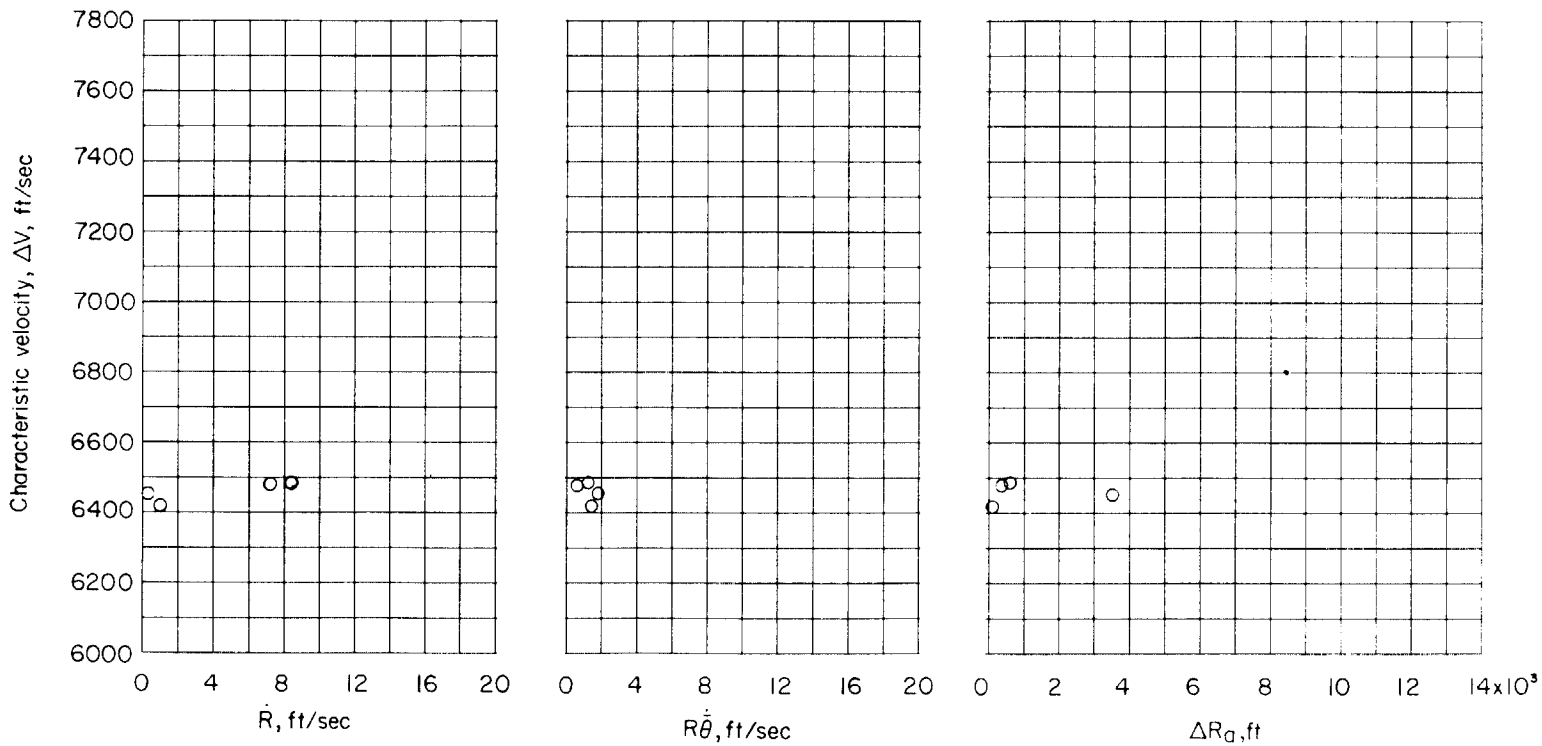
(a) $\dot{R}_o$ = 50 ft/sec; $R\dot{\bar{\theta}}_o$ = 5,435 ft/sec; h_o = 284,000 ft.

Figure 19.- Touchdown conditions of off-nominal flights with one pilot. Shading is used to distinguish data for flights having the same characteristic velocity.



