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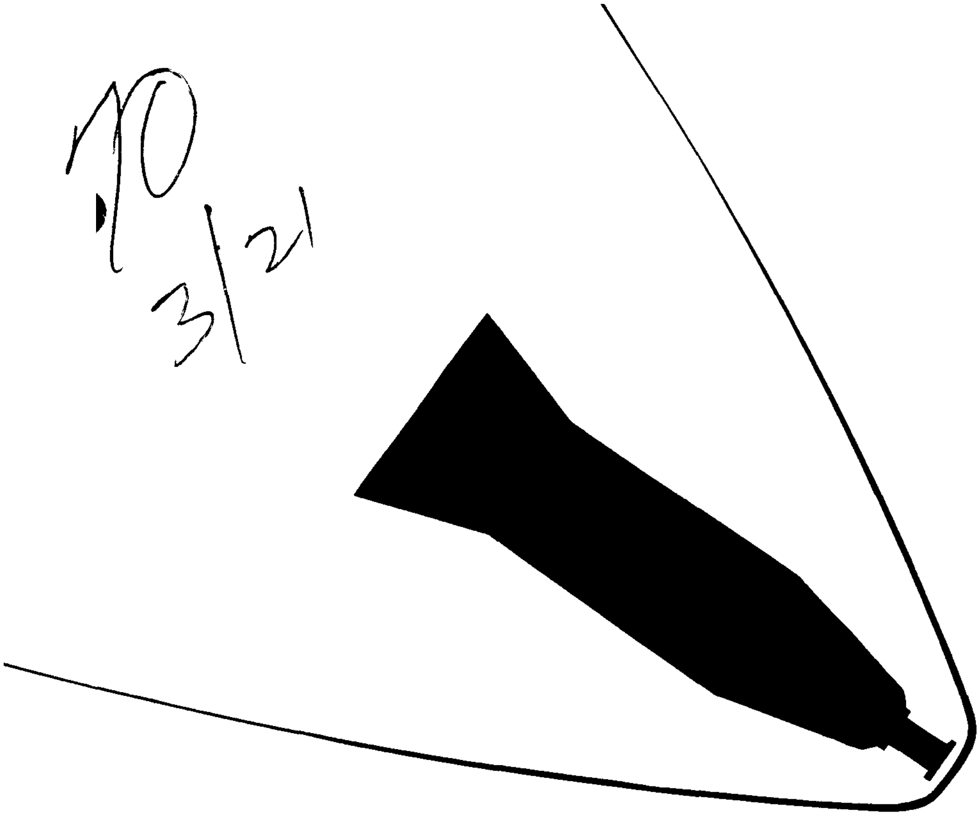

MASTER

SC-RR-65-472

AEROSPACE NUCLEAR SAFETY

CONTROLLED DEORBIT

J. A. Leonard, 9319
W. W. Joseph, 9319

SANDIA CORPORATION

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March 1966

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ABSTRACT

This report discusses controlled deorbit as a means of disposal of orbiting SNAP satellites to reduce radiological hazards. Two types of controlled deorbit systems are considered: the command initiated retro-rocket system and the command deployable high-drag device. An example situation of deorbit from an eccentric transfer orbit, similar to that flown by SNAP-10A, is examined.

SUMMARY

This report deals with one of the most important areas of concern in the field of Aerospace Nuclear Safety, namely, the disposal of orbiting nuclear power supplies. In particular, the method of disposal covered in this report is controlled deorbit. A controlled deorbit system aboard a SNAP satellite would give the ground controllers the capability of terminating the orbit of the satellite so that it would impact an ocean area.

Trajectory analyses and weight estimates have been made for two types of controlled deorbit systems:

1. Command initiated retro-rocket system
2. Command deployable high-drag device

The drag deorbit system has a basic advantage over the retro-rocket system in that it does not need an attitude control system. On a weight basis, the drag deorbit system is superior to the retro-rocket system for near earth orbits. As orbital altitude increases, the drag system loses efficiency, and at some "cross-over height" the retro-rocket system gains and retains the advantage. The cross-over height depends on the weight assumed for an attitude control system. For instance, the crossover height would be 650,000 feet for a 1000-pound satellite in circular orbit if a weight of 39 pounds is assumed for an attitude control system.

LIST OF SYMBOLS

A - area (ft^2)

C_D - coefficient of drag

g - acceleration (gravitational units)

g_o - acceleration due to gravity at surface of earth (ft/sec^2)

h - height or altitude (ft)

I_s - specific impulse (sec)

P - propellant weight as a percentage of payload weight

$P(I)$ - probability of injury from a reentering satellite

$P(I/L)$ - probability of injury given land impact

$P(I/W)$ - probability of injury given water impact

$P(L)$ - probability of a reentering satellite impacting land

$P(W)$ - probability of a reentering satellite impacting water

R - reliability of a deorbit system

R_d - range of satellite after retro-rocket initiation or deployment of drag device (degrees of arc or degrees of central angle)

R_o - radius of the earth (ft)

r - distance from center of earth to satellite (ft)

r_a - distance from center of earth to satellite at apogee (ft)

r_p - distance from center of earth to satellite at perigee (ft)

s - surface distance (statute miles)

V_a - velocity at apogee (ft/sec)

V_c - circular velocity (ft/sec)

V_p - velocity at perigee (ft/sec)

W - weight (lbs)

W/C_DA - ballistic coefficient (lbs/ft^2)

W_B - weight of balloon (lbs)

W_{PL} - payload weight (lbs)

W_{TOT} - total weight (lbs)

α - central earth angle or depression to horizon (degrees)

γ - flight path angle (degrees)

ΔV - increment of velocity (ft/sec)

μ - gravitational constant (ft^3/sec^2)

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CHAPTER I. INTRODUCTION

Disposal of orbiting nuclear power supplies is one of the important concerns in the field of Aerospace Nuclear Safety because of the radiological hazards presented by the nuclear power supplies.

An obvious means of providing radiological safety is to place the SNAP satellite in an orbit that has a lifetime long enough to permit all radioactive materials aboard to decay to safe levels. Unfortunately a long-lived orbit implies a high-altitude orbit, and this is not always consistent with the objectives of a mission. Even when a long-lived orbit is programmed, it may not be achieved because of some malfunction. Therefore, the planners of SNAP missions are obliged to consider ways to minimize hazards in disposing of the radioactive material in the power supplies.

One possible disposal method is controlled deorbit to an ocean burial. Controlled deorbit is the technique of termininating the orbit of a satellite upon command in such a way that impact will occur in a preselected area.

This report presents two methods for achieving controlled deorbit, namely, retro-rocket and high drag. The two systems are described to the extent of outlining general hardware requirements and of giving weight estimates. Furthermore, the effectiveness of each system in terms of weight penalties for a variety of orbital situations is discussed.

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CHAPTER II. SAFETY IMPROVEMENTS

The reason for choosing an ocean burial is that the hazards to the populace are significantly less from a SNAP satellite which impacts the ocean than from one which impacts land. Reference 1, which contains an analysis of hazard from sea water release of an Sr-90 capsule, concludes that the chance of an individual's ingesting a significant dose is "nonexistent or extremely remote." In Reference 2, which evaluates the probabilities of injury in the random reentry situation, the entire probability of causing injury is assigned to those fuel cells which impact land.

The reduction in hazard to be expected by using controlled deorbit as opposed to random reentry can be shown to be inversely proportional to the unreliability of the deorbit system as follows:

The probability of injury from a payload impacting the earth's surface at a random location is P(I). It can be expressed as

$$P(I) = P(L)P(I/L) + P(W)P(I/W)$$

where

$P(L)$ is the probability of land impact,

$P(W)$ is the probability of water impact,

$P(I/L)$ is the conditional probability of injury given land impact, and

$P(I/W)$ is the conditional probability of injury given water impact.

Since

$$P(I/W) \ll P(I/L),$$

the simplifying approximation

$$P(I) = P(L)P(I/L)$$

can be used. If P(L) can be decreased by controlled deorbit, then obviously the probability of injury decreases proportionately. A controlled deorbit system, designed to bring a payload from orbit to an ocean area, would change P(L) by a factor of (1 - R) where R is the reliability of the deorbit system. The equation then becomes

$$P(I) = (1 - R)P(L)P(I/L).$$

If the reliability of a deorbit system composed of commercial components is assumed to be

$$R = 0.95 \text{ (5 failures in 100)},$$

then the injury probability utilizing this system is

$$P(I) = (1 - 0.95)P(L)P(I/L)$$

$$= 0.05\, P(L)P(I/L).$$

For this case, controlled deorbit to an ocean area has reduced the probability of injury by a factor of 20 (1/0.05) over the comparable probability arising from random impact. The probability of injury can be decreased further by the selection of more reliable deorbit system components.

CHAPTER III. SYSTEM DESCRIPTION

The basic requirements for a system which would deorbit a vehicle from a purposely-achieved short-lived or medium-lived orbit differ somewhat from those of the emergency situation. The emergency situation would be a short-lived orbit perhaps arising from a loss of thrust in an upper stage rocket or from a control malfunction. The emergency would arise within minutes of launch, and controlled deorbit would be initiated within the first few orbits. Because the type of emergency cannot be forecast, the deorbit system should be entirely independent of the on-board guidance, control, and boost systems used to orbit the vehicle. Two types of deorbit systems will be discussed: the command initiated retro-rocket system and the command deployable high-drag device. The high-drag system for the emergency condition should have the following minimal capabilities: payload separation from the rocket motor, drag device, and telemetry and command. The retro-rocket deorbit system would replace the drag device with a retro-rocket and add an attitude sensing and control subsystem.

Some economy of weight could be achieved by using those systems which are already on board the satellite and are applicable to the deorbit system such as power supply, separation, and sensing and control devices. This economy carries with it the risk that such a dual-purpose component will not be operable when it is needed for controlled deorbit.

Vehicle Separation

Vehicle separation may be effected from the booster mechanically by spring-loaded latches (ground-command or timer initiated) or explosively. Explosively, separation could be accomplished by squib-actuated mild-detonating fuse (MDF) at a joint between the booster and the vehicle.

Attitude Sensing and Control

While a drag deorbit system utilizing a spherical body is insensitive to attitude, a retro-rocket deorbit system is sensitive to this variable during the application of retrothrust. To mitigate this sensitivity, an attitude control subsystem should be incorporated in the retro-rocket deorbit system. The attitude control subsystem would sense a body-centered "vertical" and the trajectory-related "forward" and keep prescribed vehicle axes aligned with this reference frame at least during the time of retrothrust. To help maintain proper attitude during application of retrothrust and to correct for thrust misalignment, spin stabilization would be a desirable feature in the attitude control subsystem.

Telemetry and Command

The telemetry and command subsystems serve to inform ground control of vehicle condition, to receive commands, and to transmit performance of commands. The necessary events that require information transmission in the retro-rocket deorbit sequence are: (1) vehicle separation from booster, (2) attitude sensing and control, and (3) deboost power application.

