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U. S. AIR FORCE
PROJECT RAND
RESEARCH MEMORANDUM

6 PHYSICAL RECOVERY OF SATELLITE PAYLOADS --
A PRELIMINARY INVESTIGATION (U)

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SUMMARY

Preliminary investigations indicate that solutions may exist for the major problems associated with the recovery of portions of a reconnaissance satellite. A method is described for recovering heat-sensitive items, such as photographic film, and the weight penalties involved are estimated. A payload of 50 lb of film, for example, could be recovered at a total expense in weight of about 225 lb.

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SYMBOLS

A	area projected in the drag direction
C_D	coefficient of drag
c	specific heat
e	eccentricity of the ellipse describing the motion of a body about the earth
g	acceleration of gravity at sea level
h	altitude
h_c	conductivity
R	radius (mean) of the earth
r	radius from the earth's center to the vehicle and/or recovery package
T	temperature
t	time
V	velocity
ΔV	incremental velocity added to the recovery package
W	weight
X	range measured on the earth's surface from addition of ΔV to impact
α	exponent of the density approximation ($\sigma = e^{-\alpha h}$)
γ	angle between the velocity of a body and the horizontal
θ	the angle of the incremental velocity with respect to the orbital velocity
μ	the gravitational constant
ν	ratio of propellant weight to gross weight for a rocket motor
ρ_{oo}	sea level air density

σ ratio of air density at altitude to that at sea level

ϕ angle between the radius vector from the earth's center to a body moving about the earth, measured from the radius vector to the apogee of the ellipse which describes the body's motion.

I. INTRODUCTION

Development of a method for recovering certain items intact from a reconnaissance satellite is INVESTIGATED ~~desirable for a number of reasons~~. The amount of information a satellite could gather would be substantially increased. Materials (film, living tissue, etc.) could be examined in detail to learn environmental effects. Also, photographic coverage of the earth's surface could be realized prior to the development of a dependable TV linkage for satellite use. A less direct benefit would be knowledge derived from working out successful recovery techniques: refinement of such techniques may be a necessary prelude to serious consideration of manned space flight.

The investigation of ~~Successful~~ recovery~~, on command,~~ of a satellite payload ~~is~~ contingent upon three conditions:

1. The trajectory of the orbiting payload must be modified so that it will intersect the earth's surface at a specified time and location.

2. Payload items must be protected from aerodynamic heating during re-entry into the earth's atmosphere.

3. The payload must be located and retrieved promptly after it impacts.

This memorandum investigates these three conditions and suggests a method for recovering useful material from an orbiting vehicle.

II. FLIGHT MECHANICS

The natural decay of a satellite orbit due to atmospheric drag will cause the body to return to the earth. The impact point could be predicted by observing the body during its re-entry into the atmosphere, but there would be no control over the location of the impact point. However, the position of the impact point can be selected--to some degree of accuracy--by changing the velocity vector of the package to be recovered. This could be done by using a rocket to apply thrust to the package after it separates from the orbiting body either to change the magnitude, or the direction and magnitude, of the velocity of the package relative to the circular orbital velocity vector. The descent range and the re-entry conditions of the package are a function of the initial orbital altitude and of the direction and magnitude of the velocity increment.

The velocity for a circular orbit at any radial distance can be expressed as

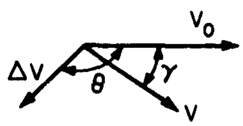
$$V_{orb} = \sqrt{\frac{\mu}{r_{orb}}}$$

where μ is the gravitational constant and is taken to be 1.41008×10^{16} ft^3/sec^2.

In general, the magnitude of the resultant velocity of the package is

$$V = \sqrt{(V_{orb})^2 + (\Delta V)^2 + 2V_{orb}\,\Delta V \cos \theta}$$

where θ is the angle of the velocity increment ΔV measured clockwise from the orbital velocity vector as shown in the following sketch.



The angle between this resultant velocity vector and the instantaneous horizontal is given by

$$\tan \gamma = - \frac{\Delta V \sin \theta}{V_{orb} + \Delta V \cos \theta} .$$

The eccentricity of the ellipse can be written as

$$e = \sqrt{1 - \left(2 - \frac{V^2 r}{\mu}\right) \frac{V^2 r}{\mu} \cos^2 \gamma}$$

where V is the resultant velocity and r is the orbital radius.

The range angle, measured at the center of the earth, from this point to the apogee of the ellipse is given by

$$\cos \phi_1 = \frac{1 - \frac{V^2 r}{\mu} \cos^2 \gamma}{e}$$

and the range angle from the apogee to the surface of the earth (r = R) is given by

$$\cos \phi_2 = \frac{1 - \frac{V^2 r}{\mu} \cos^2 \gamma \frac{r}{R}}{e} .$$

Therefore, the total descent range from the point where the velocity increment is added to the surface of the earth is given by

$$X = R\,(\phi_1 + \phi_2)$$

for a vacuum re-entry trajectory.

For the case of a velocity increment directed rearward ($\theta = 180^\circ$), these equations simplify to

$$V = V_{orb} - \Delta V$$

$$\gamma = 0$$

$$e = 1 - \frac{V^2 r}{\mu}$$

$$\phi_1 = 0$$

$$\cos \phi_2 = \frac{1 - \frac{V^2 r}{\mu}\ \frac{r}{R}}{e}$$

and

$$X = R\,\phi_2 \quad .$$

The variation of vacuum range to impact as a function of velocity increment is given in Fig. 1 for initial orbital altitudes of 150, 300, and 500 statute miles for the case of a rearward increment ($\theta = 180^\circ$). It can be seen that a velocity increment of at least 2000 ft/sec is required if the descent range is to be less than 3000 n mi for an initial orbital altitude of 300 mi. The re-entry trajectory of a typical body will intersect the earth at a point roughly 200 mi short of the vacuum distance due to the influence of the atmospheric drag.

The effect of varying the angle of the velocity increment Θ is shown in Fig. 2 for an initial orbital altitude of 300 mi and a velocity increment of 3000 ft/sec. Values of Θ of 90 deg and 180 deg give approximately equal vacuum descent ranges, but the re-entry trajectories are somewhat different.

An estimate of the sensitivity of descent range to errors in the magnitude and direction of the velocity increment can be obtained from these graphs. For a 300 mi orbit and $\Delta V = 3000$ ft/sec with $\Theta = 180^{\circ}$, the slopes are $\partial X/\partial \Delta V = -0.27$ n mi/ft/sec and $\partial X/\partial \Theta = -27.1$ n mi/deg. If the velocity increment is added at an angle of about 130 deg from the direction of motion, the slope $\partial X/\partial \Theta$ is seen to be near zero, and only a velocity error would contribute to the impact point uncertainty.

The velocity and path angle of the ellipse at a given altitude above the earth can be found from the following equations. The velocity, at an altitude h, is

$$V_h^2 = V^2 + \left(\frac{2\mu}{r_h} - \frac{2\mu}{r_{orb}} \right)$$

and the path angle is given by

$$\cos \gamma_h = \frac{(V\ r\ \cos \gamma)}{V_h\ r_h}$$

where $r_h = R + h$ and the term $(V\ r\ \cos \gamma)$ is evaluated at orbital altitude after the velocity increment has been added.

The initial conditions for re-entry into the atmosphere at a nominal altitude of 250,000 ft are presented in Fig. 3a,b. Figure 3a shows the variation of re-entry velocity and path angle for three initial orbital altitudes and several velocity increments for the case of $\theta = 180^\circ$. For comparison, the re-entry conditions for a 5500 n mi ICBM fired at various ranges using its full velocity potential are indicated by the nearly horizontal dashed line. The points on this curve cover the re-entry conditions for the missile fired on trajectories which vary from the 3000 mi low case through maximum range to the 5000 mi high trajectory. Thus, we see that the terminal portions of the trajectories for the recovery of a package from a satellite in a circular orbit around the earth are similar to the re-entry trajectories of a 5500 n mi ICBM when fired at shorter ranges on low non-optimum trajectories.