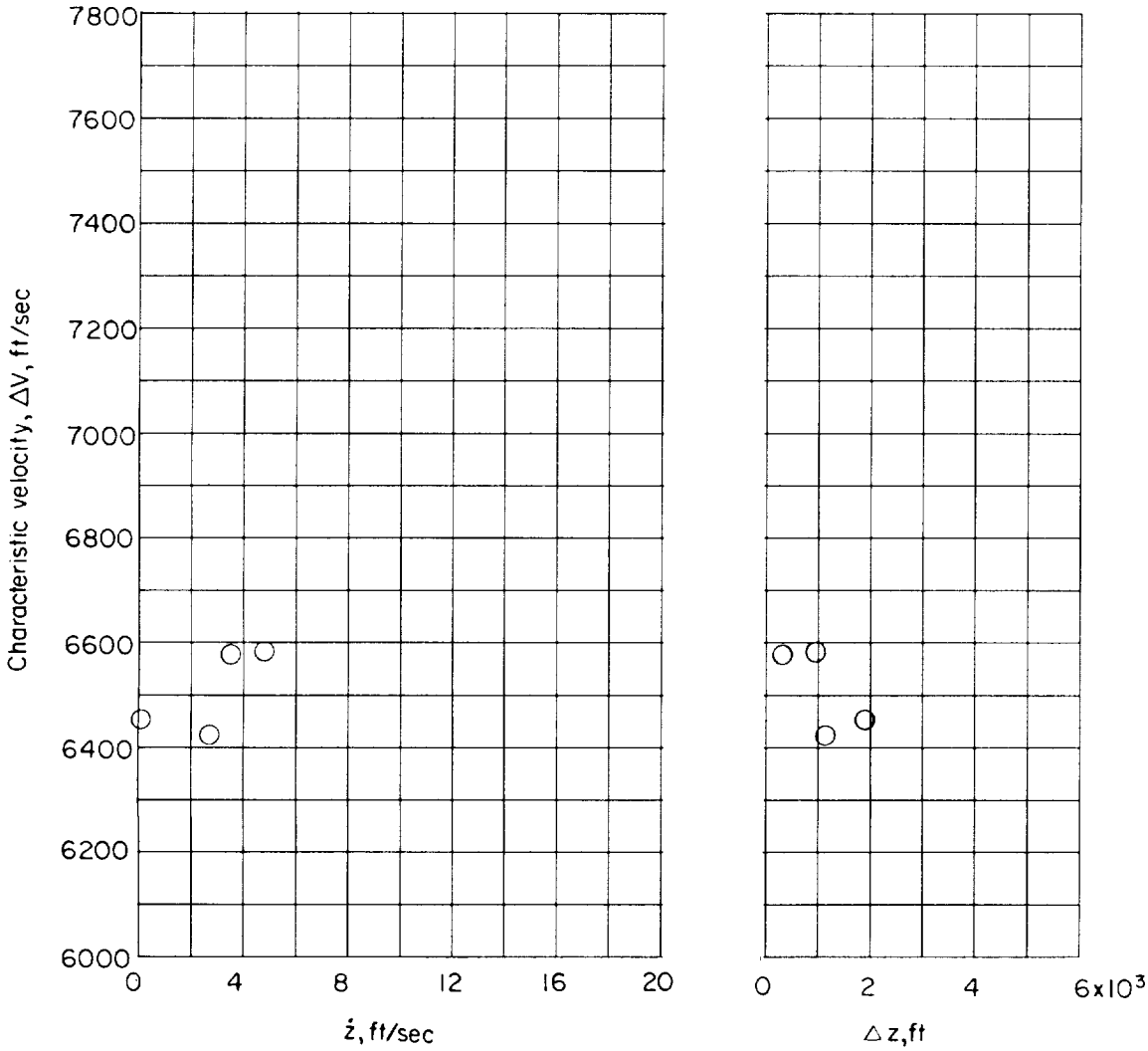
(a) Concluded.

Figure 19.- Continued.



