NASA-RP-1110 19830024557

NASA Reference Publication 1110

July 1983

Performance Requirements Analysis for Payload Delivery From a Space Station

A. L. Friedlander, J. K. Soldner,
J. Bell, G. W. Ricks,
R. E. Kincade, D. DeAtkine,
R. Reynolds, B. A. Nader,
O. Hill, G. R. Babb,
and E. C. Lineberry

NASA

25th Anniversary
1958-1983

NASA Reference Publication 1110

1983

Performance Requirements Analysis for Payload Delivery From a Space Station

A. L. Friedlander
and J. K. Soldner
Science Applications, Inc.
Schaumburg, Illinois

J. Bell, G. W. Ricks,
R. E. Kincade, D. DeAtkine,
R. Reynolds, B. A. Nader,
O. Hill, G. R. Babb,
and E. C. Lineberry
Lyndon B. Johnson Space Center
Houston, Texas

NASA
National Aeronautics
and Space Administration

Scientific and Technical
Information Branch

CONTENTS

Section | Page

Section		Page

TABLES

FIGURES

Figure		Page

Figure		Page

Figure		Page

Figure Page

ACRONYMS

ETR	Eastern Test Range, Kennedy Space Center
FV	ferry vehicle
GEO	geostationary (geosynchronous) Earth orbit
JPL	Jet Propulsion Laboratory
JSC	Johnson Space Center
L/D	lift-to-drag ratio
LEO	low Earth orbit
LOX/H_2	liquid oxygen/hydrogen
NASA	National Aeronautics and Space Administration
OTV	orbital transfer vehicle
PET	phase elapsed time (measured from any predefined reference event)
SAI	Science Applications, Inc.
STS	Space Transportation System

SYMBOLS

a	semimajor axis of orbit
α	angle of attack
A	area
C_D	drag coefficient
C_3	planetary injection energy ($\bar{V}oo^2$)
δ	angle between orbit plane intersections
$\Delta\Omega$	difference between orbit nodes
ΔV	delta velocity
Δv	impulse velocity change
e	eccentricity of orbit
g	acceleration due to gravity
γ_I	inertial flight path angle
h_a	apogee altitude
h_I	altitude of intermediate (midcourse) impulse
h_p	perigee altitude
i	inclination
I_{sp}	specific impulse
J_2	second zonal harmonic, Earth oblateness coefficient
ω	argument of perigee
Ω	longitude of ascending node on equator
P	orbit period
r_p	perigee distance
r_a	apogee distance
R_E	equatorial radius of Earth
$\bar{V}oo$	velocity vector at infinity

V_I	inertial velocity
W	weight

Subscripts

m	mission or target orbit
o	space station orbit

ABSTRACT

Operations conducted from a space base in low Earth orbit have different constraints and opportunities than those conducted from direct Earth launch. The space base allows assembly-on-orbit of payloads and upper stages or of multiple upper stages, an operational mode that relieves many size and performance constraints. A space base as a depot permits transportation of some propellant to orbit at marginal cost to the Shuttle. Both of these considerations favor some space station based options. But the base or station is in an inertial orbit, so launch window constraints are different than for our customary Earth launch sites. This performance requirement analysis has been developed to provide a reference source of parametric data and case solutions to assist potential space station users and space station and upper stage developers to assess the impacts of a space station on missions of interest to them.

This document contains the results of performance requirements analyses associated with transporting satellites to and from a space station using a generic orbital transfer vehicle(s) (OTV's). These performance analyses provide parametric data for a broad class of support missions as well as detailed data for four classes of missions which are assumed to typify support from a space station. The analyses also consider a reusable orbital transfer vehicle with and without aerobraking capability. The primary mission classes covered are: (a) geosynchronous missions; (b) planetary missions; (c) Sun-synchronous missions and (d) co-orbiting satellite support missions.

The performance requirements analyses consist of three separate but related studies: spacecraft delivery and return from a space station; maneuver strategies and impulse requirements for space station/orbital transfer vehicle missions; and OTV sizing studies.

The maneuver strategies and impulse requirements study (section 1.0) provides an understanding of the orbital transfer problem over a wide range of mission applications; also, a database of numerical data for the ΔV impulse requirements associated with different maneuver strategies is provided.

Section 2.0 of this document, Space Station/Orbit Transfer Vehicle Missions, contains integrated flight profiles for each of the mission classes previously noted. These mission profiles reflect implementation of operational planning requirements and considerations. Key guidelines, assumptions, and implementation rationale associated with the flight design are also identified.

Finally, the results of an OTV sizing study to accomplish the various OTV mission classes are contained in section 3.0. Several configuration options are investigated and compared.

This document represents a more comprehensive and rigorous execution of a March 1982 series of studies done by Science Applications, Inc. (SAI), Jet Propulsion Laboratory (JPL), and the Johnson Space Center (JSC) which examined the same topics in a preliminary assessment of a space station as a transportation system node. The results of those studies warranted the more systematic effort.

1.0 MANEUVER STRATEGIES AND IMPULSE REQUIREMENTS

1.1 INTRODUCTION AND SCOPE OF ANALYSIS

One potential benefit of a space station in Earth orbit is its use as a transportation node for the delivery and servicing of payloads placed into various mission orbits. This space-based operations concept would employ a high energy orbital transfer vehicle (OTV) to perform all necessary propulsive maneuvers in the sequence of orbit transfers from and return to the space station. The OTV design may also include aerodynamic devices to accomplish some part of the transfer sequence by aerobraking maneuvers. Aerobraking can be a very effective technique for reducing propulsive requirements, particularly in the return phase. To first order, the propulsive requirements can be measured by the total velocity change impulse (ΔV) needed to accomplish an orbit transfer sequence. It is important, therefore, to understand the characteristics of the orbit transfer problem and to obtain numerical data on ΔV impulse requirements over a wide range of mission applications. The overall objective of section 1 is to provide such an understanding and database.

The intent is to describe maneuver strategies and impulse requirements in a presentation format that facilitates mission analysis in terms of specific results yet allows more generalized conclusions to be drawn also. Since the orbit transfer parameter space is multidimensional and covers a wide range, it seemed advisable to place some constraints on the problem. The study groundrules and assumptions established at the outset are as follows:

A. The space station is in a 200 n. mi. circular orbit; orbit inclination will be treated parametrically for several discrete values in the range 28.5 to 98°; orbit orientation measured by the nodal longitude ascending on the equator will be treated as a time-related variable over the full possible range of 0 to 360°.

B. The mission (destination) orbit is either circular or elliptical, but in the latter case only special unique orbits are examined; circular orbit period (altitude), inclination, and node relative to the station orbit will be treated parametrically; elliptical orbits will be either coplanar with the station or the semisynchronous, Molniya-type orbit at 63.5° inclination.

C. ΔV impulse requirements will be given for both delivery and return orbit transfers.

D. Transfer options considered will include all-propulsive maneuvers and aerobraking-assisted maneuvers.

E. ΔV requirements will be based on optimal two-impulse and/or three-impulse transfers; an upper limit of approximately two days (48 hours) will generally be imposed on both delivery and return transfers, but the sensitivity to transfer time will also be shown; a single atmospheric pass is assumed in the aerobraking-assisted maneuvers.

F. Rendezvous time-phasing requirements are not included in the optimal transfer analysis, but are considered in section 2.0 of this report.

The guidelines stated above are believed to be practical, while not too restrictive, and serve to make the analysis and presentation formats tractable. The selection of 200 n. mi. for the space station's orbital altitude is based on previous studies that addressed the question of optimal altitude as a trade-off between Shuttle payload performance and orbit decay due to atmospheric drag. Small variations about the 200 n. mi. reference, and even slight eccentricity, would have no significant effect on the mission ΔV requirements to be presented herein. Station orbit inclinations of 28.5, 57 and 98^{o} represent, respectively: (a) due east ETR Shuttle launches having maximum payload performance, (b) ETR range safety limit, and (c) Sun-synchronous operations.

The range of circular mission orbits encompass low-altitude, geostationary, and high-altitude applications with orbit inclinations varying from equatorial through sun-synchronous. Several discrete orbit periods ranging from 1.53 to 72 hours (200 to 43,970 n.mi. altitude) will be examined in detail, but the presentation formats will also allow interpolation of ΔV requirements for other cases that may be of interest. The special navigation-type elliptical orbit (Molniya) considered here has a period of 12 hours, an inclination of 63.5^{o}, and has its perigee location displaced 270^{o} from the ascending node. For convenience the perigee altitude is set at 200 n. mi., although other choices up to 400 n. mi. would have small effect on transfer requirements.

As will be discussed in the subsection dealing with orbit geometry and transfer characteristics, the Earth's oblateness causes the orbit's ascending node to regress with time along the equator at a rate depending on altitude and inclination. Hence, even though the space station and mission orbit inclinations may be fixed, their orbital plane intersection and consequently the total plane change requirement will vary with time. This factor cannot be ignored since its effect will be manifested as a trade-off between increased requirements for "launch-on-demand" vs. launch time delay while waiting for the optimal plane orientations to occur. It should be noted that the option of propulsively adjusting the space station's orbit node is much too expensive to consider. The plane orientation factor will be accounted for in a global sense by allowing the ascending node difference between the station and mission orbits to take on all possible values in the range 0 to 360^{o}.

Graphic data of total ΔV impulse will be shown separately for both delivery and return transfers. This allows greater flexibility for the mission analyst to construct his own round trip scenario accounting for mission-unique timing considerations, propulsion staging, and payload definitions. However, by way of illustration, several examples of the use of the ΔV data in round trip mission scenarios will be described.

The groundrule of optimal (minimum ΔV) transfers requiring either two or three propulsive impulses reflects practical maneuver strategies. Other nonoptimal strategies, perhaps at small ΔV penalty, could be derived to satisfy unique constraints, but these would depend on specific vehicle/mission definitions. In the case of optimal three-impulse transfers needed for large plane change situations, the transfer time can become very long (even approaching infinity); we have therefore imposed a transfer time constraint in generating ΔV requirements to better represent practical implementations.

The order of material presentation is given by the table of contents. Basic characteristics of orbit transfer will be described first, followed by separate subsections of graphical data formats for the circular and elliptical mission orbits. The final subsection describes the use of this data for several example mission applications.

1.2 CHARACTERISTICS OF ORBIT TRANSFER

Orbit transfer may be defined as the motion of a vehicle (i.e., a trajectory) that starts from a prescribed initial orbit motion and ends at a prescribed final orbit motion. Suppose that all motion is governed by an inverse-square central force field, ignoring for the moment small perturbing influences. The initial and final orbits may have general conic section geometry (circle, ellipse, parabola, hyperbola) in three-dimensional space and need not be coplanar. The same is true of the transfer trajectory; impulsive trajectories may consist of N subarcs connected by N + 1 impulse points. The transfer time may be "open", fixed, or constrained to be less than a prescribed limit. Rendezvous is a special class of orbit transfer wherein a time relationship determines a body's position in its orbit. Finally, optimal orbit transfer usually refers to that particular solution which minimizes the sum of all ΔV impulses.

The optimal transfer problem has received much attention since the late 1950's and considerable progress has been made in determining solutions to various problem categories. Both analytic and numerical methods have been applied. Some special cases like circle-to-circle noncoplanar transfers and circle-to-ellipse coplanar transfers are well understood. Other problems like noncoplanar circle-to-ellipse transfers do not yield general analytic solutions; numerical search methods and parametric studies typically are employed to obtain specific answers, multivariable solution maps, and partial conclusions regarding optimal transfer characteristics.

The 1969 paper by Gobetz and Doll is regarded as the most comprehensive survey and description of impulsive orbit transfer (ref. 1). This fundamental reference spawned subsequent research which may be found in the open literature and in a few textbooks on the subject. No attempt to summarize this body of knowledge will be made here. Instead, application to the problem at hand will be described by example graphic illustration and discussion of key results, including the extension to aerobraking-assisted orbit transfers. First, however, some definitions of orbit geometry and perturbations apropos to a space station transportation node are discussed.

1.2.1 Orbit Definitions and Geometry

Figure 1-1 shows the relationship between orbit period and altitude for both circular and elliptical mission orbits. The minimum distance circular orbit at 200 n. mi. altitude, equal to the space station's altitude, has a period of 1.53 hours. A slightly longer 2-hour period orbit has a circular altitude of

907 n. mi. Geosynchronous (24-hour) orbit altitude is about 19 310 n. mi. The farthest mission orbit considered in this analysis has a 72-hour period and altitude of 43 970 n. mi. Apogee altitude of the 12-hour elliptical orbit is approximately 21 600 n. mi.

Orbit orientation angles are illustrated by the celestial sphere projection shown in figure 1-2 with the reference plane being the Earth's equator. Orbit inclination angles are denoted by i_o for the space station and by i_m for the mission orbit. In the general case, the two orbits will not intersect at the equator; i.e., their ascending node longitudes will be different. This difference is denoted by $\Delta\Omega = \Omega_m - \Omega_o$ which is allowed to take on any value in the full range of 360^o.

The consequence of ascending node misalinement is that the station and mission orbit planes intersect at an angle δ which can be much larger than the simple difference between their inclinations. This plane change angle is a very important parameter in determining ΔV impulse requirements. Spherical trigonometry gives the governing relationship,

$$\cos\delta = \cos i_o \cos i_m + \sin i_o \sin i_m \cos\Delta\Omega \qquad (1)$$

which applies to either circular or elliptical orbits. Note that the range of plane change can extend as high as 180^o, (if $\Delta\Omega = 180^o$ and $i_o + i_m = 180^o$) but is usually lower for most cases. For example, if $i_o = 28.5^o$ and $i_m = 57^o$, then the plane change requirements are 28.5, 61.4 and 85.5^o, respectively, for $\Delta\Omega$ = 0, 90 and 180^o. Figure 1-3 shows a graphical contour plot of equation (1) for the space station orbit at 28.5^o inclination. Other representations of this equation will be given later. It is important to note that for transfers between circular orbits of given size the optimal total impulse depends only on the plane change angle.

Another orientation angle becomes important when the mission orbit is elliptical. The argument of perigee, denoted by ω in figure 1-2, locates the position of perigee relative to the ascending node on the equator. The orbit's apseline or major axis between perigee (r_{pm}) and apogee (r_{am}) is thus orientated in three-dimensional space by the angles ω and i_m. Transfer analysis between nonplanar circular and elliptical orbits becomes more convenient if the apseline is measured with respect to the intersection point of the two orbit planes; e.g., the ascending node of the mission orbit on the station orbit plane. The relative argument of perigee ω_r is readily calculated using spherical trigonometry. For fixed orbit sizes, transfer impulse requirements are determined solely by the two relative orientation angles (δ,ω_r) which are independent of the originally chosen equatorial reference frame.

1.2.2 Earth Oblateness Perturbations

The fact that Earth is not a perfect sphere causes small perturbations in all orbit elements of an otherwise Keplerian conic orbit. Other perturbing influences such as lunar and solar gravity also exist but these may be ignored to first order for the range of orbits considered here. The main perturbation that cannot be ignored is the secular precession of the orbit in an inertial coordinate frame. This secular or nonperiodic, time-prevailing effect is due to the oblate figure of the Earth as represented principally by the second zonal harmonic coefficient J_2. The two main effects on the orbit are regression of the line of nodes and rotation of the line of apsides (for elliptical orbits). Expressed in units of degrees per day, the averaged secular perturbation rates are functions of orbit inclination, size and shape as given by the equations

$$\dot{\Omega} = -B \cos i \tag{2}$$

$$\dot{\omega} = 1/2B(5 \cos^2 i - 1)/2 \tag{3}$$

$$B = 540 J_2 \left(\frac{R_E}{a(1-e^2)} \right)^2 \left(\frac{24}{P} \right) \tag{4}$$

where $J_2 = 1.0827 \times 10^{-3}$, $R_E = 3442$ n. mi. is the equatorial radius of Earth, a and e are the semimajor axis and eccentricity of the orbit, and P is the orbit period in hours.

Figure 1-4 shows the nodal change with time for space station orbits of various inclinations. At 28.5° inclination the nodal regression rate is 7.2°/day; hence a time period of 50 days is required for the node to move westward along the equator before returning to its original location. Higher inclination orbits have smaller regression rates, and longer time periods are necessary for precession through 360°. Note that for the retrograde orbit at 98° inclination, the node moves eastward along the equator and completes a full revolution in 327 days.

The nodal regression rate for circular mission orbits is shown in figure 1-5 as a function of orbit period (altitude) and inclination. This rate is seen to fall off quite rapidly with orbit size and is less than 1°/day for orbit periods greater than four hours (3464 n. mi. altitude).

The significance of nodal regression on orbit transfer requirements may be illustrated by the following example. Consider payload retrieval from a circular 57° inclination orbit and return to a space station at 28.5° inclination with negligible stay time before return. If the two orbits had identical ascending nodes and there were no oblateness perturbation, then the plane change requirement would be 57 - 28.5 = 28.5° for both delivery and return transfers. Suppose, however, that at some initial time reference the mission orbit node led the station node by 90°. An immediate launch would require a plane change of $61.4° = \cos^{-1}$ (cos 28.5 cos 57) as determined by equation (1). Alternatively, one could wait 270 ÷ 4.8 = 56.25 days for nodal realinement; the nodal regression rates of the two orbits being 7.2 and 2.4°/day as determined from figure 1-5. Suppose that an immediate launch were necessary and that the three-impulse transfer time were two days each for delivery and return phases. The effective plane changes required for delivery and return would then be 59.2 and 71.6°, respectively, which correspond to the nodal misalinements of 85.2 and 114.0°.