Figure 3b is included to show the effect of varying the angle of the velocity increment for the case of a 300 mi orbital altitude and a velocity increment of 3000 ft/sec. The dashed curve, representing an orbital altitude of 300 mi with varying velocity increments added at $\theta = 180^\circ$, is repeated from Fig. 3a. It is seen that the re-entry conditions vary symmetrically around $\theta = 180^\circ$ while the total vacuum descent range as shown in Fig. 2 does not. It is interesting to note that the re-entry velocity resulting from a 3000 ft/sec increment added at angles of less than about 110 deg or more than about 250 deg is greater than the local circular orbital velocity.

No re-entry trajectories have been calculated for the initial conditions corresponding to this package recovery study, but a case corresponding to the maximum range ICBM point has been used to compute the temperature history shown in Fig. 4.

Reference 1 presents an approximate analytical solution of the equations of motion during the re-entry. These equations give reasonable values for the various trajectory variables until the curvature of the path becomes large.

The maximum deceleration experienced by the body is given by

$$\left(\frac{dV}{dt}\right)_{max} = \frac{\alpha V_o^2 \sin \gamma_o}{2e}$$

and the velocity at which it occurs is approximately

$$V = 0.607 V_o$$

where V_o and γ_o are the velocity and path angle respectively, at the nominal re-entry altitude.

The altitude for maximum deceleration can be found from the equation for the density ratio which is

$$\sigma = \frac{\alpha}{g \rho_{oo}} \frac{1}{\left(\frac{C_D A}{W \sin \gamma_o}\right)}$$

where α is the constant in the exponent in the isothermal atmosphere approximation $\sigma = e^{-\alpha h}$. The value of α is about 4×10^{-5} per foot.

III. PROTECTION DURING RE-ENTRY

Two steps can be taken to protect the payload from excessive temperature rises. The aerodynamic heating of the outer shell can be reduced through control of the re-entry flight path, and the payload can be insulated from the outer casing. First, let us consider the problem of controlling the re-entry flight path.

For "dump ranges" (ranges on the surface of the earth from point of impulse to impact) of a few thousand miles the descending trajectory resembles that of an ICBM. The re-entry heating problems of ICBM warheads have been examined in detail by Carl Gazley, Jr. (Ref. 1). His work demonstrates that as the ratio $\frac{C_D A}{W \sin \gamma}$ is increased, maximum deceleration occurs at successively higher altitudes and the total heating per unit area decreases. The possibility suggests itself that a large enough parachute would limit the total heating to such an extent that conventional parachute fabrics would retain their strength.

This has been investigated by Gazley, in an unpublished work, for the example of a re-entering warhead attached to a parachute 100 ft in diameter with a total weight of 3500 lb. Figure 4 shows the heating experienced by such a device. The case was assumed to be an insulated skin of 0.05 in. stainless steel. The parachute was constructed of fiberglas cord and cloth (designed with a factor of safety equal to 2) and covered with an additional layer of cloth as a shield against the high transient heating load.

This method of supplying a large drag area is a somewhat arbitrary choice. To be suitable the device used should be able to expand its linear dimensions by one or two orders of magnitude and yet weigh only a fraction of the total package weight. One might consider such devices as balloons (the parachutes might use balloons to ensure proper opening) or structures which telescope or fan out. However, as the design for any such structures approaches minimum weight it begins to resemble the design of a foil parachute. The inherent advantages of fabric (i.e., efficient packaging, non-continuous fracture and heat paths, etc.) suggest a parachute of conventional construction.

The behavior of parachutes is known for reasonably dense gases and for speeds up to low supersonic Mach numbers (Ref. 2). For estimating performance in rarified gases at hypersonic Mach numbers a "reasonable extrapolation" was made. Obviously, experimental investigation will be necessary before performance can be predicted with confidence.

Peak aerodynamic heating occurs in a region of the atmosphere in which little knowledge has been accumulated but where considerable interest is now being directed. By conventional criterion the flow over the package will be well within the all-laminar regime (ignorance concerning transition will not be critical). The mechanism of heat transfer to the parachute cannot be well defined until the flow is better understood. The flow and heat transfer have been assumed similar to those for a blunt body for the temperature studies in this memorandum.

The shroud lines were treated as infinite, inclined cylinders for the determination of heating. The covering fabric on the lines would probably lose most of its strength at maximum temperature, but all that is required is that it hold together for about one minute.

Figure 4 indicates directly how the temperature of a recovery package might be controlled. The time-temperature curves could be interpreted as applying to such a package if the parameter $\frac{C_D A}{W}$ were equal for the two cases (from Fig. 3 the angle γ is seen to be similar to that for an ICBM).

In the case of the re-entering warhead the parachute would be detached about 30 sec after re-entry to lessen warhead vulnerability to interception. (The effect on the warhead skin temperature of retaining the parachute is indicated on Fig. 4 by a dashed line.) The recovery package would retain its parachute until touchdown, not primarily because of temperature considerations but because a longer descent time and lower impact velocity (about 17 ft/sec) would aid in the successful recovery of the package.

The skin temperature is a function not only of the flight path $\left(\frac{C_D A}{W \sin\gamma}\right)$, and the skin properties, but also of the absolute size of the package (due to the dependence of the heat-transfer coefficient upon the Reynold's number). This is, of course, included in the calculations appended to this memo but it is actually a second-order effect for these qualitative considerations.

The payload can be insulated from the heated surface with conventional materials and techniques. Due to the transient nature of the heating the insulating material and the outer casing are more efficient if used in a number of layers rather than a single layer. (This is in agreement with the results shown in Ref. 3.)

Figure 5 illustrates the heating of an inner case as a function of time for an outer skin raised to a constant elevated temperature. The distribution of metal between the outer case and successive layers within the fiberglas insulation is varied as follows: (a) outer and inner cases only; (b) one intermediate layer; and (c) two intermediate layers. This figure demonstrates the ability to increase the insulating effectiveness to allow for increased outer-case temperatures without the addition of appreciable weight. In essence the heat capacity of the insulation is increased with no commensurate increase in conductivity.

In reality the temperatures of the outer case will begin to decrease after the first half minute (i.e., dashed line in Fig. 4). The temperatures of the outer case, interlayers, and inner case are plotted as functions of time in Fig. 6.

It is worth noting that most of the heat input to the intermediate layers and the inner case occurs after the outer case has started to cool. Figure 7 shows the outer-case temperature for the transient condition and the corresponding equilibrium temperature (determined for the altitude and velocity as functions of time during descent). It appears that a possible aid to the protection of the package might consist of the ejection of the front outer layer after the initial heating has concluded. The next layer exposed would remain close to the equilibrium temperature for the remainder of the flight path and the heating problem would be correspondingly reduced. This technique is conceptually very simple; mechanically, it implies that the $\frac{C_D A}{W}$ of the ejected layer is less than that of the remaining package plus parachute (to eliminate interference with the parachute).

IV. PACKAGE LOCATION AFTER IMPACT

Although it appears quite possible to bring the package through the atmosphere in an undamaged condition there remains the problem of location and recovery after impact. This problem is lessened considerably if the parachute is retained till touchdown. A total descent time from dump signal to impact of from 40 to 60 minutes allows ample time for the package to be tracked, its final position predicted, and recovery proceedings initiated. (A small radio beacon could be included for small cost in weight and it would facilitate tracking.)

The ease with which a package could be recovered is a direct function of the predictability of its impact point. Figure 2 indicates the sensitivity of range to the elevation angle of the incremental thrust. As shown, the effects of errors in this angle are minimized if the angle is chosen for minimum range. Angular errors of as much as ± 20 deg could easily be tolerated if distance errors of as much as 100 n mi were acceptable.

Errors in azimuth of the incremental thrust will result in proportional errors on the ground (e.g., for $h = 150$ mi and $\Delta V_i =$ 2000 ft/sec an angular error in azimuth of ± 10 deg results in a ground error of ± 35 n mi).

In the unlikely event of a high wind which is invariant with altitude these deviations could be approximately doubled. (This effect can be estimated ahead of time.)

V. SYSTEM WEIGHTS

An estimate has been made of the weight penalties involved if the equipment necessary for successful recovery were included in a satellite stage. The components of such a recovery system are illustrated in Fig. 8.