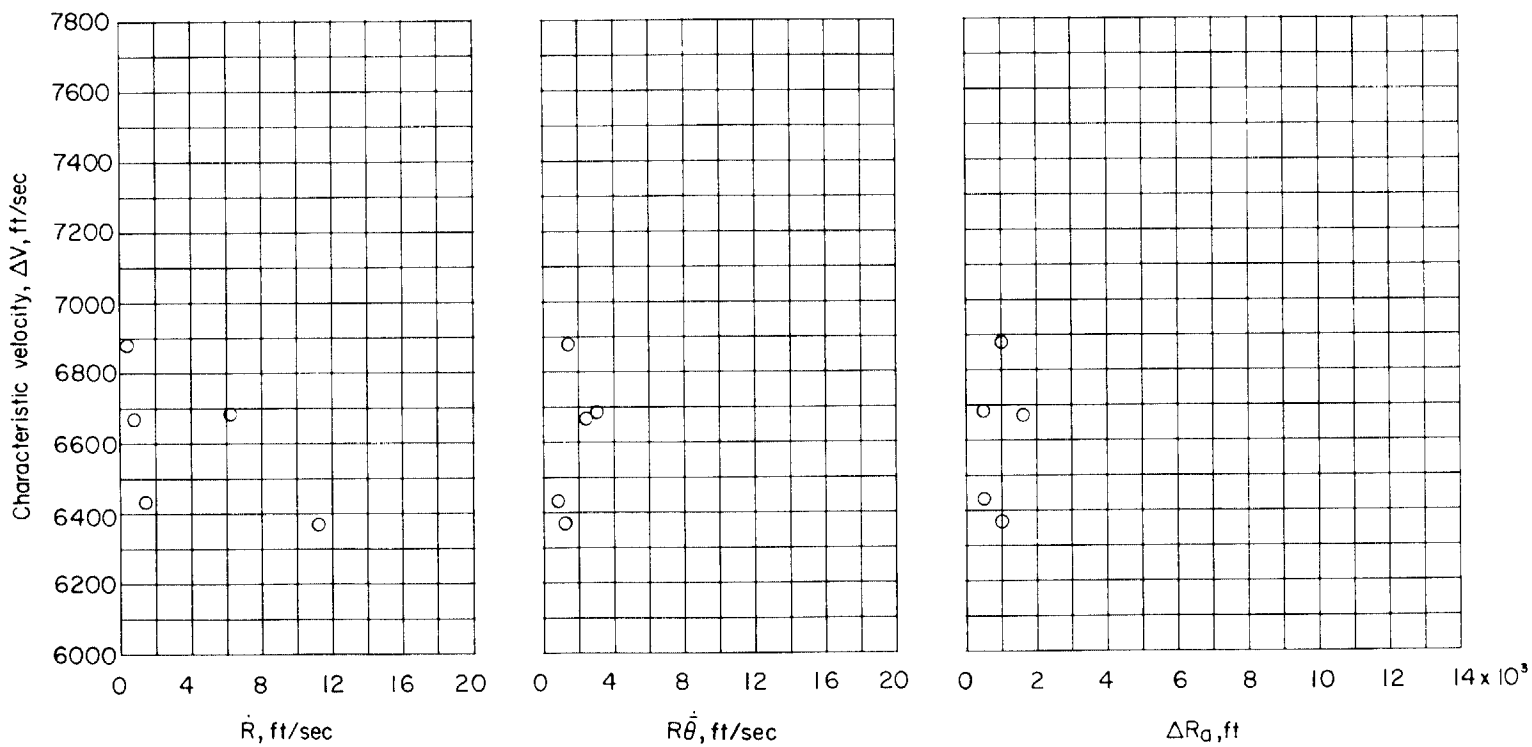
(b) $\dot{R}_o$ = -50 ft/sec; $R\dot{\theta}_o$ = 5,335 ft/sec; h_o = 244,000 ft.

Figure 19.- Continued.