Retro-Rocket Deorbit

The requirements for the retro-rocket engine needed for the controlled deorbiting discussed herein include:

1. High specific impulse, to minimize propellant weight
2. High propellant density, to minimize packaging volume
3. High ratio of propellant weight to rocket engine weight, to minimize retro-rocket subsystem weight
4. High reliability, which implies simplicity of operation
5. Economy, which implies simplicity of design

6. Excellent chemical stability without degradation of performance characteristics with time

7. Safe while being handled, transported, and flown, i.e., low explosion and toxicity hazard.

In addition, the only control needed for the retro-rocket is the initiation command, because neither pulsed operation nor thrust termination is necessary or desirable.

This report has not been predicated upon the use of solid or liquid propellant rockets to obtain the retrothrust but has merely assumed an instantaneously applied ΔV* obtained from a propellant with a given specific impulse.

Drag Deorbit

The requirements for a drag device to supplant a retro-rocket for controlled deorbiting include:

1. High drag to weight ratio

2. Low weight to minimize contribution to deorbit system weight penalty

3. Low packaging volume

4. High reliability which implies an easily deployable device.

Depending on the type of drag device used, the only control needed may be the initiation command to deploy. This report considers the drag device to be an inflatable balloon that supplies drag over a relatively long period of time as contrasted to the instantaneously applied ΔV of the retro-rocket.

*Throughout this report ΔV is used to denote an increment of velocity added to or subtracted from the magnitude of the orbital velocity vector of a satellite, i.e., unless otherwise stated, ΔV is assumed collinear with the orbital velocity vector.

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CHAPTER IV. OPERATION OF DEORBIT SYSTEMS

Orbital Mechanics

Several relationships which are useful in orbital mechanics will be presented at this time without proof. A review of these relationships may prove helpful in recognizing the many parameters which must be considered in the controlled deorbit problem. For this presentation, a spherically homogeneous earth and no atmosphere or other sources of perturbing forces are assumed.

Circular velocity is that velocity required to maintain a circular orbit at a given height:

$$V_c = \sqrt{\frac{\mu}{r}} \tag{1}$$

where r is the distance to the center of the earth and μ is the gravitational constant,

$$\mu = g_o R_o^2, \tag{2}$$

g_o being the acceleration of gravity at the surface of the earth and R_o the radius of the earth.

For eccentric orbits, the velocity at apogee and perigee (V_a and V_p) are inversely related to the orbital radius at apogee and perigee (r_a and r_p):

$$\frac{V_a}{V_p} = \frac{r_p}{r_a} \tag{3}$$

and

$$V_a = \sqrt{\frac{\mu r_p}{a r_a}}\,, \tag{4}$$

$$V_p = \sqrt{\frac{\mu r_a}{a r_p}}\,; \tag{5}$$

where a is the semimajor axis of the orbital elipse,

$$a = \frac{r_a + r_p}{2}\,. \tag{6}$$

The terms "eccentric orbit" and "elliptical orbit" will be used synonymously to mean a noncircular orbit.

Perturbation Theory as Applied to Deorbit

Instantaneous ΔV, Circular Orbit

Any force applied to an orbiting body will perturb its orbit. In case of a circular orbit, if an increment of velocity (ΔV) is added instantaneously to orbital velocity, that point in the orbit becomes the perigee of a new eccentric orbit of higher energy. For this study, the thrust of a retro-rocket can be assumed to result in an instantaneous ΔV. Again, for the case of a circular orbit, if a ΔV is subtracted from the orbital velocity, that point in the orbit becomes the apogee of a new lower energy orbit. If the reduction in velocity is sufficient, the perigee height of the new orbit will be so low that the satellite will graze the surface of the spherical, no-atmosphere earth. The first passage through the new perigee would occur exactly one-half orbit after the application of ΔV.

From the above discussion, it can be seen intuitively that one-half orbit (180 degrees) after application of ΔV represents the maximum range that is useable for controlled deorbit. Maximum range implies the minimum ΔV and, therefore, the minimum weight penalty for a retro-rocket. While the application of more ΔV would result in a shorter range to impact, the application of less ΔV would fail to reduce the perigee height to zero and the satellite would continue in orbit. There exists, therefore, a unique ΔV for any given circular orbit which will perturb that orbit into an earth intersecting orbit of zero perigee height. This ΔV can be calculated from Equations 1 and 4 by calculating the velocity of the circular orbit, V_c, and the velocity at apogee, V_a, of the earth intersecting orbit ($r_p = R_o$). The difference between these velocities is ΔV.

Restoration of the atmosphere changes the basic problem very little. Perigee altitude, instead of zero, need only be reduced to the height at which atmospheric capture will take place. The "capture altitude" is a function of the ballistic coefficient of the satellite, its velocity at perigee, and the parameters of the model atmosphere.

Figure 1 is a plot of ΔV as a function of altitude for 180-degree deorbit range, without atmosphere, where the earth model is a sphere of radius, R_o, 20, 855, 107 feet. For comparison purposes, Figure 1 also includes a plot of the same function using an atmosphere and an oblate earth. This curve was derived through use of a CDC 3600 digital computer and Sandia Corporation's TTA* program. The TTA program models the earth as a rotating oblate spheroid of radius 20, 855, 107 feet at the poles (used above) and radius 20, 925, 540 feet at the equator. The atmosphere model used in computation was the theoretical 1959 Air Research and Development Command (ARDC) atmosphere. The TTA program in conjunction with the 1959 ARDC atmosphere has been used for all computer work in this report.

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*A generalized rigid-body trajectory program for digital computer.

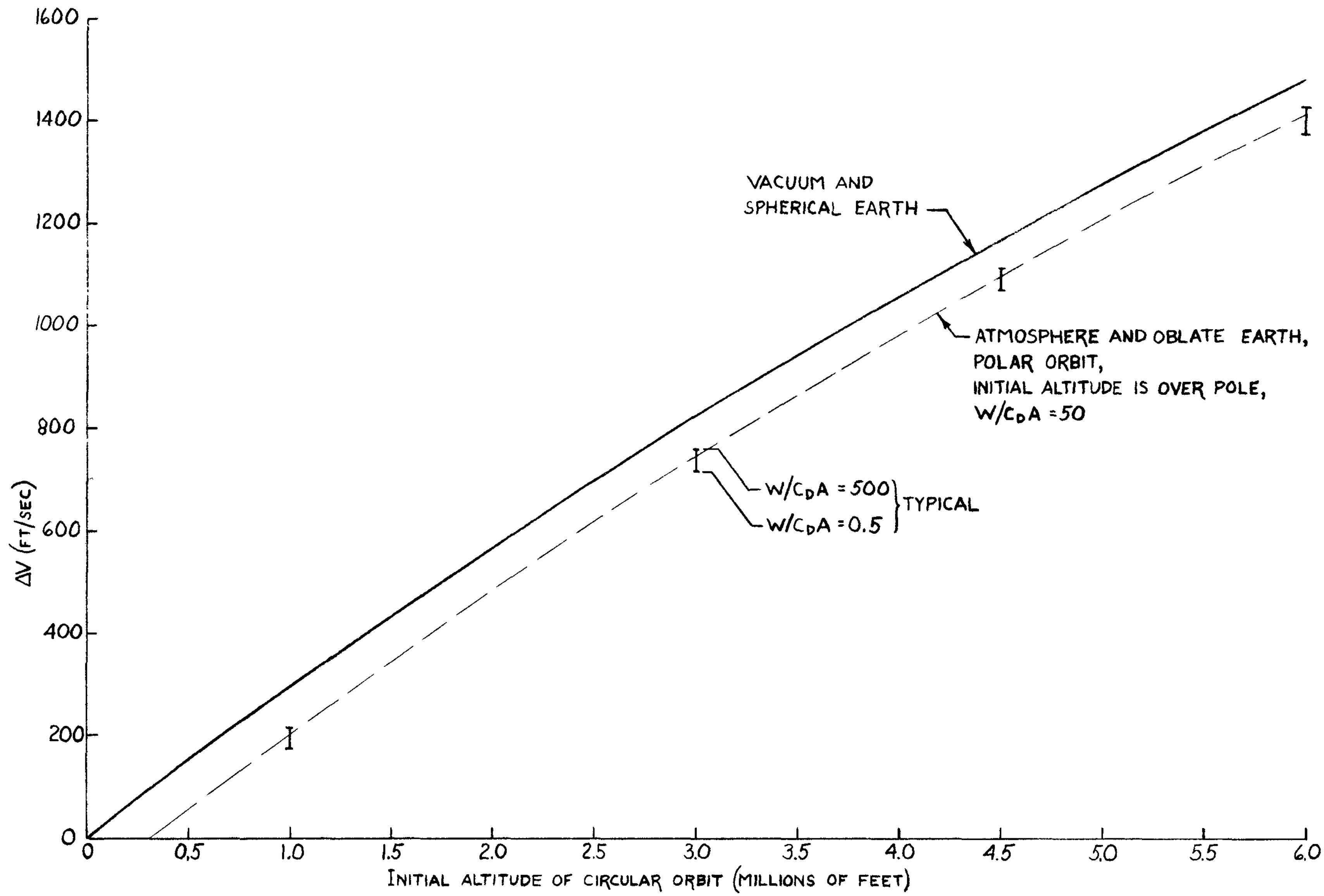

Figure 1. ΔV Required for 180-Degree Deorbit Range from Circular Orbit

The curve for ΔV, in the atmosphere case, in Figure 1 is based on a rigid body with a ballistic coefficient ($W/C_D A$) of 50 lbs/ft^2. Superimposed on this curve at altitudes of 1, 3, 4-1/2, and 6 million feet, the range of ΔV for $0.5 \leq W/C_D A \leq 500$ is shown. This demonstrates that ΔV is a fairly weak function of $W/C_D A$, at least for a deorbit range of 180 degrees.

Examination of Figure 1 shows, as may be expected, that when deorbiting through the atmosphere less ΔV is necessary to achieve deorbit in a given range.

Instantaneous ΔV, Eccentric Orbit

The perturbation of an eccentric orbit by the application of an instantaneous ΔV is similar to the perturbation of a circular orbit. In general, a ΔV added or subtracted at one apsis will raise or lower the height of the opposite apsis. For instance, a ΔV added to orbital velocity at the apogee of the orbit would result in a new orbit of greater height at perigee and of the original height at apogee.

The perturbation of interest for the controlled deorbit problem is the one caused by a reduction in velocity at apogee. This results in a reduction of perigee height one-half orbit later. Figure 2 portrays this effect. Since perigee is already the closest point to the earth, it seems reasonable that reducing perigee is the least expensive way to cause the orbit to intersect the earth.