The natural regression of the node can be used to maintain an orbit in sunlight the year round. A retrograde orbit is necessary for the eastward nodal motion of 0.985°/day. Figure 1-6 shows the required inclination as a function of altitude for Sun-synchronous circular orbits.

Figure 1-7 illustrates the apseline rotation of several elliptical orbits, all of which have a 200 n. mi. perigee altitude. The rotation rate is highest for orbits of low inclination and short period. For example, at 28.5° inclination, the perigee location advances at the rates of 1.2, 0.5 and 0.2°/day, respectively, for orbit periods of 6, 12, and 24 hours. The rotation rate approaches zero at the so-called "critical inclination" of 63.4°. Perigee recedes towards the ascending node for prograde orbit inclinations above the critical value.

1.2.3 All-Propulsive Transfer Strategies

The simplest and least expensive type of orbit transfer occurs when the departure and target orbits lie in the same plane. In this case, the well-known Hohmann transfer provides the minimum impulse solution in almost all situations of practical interest. The Hohmann transfer between coplanar circles is an ellipse cotangential to the circles along the apseline of the ellipse; i.e., a 180° transfer angle between perigee (inner circle radius) and apogee (outer circle radius) of the ellipse. This two-impulse maneuver requires two accelerating impulses to go to a larger circle, or two decelerating impulses to get to a smaller one. Delivery and return transfers are symmetric in that the sum of the two impulses are identical in each case.

Coplanar transfers between circular and elliptical orbits are also optimized, in most cases, by two-impulse Hohmann transfers. When the ellipse lies outside of the circle, then the optimal transfer always connects the circular radius to the apogee of the ellipse; for inner ellipses the connection is to the perigee. The optimal transfers for intersecting orbits connects the circle to the apogee of the ellipse. If the circle is tangent to the ellipse at either perigee or apogee, the Hohmann transfer degenerates to a one-impulse maneuver. Finally, the total impulse requirement is the same for delivery and return transfers.

Figure 1-8 shows the ΔV impulse requirement as a function of mission orbit period for circular and elliptical coplanar transfers. In this example the station orbit is circular at 200 n. mi. altitude and the ellipse perigee is also at this altitude. The total impulse for circle-circle transfer reaches a maximum value of about 13 500 ft/sec at a mission orbit period near 100 hours (55 000 n. mi. altitude), and then falls off gradually for transfers to higher orbits. It is generally less costly to transfer to a coplanar ellipse of the same energy (orbit period) as the circular mission orbit; this is the case also for the same apogee distance.

An interesting result for coplanar circular transfers is that a three-impulse bielliptic Hohmann transfer yields lower ΔV than the two-impulse strategy when the mission orbit period exceeds 63 hours (40 000 n. mi. altitude). The optimal three-impulse transfer is through infinity (Earth parabolic escape and return) as shown by the broken line curve in figure 1-8. Of course, this would take infinite transfer time and be somewhat impractical. Bielliptic transfers to a finite distance with a practical time constraint offer very little saving over the two-impulse strategy for coplanar transfers. However, such will not be the case when large plane changes are needed.

Only three possible transfer modes can be optimal for noncoplanar circular orbits: two-impulse usually referred to as "generalized" or "tilted" Hohmann transfer; three-impulse bielliptical transfers made up of two tilted Hohmanns; and transfer through infinity. The two- and three-impulse strategies are logical extensions of their planar solutions with all maneuvers made on the common line of nodes between the initial and final orbits, and at the apse points of the transfer ellipse. Each maneuver generally includes an out-of-plane component to change orbit inclination, but is otherwise circumferentially directed as in the coplanar problem. The optimal plane change split between maneuvers dictates that most of the plane change should be made at the outermost distance. The transfer through infinity is a limiting case of the three-impulse transfer where all the plane change is made at the second maneuver point at infinite distance and at infinitely small ΔV cost.

Figure 1-9 shows the boundary curves of the three optimal transfer regions as mapped in the parameter space of plane change angle and mission orbit period. If the required plane change exceeds 60°, transfers through infinity are globally optimal for all target orbits. Such transfers may also be optimal at lower plane change angles depending on the mission orbit period, and are always optimal if the orbit period exceeds 63 hours regardless of the plane change. The region in which two-impulse transfers are optimal extends to as high as 40° plane change for orbit periods in the range 7 to 15 hours, but for nearby or very distant orbits these transfers are best only for relatively small plane changes. This leaves the three-impulse strategy to fill the gap for circle-to-circle transfers of moderate plane change not too far removed from the initial orbit. Actually, the three-impulse strategy proves to be quite important since transfers through infinity are not of any real interest except as a reference lower impulse limit. Results given later in this section will extend the three-impulse boundary to all values of plane change above 60° with a constraint placed on maximum transfer time.

1.2.4 Aerobrake-Assisted Transfers

An orbit transfer vehicle equipped with aerodynamic surfaces may utilize Earth's atmosphere for purpose of energy management. Significant reduction in propulsion-supplied ΔV impulse may result for both delivery and return mission phases depending on specific orbit transfer requirements. In the present context, aerodynamic augmentation refers to a drag device only. While the aerobrake design may have a small amount of lift capability, this is used for trajectory control purposes rather than plane change maneuvers; an aeromaneuvering vehicle must have moderate-to-high L/D to effect any significant plane change. Hence, the aerobrake-assisted transfer involves only in-plane velocity reduction during the brief pass through the upper atmosphere.

Figure 1-10 illustrates the potential savings in ΔV impulse provided by aerobraking. The effective drag velocity increment is calculated as the reduction in perigee velocity and shown as a function of the pre-entry orbit parameters for an assumed ballistic coefficient of 23 kg/m^2 (4.7 lb/ft^2). The typical range of perigee altitudes for single-pass aerobraking operations is 82-100 km (44-54 n. mi.). Note that the drag velocity increment at constant perigee is virtually insensitive to preentry apogee altitudes above 20 000 n. mi. However, the atmosphere capture (non-skipout) limit is very sensitive to perigee altitude. The analysis of aerobrake-assisted transfers will assume single-pass operation with an effective upper limit drag reduction of about 8000 ft/sec. It is recognized that specific design temperature limitations may require higher altitude operations and multiple passes to achieve the maximum ΔV potential of aerobraking, but at a cost in extended mission time.

A comparison of all-propulsive and aerobrake-assisted impulse requirements is shown in Figure 1-11 for the return phase of coplanar circular transfers. The ΔV savings is 550 ft/sec for return from a 2-hour orbit, 6220 ft/sec from a 12-hour orbit, and 8820 ft/sec from a 72-hour orbit. Even larger savings will be seen to exist in the case of noncoplanar return transfers.

1.3 TRANSFER REQUIREMENTS FOR CIRCULAR ORBITS

This section presents ΔV impulse requirements for optimal transfers between a low altitude space station and various mission orbits having circular orbit periods ranging from 1.5 to 72 hours. The station orbit is circular at 200 n. mi. altitude. Orbit inclinations with respect to the equatorial plane are treated parametrically over a wide range of possibilities. Working graphs show total impulse and transfer time for both delivery and return mission phases with direct comparison of all-propulsive and aerobrake-assisted transfer strategies.

1.3.1 Method of Analysis

Schematic diagrams of the various transfer strategies are shown in figures 1-12 and 1-13. These are shown in planar perspective for drawing convenience, but it is understood that the station and mission orbits are not generally coplanar. The common line of nodes is indicated, and all impulsive maneuvers are made along this line with 180° transfers between impulses. The generalized Hohmann subarcs, with plane change allowed at each impulse point, are known to yield the optimal solution in the sense of minimum ΔV sum. All-propulsive strategies include both two-impulse and three-impulse transfers; the latter employs an intermediate impulse indicated by point 2 at an altitude h_I equal to or greater than the mission orbit altitude. Aerobraking in the Earth's upper atmosphere is "free" energy management in the sense that no ΔV is chargable at that point to the propulsive impulse requirement. Note that aerobrake-assisted delivery transfers are always of the three-impulse type whereas either two- or three-impulse return transfers may be considered. The all-propulsive delivery and return transfers are symmetric and have identical solutions. This symmetry is not generally true for aerobraking.

Total ΔV can be expressed as a function of the following parameters: (1) the altitudes of the station and mission orbits; (2) the intermediate altitude in the case of three-impulse transfer; (3) the individual plane changes at each propulsive impulse point; and (4) the altitude of the aerobraking pass in the atmosphere. Since all maneuvers are of the nodal type, the total plane change requirement must equal the sum of the individual plane changes; this is the fundamental relationship on the condition of optimality.

With no significant effect on the solution, the aerobraking altitude may be fixed at any reasonable value (e.g., 44 n. mi.). No plane change occurs during aerobraking since it has been assumed here that this is a drag-only maneuver.

Given the station and mission orbit altitudes and the total plane change between orbits, the optimization problem involves the determination of the plane change split between impulses and the intermediate altitude of a three-impulse transfer. An upper limit on this altitude, or alternatively on the total transfer time, may be imposed for practical reasons. The basic algebraic equations of the problem and those derived from the calculus are fairly straightforward, but will be omitted here for reasons of brevity. The overall system of equations contain the free variables as radicals. A closed-form solution is apparently not feasible, although a series expansion solution has been derived elsewhere. An alternative solution, and the one adopted here, is direct numerical iteration of the nonlinear algebraic equations. Convergence is well-behaved and rapid enough with a reasonable first guess of the variables and with the use of tracking from one solution to the next nearby case. A FORTRAN program was written for this purpose and run on a PDP 11-34 computer. Graphic data formats were also computer-generated.

1.3.2 Working Graphs

Equation (1) of the previous section gave the plane change requirement as a function of station and mission orbit inclinations and the nodal misalinement of the two orbits. This relationship is graphed in figures 1-14 through 1-16 for specific station inclinations of 28.5, 57 and 98°. The mission analyst may use these graphs where convenient, or simply solve equation (1), to obtain the plane change angle δ which serves as the basic parameter for expressing the variation of ΔV impulse requirements.

A summary presentation of results is given first by figures 1-17 a and b corresponding to the delivery and return phases of circular orbit transfer. Contours of constant ΔV are mapped as a function of plane change angle and mission orbit period. The data shown reflect the better choice of two-impulse or three-impulse solutions with a 48 hour transfer time limit. Each graph provides an "accessible region" comparison of the all-propulsive and aerobrake-assisted transfer modes. For example, consider the delivery phase with a ΔV limit of 16 000 ft/sec. The all-propulsive transfer capability extends from a 40° plane change for a 1.53-hour orbit to an 80° plane change for a 40-hour orbit. Aerobraking is superior for mission orbit periods less than 12 hours, and can achieve 180° plane change for orbit periods less than 3.2 hours. An interesting feature of the all-propulsive transfer mode is that for small values of plane change the shorter period mission orbits yield lower ΔV; in the range 30 to 40°

the ΔV requirement is not very sensitive to orbit period; for larger plane changes the trend reverses and it is actually easier to get to orbits farther away from the space station.

Figure 1-17b shows the clear advantage of aerobraking on return transfers to the station. An impulse capability of 12 000 ft/sec allows return access from almost all orbit sizes and inclinations even up to a 180° plane change. In contrast, a 12 000 ft/sec all-propulsive capability would limit the plane change to only 30° and the coplanar mission orbit period to under 17 hours (14 000 n. mi. altitude). Halving this capability to 6,000 ft/sec still allows return from a geostationary orbit with plane changes as much as 50°, whereas all-propulsive access is limited to four-hour period coplanar orbits and plane changes under 14° for nearby orbits.

The set of graphs presented in figures 1-18 through 1-24 provide easier to read data formats of total impulse and transfer time requirements. Graph pairs are given for seven specific mission orbits having orbit periods ranging from 1.53 to 72 hours. All-propulsive and aerobrake-assisted requirements, both two- and three-impulse, are compared for both delivery and return mission phases.

The transfer time limit of 48 hours was chosen somewhat arbitrarily, although it is not an unreasonable constraint. It is of some importance, however, to indicate the sensitivity to transfer time for three-impulse maneuvers having large plane change angles. This is shown in figures 1-25 through 1-31 for a 90° plane change requirement. Note that the 48 hour nominal limit generally occurs in the downside sensitivity region; i.e., after the knee of the curve. Hence, the penalty for halving the limit is usually greater than the gain for doubling the limit. If mission/vehicle design permits a longer transfer time of four days or 96 hours, then, depending on mission orbit period, the ΔV reduction is 1 to 8 percent for the all-propulsive mode, 4 to 7 percent for aerobrake-assisted delivery, and 5 to 25 percent for aerobrake-assisted return. As a general characteristic, the advantage of longer transfer time becomes more significant as the plane change angle and mission orbit distance increase.

1.4 TRANSFER REQUIREMENTS FOR ELLIPTICAL (MOLNIYA) ORBITS

This section presents ΔV impulse requirements for optimal transfer between a low altitude space station and the special, semisynchronous elliptical orbit of 12-hour period and 63.5° inclination. Transfer options considered here include all-propulsive delivery and return and aerobrake-assisted return. Results are given for space station orbit inclinations of 28.5, 57 and 98°.

1.4.1 Method of Analysis

A schematic diagram of the transfer strategies is shown in figure 1-32. Only three-impulse maneuvers are described as the station and mission orbit planes will generally be separated by a large enough angle so that two-impulse maneuvers are nonoptimal. For all-propulsive delivery, the first impulse establishes an intermediate outbound transfer toward the region of nodal intersection of the two orbit planes. This impulse is applied in the near-tangential direction at or very close to the perigee of the intermediate transfer. Only a small amount of plane change occurs at this point. The second impulse supplies most of the necessary plane change at large distance, typically between 4 and 18 Earth radii depending on the orbit orientations. The third impulse completes the plane change and orbit energy match and is made somewhere between apogee and perigee on the downside of the mission orbit. Unlike circular orbit transfer, the impulses are not generally made on the line of nodal intersection nor at apse points of the transfer ellipses.

Geometric considerations would argue that the optimal all-propulsive return should be the complementary image of the delivery transfer. That is, the mission orbit arrival point becomes the departure point, the midcourse impulse point has the same spatial location, and the return point is the same as the station departure point. The transfer arcs through these renumbered impulse points are simply the complementary or "unflown" portions of the delivery transfer arcs. The sum of impulses is the same for delivery and return. This is, of course, the ideal situation with no accounting for stay time in the mission orbit or the real position-time relationships of vehicles in these orbits. Practical round trip sorties would require appropriate rendezvous phasing maneuvers at some expense in total trip time.

Return with aerobraking is similar to the all-propulsive strategy except for inclusion of perigee lowering at impulse point 2, the subsequent atmospheric phase, and the additional 180° subarc back to the station orbit. The total impulse savings of aerobraking may be closely approximated by analytic formula given the characteristics of the all-propulsive solution.

Numerical solutions of the delivery phase, all-propulsive transfers, were obtained using a multiimpulse optimization code. This FORTRAN program is based on the calculus of variation (primer vector) theory and employs a conjugate gradient search procedure for determining the optimal impulse times and positions. Supplementary data and checkpoints were obtained from the solution maps given in reference 2.

1.4.2 Working Graphs

The plane change relationship between station and mission orbit inclinations and nodal difference given by equation (1) is graphed in figure 1-33. For a space station at 28.5° inclination, the plane change angle varies between 35 and 92° as the nodal misalinement increases from zero to 180°. The corresponding range of plane changes for station orbits at 57 and 98° is, respectively, 6.5 to 120.5° and 34.5 to 161.5°. Hence, transfer to the Molniya orbit from these space station locations is never coplanar and the plane change requirement could be very large.

Figures 1-34 through 1-36 show the total ΔV impulse as a function of orbit nodal difference for delivery and return transfers. Minimum impulse requirements of the all-propulsive mode are about 9300 ft/sec for the 57° station, 13 800 ft/sec for the 98° station, and 14 900 ft/sec for the 28.5° station. Note that the optimal nodal difference is near zero for the stations at 28.5 and 57° inclinations, but is closer to 90° for the station at 98° inclination. Maximum impulse requirements corresponding to the condition of 180° nodal misalinement are in the range 18 000 to 19 800 ft/sec for the three station orbits.

Aerobraking is seen to offer a very substantial reduction in ΔV for return transfers from the Molniya orbit to the space station. The range of this savings for various orbit geometries is 7350 to 9250 ft/sec.

1.5 EXAMPLE APPLICATIONS - DELIVERY AND RETURN WITH STATION AT 28.5°

Use of the graphic data formats to quickly estimate ΔV impulse requirements will be illustrated for three examples of round trip orbit transfers. The mission applications are: Sun-synchronous circular orbit; geostationary equatorial orbit; and elliptical Molniya orbit. The space station orbit will be assumed to have a 28.5° inclination.

1.5.1 Sun-Synchronous Mission Orbit

The mission objective is to service a satellite in a 200 n. mi. altitude, near-polar circular orbit and then return to the space station base. A one-day service time will be assumed. Figure 1-6 shows that the Sun-synchronous inclination of the satellite's orbit is 97°. It will be assumed that launch is not time-critical so that the necessary waiting period for optimal alinement of orbit nodes is allowable. Figure 1-5 gives the nodal regression rates of the station and satellite orbits as 7.2°/day and approximately 1.0°/day. If the optimal nodal alinement at launch were near $\Delta\Omega = 0$, then the plane change requirement would be 68.5° as determined from equation (1). Figure 1-18 indicates that a large ΔV savings is obtained by using a three-impulse transfer instead of two impulses, even though the one-way transfer time is much longer; 48 hours instead of 0.76 hour (delivery) or 1.48 hours (return). Furthermore, great benefit is obtained by aerobraking for both delivery and return transfers. Aerobraking capability will therefore be assumed for this mission application.