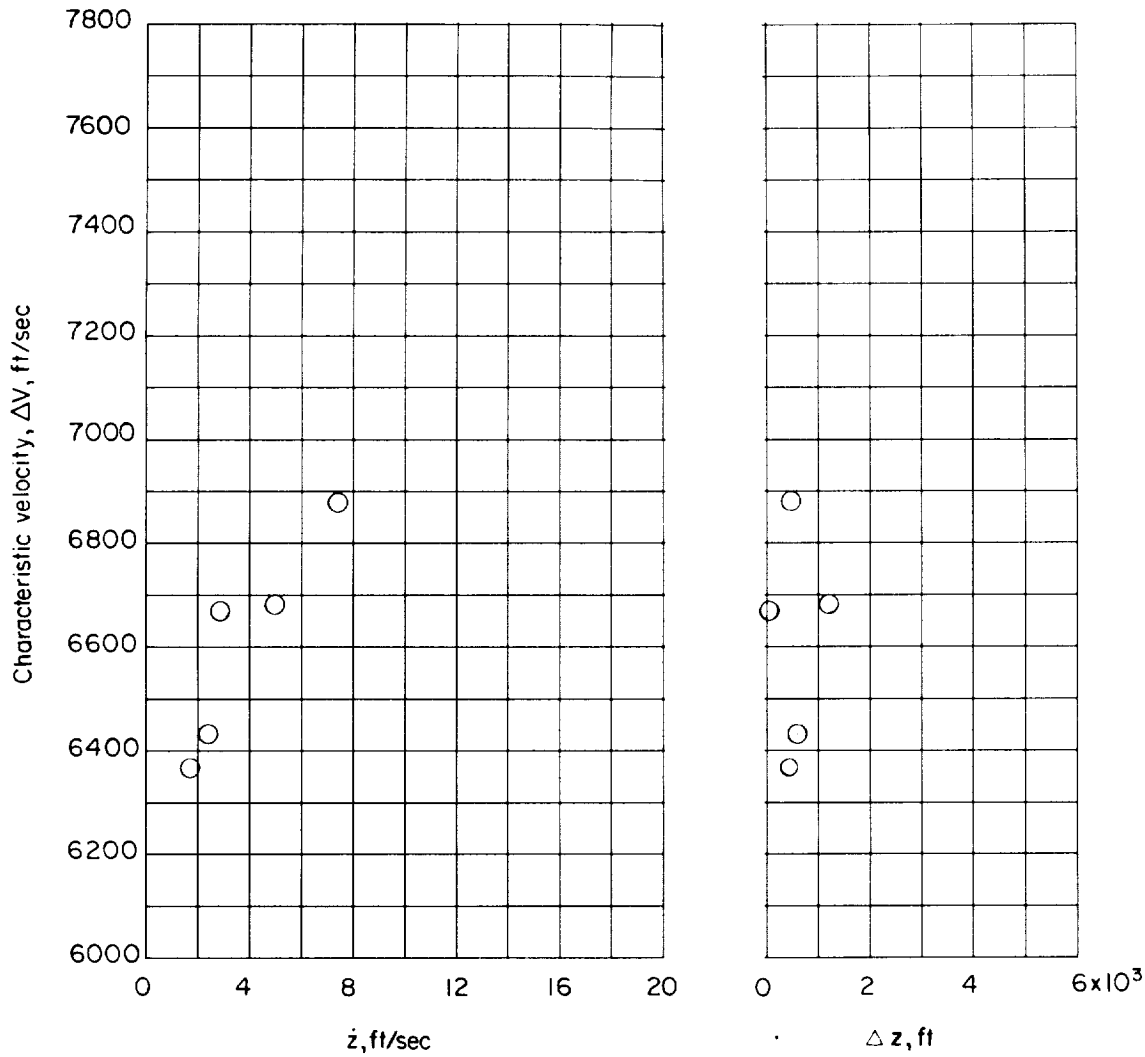
(b) Concluded.

Figure 19.- Continued.



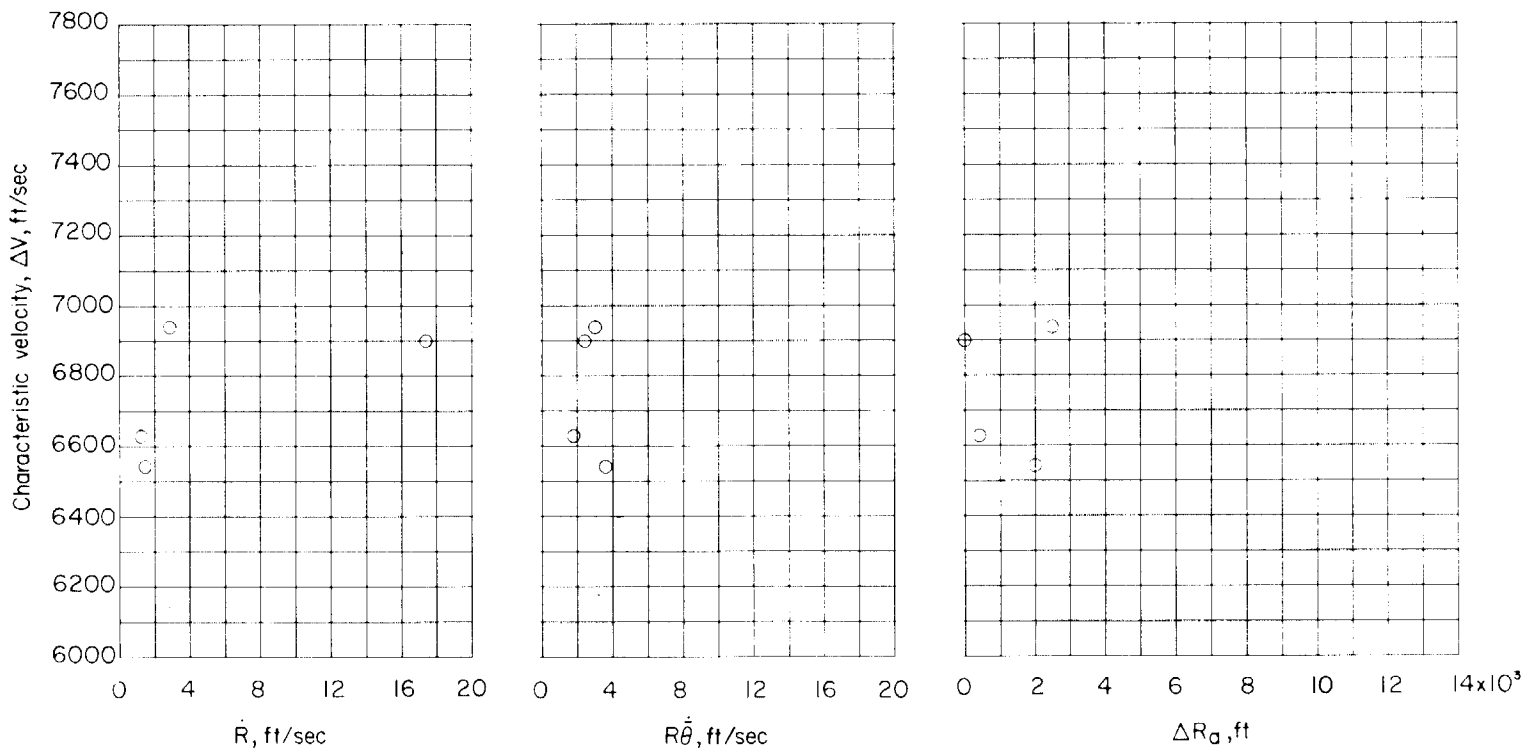
(c) $\dot{R}_o$ = -50 ft/sec; $R\dot{\bar{\theta}}_o$ = 5,335 ft/sec; h_o = 264,000 ft.