As in the case of deorbit from circular orbit in a vacuum, there exists, for any initial orbit, a unique ΔV which, if applied at apogee, will result in a perigee height of zero. This ΔV can be calculated from Equation 4 by calculating V_a for both the initial orbit and the earth intersecting orbit. The difference between the two V_a's is ΔV.

Application of ΔV at points other than the apses or at angles other than along the velocity vector results in rotation of the axes of the orbital ellipse, or a shift in the orbital plane, as well as changes to the height of apogee and perigee. These will not be discussed in detail in this study, however, since the main interest will be in deorbit systems that impose, on the satellite, a minimum weight penalty which implies a collinear retrofire at apogee with ΔV sized for one-half orbit range to impact.

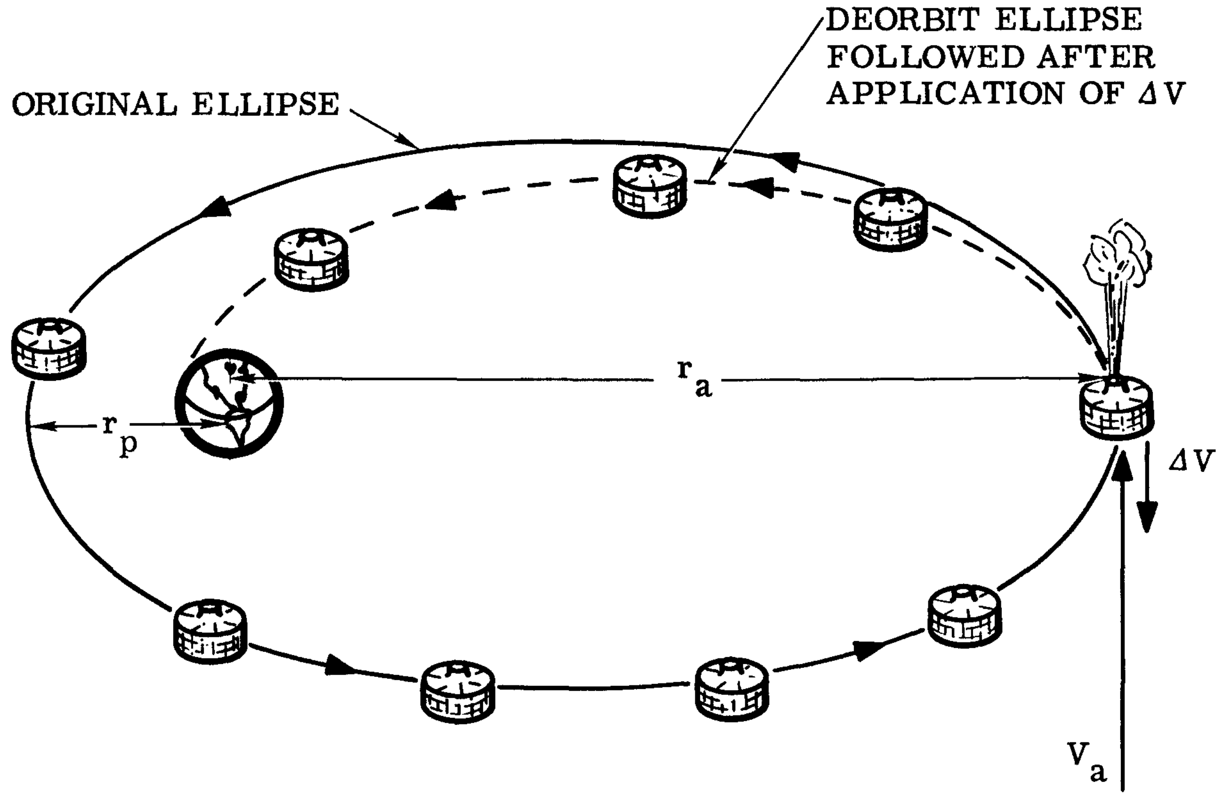

Figure 2. Illustration of 180-Degree Deorbit with ΔV Applied at Apogee

Figure 3 shows a typical case of retro-rocket deorbit from a polar orbit. The application of ΔV at apogee is shown by the instantaneous drop in velocity at 2850 seconds.

Perturbation by Drag, Circular Orbit

Aerodynamic forces acting on a satellite will cause it to lose altitude and descend in a spiral path at greater and greater velocity until the density of the air reaches the level at which the drag force overcomes the component of gravity along the flight path. At this point reentry deceleration and the other typical reentry phenomena take place.

Figures 4 and 5 show the relationships of the various parameters which influence drag deorbit from a circular polar orbit. A circular polar orbit (around oblate earth) is defined to be one in which the velocity equals local circular velocity and the flight path angle is zero over the equator or a pole (the only places where local horizontal is perpendicular to the radius vector).

The initial latitude for the computation was over the equator. This simplifies programming and also results in a conservative (long) range to impact, because the oblateness of the earth has the effect of raising the height of the orbit as the satellite travels away from the equator. Coefficient of drag used for the computation was 2.0, assuming free molecular flow.

In Figures 4 and 5 the range to impact shown is in degrees of arc (degrees of latitude for a polar orbit) starting from the point at which the drag device is deployed. Figure 4 shows range as a function of ballistic coefficient (W/C_DA) for a variety of initial altitudes. Note that on a log scale the plots at constant initial altitude are approximated by parallel, straight lines. This enables one to derive an empirical relation between range, height of circular orbit, and ballistic coefficient:

$$R_d = (W/C_DA)^{0.408} (0.00183h - 685)$$

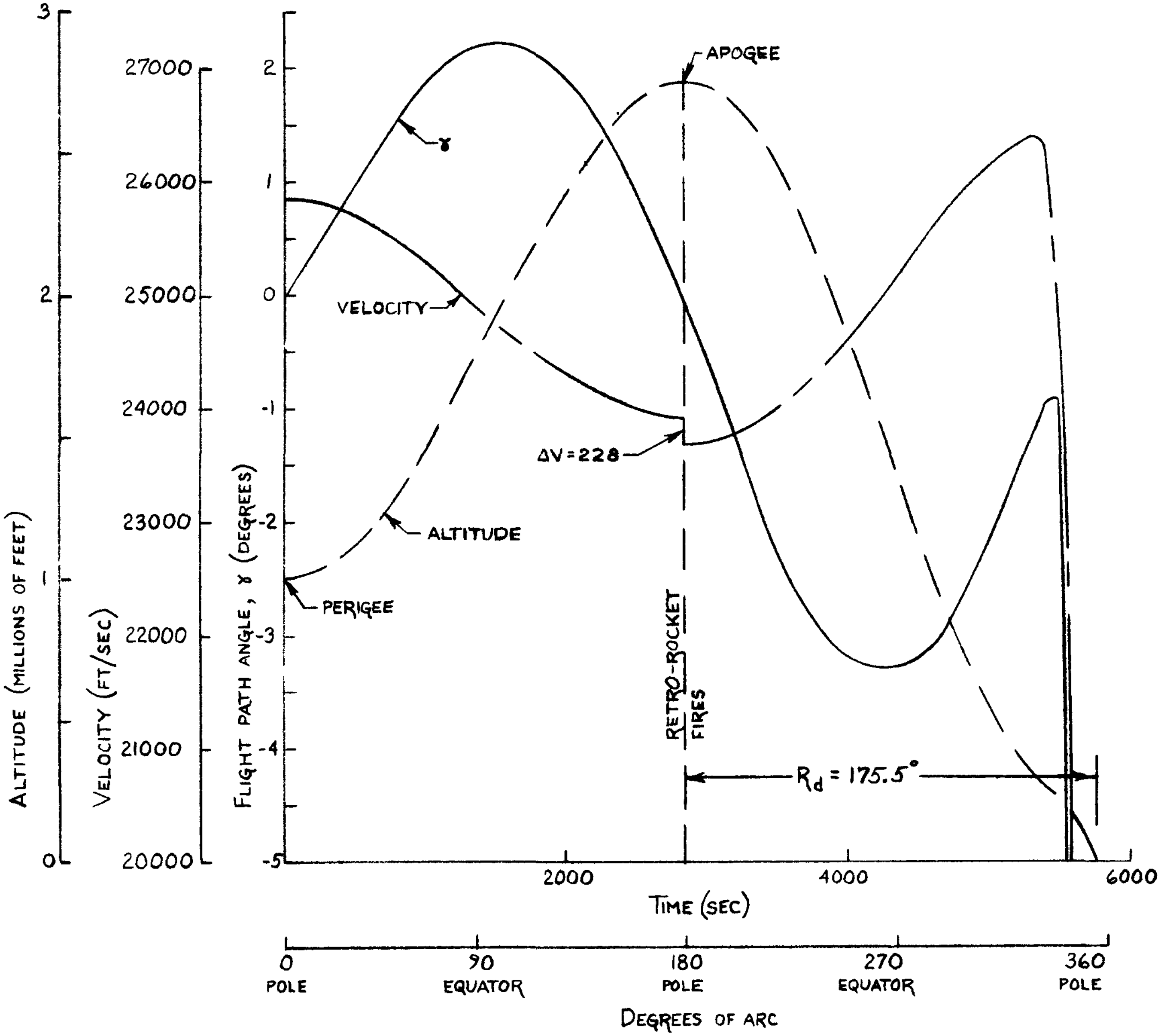

Figure 3. Typical Retro-Rocket Deorbit Profile - Polar Orbit

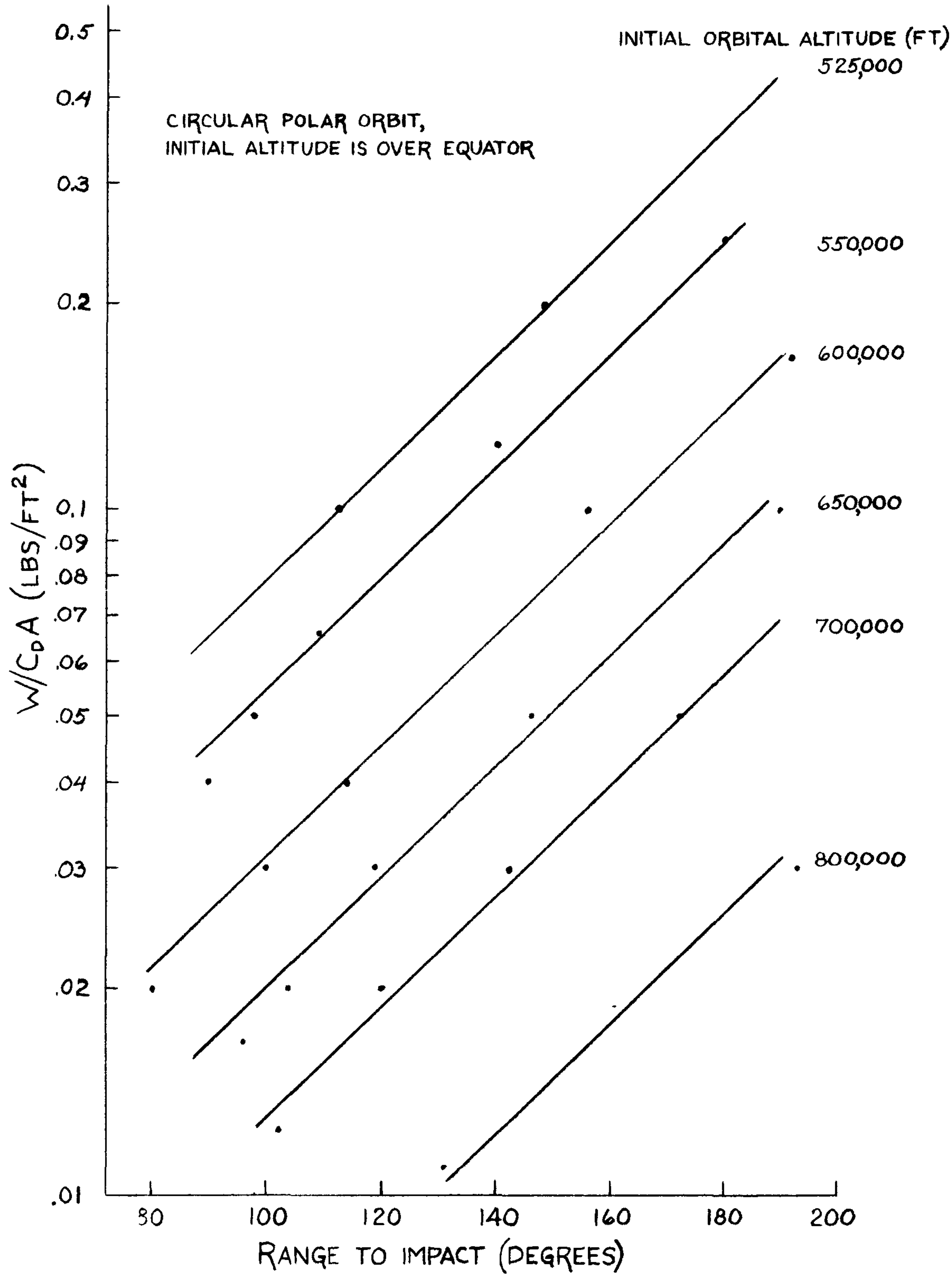

Figure 4. Drag Deorbit. W/C_DA vs Range to Impact

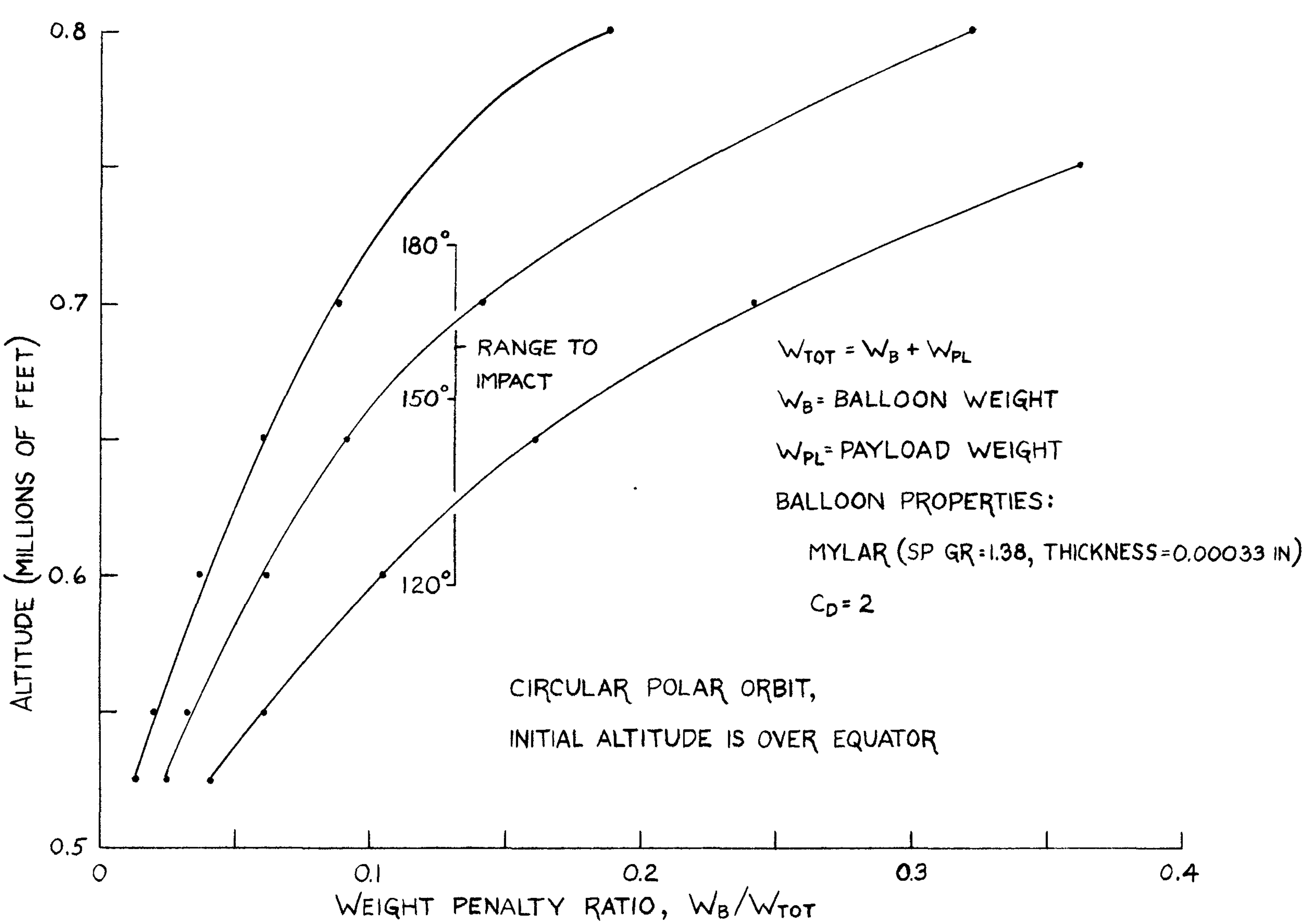

Figure 5. Drag Deorbit Weight Penalty Ratio for Various Ranges to Impact

where

R_d = range after deployment (degrees of arc)

W/C_DA = ballistic coefficient (lbs/ft^2)

h = initial height (ft) over the equator.

This relation gives good correlation over the ranges presented in Figure 4.

Figure 5 shows the relative weight of the drag device as a function of circular orbital altitude for a family of ranges to impact. In calculating weight, the following assumptions have been made:

1. The drag device is a sphere (balloon).

2. The material is Mylar, specific gravity 1.38, thickness 0.00033 inch.

As a way of checking the practicality of such a light device, the balloon was allowed to "tear off" during reentry. To do this, the area used in computation was reduced to that of the payload alone at the time when reentry acceleration exceeded 4 g's. In all cases, the range after the removal of the balloon was about 9 degrees. This indicates that the satellite was already irrevocably captured. One can conclude from this that the ability to survive reentry conditions is not an important criteria in the design of the drag device.

It can be seen in Figure 5 that the growth of the weight penalty ratio,

$$\frac{W_B}{W_B + W_{PL}},$$

is rather rapid as orbital altitude increases. The maximum weight penalty ratio, is, of course, unity. This limit represents an absolute upper bound on the practicality of drag deorbit since the payload weight has been reduced to zero and all weight is balloon weight. Actually an upper bound on weight penalty ratio would be placed arbitrarily at a much lower level. Extrapolation of Figure 5 shows that unity weight penalty will probably be reached at an altitude of less than one million feet for the 180-degree case and even lower for the shorter ranges.

Perturbation by Drag, Eccentric Orbit

In the retro-rocket deorbit situation ΔV is a very strong function of perigee height, but drag deorbit (like orbital lifetime) is dependent on both height of apogee and perigee. If perigee is too high, the density of the atmosphere will be insufficient to capture the satellite. If apogee is too high, the velocity at perigee will be so great that the drag forces cannot reduce the velocity quickly enough and the satellite again ascends.

For a drag deorbit system to be effective, that is, for a high drag device to cause reentry of a satellite in substantially less than one revolution after dedeployment, all of the velocity in excess of local circular velocity must be removed by deceleration resulting from drag at or shortly after the perigee point of the original orbit. This will prevent the satellite from ascending, and subsequent deceleration will reduce the velocity to suborbital levels, forcing reentry.

Figure 6 shows a typical case of drag deorbit from an eccentric orbit. Figure 7 shows the relationship between apogee height, perigee height, and deorbit range. In these graphs the heights shown for both perigee and apogee are over the poles. Range is measured in degrees of latitude from the point of deployment. The figures are for the ballistic coefficient ($W/C_D A$) of 0.062 which, for the balloon properties that have been assumed, corresponds to a weight penalty ratio of 7.8 percent. The drag device is deployed (the area used in computation of the trajectory is instantaneously increased) after the orbit has gone below one million feet and passed the equator on the descending node. These conditions were selected to initiate deployment as close as possible to the minor axis in every case presented.