Since both orbits are subject to significant nodal regression over several days time, particularly the station, the optimal nodal difference at launch could be somewhat offset from zero. Taking the average value of delivery and return plane change angles as a performance measure, a quick calculation shows that the station node should lead (be east of) the satellite node by about 20° at the time of launch. Then, with the westward motion of station node and eastward motion of satellite node, the respective nodal difference of the delivery and return transfers are:

$$\Delta\Omega_D = (1.0 \text{ deg/day}) \times (2 \text{ days}) - 20 \text{ deg} = -18 \text{ deg}$$

$$\Delta\Omega_R = (1.0 \text{ deg/day}) \times (3 \text{ days}) - 20 - (7.2 \text{ deg/day}) \times (5 \text{ days}) = 19 \text{ deg}$$

Equation (1) gives the plane change angles of the two transfers as 69.9 and 70.0°. Figure 1-18a may be read at these angles to give the aerobrake-assisted delivery and return ΔV impulses of approximately 11 900 ft/sec each. Note that the all-propulsive three-impulse requirement would be almost double this value.

The previous calculation for the best nodal position at launch is a moot point in this case since the ΔV requirement is fairly insensitive to small variations in the $\Delta\Omega = 0$ region. For example, if $\Delta\Omega$ were chosen to be zero at launch, then the two transfer plane change angles would have been 68.52 and 74.87°, and the corresponding delivery and return impulse totals would have been 11 860 and 12 030 ft/sec -- not very different. One final point on this example is the distribution of ΔV impulses for both delivery and return transfers. These data were not previously graphed, but the computer printout gives the following result: 9539 ft/sec at station launch (or satellite orbit departure); 2085 ft/sec at the second midcourse point; and 277 ft/sec at satellite arrival (or station return).

1.5.2 Geostationary Mission Orbit

The mission objective in this example is to deliver a satellite to a 24-hour equatorial orbit and return to the station base as soon as possible. Each transfer must execute a plane change of 28.5°. Figure 1-23 indicates that a two-impulse, all-propuslive delivery is optimal and that a two-impulse, aerobrake-assisted return is quite adequate (1000 ft/sec more than the three-impulse return). The corresponding transfer times are 5.3 hours and 6.0 hours.

Figure 1-5 shows the nodal regression in the geostationary orbit is negligible over short time intervals. With the station nodal regression at 7.2°/day (0.3°/hr), the arrival point at GEO on a 180° transfer will lead the station's (opposite) nodal position on the equator by 1.6°. One could probably return immediately with little ΔV penalty, although the strict optimal condition is departure at a nodal intersection. Ignoring any rendezvous phasing problems, the next such intersection occurs $178.4 \div 15.3 = 11.66$ hours after GEO arrival. This value needs to be adjusted slightly to account for the six-hour return transfer; departure from GEO should occur 11.54 hours after arrival.

Figure 1-23a read at a plane change angle of 28.5° gives the two-impulse, all-propulsive ΔV requirement at 13 800 ft/sec; the impulse distribution is 7970 and 5830 ft/sec. The sum of impulses for the two-impulse, aerobrake-assisted return is 6325 ft/sec with the distribution of 6045 and 280 ft/sec.

1.5.3 Molniya Mission Orbit

The objective of this mission example is to carry out an immediate transfer to the semisynchronous elliptical orbit at 63.5° inclination, remain there for three days, and then return to the space station base. Assume that at the time of launch demand the station node leads the target orbit node by 120°. Figure 1-33, or equation (1), shows the plane change angle to be approximately 80° for the delivery transfer.

All-propulsive delivery at $\Delta\Omega = -120°$ requires a total ΔV impulse of about 17 100 ft/sec as read from figure 1-34. The one-way transfer time is 48 hours. The three-impulse distribution (computer printout) is 9470, 3620 and 4050 ft/sec. The midcourse impulse is made 11.9 hours after launch at 51 400 n. mi. altitude. Arrival at the target orbit is at a true anomaly of 219° and altitude of 11 800 n. mi.

With return to the station seven days after launch, the station's orbit node would have regressed about 50°. The nodal difference between the two orbits on the return transfer is $\Delta\Omega = -70°$. The ΔV requirement is 6500 ft/sec as read from figure 1-34 assuming an aerobrake-assisted return. Computer results show the impulse distribution is 3700, 2520, and 280 ft/sec. The departure impulse on the target orbit is at a true anomaly of 226° and altitude of 9820 n. mi. The midcourse impulse occurs 36 hours after departure at an altitude of 52 200 n. mi.

1.5.4 Mission Nomograms and Performance Envelopes

The preceding paragraphs have outlined the procedure and data sets for quickly estimating mission performance, given a set of initial conditions. For user convenience and rapid data access, these data have been consolidated and are presented in the form of mission nomograms in figures 1-37(a) through 1-37(d). The data points that are highlighted correspond to the mission examples in paragraphs 1.5.1 through 1.5.3.

Figure 1-37(a) is a repeat of figure 1-5 (nodal regression rate) with higher resolution for purposes of extracting specific values. Figures 1-37(b) through 1-37(d) trace the example mission.

Figure 1-37(b) is the nomogram for payload deploy in a low Earth orbit. The examples are for a typical Sun-synchronous mission. From the plot at the upper left, the required orbit nodal difference for balancing the outbound and return plane change is seen to be a function of the stay time in the mission orbit; therefore, once the stay time is determined, the approximate optimized mission parameters can be determined from the following steps:

A. Using the mission stay time as a parameter, the plot in the upper left is used to determine the required nodal difference for the inclination of interest, in this case 97°;

B. Once the required orbit nodal difference has been determined, the figure in the upper right is used to extract the plane change angle that results;

C. Having found the approximate plane change angle, the plots on the lower left and right can be used to determine estimate of the mission performance parameters.

Figure 1-37(c) is the nomogram for payload deploy in a 24-hour period orbit, the example being for a typical geosynchronous mission. Since the geostationary mission node vanishes (is undefined) for an equatorial orbit the effect of mission stay time can be described in terms of movement of the space station node.

This is shown in the upper left plot; this, however, becomes trivial since the plane change for an optimum time departure is constant at 28.5° -- independent of initial orbit node difference -- as shown in the upper right; as previously described, the mission performance parameters can then be determined.

Figure 1-37(d) is for a twelve hour elliptic (Molniya) mission. Using the figure in the upper left, the nodal difference between departure and return is shown as a function of mission stay time. These charts may be used in the following manner:

A. Given a mission stay time, the nodal difference between delivery and return is determined from the upper left plot.

B. For any prespecified initial nodal difference between the space station orbit and the Molniya orbit, the required plane change for delivery is determined from the upper right figure.

C. The plane change for return can be determined also from the upper right figure by incrementing the delivery orbital nodal difference by the amount established as a function of mission stay time. When the space station node for delivery leads the mission orbit the retrieval nodal difference will move to the left; when the space station node for delivery is behind the mission orbit, the retrieval node difference moves right.

In addition to these nomograms, the performance requirements (reflecting the appropriate choice for the optimum maneuver strategy) are shown in figures 1-38 through 1-41 in terms of the mission envelope that can be captured. Figures 1-38(a) through (c) reflect outbound phase mission envelopes for space station inclination of 28.5, 57 and 90°, respectively, utilizing an all propulsive OTV; figures 1-39(a) through (c) reflect the outbound mission envelopes utilizing aero-braking assist. Figures 1-40(a) through (c) address the return phase employing an all-propulsive OTV, and figures 1-41(a) through (c) depict the mission envelope for an aero-braking assist return.

1.6 REFERENCES

(1) Gobetz, F. W. and Doll, J. R.: A Survey of Impulsive Trajectories, AIAA Journal, vol. 7, no. 5, May 1969, pp. 801-834.

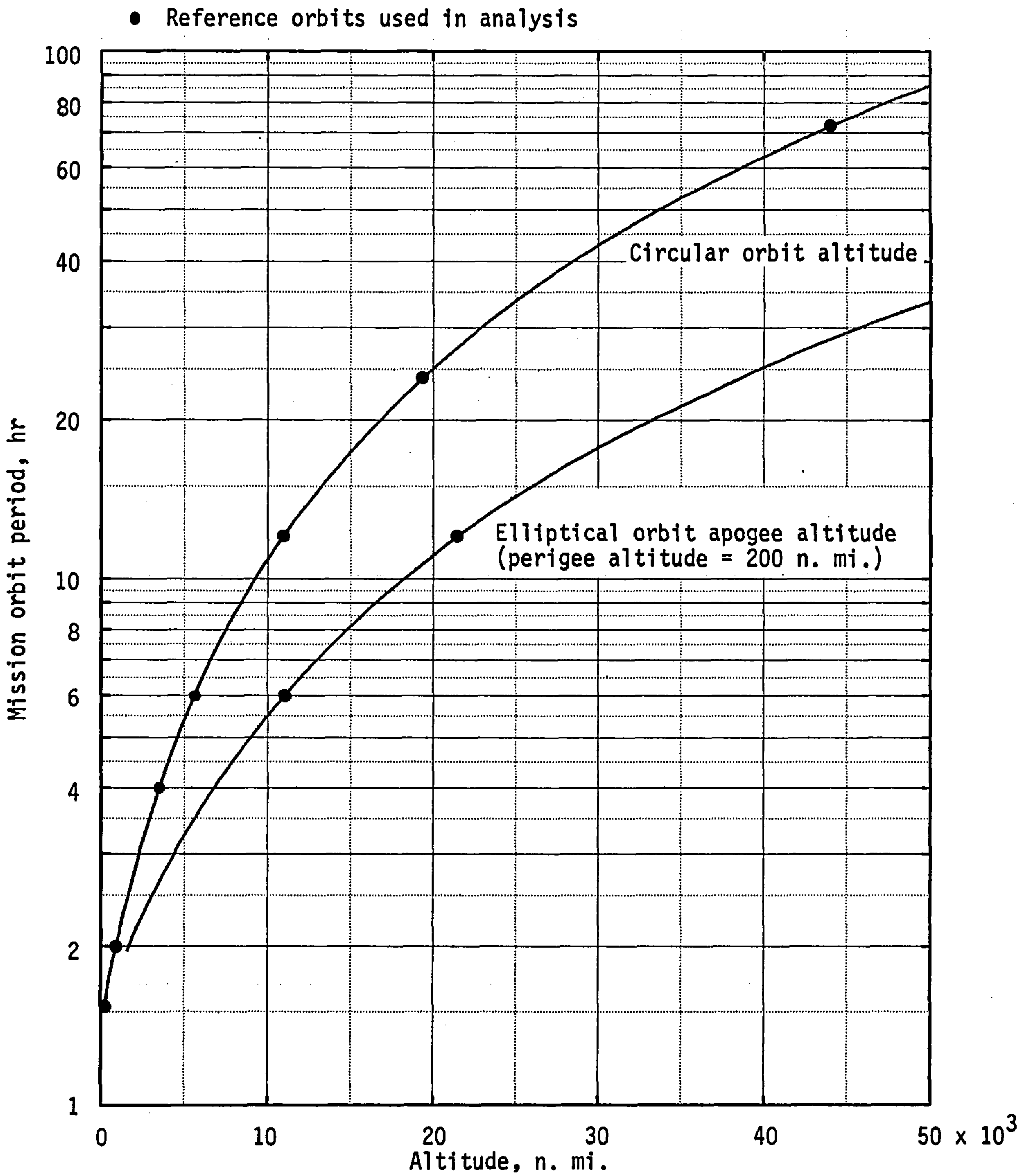


Figure 1-1.- Relationship between mission orbit period and altitude.

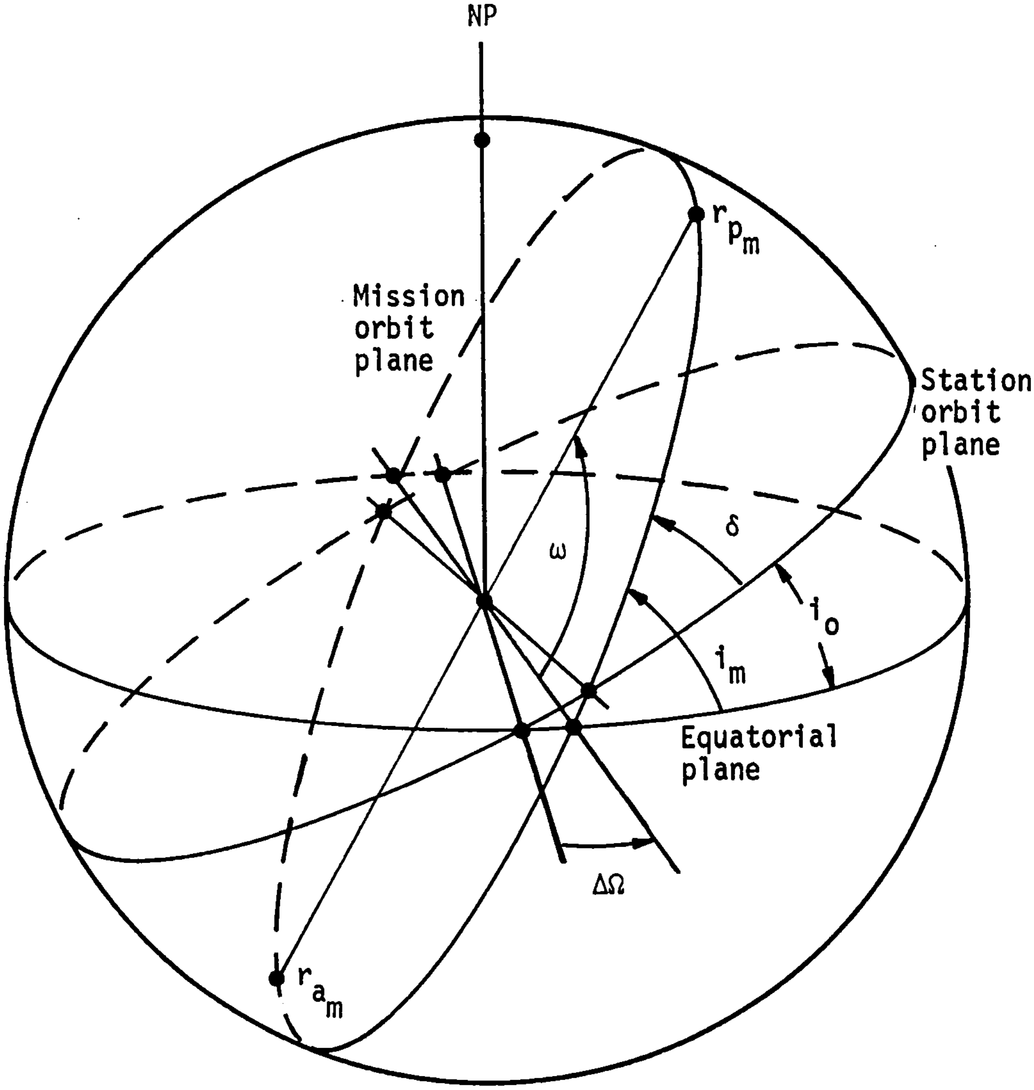


Figure 1-2.- Space station and mission orbit definitions.