The prime item for which all this effort is being made is assumed to be exposed photographic film. Total weights derived for this assumption will be conservative for payload items of greater density and decreased temperature sensitivity. (The film is assumed to be damaged if its temperature exceeds 100°F.)

The component weights for a film weight of 50 lb are tabulated in Table 1 for an orbital altitude of 150 st mi and a dump range of 2500 n mi. The assumptions and procedures for determining these weights are included in Appendix B, as is the determination of total package weight as a function of film weight, altitude, and dump range. This variation is shown in Fig. 9.

Table I

ITEM	WEIGHT (lb)
Film	50
Reel	5
Beacon (plus power, switching, etc.)	10
Case + Insulation	45.8
Parachute (plus positive opening device)	25.8
Solid Rocket	91
TOTAL	227.6

VI. DISCUSSION OF RESULTS

The problem of modifying the trajectory of an orbiting body so that it will intersect the earth's surface at a specified time and location appears to yield to simple, straightforward techniques. The method investigated here, that of using a rocket to change the velocity vector of the package to be recovered, is particularly suitable when the atmosphere and/or the orbital properties are not known accurately.* Use of a rocket assumes at least a gross control of the body's attitude--it would be sufficient to be able to point approximately downward.

The problem of protecting payload items from aerodynamic heating during re-entry into the earth's atmosphere calls for two approaches, both of which appear to be feasible. The payload can be insulated from the heated surface with conventional materials and techniques; and the aerodynamic heating of the outer shell can be reduced through control of the re-entry flight path by means of a double-layered fiberglas parachute. Fibers with better high-temperature characteristics than glass, such as quartz, should be investigated. More knowledge is also needed about the behavior of light, high-drag devices in a hypersonic, rarified gas stream and about the reaction of parachute elements (double-layered or otherwise) to highly transient heating loads.

*When sufficient knowledge and control are available, one might consider flying an orbiting body at a critically low altitude and commanding significant increases in drag to effect a return.

The final problem, that of locating the payload after impact, does not appear to be of unusual difficulty. A small radio beacon and, possibly, flotation gear contained in the recovery package appear to offer a reasonable solution. The long descent times might permit airborne units to be in the immediate vicinity when the package reaches the surface.

In conclusion it may be said that the successful recovery of useful material from orbiting vehicles appears to be an attractive possibility. Because of the preliminary nature of this study no conclusive judgment can be made of the method outlined here for returning payloads, but the inherent simplicity of this method suggests that it be given further attention.

Appendix A

DETERMINATION OF PROTECTIVE CASE TEMPERATURES

The efficacy of an insulating layer when protecting against highly transient heating loads is a function of two of its properties, the conductivity and the heat capacity. Low conductivity and high heat capacity are desirable, but are generally incompatible in a homogeneous material.

The heat capacity of an insulating layer can be increased by the insertion of discrete layers of a material of high conductivity and large heat capacity. If these layers are oriented perpendicular to the direction of heat flow, and if the structure of the insulating material is fine compared to the interval between layers the conductivity of the insulating layer will not be increased.

This effect can be demonstrated as follows--a step function is assumed for the temperature of an outside wall, (i.e., for $t < 0$, $T_w = 80^oF$ and for $t \leq 0$, $T_w = 700^oF$). The effectiveness of an insulation, composed of a total of 2 in. of fiberglas and 0.10 in. of steel, is investigated for the following conditions: (a) a conventional arrangement of a 2 in. slab of fiberglas backed by a 0.10 in. skin of steel; (b) 2 slabs of fiberglas, each 1 in. thick, separated by a 0.05 in. sheet of steel, and a 0.05 in. inner case; and (c) 3 slabs of fiberglas, each 0.667 in. thick, separated by 0.033 in. steel sheets, and an inner case of 0.033 in. steel.

For each of these three conditions the temperature of the inner case was determined as a function of time using the following physical properties--

conductivity of fiberglas, $h_c = 0.024 \frac{BTU}{HR\ FT\ {}^oR}$

specific heat of fiberglas, c = 0 (assumed)

conductivity of steel, $h_c = \infty$ (assumed)

specific heat of steel, c = 0.12

The results of this calculation are plotted in Fig. 5. The definite but transient advantage of multiple layers is demonstrated.

The insulating effectiveness of category (c) has been investigated for an outer wall exhibiting the temperature history of an insulated skin (of equal thickness, i.e., 0.05 in.) re-entering the atmosphere along the trajectory plotted in Fig. 10. This trajectory was determined for a re-entering warhead but, as can be seen in Fig. 3, it falls well within the range of trajectories expected from re-entering orbiting bodies.

The following assumptions governed the determination of the outer wall temperature. It was conservatively assumed that the surface areas effective in convection and radiation were equal. (The expression for the convective heat transfer coefficient is based on area projected in the velocity direction, while that for radiation is based on total area.) Atmospheric properties were obtained from Ref. 4.

The temperatures of the outer case and of the inner layers are plotted as functions of time in Fig. 6. It appears possible to protect a re-entering payload using relatively simple techniques.

Appendix B

DETERMINATION OF SYSTEM WEIGHTS

The techniques and mechanisms employed in recovering film from a satellite can be examined in terms of the orbiting weights that are implied. Component weights have been estimated for an elementary system (see Fig. 8) using the following assumptions:

The film and reel and the radio beacon, plus its power, switching, etc., were assumed to have over-all specific gravities of approximately 0.5. With efficient packaging they might be enclosed within a surface area equal to twice that for a sphere of equal volume. Of this area three-quarters is covered by complete insulation (an outer case of 0.05 in. steel followed by three equal layers of 0.667 in. fiberglas and 0.033 in. steel) and the remaining one-quarter is covered by the inner 0.033 in. steel only. The density of steel is taken as 485 lb/ft^3 and that of fiberglas as 7 lb/ft^3.

Starting with an arbitrary film weight, increasing it by 10 per cent to allow for the reel, and adding 10 lb for the beacon, etc., the total weight of the insulated package can be determined. The size and weight of the required parachute can be determined directly knowing the package weight, the area-weight ratio of the parachute, and stipulating that the ratio, $\frac{C_D A}{W}$, for the package-parachute combination should equal 4. (C_D = 1.4, Ref. 3).

Parachute weights were determined for double-layered elements throughout and a 100 per cent margin of safety on the inner, load-carrying members. A ratio of 0.06 lb/ft^2 was used. The parachute weights determined in this fashion were increased by 10 per cent to allow for positive opening devices.

If the total weight of the package plus parachute is known, the size of rocket can be determined that is needed to accomplish the specified deceleration. The specific impulse for a solid rocket was assumed to be 200 sec and the ratio of propellant weight to total rocket weight (ν^*) to be 0.6.$^{\Delta}$

The aforementioned methods were used to estimate total weights as functions of film weight, altitude and dump range. This relationship is plotted in Fig. 9. Table I itemizes the component weights for a system returning 50 lb from a 150 mi orbit with a dump range of 2500 n mi.

The weight estimates do not include effects on the design of the orbiting stage. The structure of the orbiting vehicle should accommodate the recovery system so that its components would require a minimum of readjustment before firing.

$^{\Delta}$These assumptions introduce a considerable degree of conservatism.