One of the interesting features of Figure 6 is that after deployment of the balloon the flight path angle, γ, never quite gets positive again (which means the satellite's altitude never increases). The range to impact for this particular orbit is 176 degrees, which is approaching the maximum effective range. Ranges

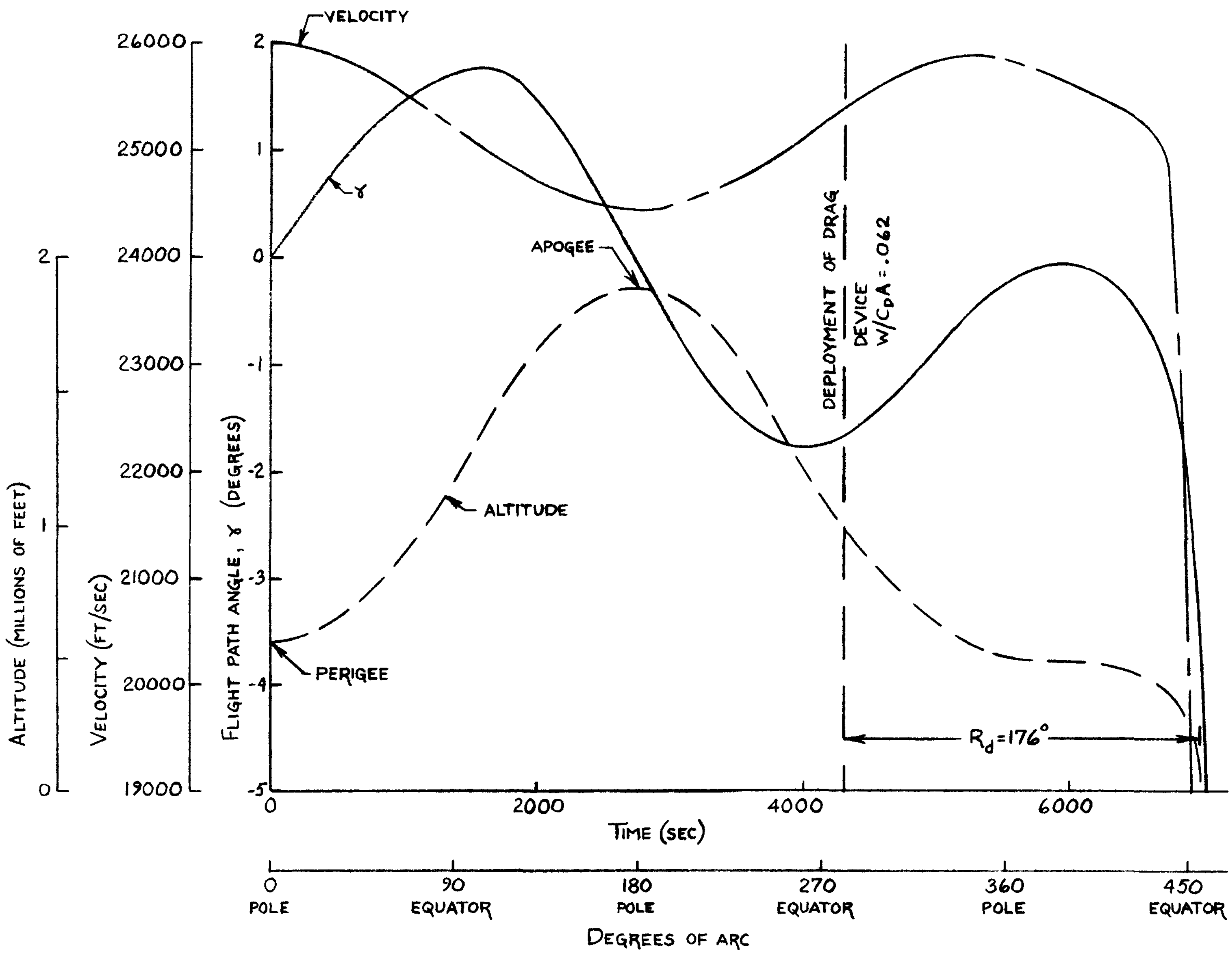

Figure 6. Typical Drag Deorbit Profile - Polar Orbit

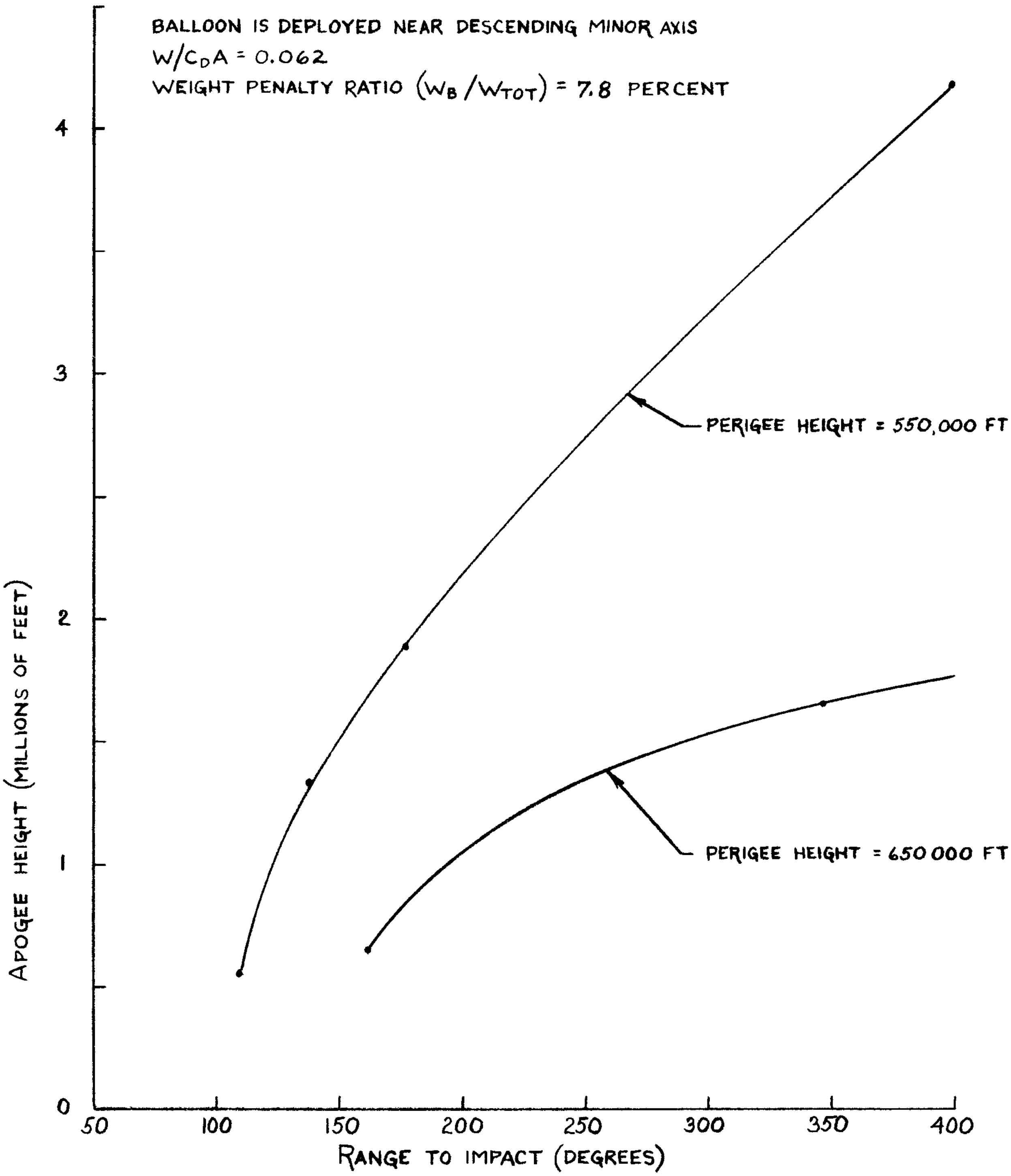

Figure 7. Drag Deorbit (Eccentric Orbit). Apogee Height vs Range for Various Perigees

much greater than this are subject to very large dispersions and are not desirable for this reason. A rule of thumb for sizing drag devices might be stated from these observations: an effective drag deorbit device for eccentric orbits must have sufficient area to prevent the flight path angle from becoming greater than zero.

Figure 7 shows that, for a perigee height of 550,000 feet, the deorbit range is less than 180 degrees for apogees as high as 2.0 million feet (for the drag area chosen). But for a perigee of 650,000 feet, the range is greater than 180 degrees if the apogee is higher than one million feet. Figure 7 serves to show that drag deorbit is most effective for near earth orbits but also that effectiveness decreases rapidly as apogee or perigee increases.

Emergency Situation Example

To examine the emergency situation that can arise, consider a satellite launched into a polar orbit from the Western Test Range. The mission, similar to that flown by the SNAP-10A, differs in that the payload consists of a SNAP isotopic power source and a retro-rocket deorbit system with a combined weight of 1000 pounds. The payload is to be placed in a 700-nautical-mile circular orbit with an orbital lifetime of 3800 years. This long-lived orbit would allow time for the radioactive materials of the SNAP system to decay to harmless levels prior to reentry.

During the first 15 seconds of flight, the missile rises vertically from the pad. Then a preselected pitch program is initiated which results in a transition from vertical to southward flight. Second-stage burn lasts for 230 seconds after a first-stage burn of 265 seconds. Until 14 seconds before termination of second-stage thrust, orbital velocity has not been achieved and the instantaneous impact point of the vehicle has moved south through the Pacific Ocean, across Antarctica, and then north into the Indian Ocean. Second-stage thrust ceases with the vehicle in an elliptical transfer orbit at a perigee height of 87 nautical miles. After coasting for approximately 50 minutes to an apogee height of 700 nautical miles over Madagascar, a short application of thrust adds enough velocity to the vehicle to circularize the orbit at this height.

The emergency arises if the last application of thrust does not occur and the payload and rocket are left in the transfer orbit with a period of approximately 94 minutes and an orbital lifetime of 1 to 4 months. Figure 8 is an altitude profile versus latitude for the transfer orbit and the deorbit trajectory. To obtain maximum benefit from the ΔV provided by the retro-rocket deorbit system, the payload must be separated from the booster. The separation command may be initiated either from the ground or by the deorbit system itself when a malfunction has been sensed. In addition, the attitude of the payload must be controlled for proper retro-rocket alignment during firing. Because the malfunction occurs near apogee, by the time separation and attitude control have been performed the payload will be some distance beyond apogee. Therefore, unless the retro-rocket can supply more (up to 10 times more) than the ΔV needed at apogee for a 180-degree deorbit, it is necessary to wait at least until the next apogee before commanding retro-rocket firing. For these requirements (ΔV application at apogee and 180-degree deorbit) Figure 9 indicates a ΔV of 91 ft/sec. This figure shows the relationship between ΔV and range to impact in degrees for the example payload and indicates quite pointedly why a 180-degree range to impact was selected for this study. A greater value results in dispersion disproportionate to a small change in ΔV. A smaller value is not economical of ΔV.

With retrothrust applied as the payload reaches apogee in the second orbit (16-degree south latitude, 21-degree east longitude), impact will occur southwest of the Hawaiian Islands at approximately 16-degree north latitude, 170-degree west longitude. Overshoot will move the impact point into the South Pacific Ocean away from land masses.