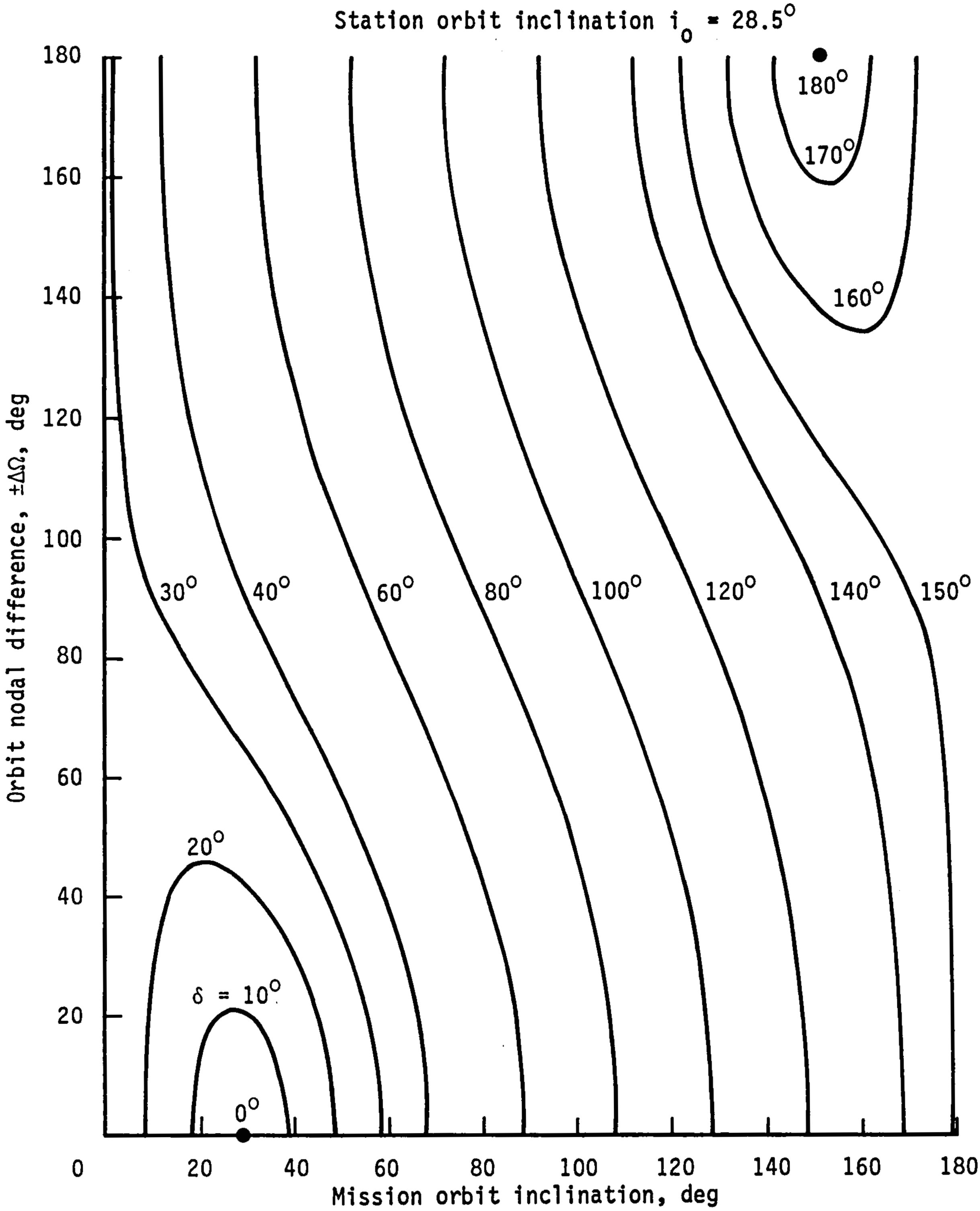


Figure 1-3.- Plane change contours for space station at 28.5 degrees inclination.

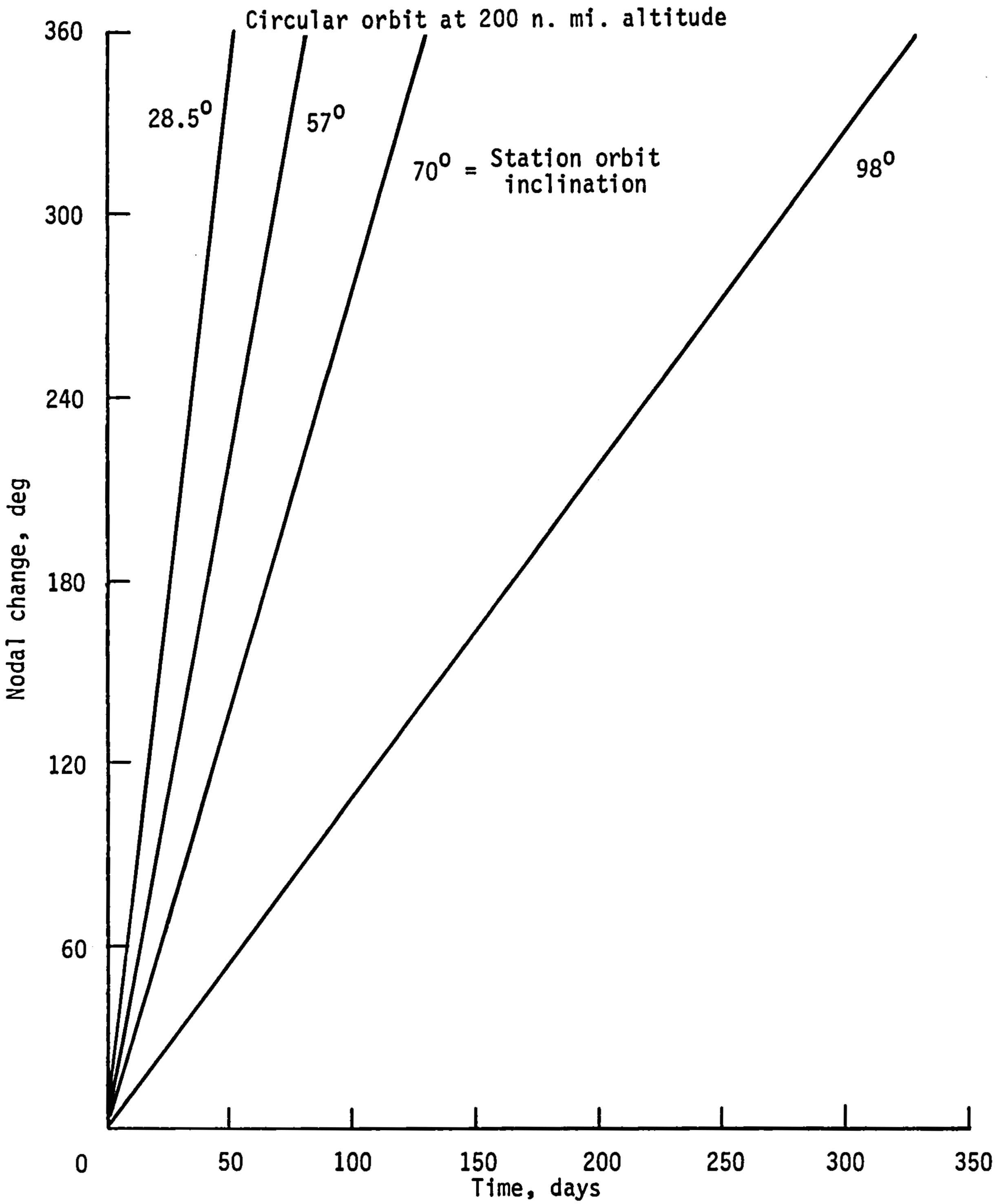


Figure 1-4.- Sun-synchronous circular orbit inclinations.

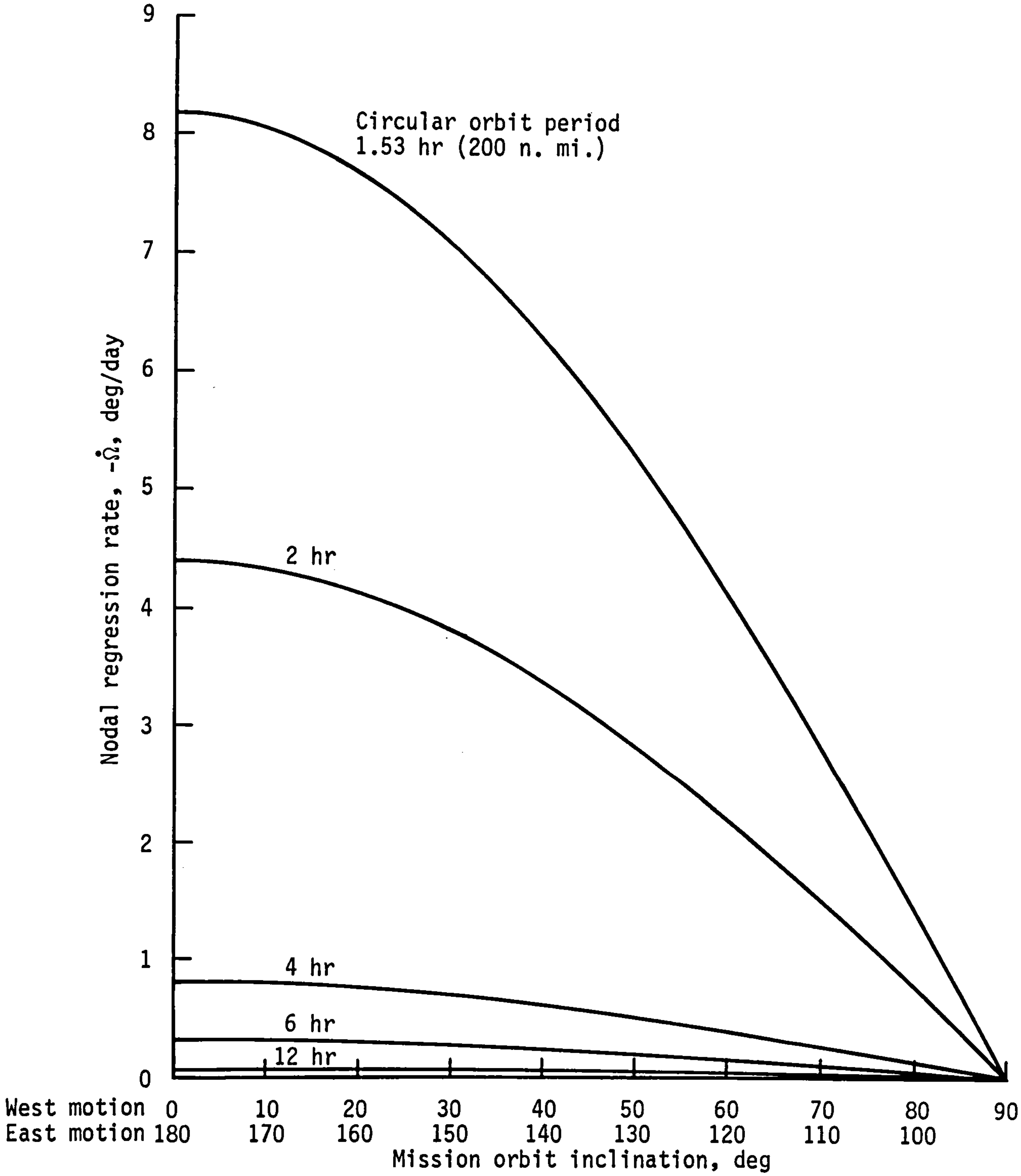


Figure 1-5.- Nodal regression of circular mission orbits.

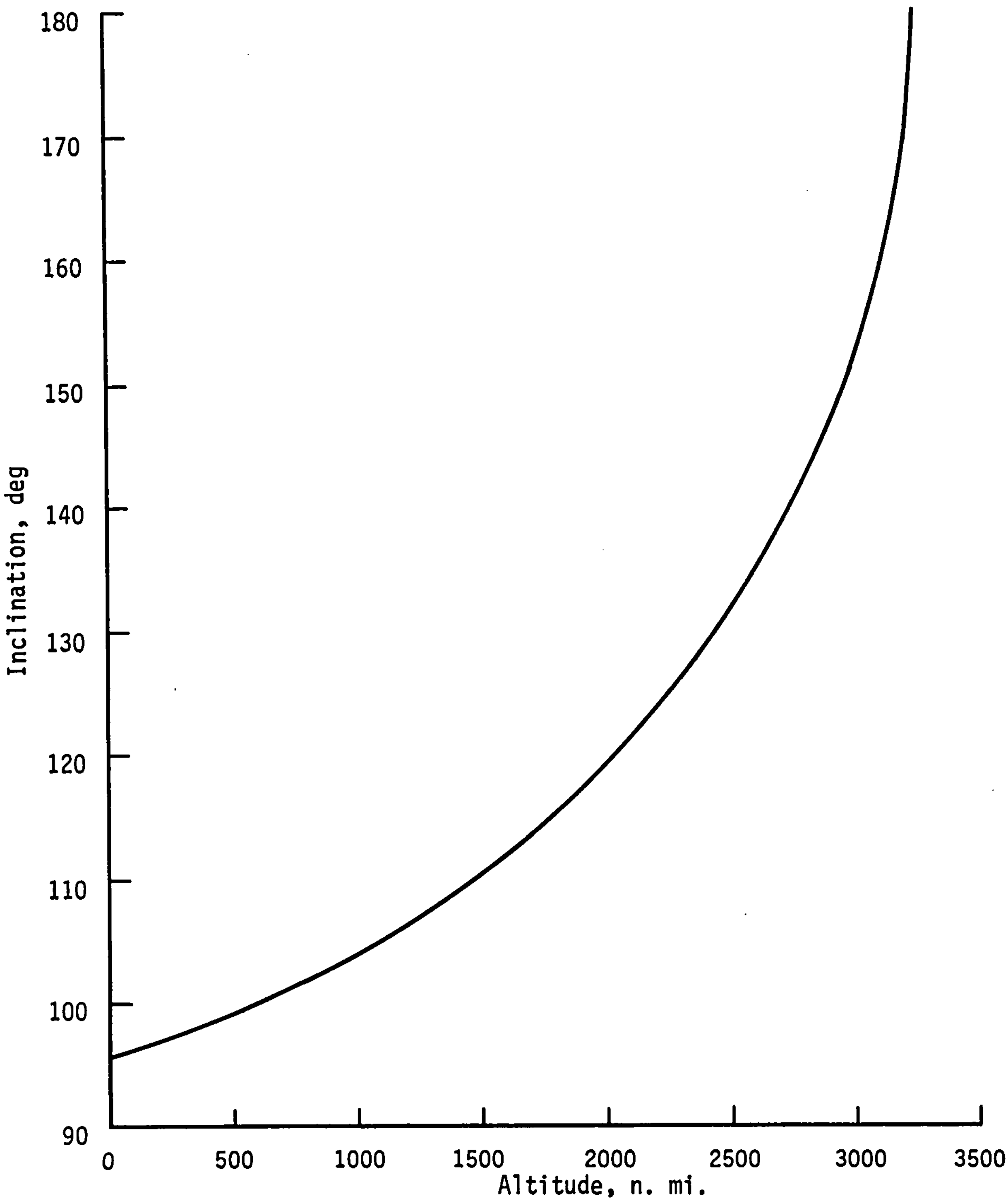


Figure 1-6.- Sun synchronous circular orbit inclinations.

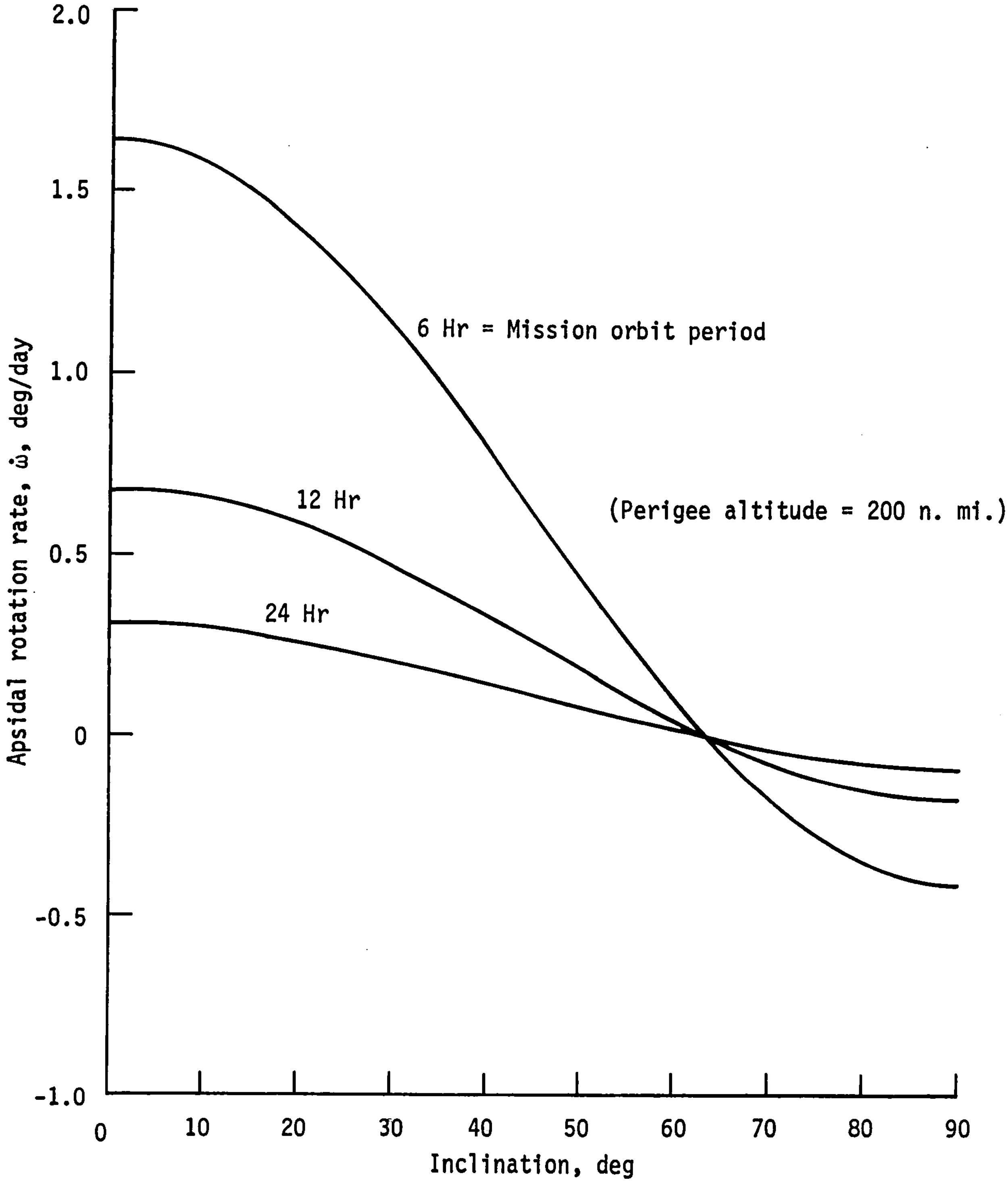


Figure 1-7.- Apseline rotation of elliptical orbits.