Figure 19.- Continued.



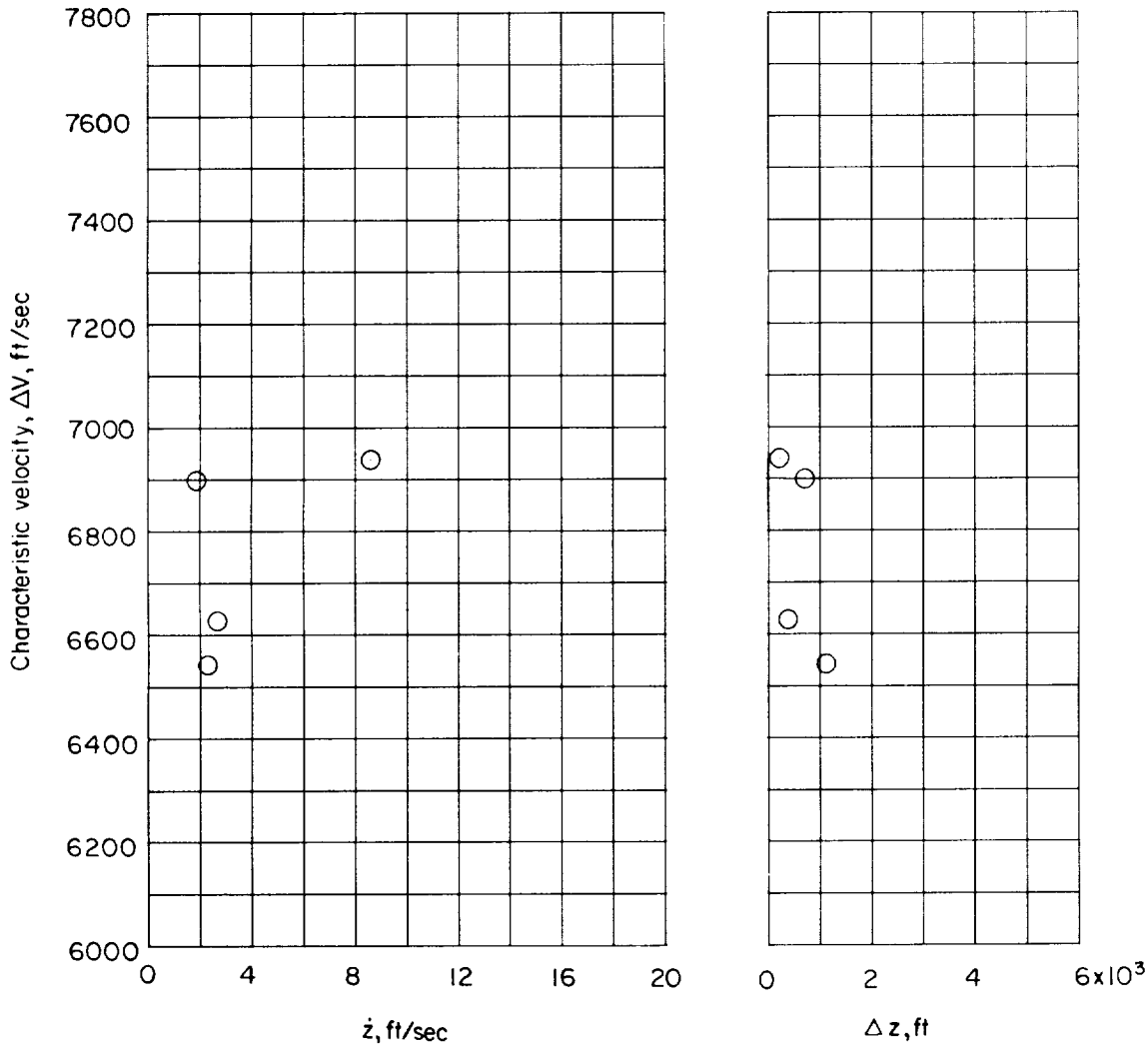
(c) Concluded.