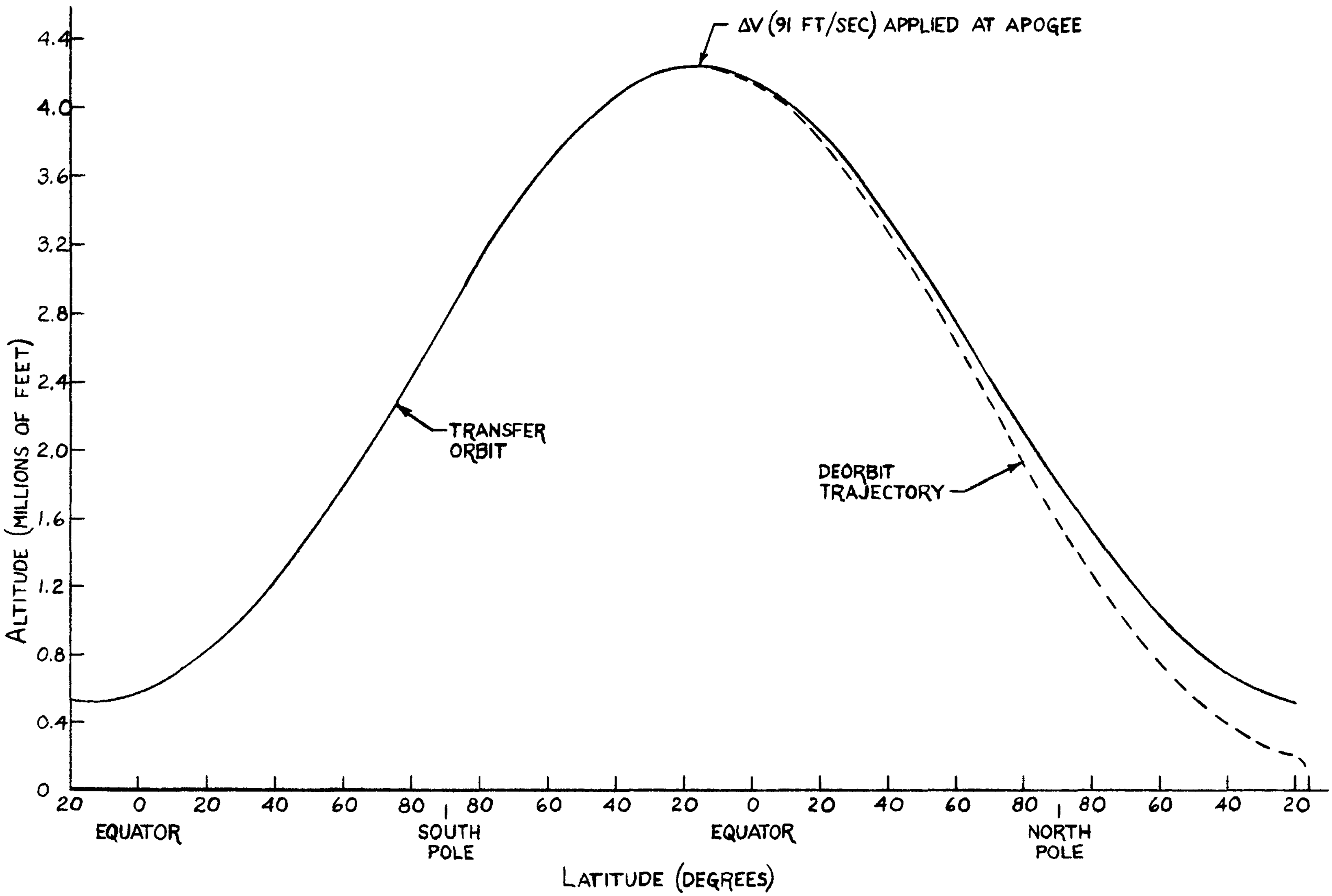

Figure 8. Trajectory Profile - Example Problem

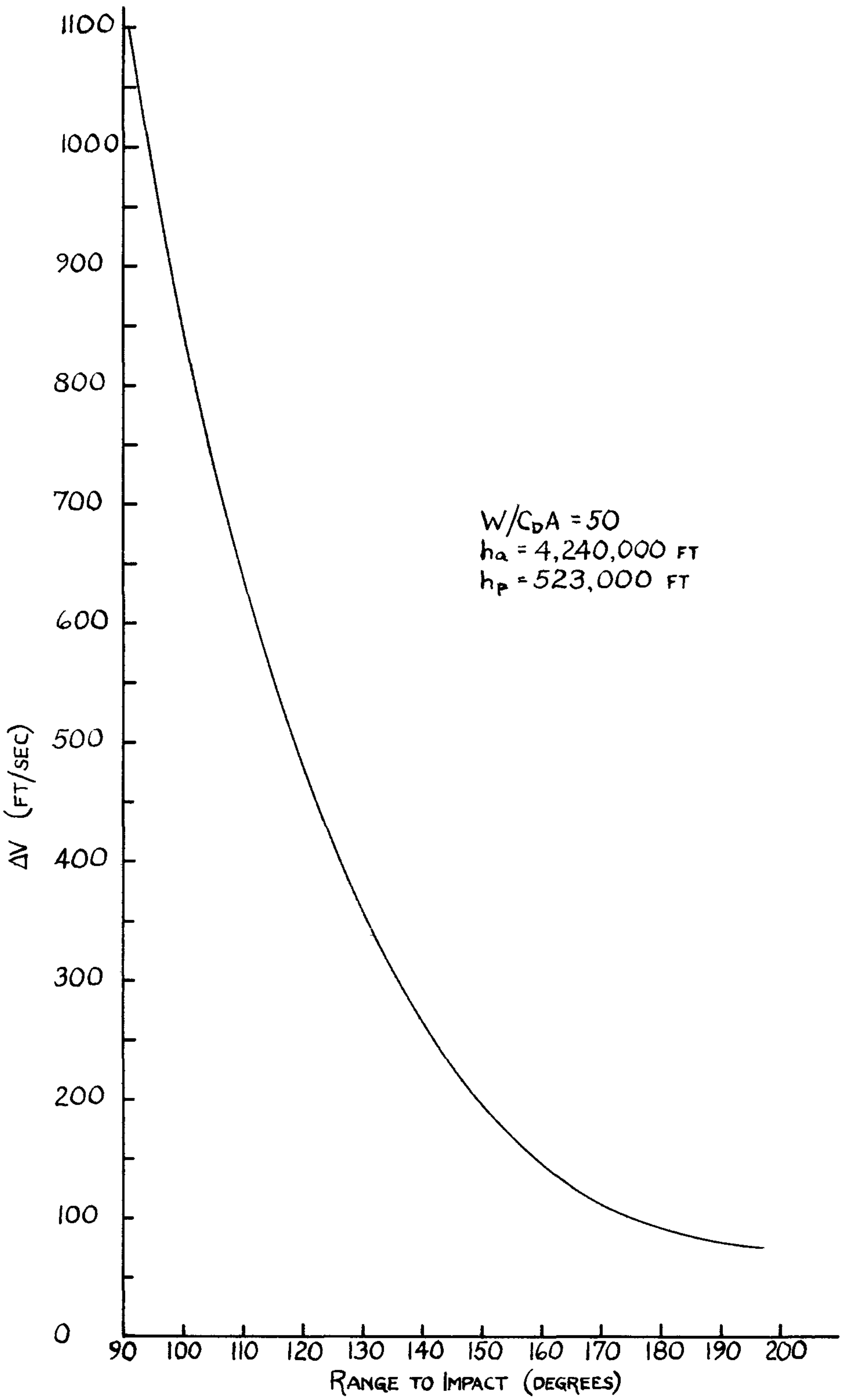

Figure 9. ΔV vs Range to Impact - Example Problem

Utilization of the following USAF satellite control facilities and NASA ground tracking and communication facilities appears feasible:

Kaena Point, Hawaii, Tracking Station (HTS)

Kodiak, Alaska, Tracking Station (KTS)

Thule, Greenland, Tracking Station (TTS)

Pretoria, South Africa, Tracking Station (PTS)

Figure 10 indicates the distance to the horizon for a satellite at various altitudes. Plotting this distance from the payload at apogee (4, 240, 000 feet), Figure 11 indicates that tracking and communication is feasible by PTS when the payload reaches apogee the first and second times. Therefore ground control is aware of the failure to circularize immediately and may command payload/rocket motor separation at this time. Additional plots on Figure 11 utilizing Figures 8 and 10 indicate that TTS, KTS, and HTS are all in a position to track and communicate with the payload at some time during the first orbit. In addition, they will have been informed of the failure by PTS. Depending on the specifics of the particular deorbit system, any of these ground stations may command the initiation of attitude control during this orbit. Retrothrust initiation can be commanded from PTS at second orbit apogee. As a backup, if the deorbit system is so designed, the TTS could initiate a timer that would command retrothrust at the next (third) apogee with impact still in the North Pacific Ocean at approximately 16 degrees north latitude, 164 degrees east longitude. Since TTS is well within the 23-degree line of observation of the payload as it passes over the North Pole (Figures 8 and 10), TTS is in a position to track, communicate with, or control the payload during each orbit. In fact, any orbiting vehicle passing over the North Pole at an altitude of at least 600, 000 feet can be "seen" by TTS. Table I summarizes the capabilities of the four ground stations to communicate with the example problem payload during the first two orbits.