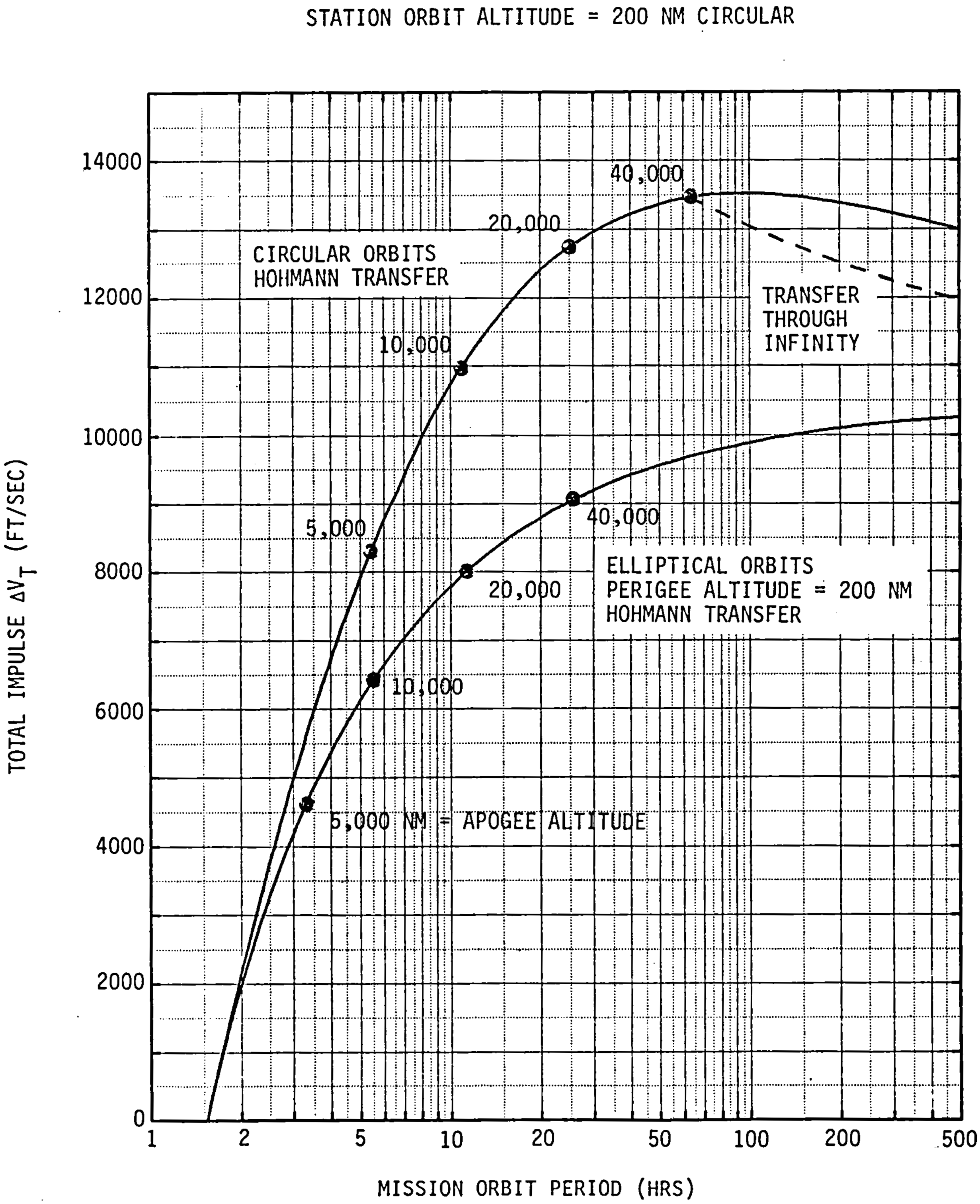


Figure 1-8.- Impulse requirements for coplanar orbit transfer.

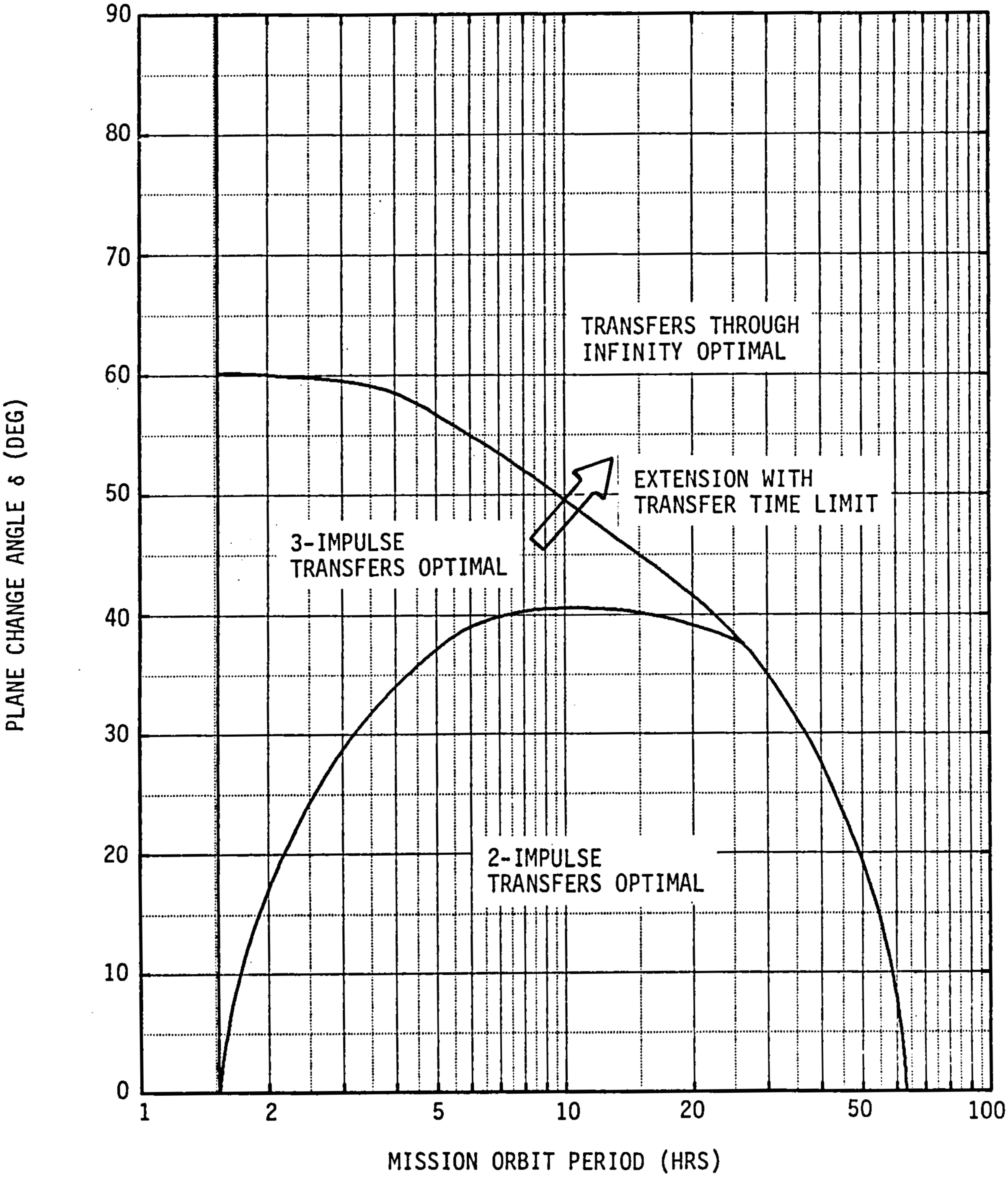


Figure 1-9.- Optimal transfer for noncoplanar circular orbit transfers.

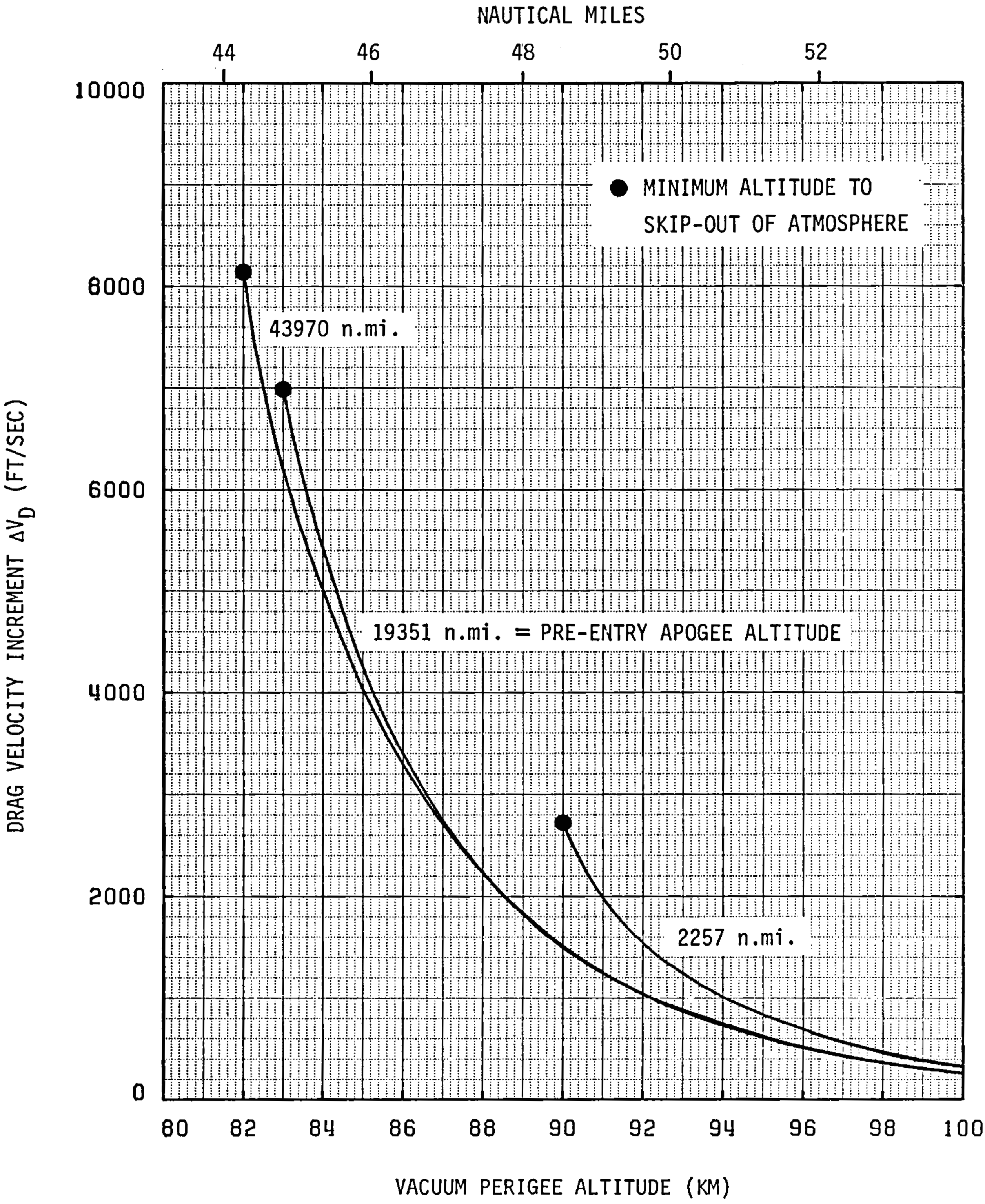


Figure 1-10.- Aerobrake potential B = 23 kg/m^2.

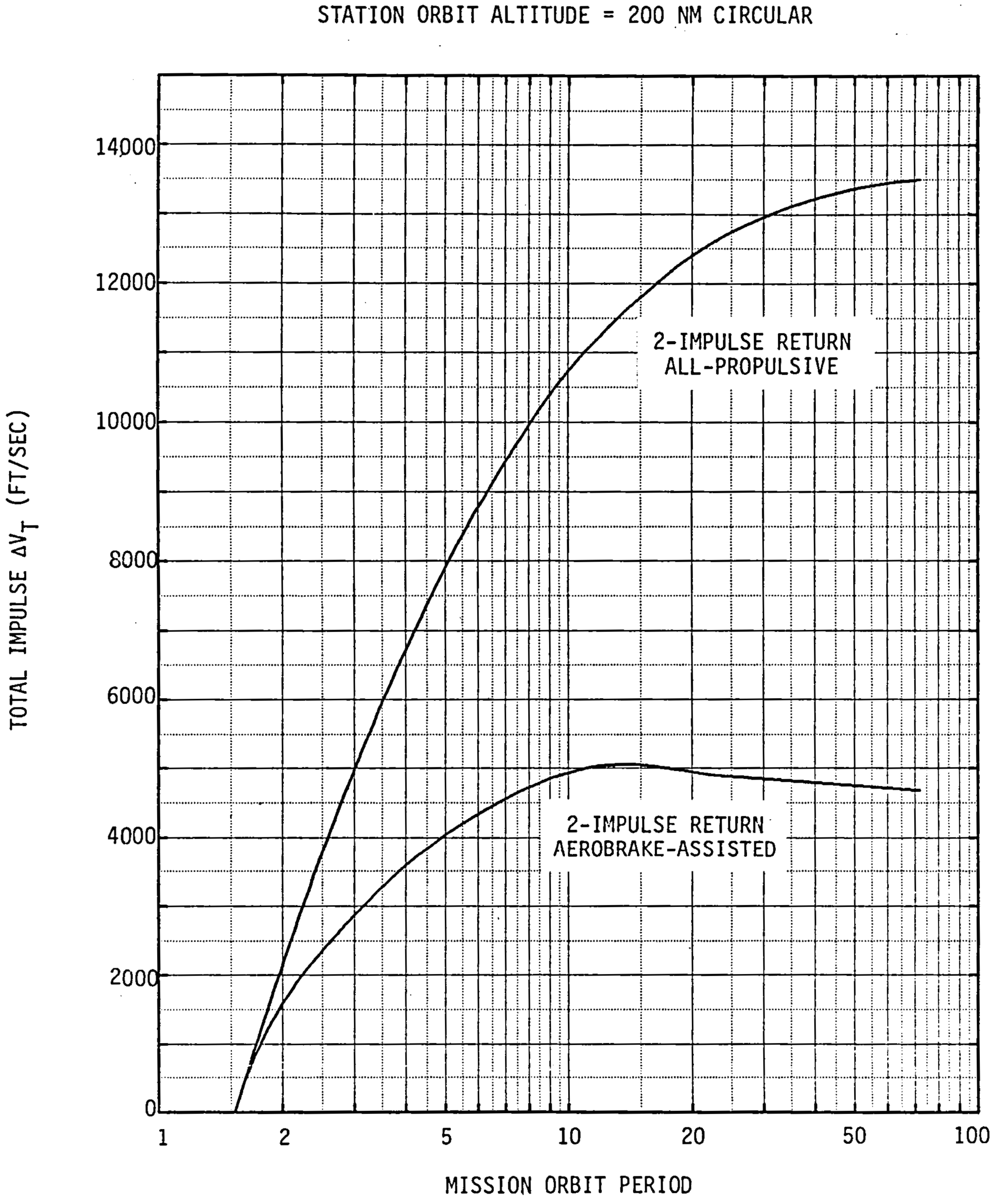


Figure 1-11.- Return transfer requirements for coplanar circular orbits.

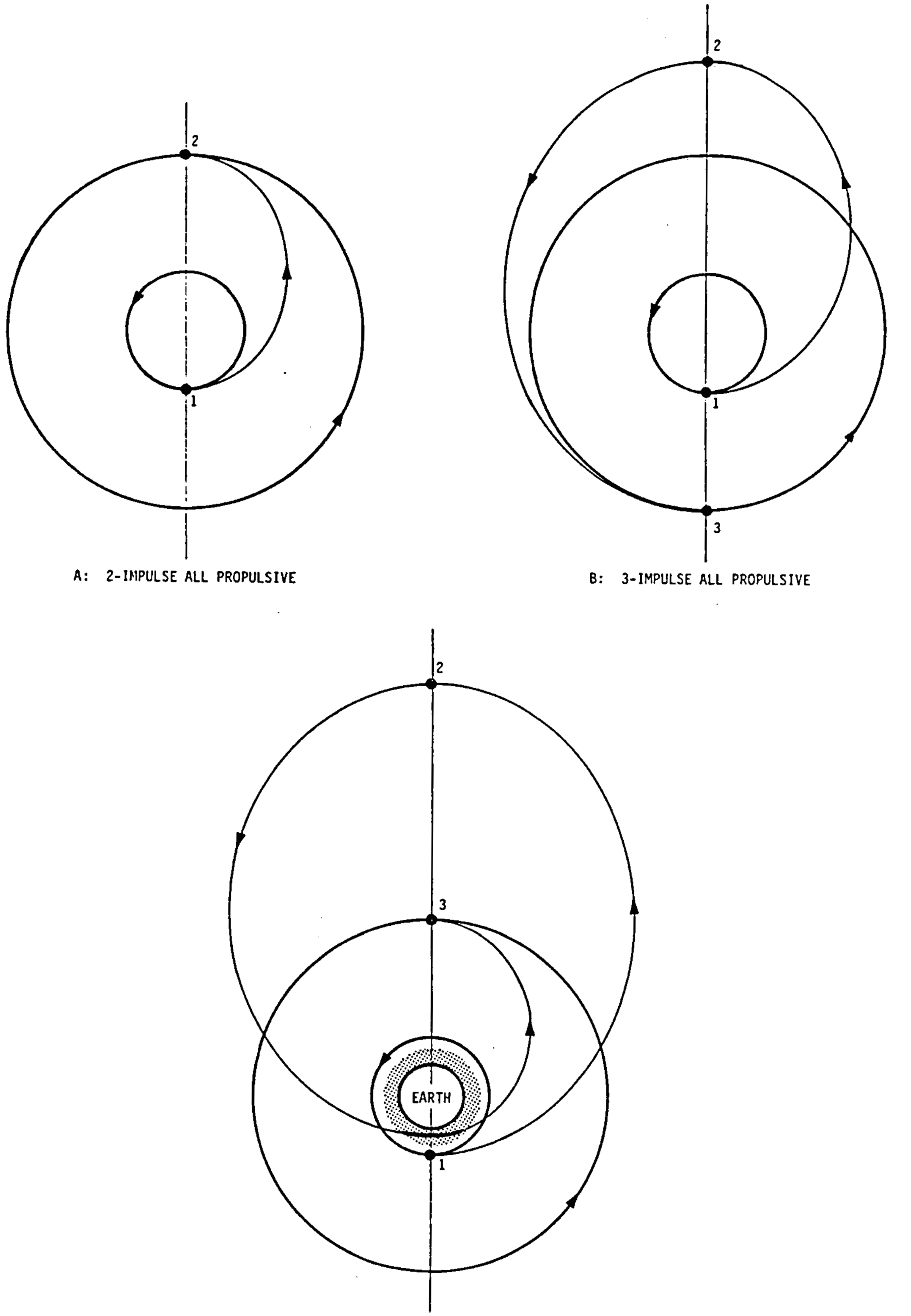


Figure 1-12.- Circle-to-circle transfer strategies - delivery phase.