Figure 19.- Continued.



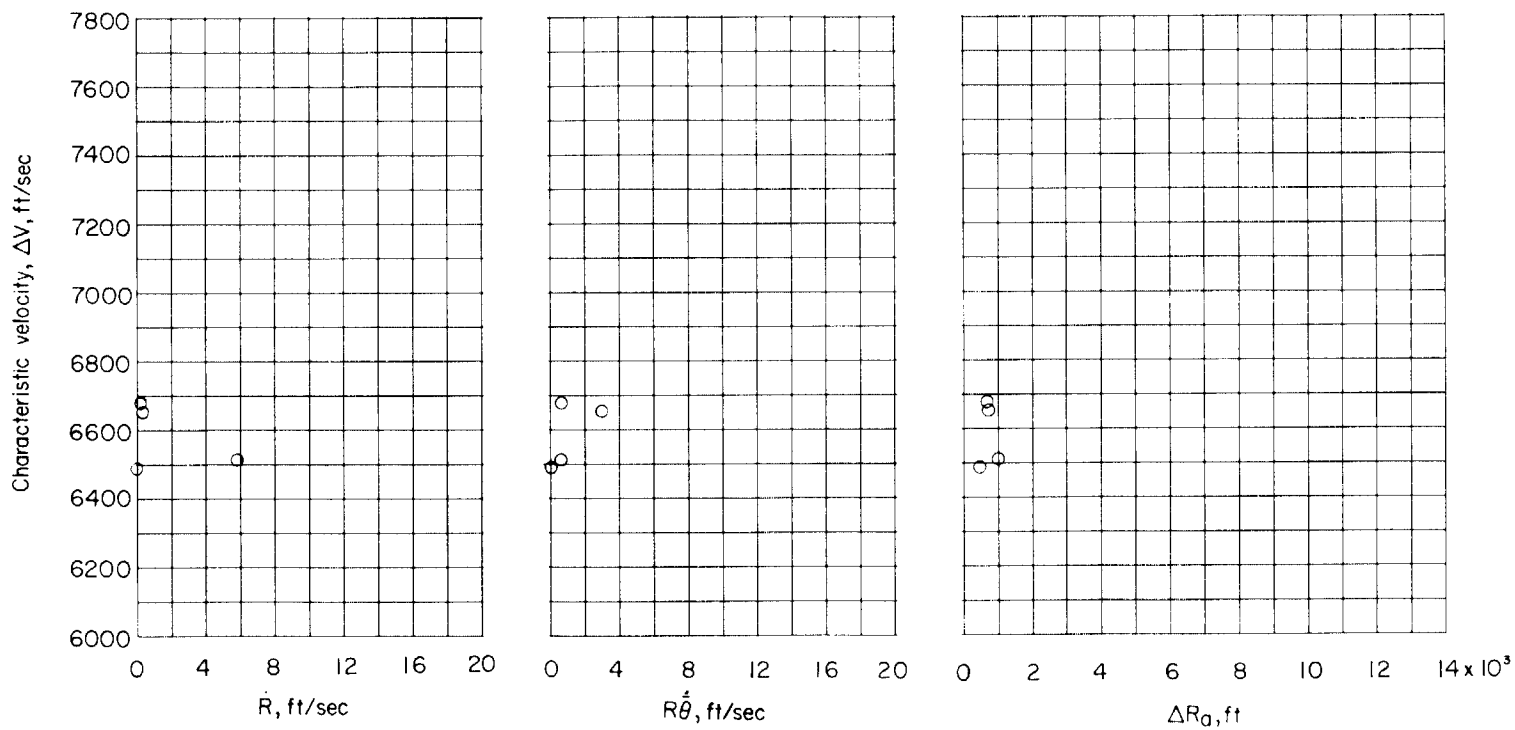
(d) $\dot{R}_o$ = -50 ft/sec; $R\dot{\bar{\theta}}_o$ = 5,435 ft/sec; h_o = 264,000 ft.