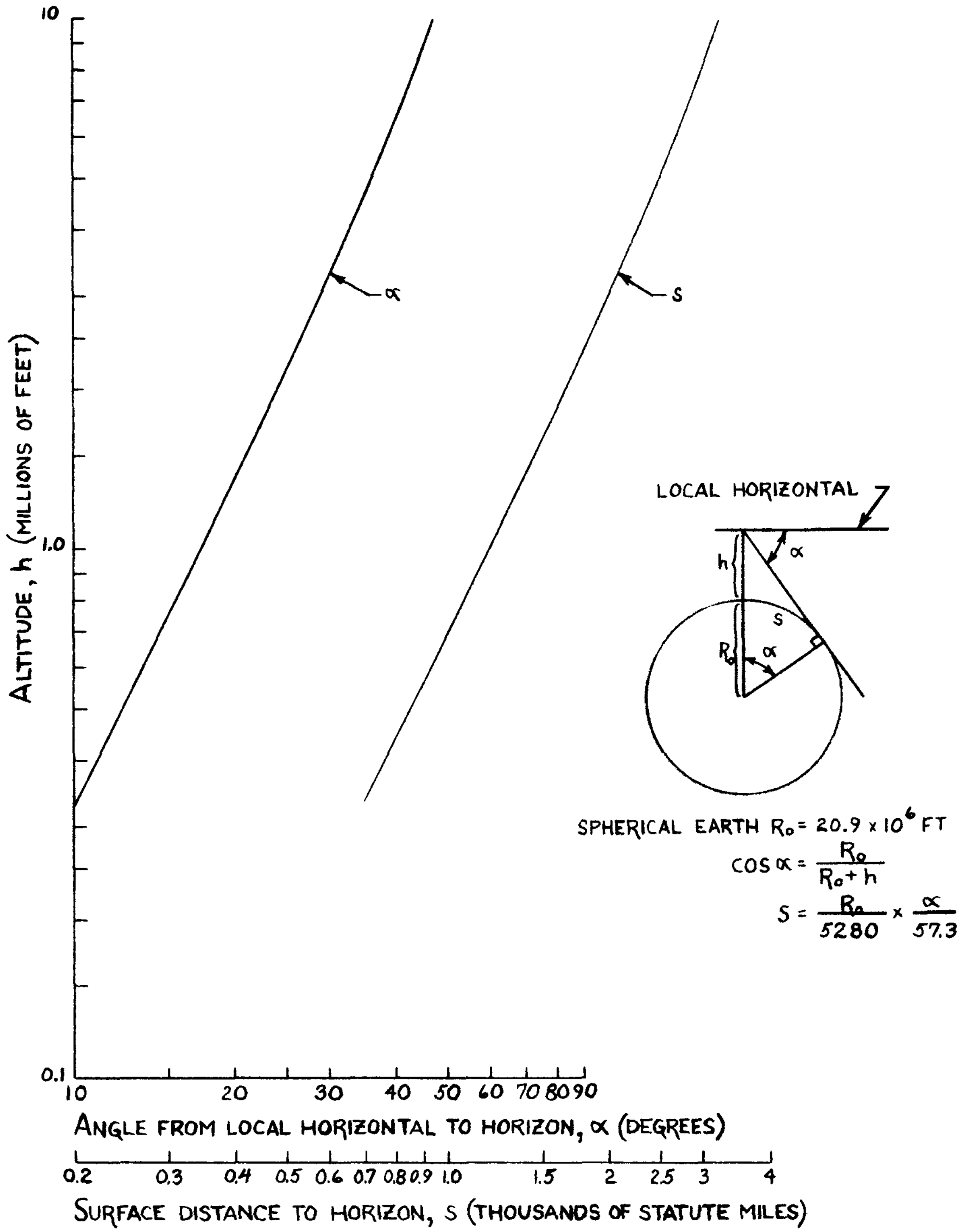

Figure 10. Altitude vs Distance to Horizon

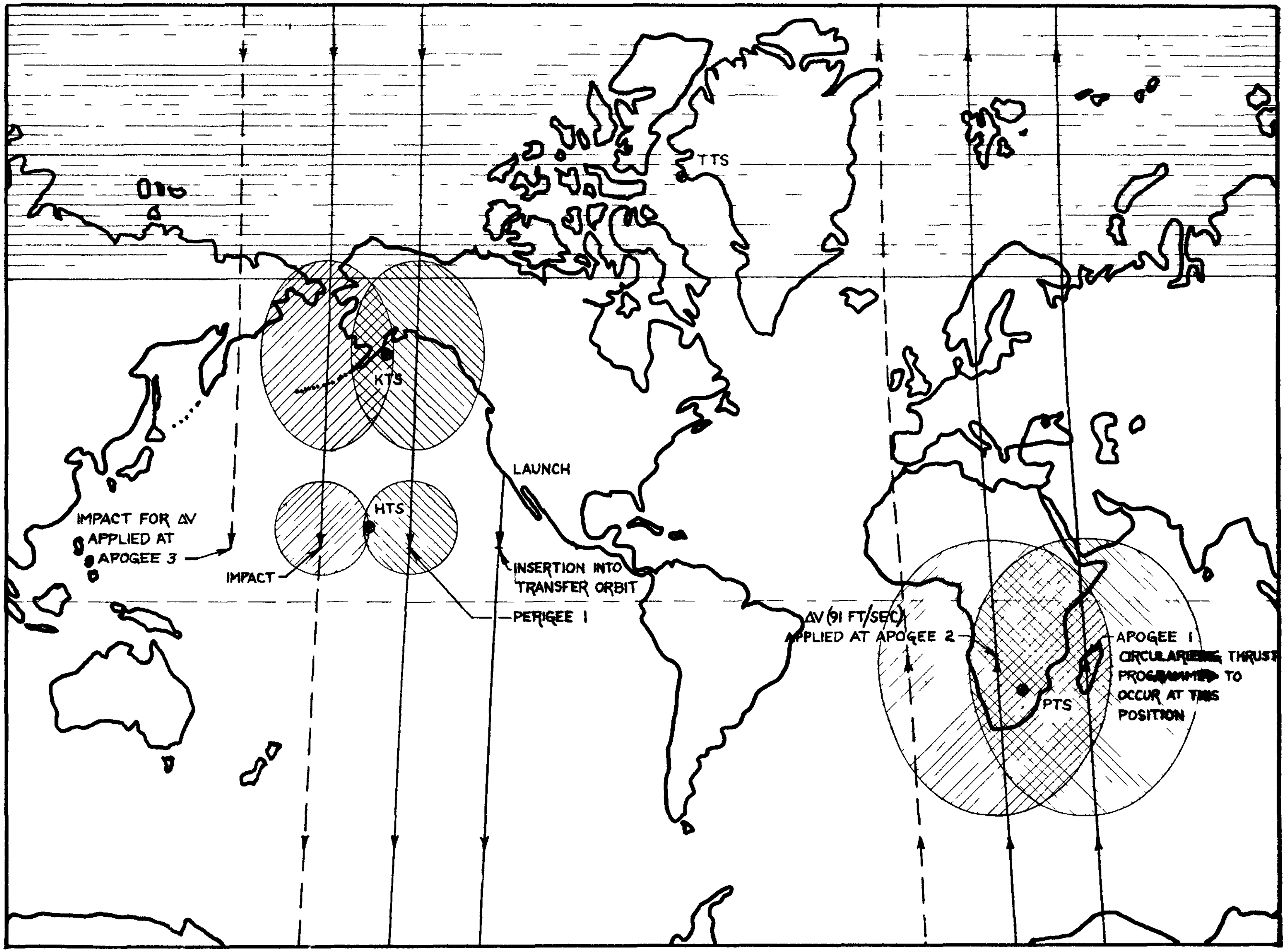

Figure 11. Example Problem. Satellite Ground Path and Tracking Station Coverage

TABLE I

Example Problem

Tabulation of Ground Station Line of Sight Capability During First Two Orbits

Orbit	Ground Station	Within Line of Sight
1	Pretoria, South Africa (PTS)	Yes
1	Thule, Greenland (TTS)	Yes
1	Kodiak, Alaska (KTS)	Yes
1	Kaena Point, Hawaii (HTS)	Marginal
2	PTS	Yes
2	TTS	Yes
2	KTS	Marginal
2	HTS	Marginal

CHAPTER V. DEORBIT SYSTEM WEIGHT

It was noted in Chapter III that the minimum capabilities of a high-drag deorbit system differ from those of a retro-rocket deorbit system. Figure 12 is a block diagram comparison of these requirements. For either system, a portion of the weight penalty results from components that change only slightly with payload weight. These include the components of the payload separation and telemetry and command subsystems. In a drag deorbit system, the weight of the drag device increases directly with payload weight; while, in a retro-rocket deorbit system, the weight of propellant increases in a similar manner. The weight of the retro-rocket engine structure and attitude control components also varies with payload weight but to a lesser degree than the weight of the retro-rocket propellant. Relative merits (on weight basis) of the two systems (drag and retro-rocket) can be indicated by comparing the weights of the subsystems that are not common to both, i.e., the drag device versus the retro-rocket and attitude sensing and control subsystems. Figure 13 shows these weights as a function of orbital altitude for circular polar orbits; a deorbit range of 180 degrees is assumed. The weight of the retro-rocket deorbit system is plotted as a family of curves so that any weight may be assumed for the attitude control system. The intersection of one of these lines with the curve indicating the weight of the drag device defines an altitude at which the weight required to deorbit in 180 degrees is the same for either system. For example, to deorbit a 1000-pound payload (including deorbit system) in 180 degrees from a circular orbit at 650, 000 feet, a drag device weighing 60 pounds is required. This weight is equal to that of a retro-rocket deorbit system with a 39-pound attitude sensing and control subsystem.