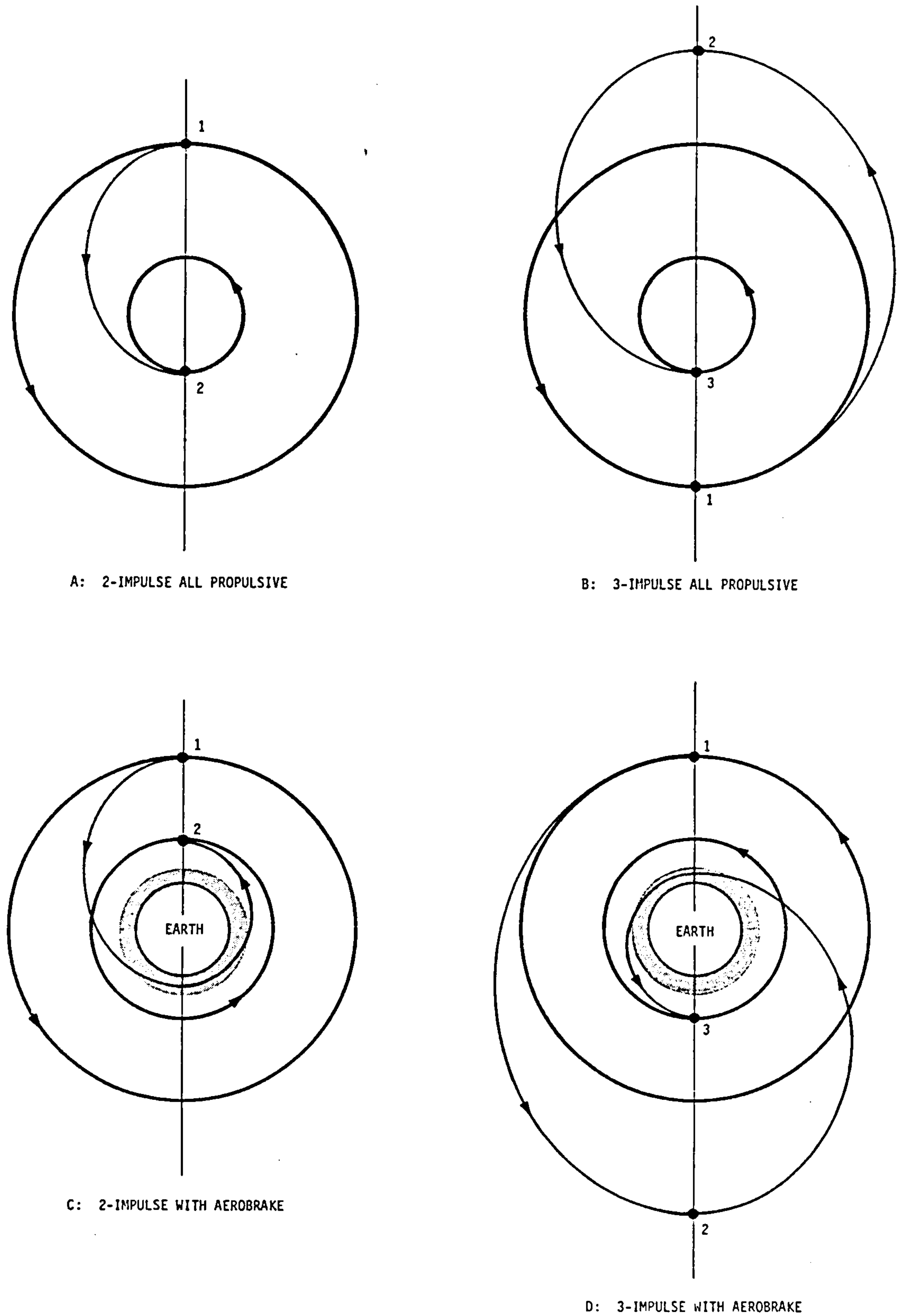


Figure 1-13.- Circle-to-circle transfer strategies - return phase.

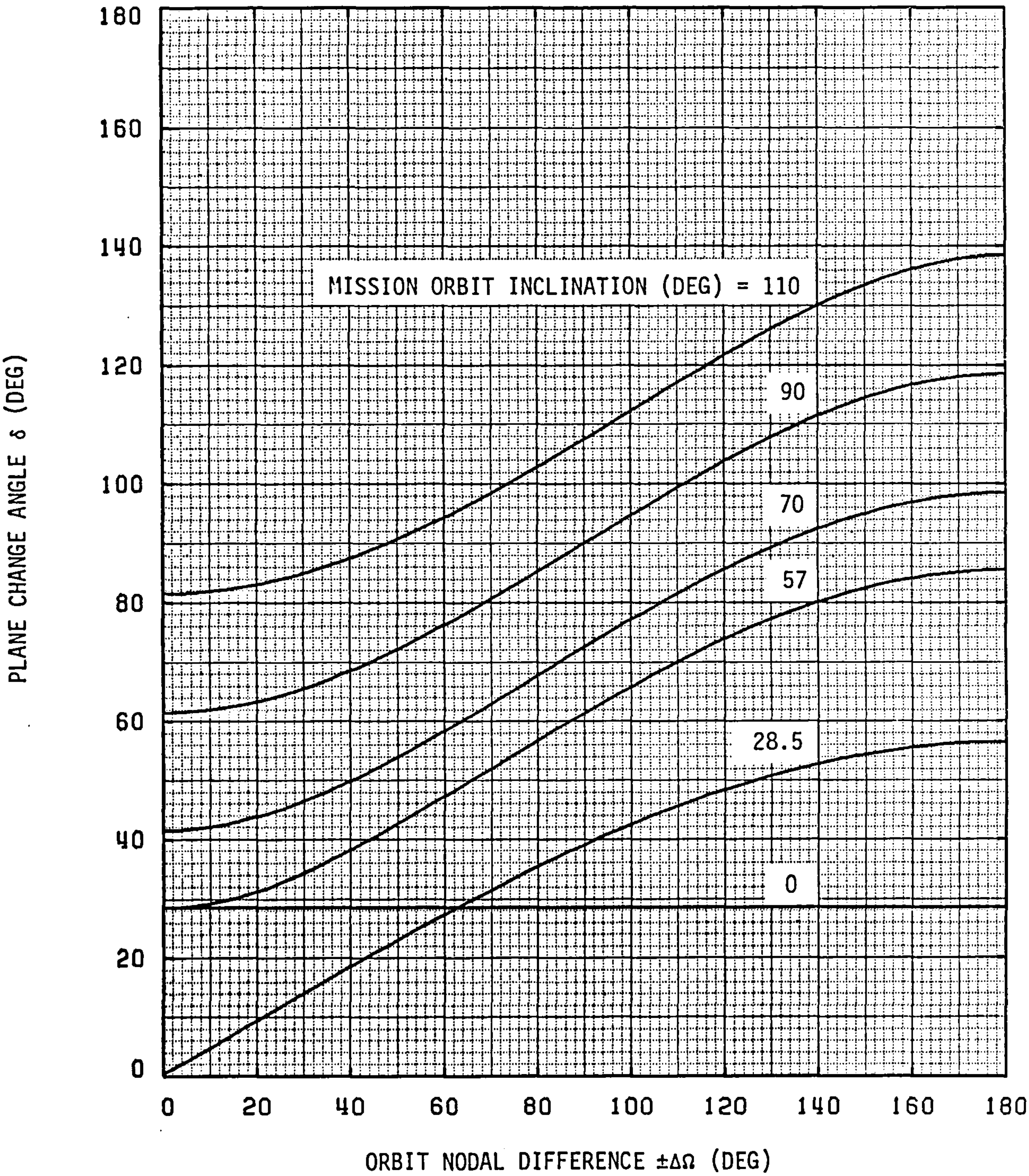


Figure 1-14.- Plane change requirements for station orbit at 28.5 degrees inclination.

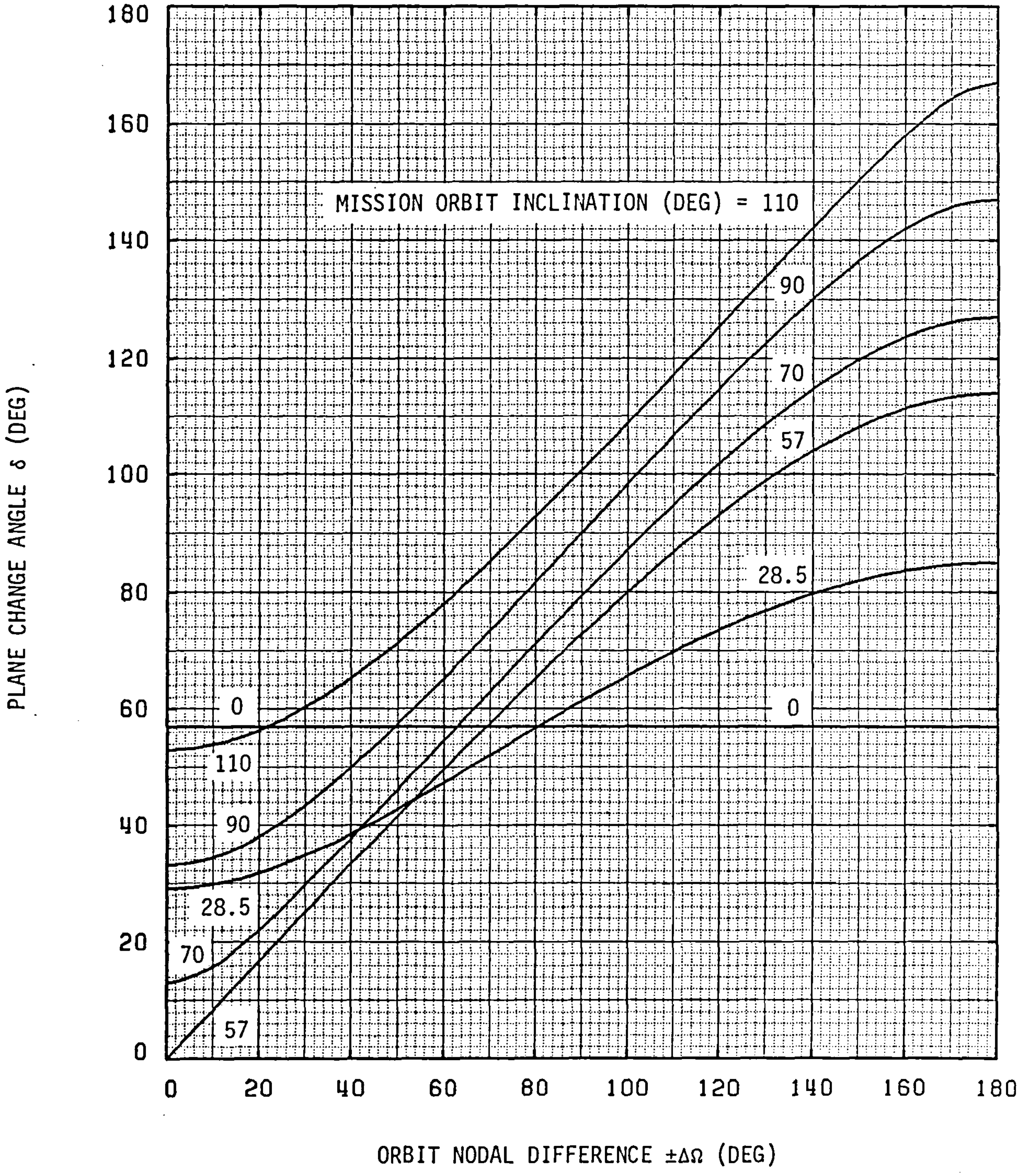


Figure 1-15.- Plane change requirements for station orbit at 57 degrees inclination.