Figure 19.- Continued.



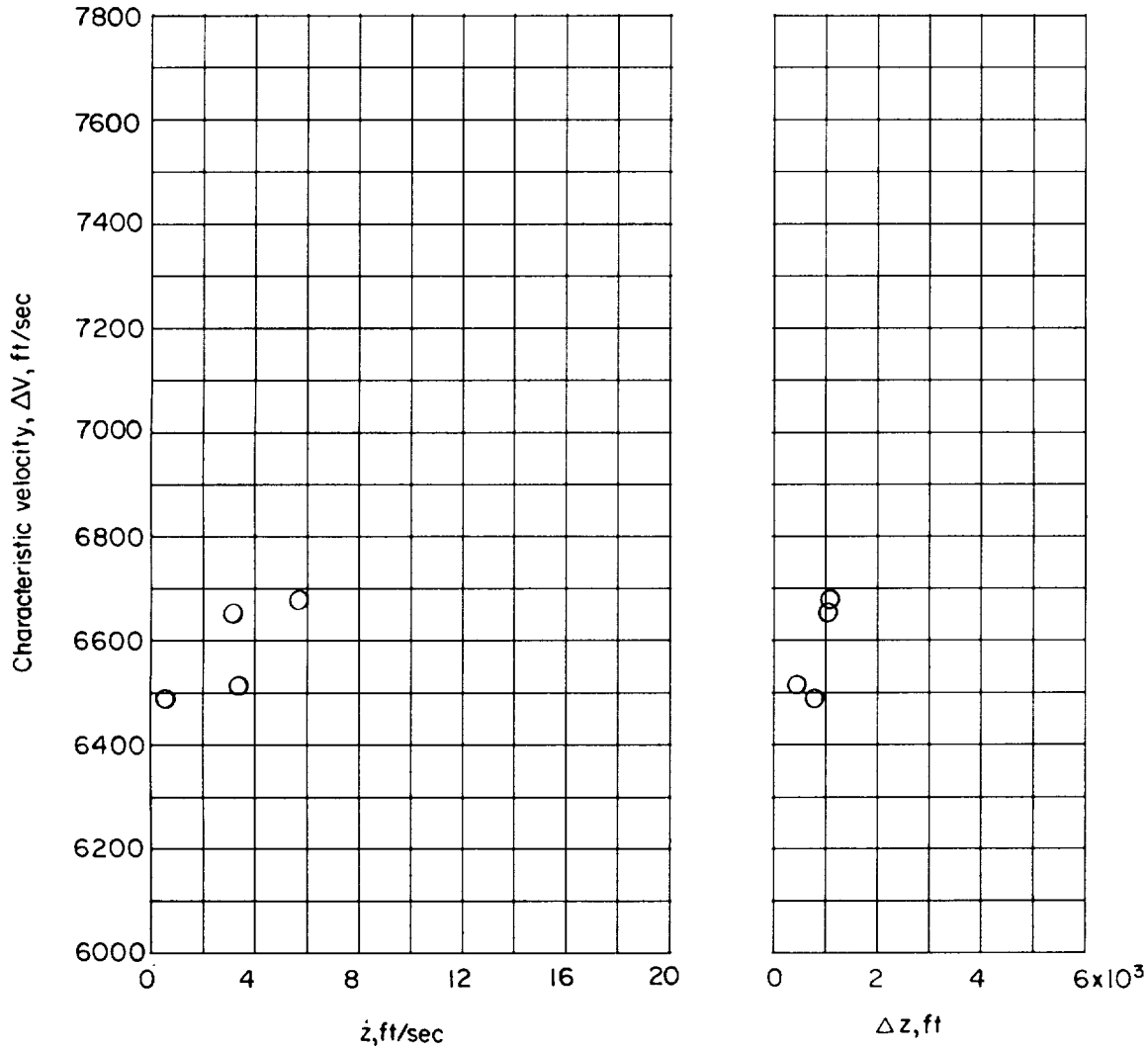
(d) Concluded.