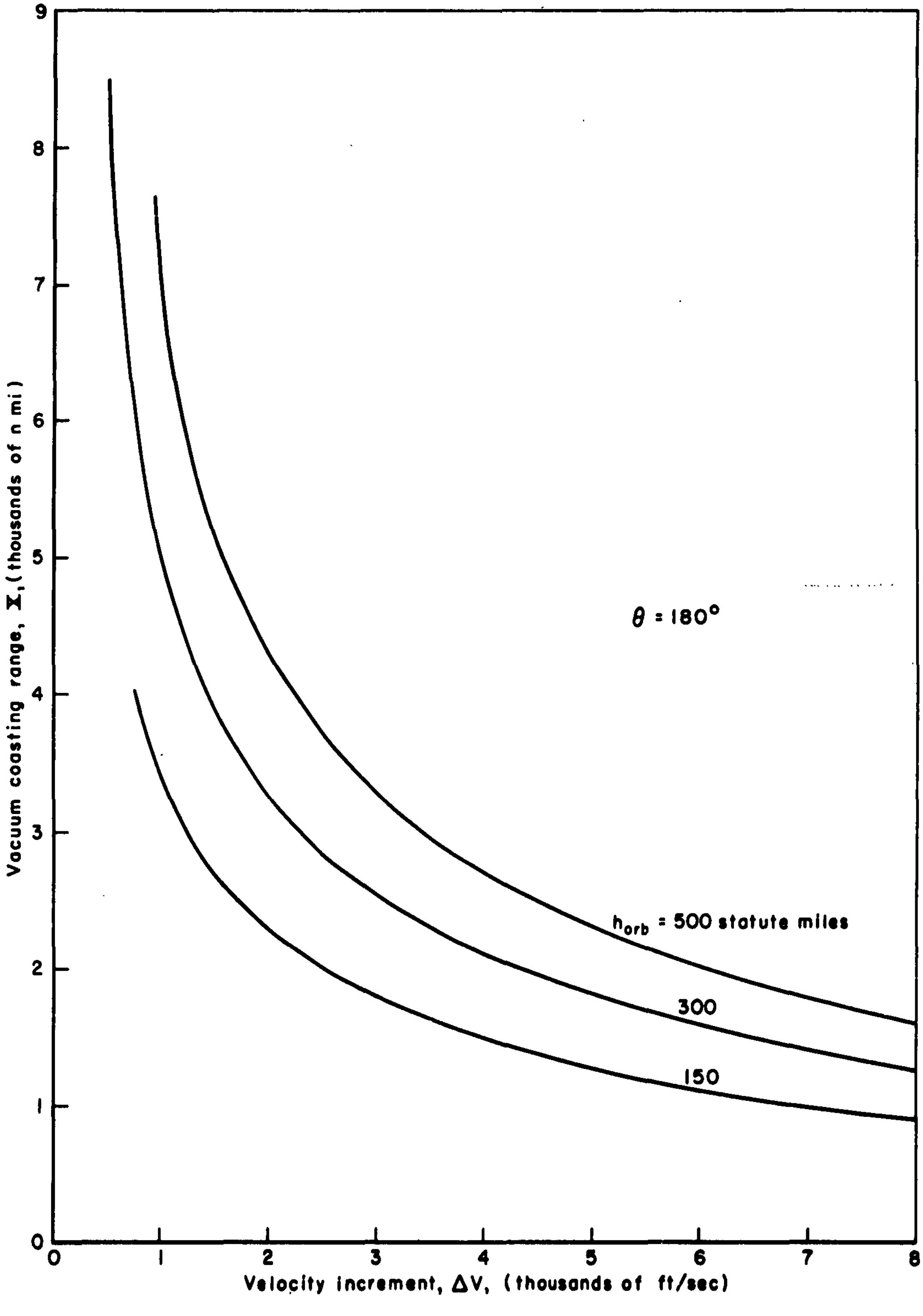


Fig. 1—Ground range vs velocity increment for package drop from a satellite

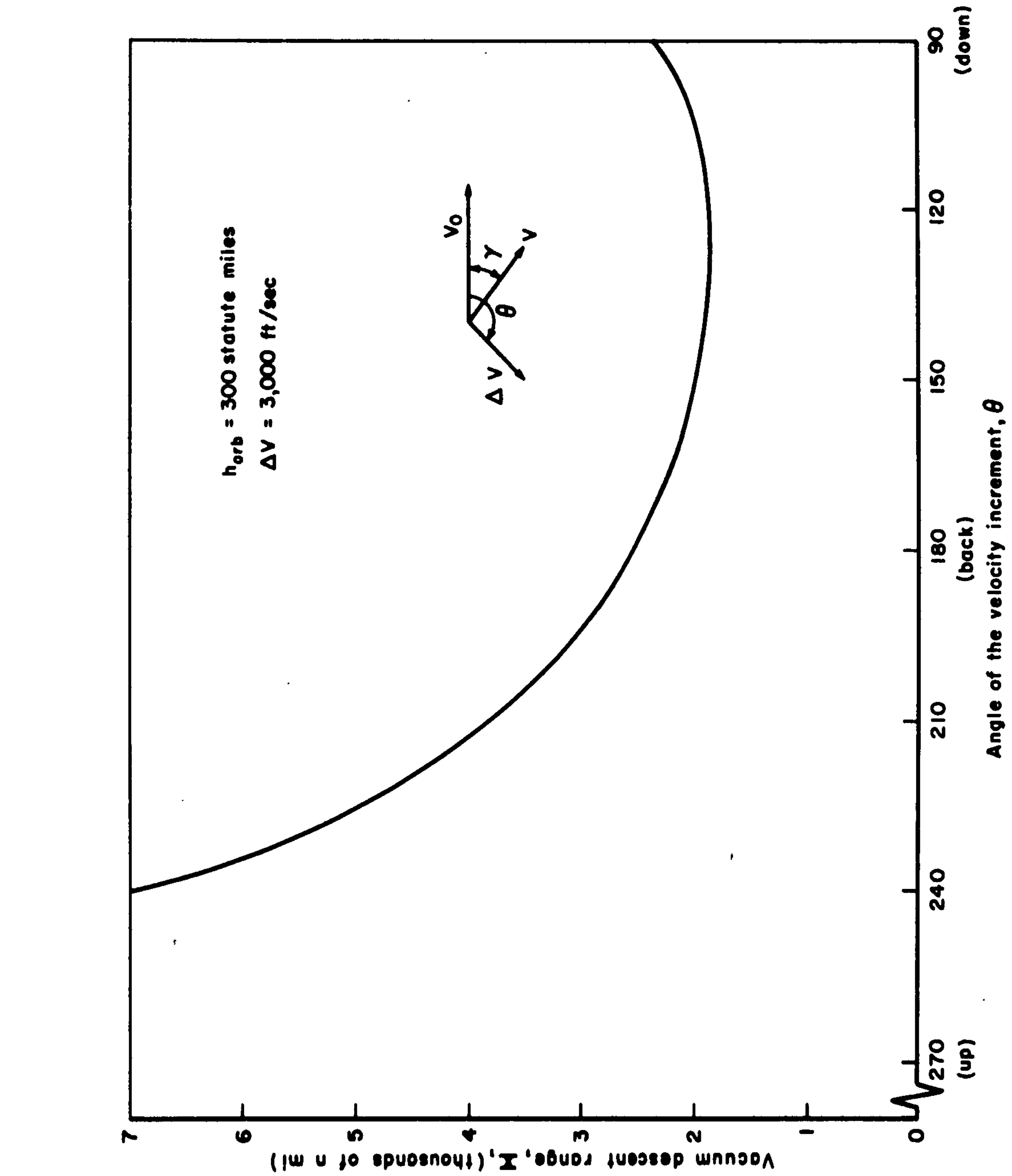


Fig. 2—Vacuum descent range vs angle of velocity increment