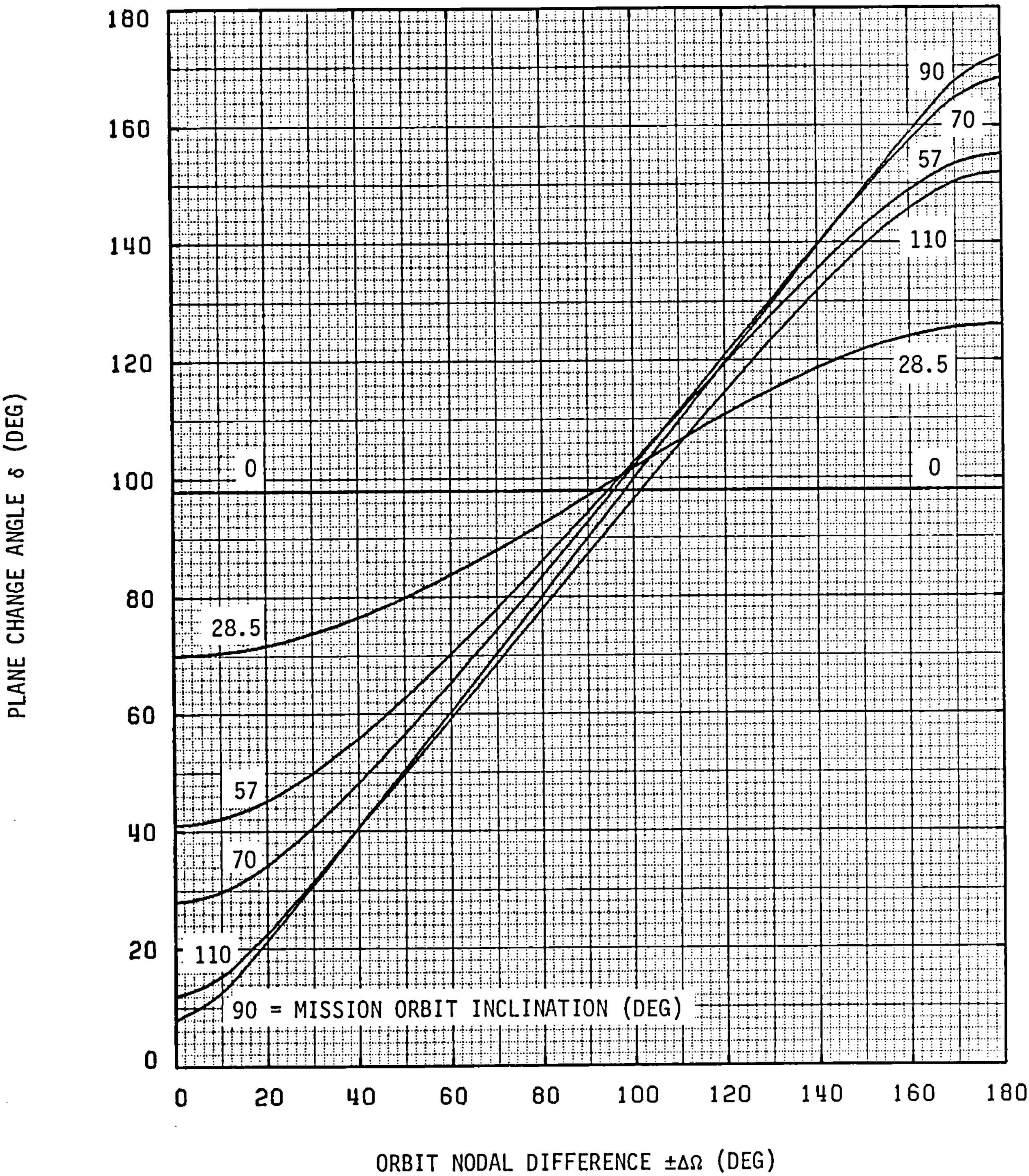
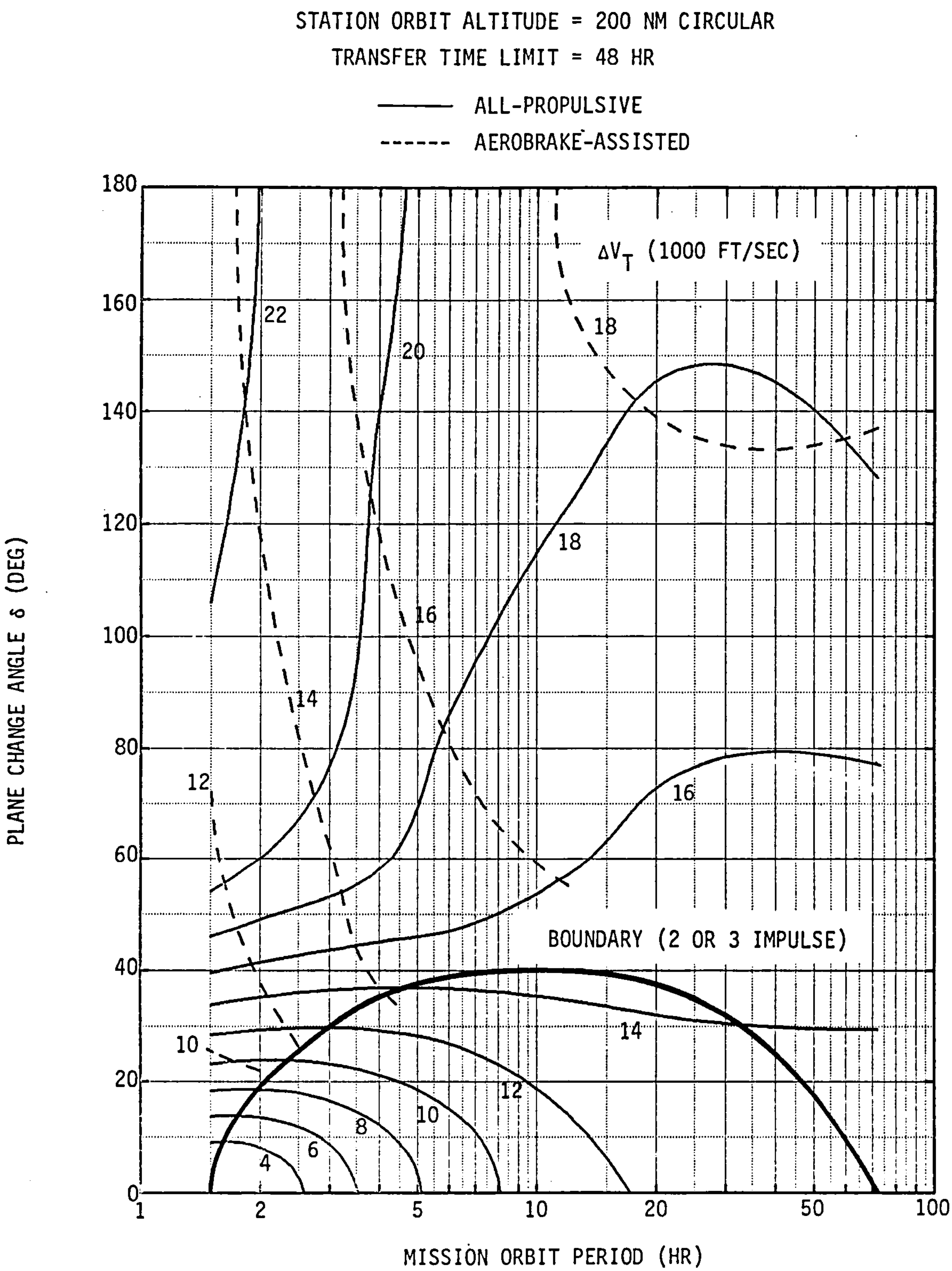
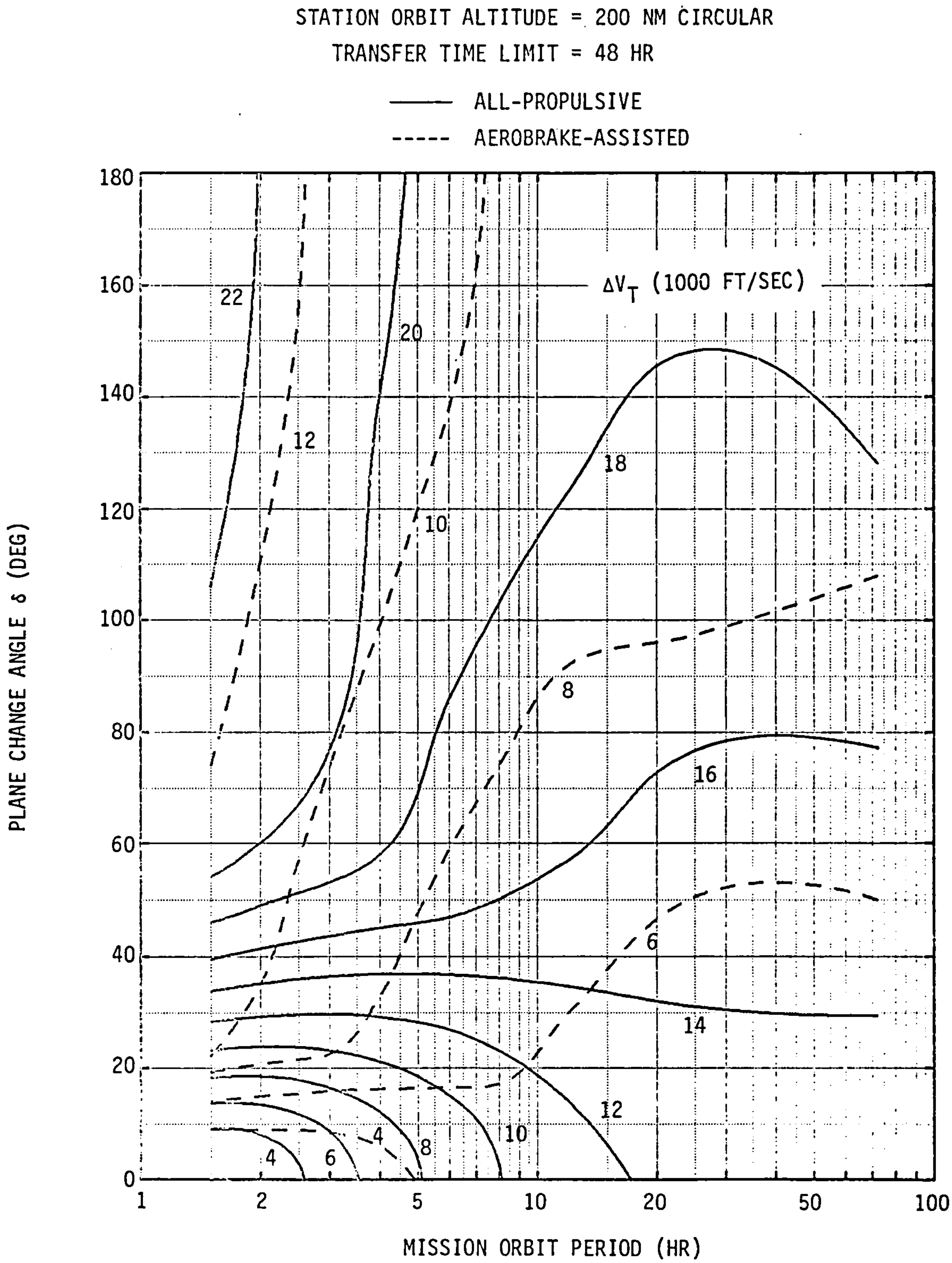


Figure 1-16.- Plane change requirements for station orbit at 98 degrees inclination.



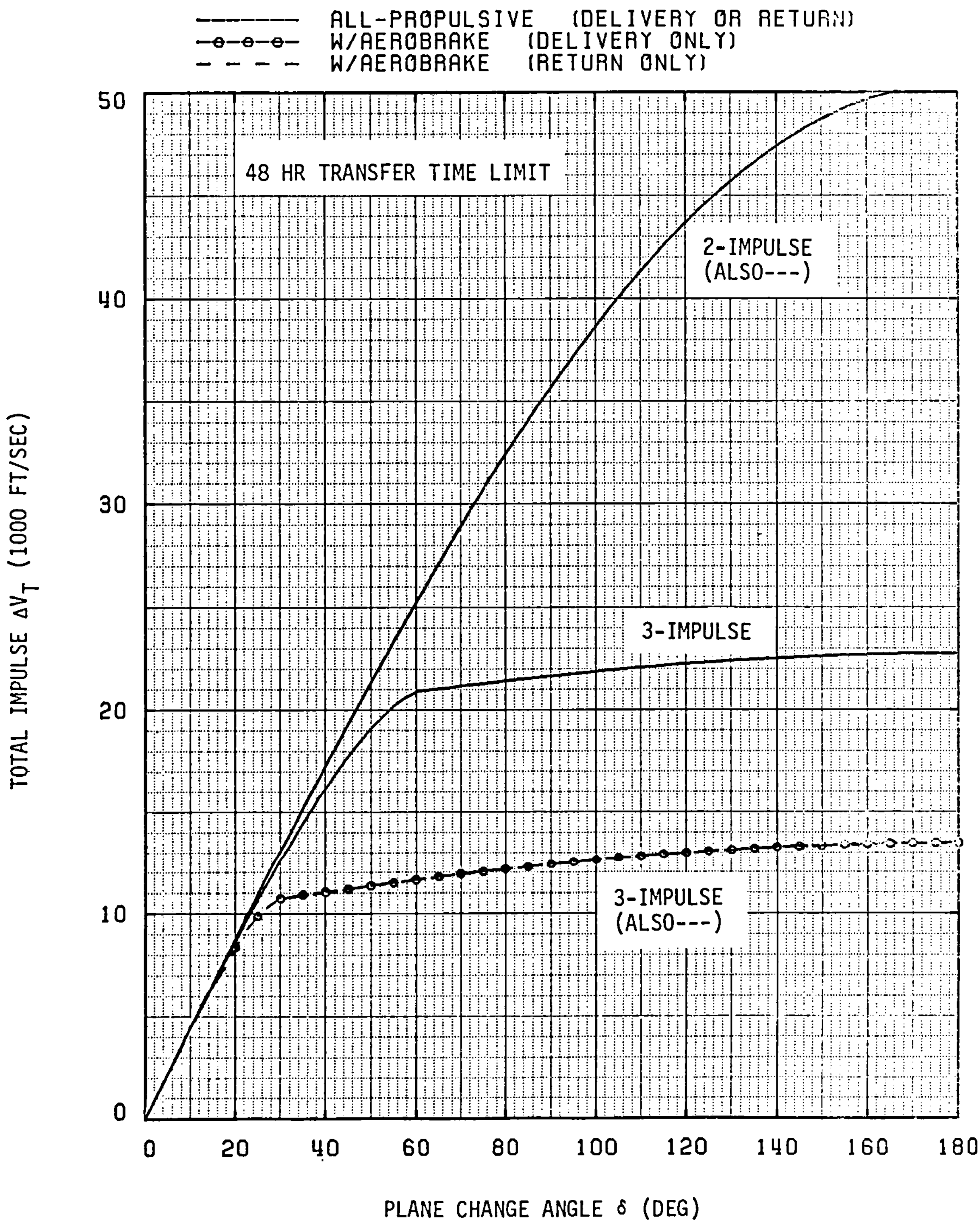
(a) Delivery phase.

Figure 1-17.- ΔV contours for circular orbit transfer.



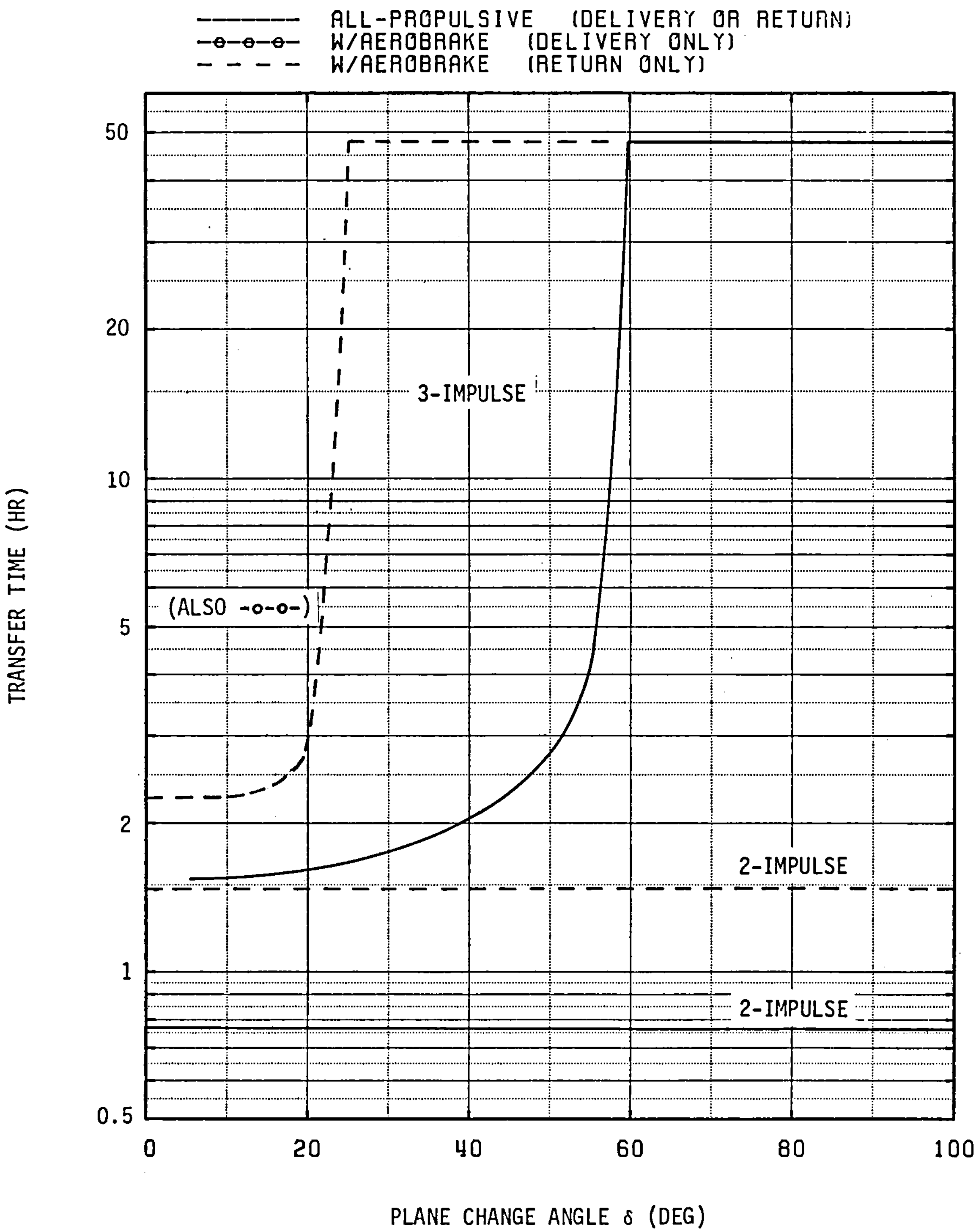
(b) Return phase.

Figure 1-17.- Concluded.



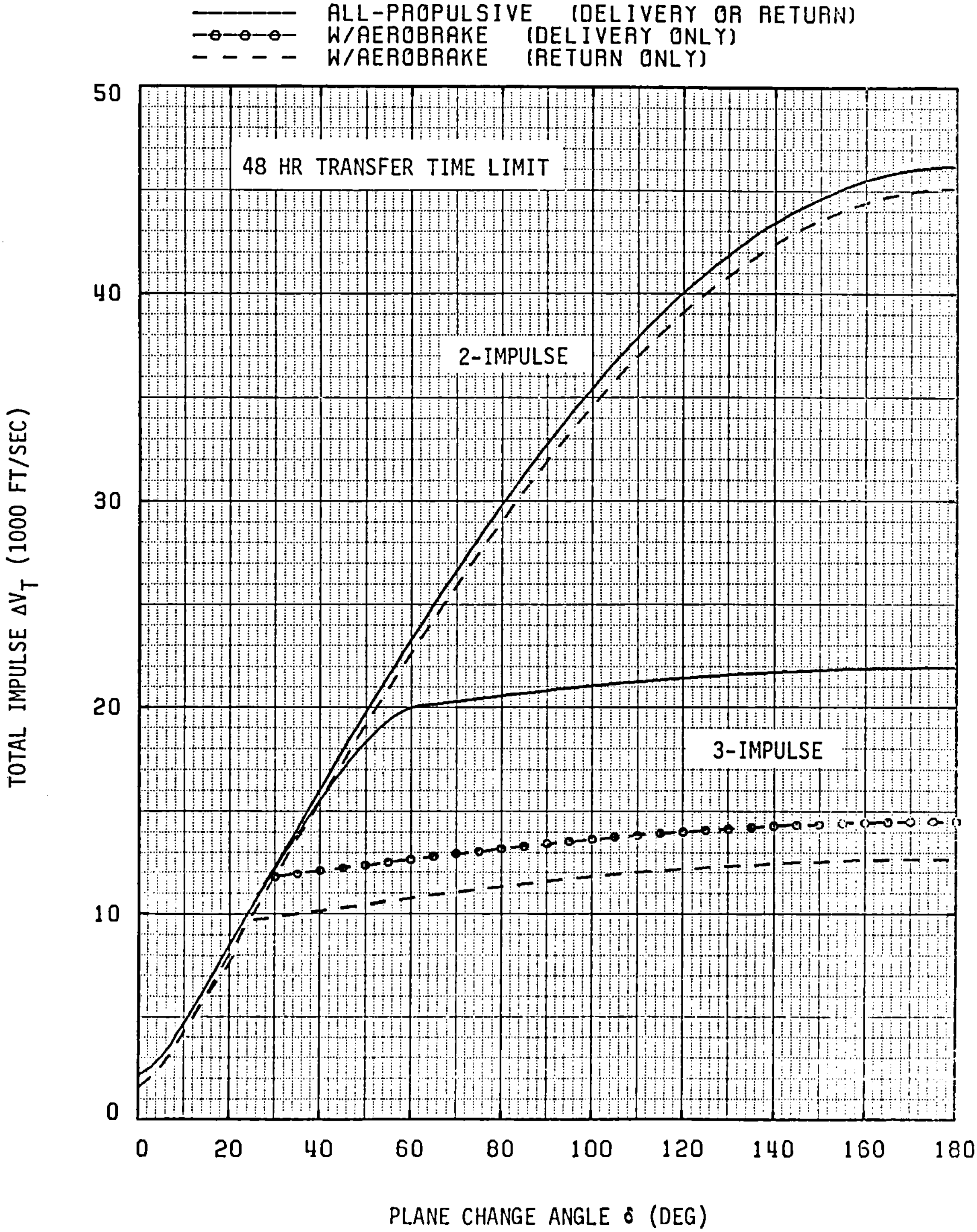
(a) Total impulse ΔV.