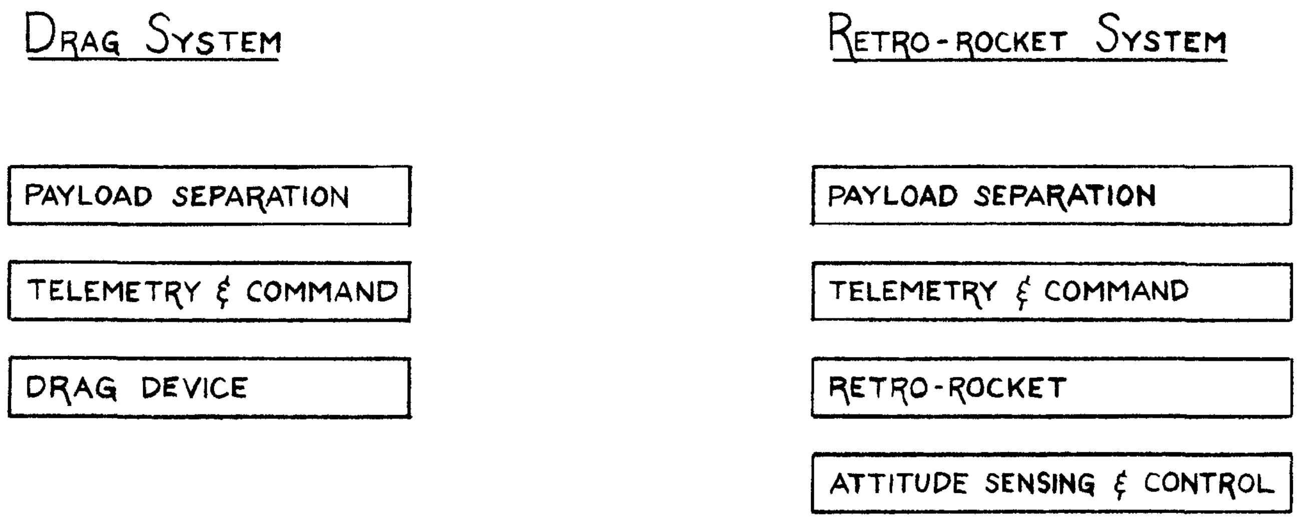

Figure 12. Comparison of Deorbit Systems. Subsystem Requirements

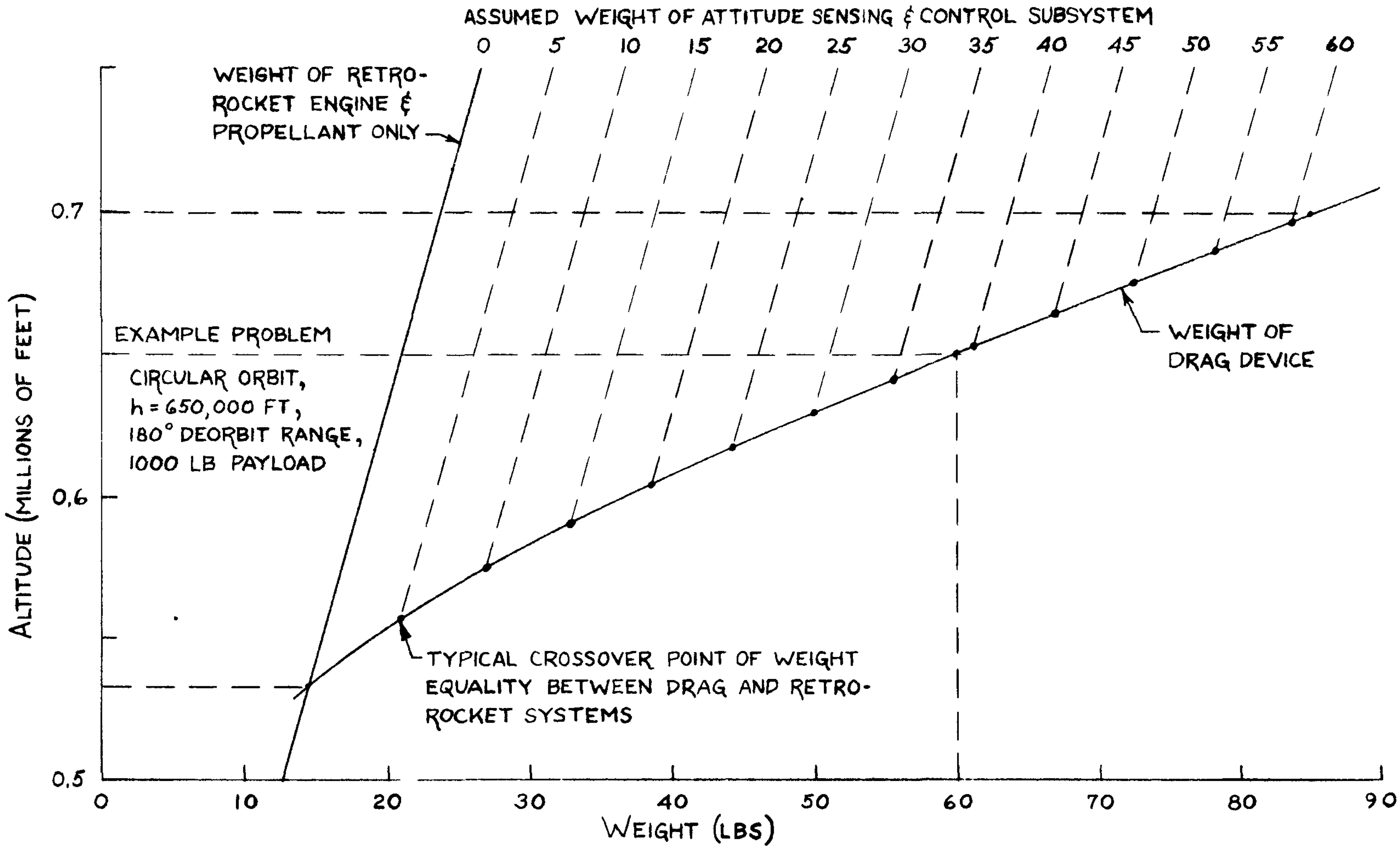

Figure 13. Weight Comparison. Drag Device vs Retro-Rocket Subsystems

In Figure 13, the curve for weight of the drag device is taken from Figure 5, for a 1000-pound payload with a 180-degree range to impact. The line representing weight of propellant and rocket engine was derived from the equation

$$\frac{W_{Propellant} + W_{Payload}}{W_{PL}} = e^{\frac{\Delta V}{g_o I_s}} \qquad (7)$$

found in Reference 3, where

ΔV = values selected from Figure 1 (atmosphere curve)

g_o = acceleration of gravity at the surface of earth

I_s = specific impulse of retro-rocket propellant, assumed 240 sec

In addition, the weight of the propellant was assumed to be 0.55 of the weight of the engine. This assumption, which is based on discussion presented in Reference 3, is considered typical for small engines.

Figure 13 indicates two other important points:

1. For deorbiting in 180 degrees from circular orbits of less than 534,000 feet, the drag device is superior on a weight basis regardless of the weight of the attitude control system.
2. For deorbiting in 180 degrees from circular orbits greater than 700,000 feet, the retro-rocket is superior unless the attitude sensing and control subsystem becomes inordinately heavy (heavier than 60 pounds).

Equation 7 can be rearranged so that retro-rocket propellant (as a percentage of payload weight) can be expressed as a function of ΔV, i.e.,

$$P = 100\, e^{\frac{\Delta V}{g_o I_s}} - 100 \qquad (8)$$

where

$$P = \text{percentage of payload weight.}$$

Figure 14 is a plot of this function for the range of ΔV pertinent to this study. This figure indicates that, if the minimum ΔV required to deorbit a payload is known, the minimum retro-rocket propellant weight (expressed as a percentage of the payload weight) may be easily determined.

The example problem of Chapter IV is used to achieve the following weight estimate of the component parts of the 1000-pound payload requiring a ΔV of 91 ft/sec to deorbit in 180 degrees.

Components	Weight (lbs)
Payload separation	10
Telemetry and command	25
Attitude sensing and control	38
Retro-rocket propellant and structure	28
	101
SNAP system	899
	1000

This tabulation indicates that, for the example situation, a retro-rocket deorbit system could be included at a weight penalty to the payload of approximately 10.1 percent

$$\left(\frac{101}{1000} \times 100\right) .$$

A comparable weight estimate for the drag deorbit system cannot be made because the orbit of the example problem is above the practical limits of a drag deorbit system (Chapter IV).

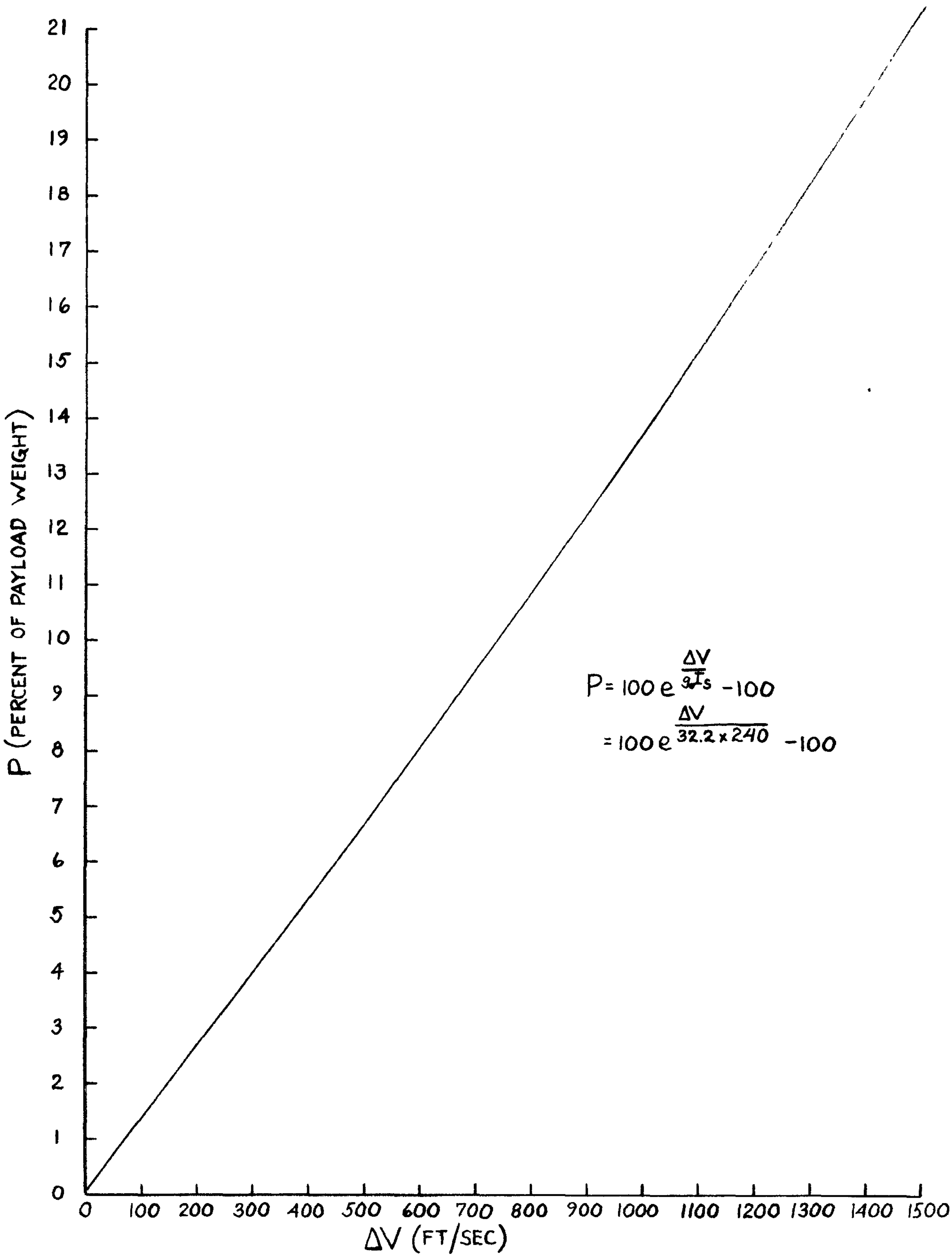

Figure 14. Propellant Weight vs ΔV

CHAPTER VI. CONCLUSIONS

A controlled deorbit system increases SNAP satellite safety and mission flexibility. There are several situations that may serve to exemplify this increased flexibility.

1. Missions may be planned for lower altitudes than those deemed acceptable from the viewpoint of radiological safety with random reentry. At a predetermined time in the orbital decay, the satellite could be deorbited to an ocean area.

2. When a nonnominal orbit is achieved, it may be possible to accomplish some or all mission objectives prior to deorbiting instead of resorting to self-destruct systems.

3. When a mission has been accomplished or a satellite is of no further use, the controlled deorbit system can function to remove the debris from space. This use is not necessarily for reasons of safety.

4. For political, economic, or scientific reasons, it may be desirable to recover a satellite. A controlled deorbit system makes this attempt feasible. However, this study does not consider satellite recovery because the primary aim is nuclear safety which may be achieved with a low-cost deorbit system whose design requirements do not include the accuracy and reliability needed for recovery attempts.

This study considers the most critical condition requiring controlled deorbit as that emergency arising from a rocket motor failure or a control malfunction which could result in a short-lived orbit with the radiological hazards associated with random reentry. The emergency would arise within minutes of launch, and controlled deorbit would be initiated within the first few orbits.

This study does not contend that a controlled deorbit system is the only solution to the problem of disposal of radioactive material in orbit. Other techniques, such as reentry burnup, intact reentry with burial, or reboost, are possible. Some of these techniques could be used jointly or as backups for each other.

A controlled deorbit system does inflict certain disadvantages on a satellite. The most obvious detriment is the weight of the deorbit system which must be boosted into orbit with the satellite. This weight would include the component weights for telemetry and command, attitude sensing and control (for retro-rocket deboost), deboost power and control, and satellite separation from the booster. Electrical and mechanical relationships between the satellite and the deorbit system will affect adversely the performance reliability of the satellite. This degradation results from the increased complexity of the total system. These disadvantages may be minimized by prudent design.

These disadvantages must be weighed against the advantages listed above in deciding on the most effective means of achieving the safety criteria for any given mission.

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