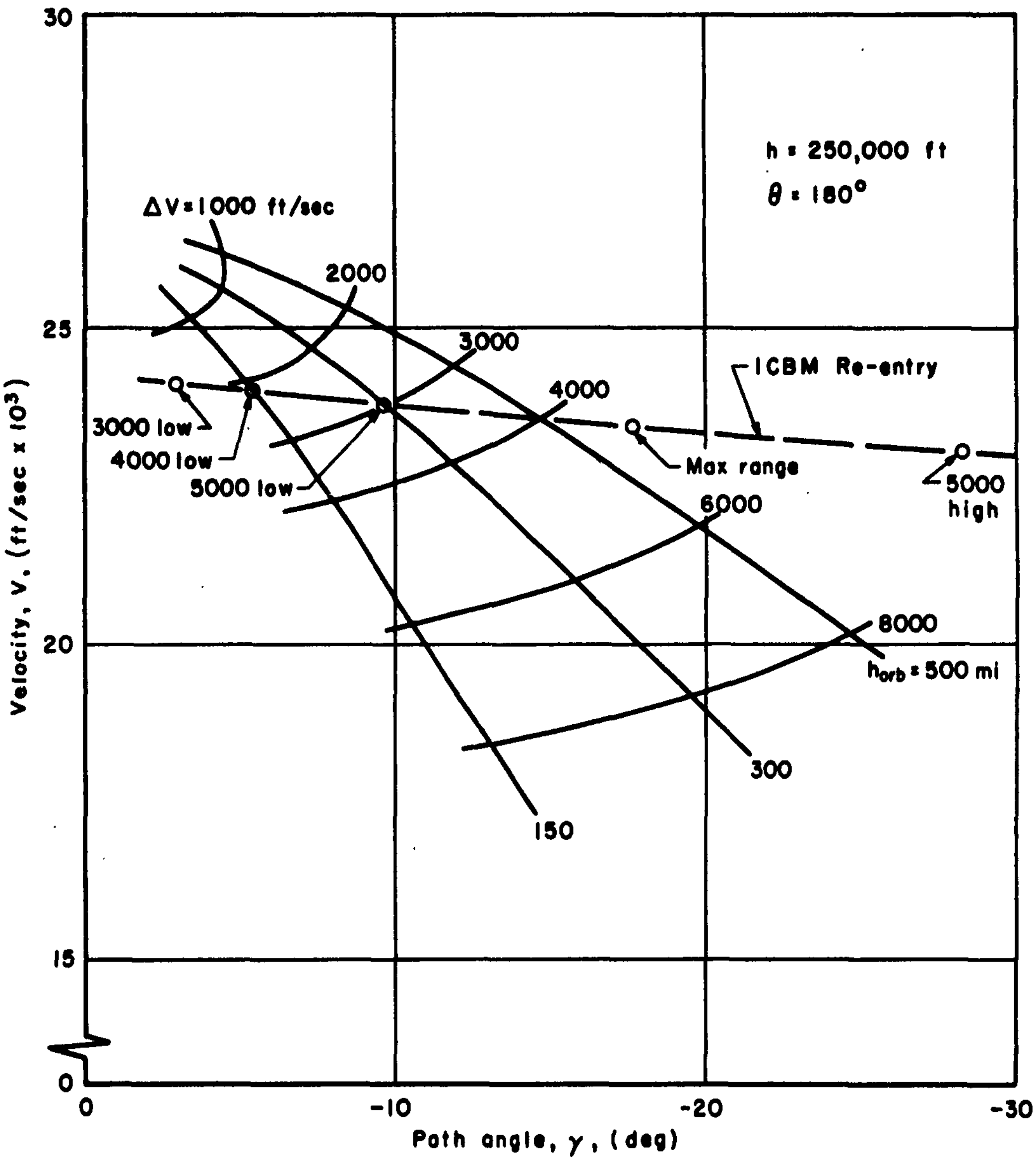


Fig. 3a — Re-entry initial conditions

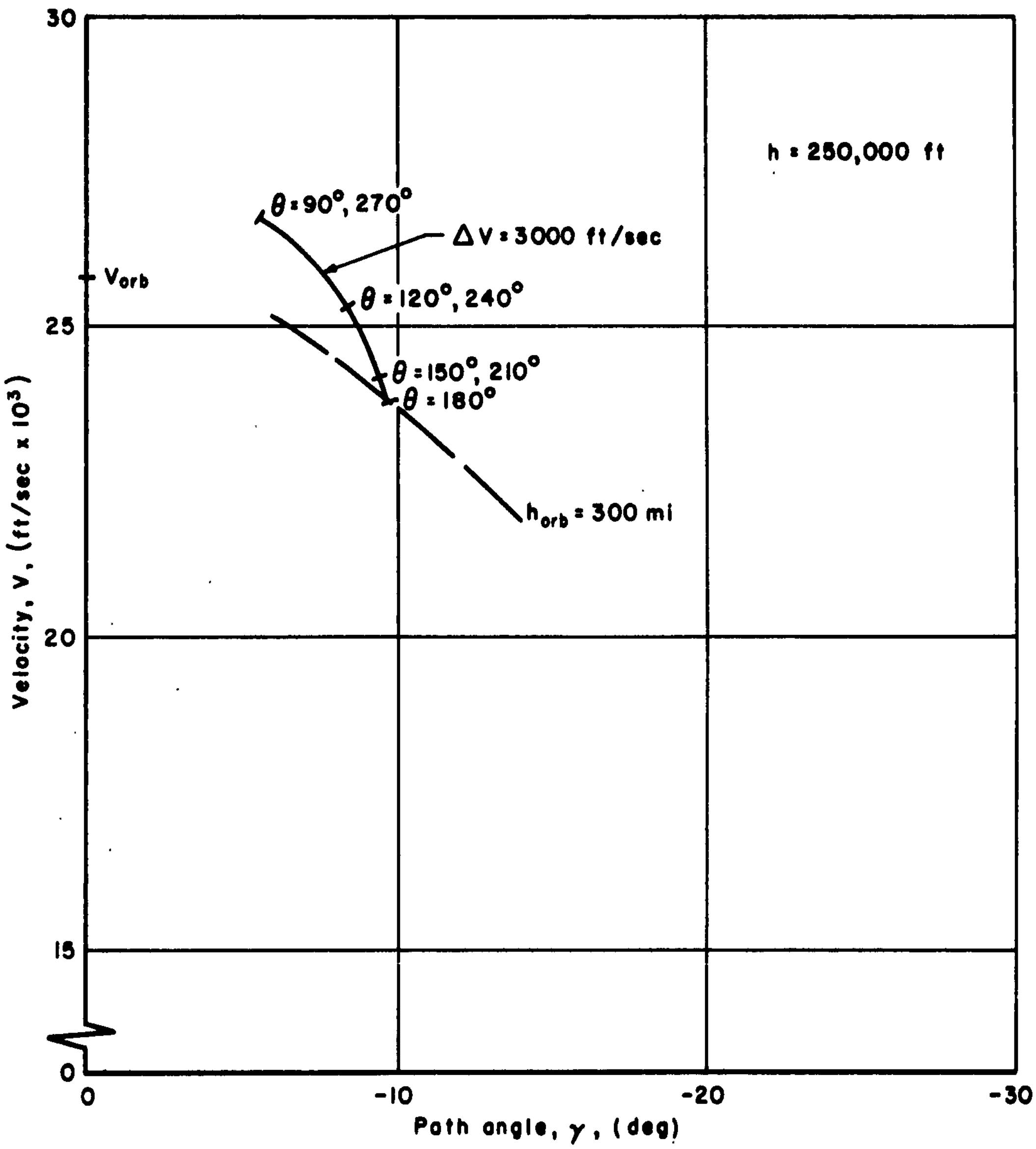


Fig. 3b — Re-entry initial conditions

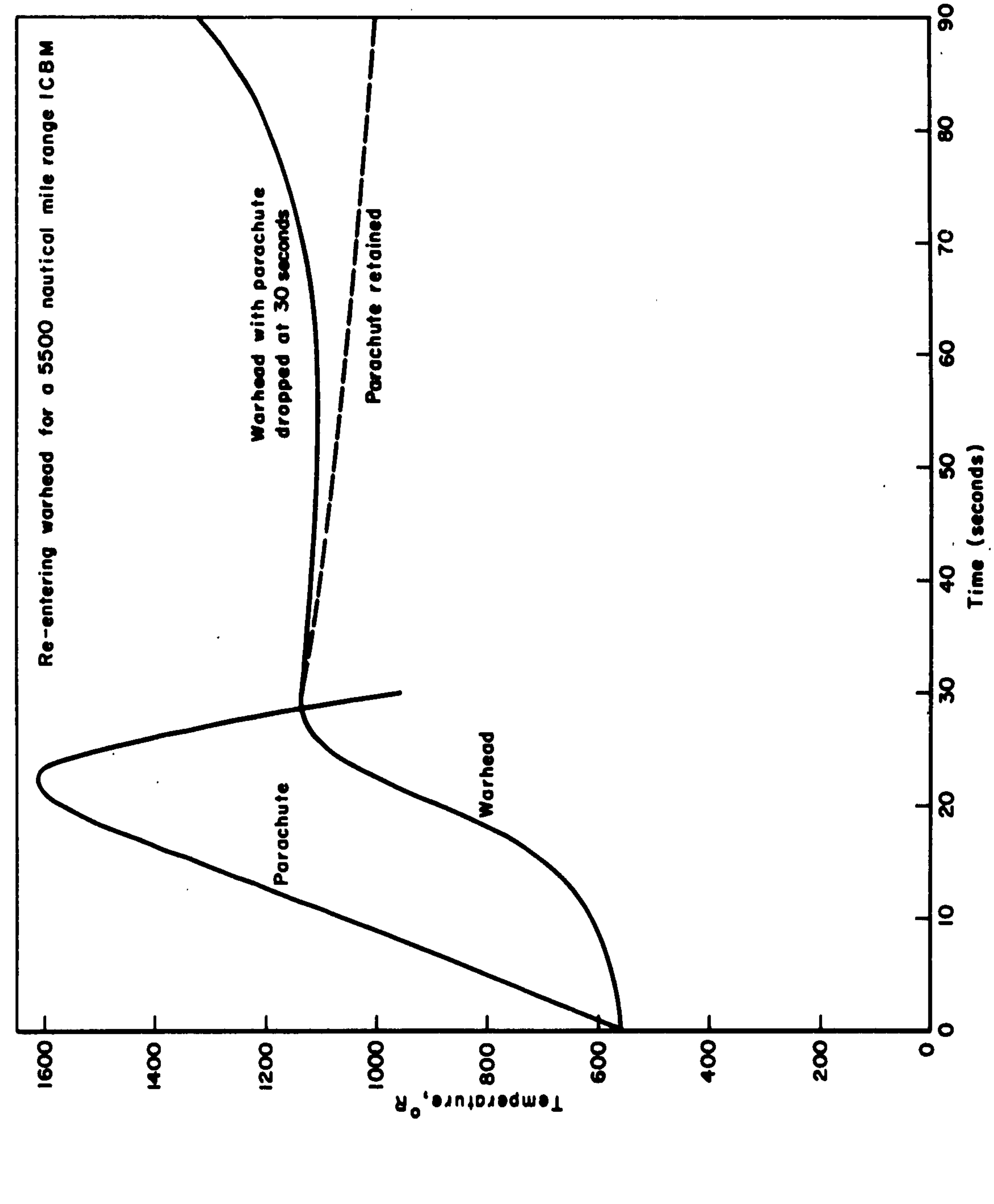


Fig. 4 — Warhead and parachute temperature vs time