Figure 19.- Continued.



(e) $\dot{R}_o$ = 50 ft/sec; $R\dot{\bar{\theta}}_o$ = 5,435 ft/sec; h_o = 264,000 ft.

Figure 19.- Continued.



(e) Concluded.

Figure 19.- Concluded.

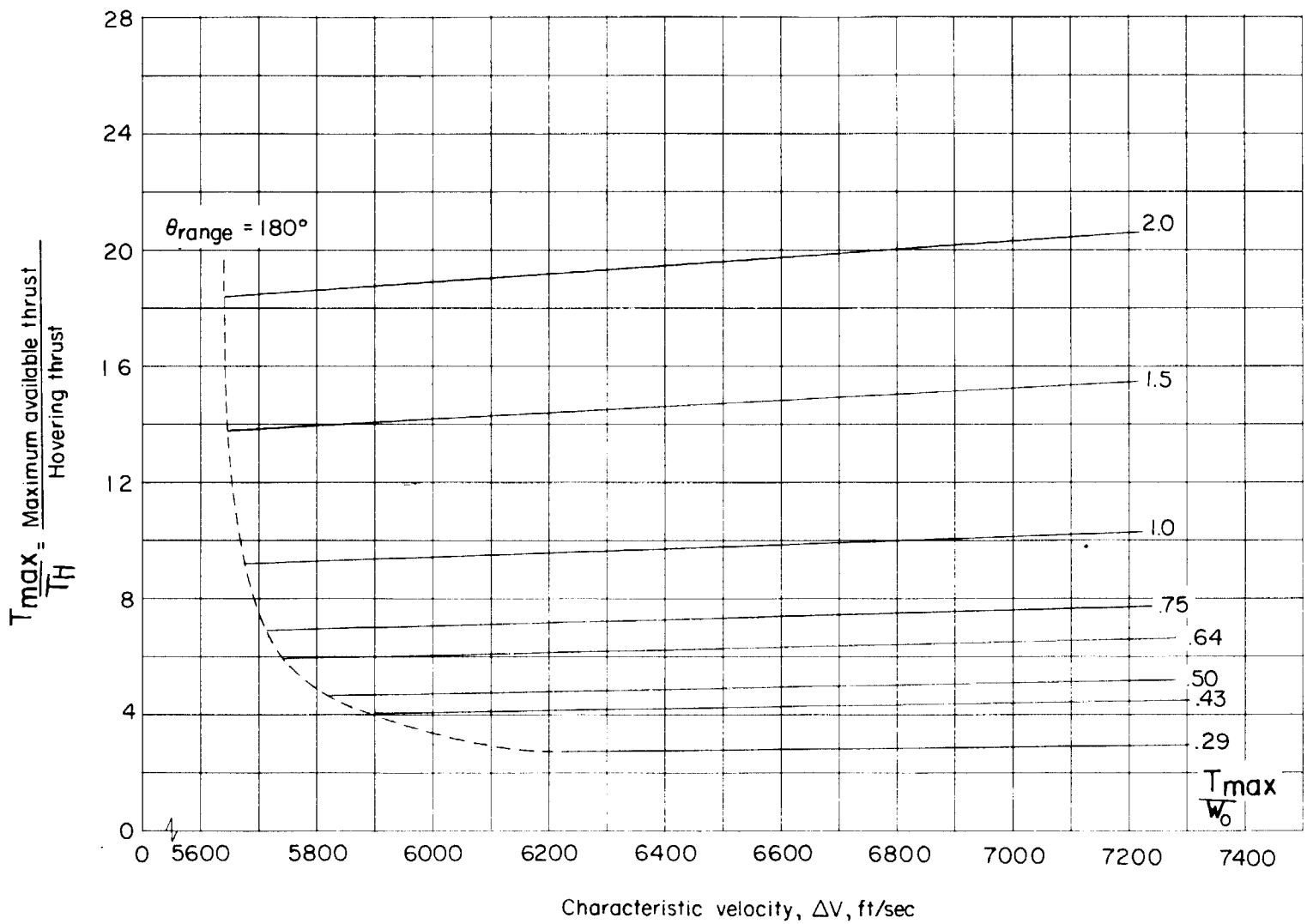


Figure 20.- Throttleability range necessary for hovering. I_{sp} = 420 sec.

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