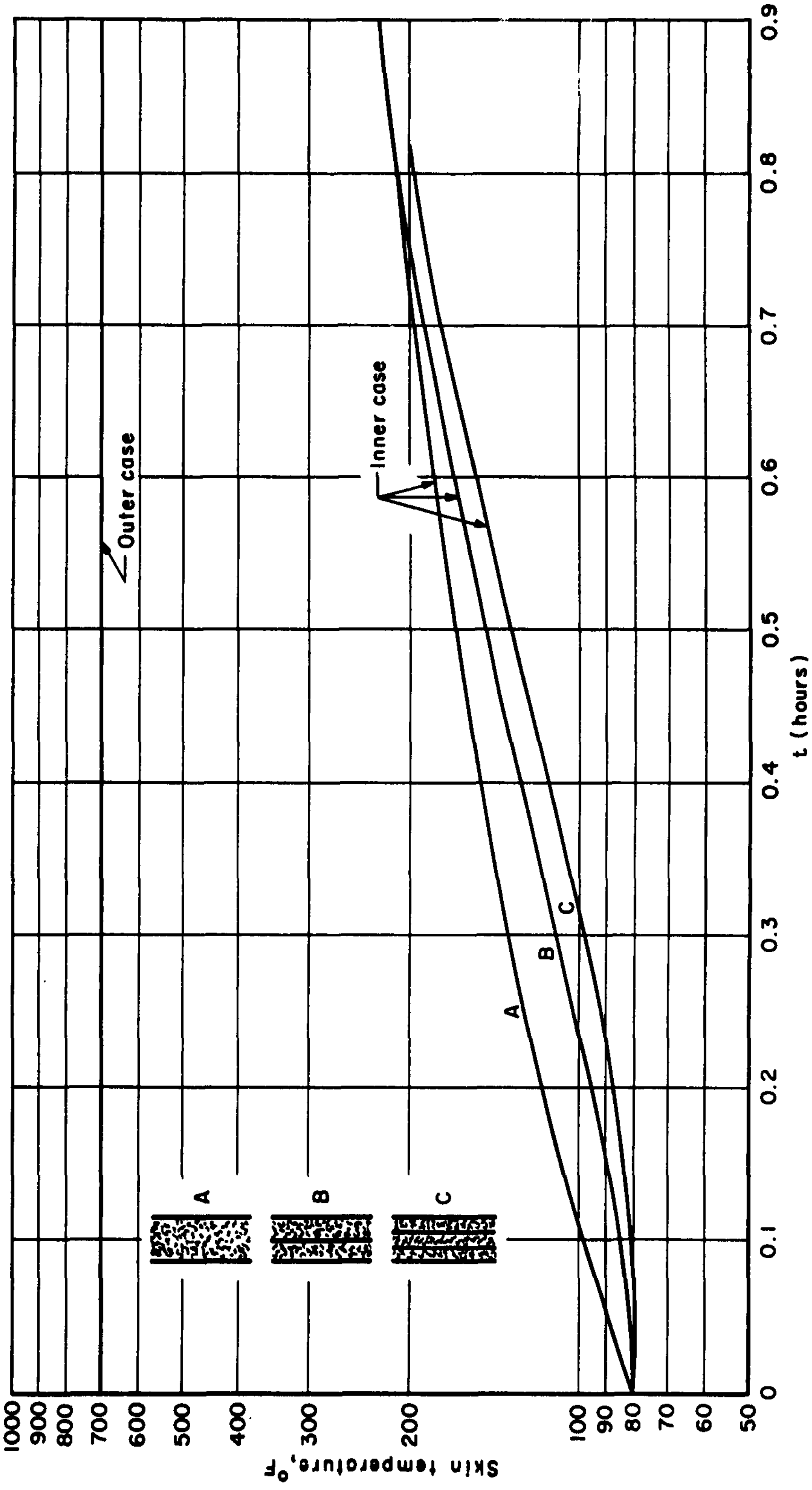


Fig. 5 — Temperature vs time for multi-layered insulation

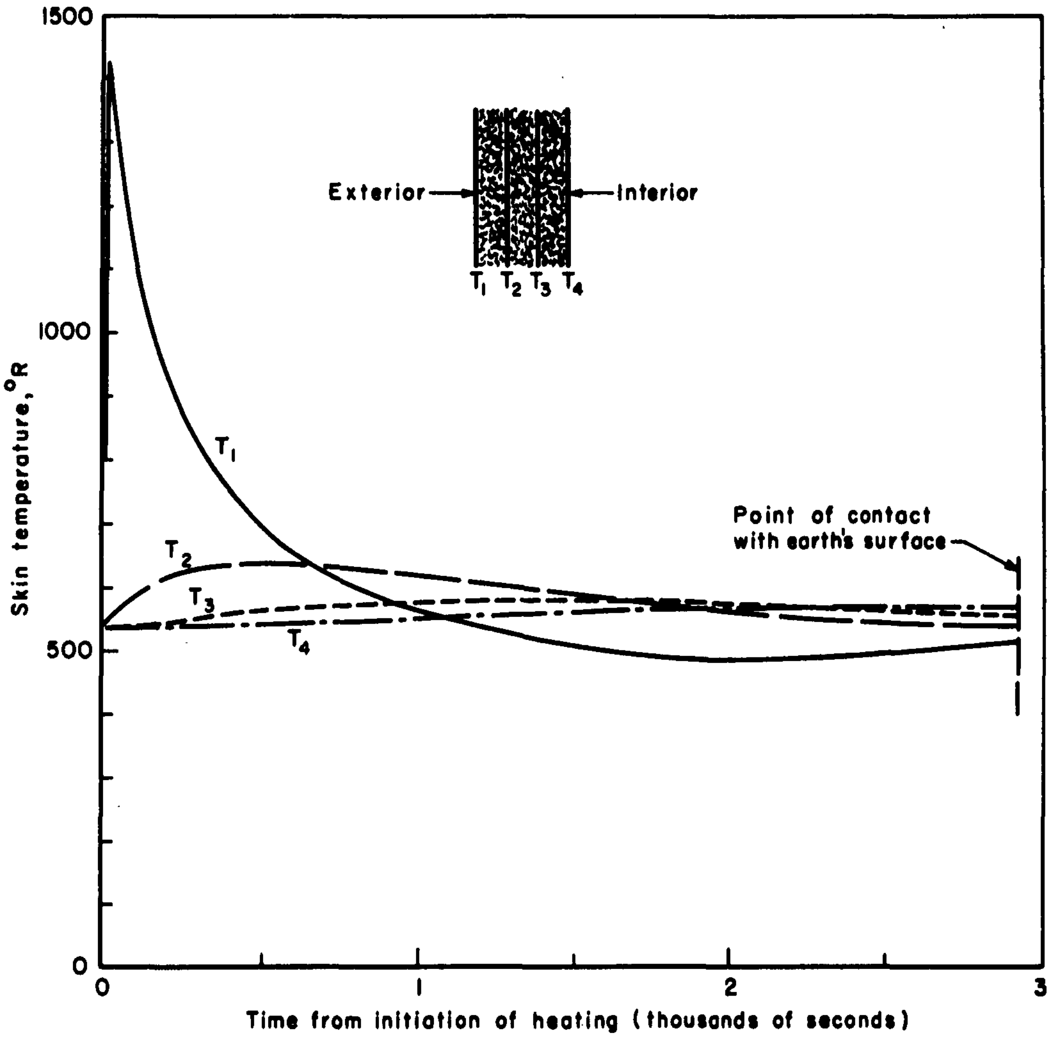


Fig. 6 — Temperature vs time for an insulated re-entering body

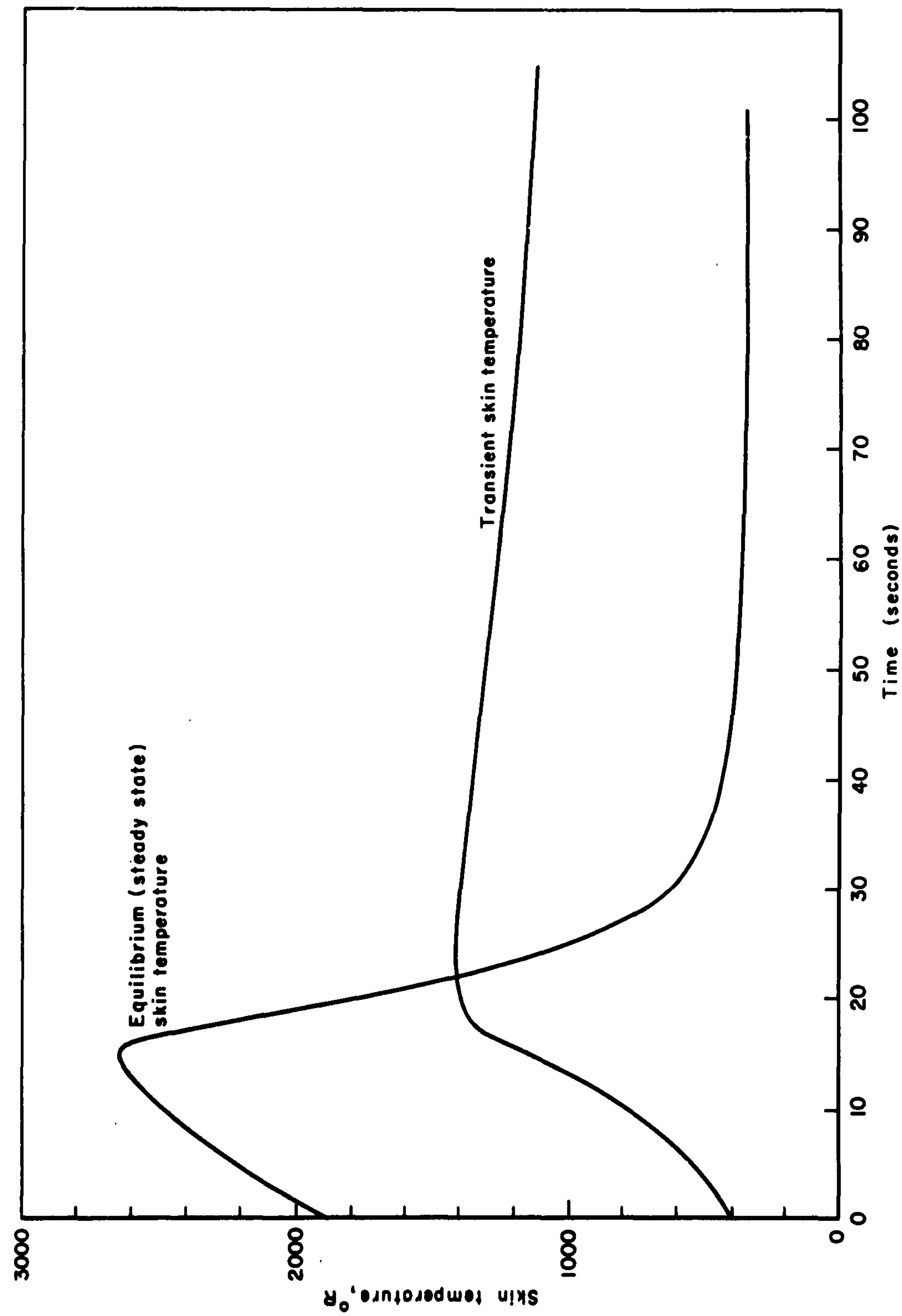


Fig. 7 — Transient and equilibrium outer skin temperatures during re-entry

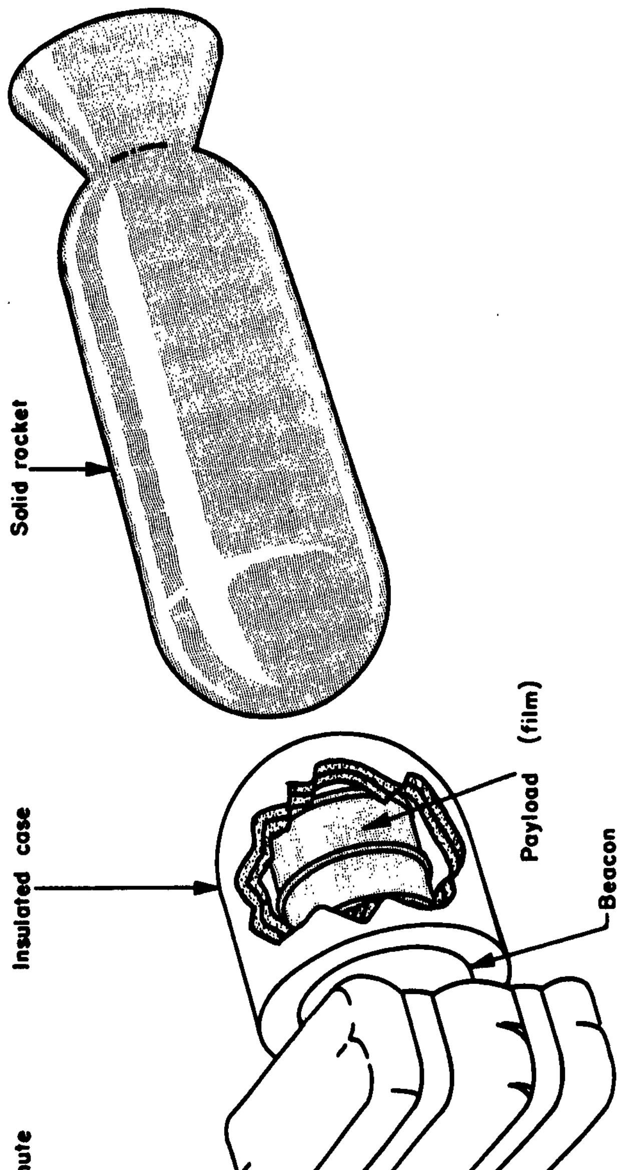


Fig. 8 — Basic elements of a recovery package

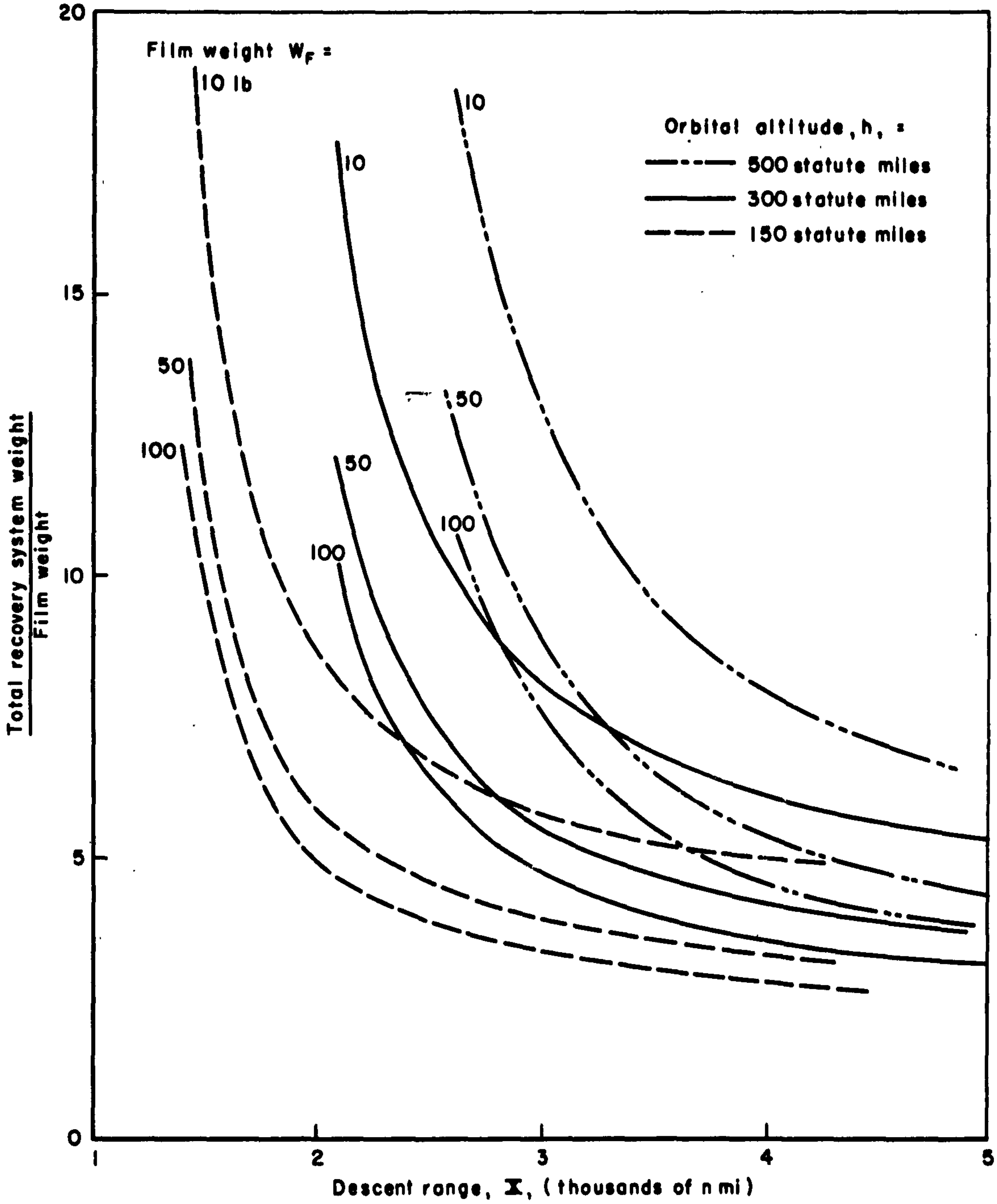


Fig. 9—Ratio of total weight to film weight vs range for varying altitude and film weight

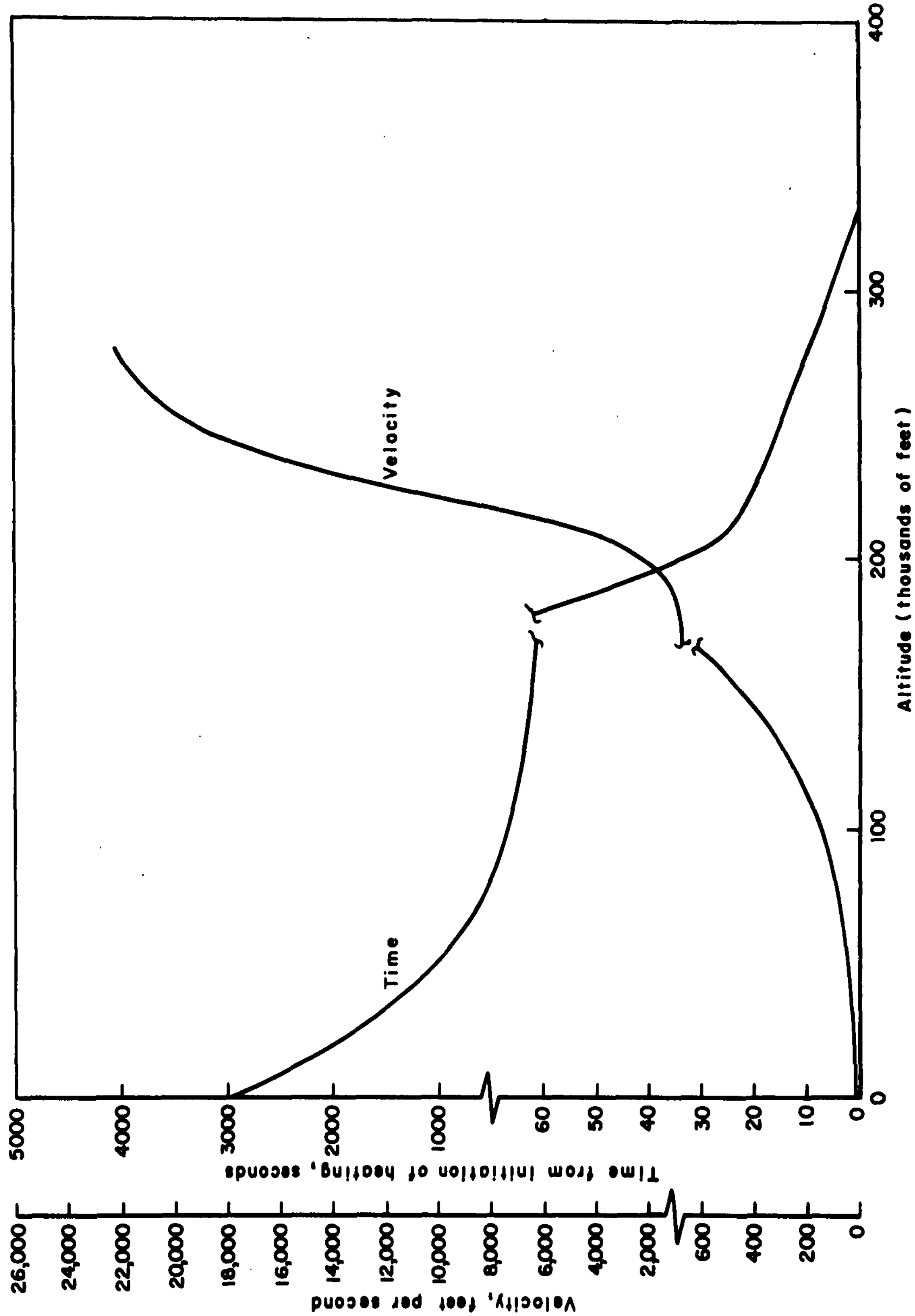


Fig. 10—Re-entering trajectory

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