Figure 1-18.- Effect of maneuver strategy options for a 1.53-hour circular mission orbit.



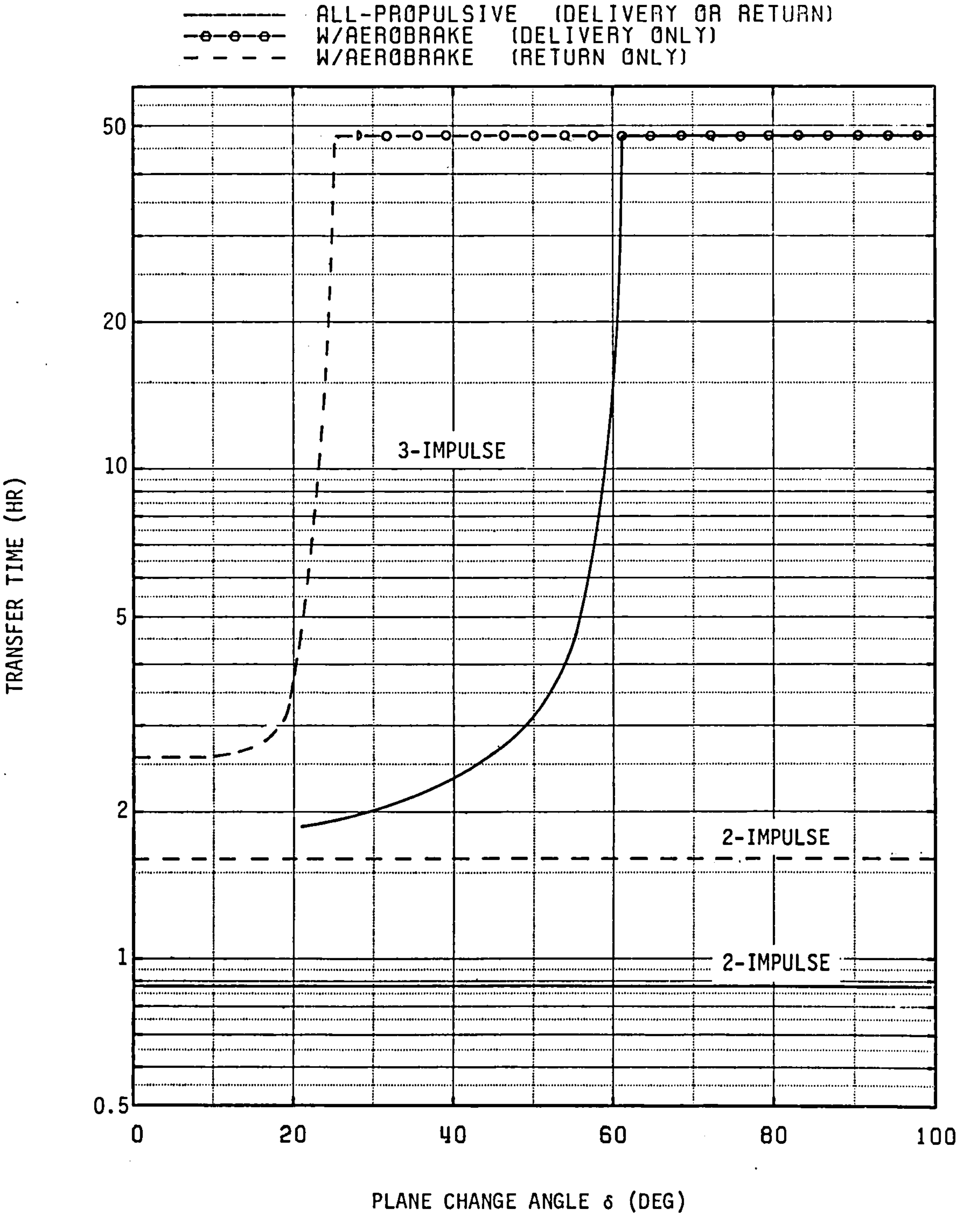
(b) Transfer time.

Figure 1-18.- Concluded.



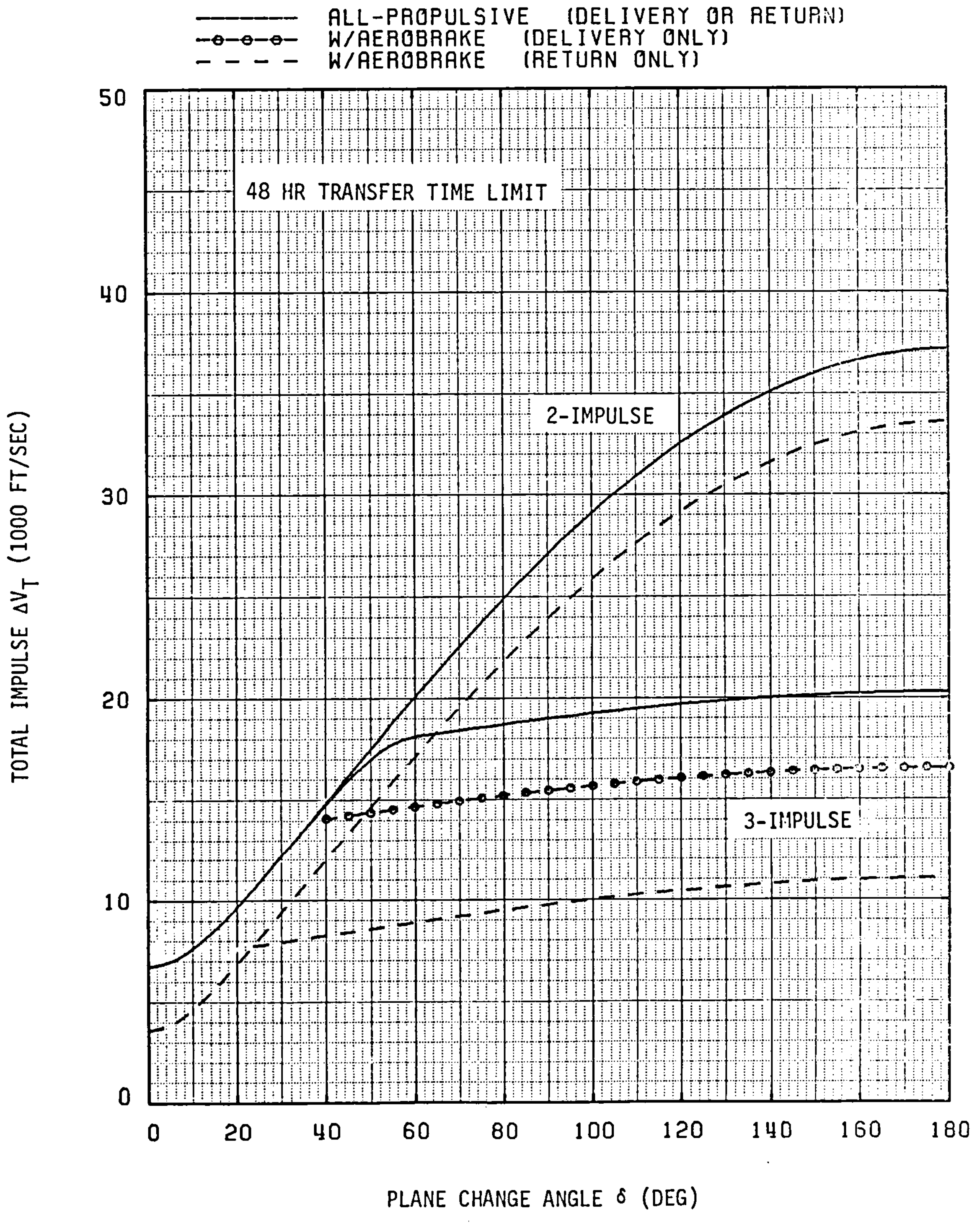
(a) Total impulse ΔV.

Figure 1-19.- Effect of maneuver strategy options for a 2-hour circular mission orbit.



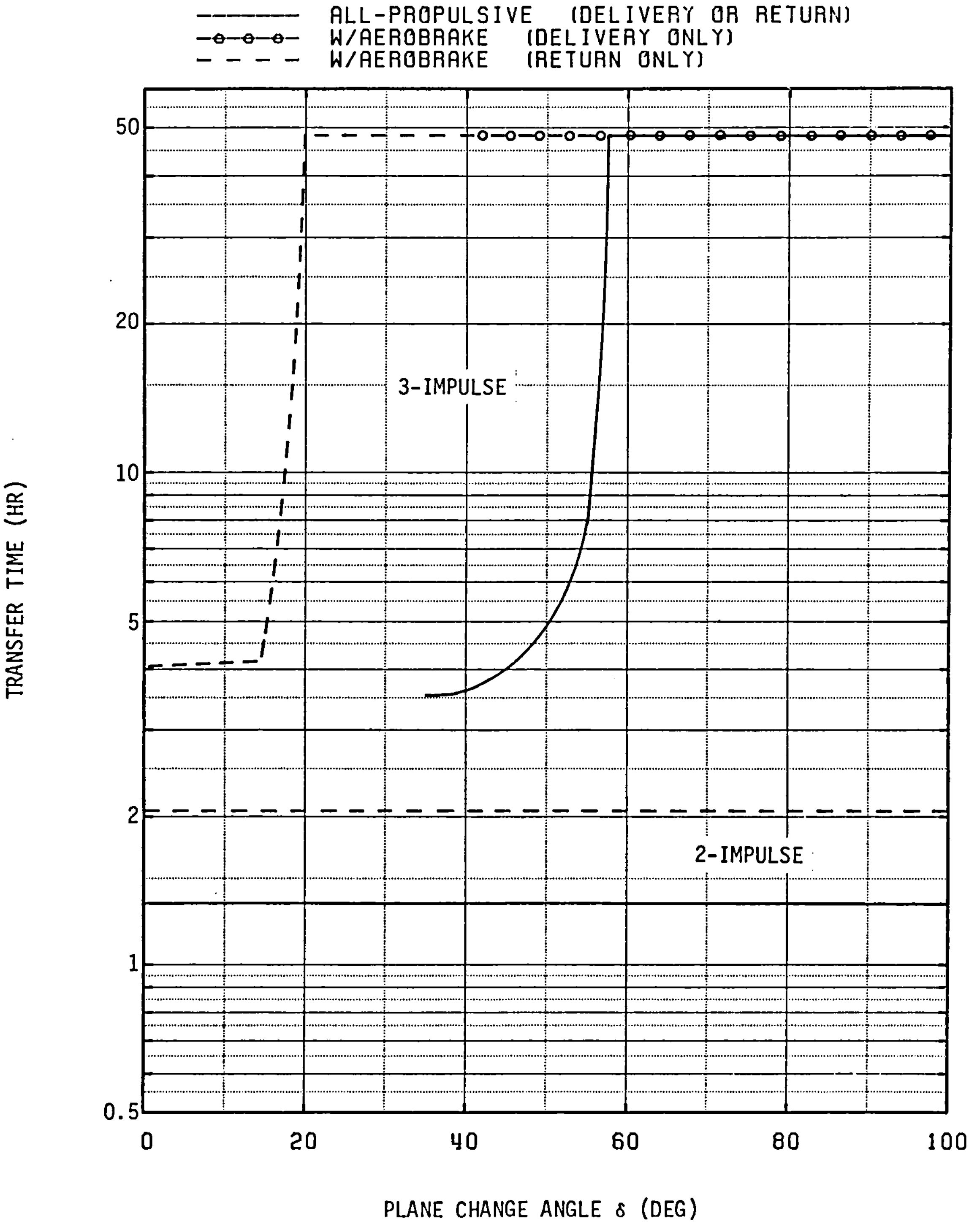
(b) Transfer time.

Figure 1-19.- Concluded.



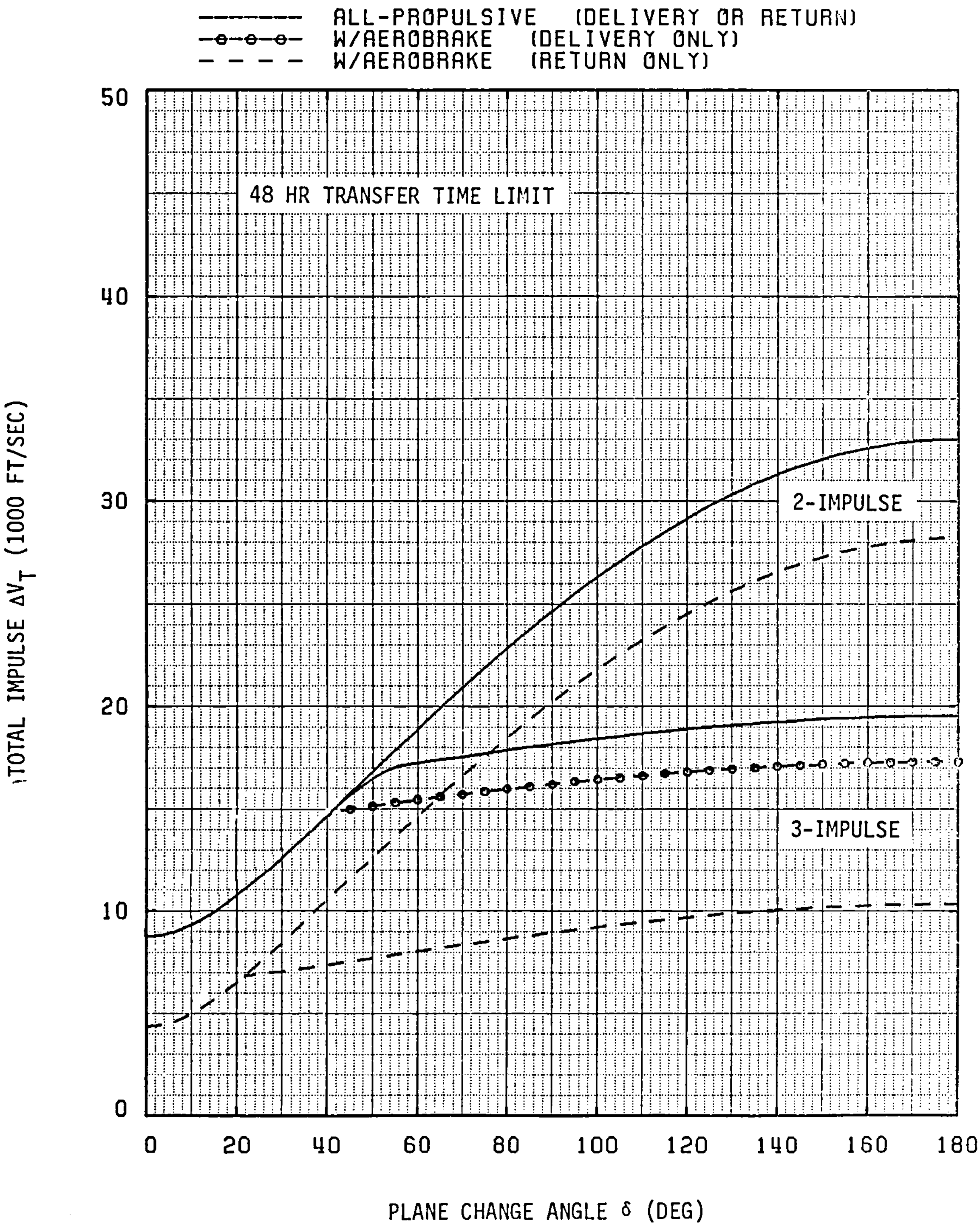
(a) Total impulse ΔV.

Figure 1-20.- Effect of maneuver strategy options for a 4-hour circular mission orbit.



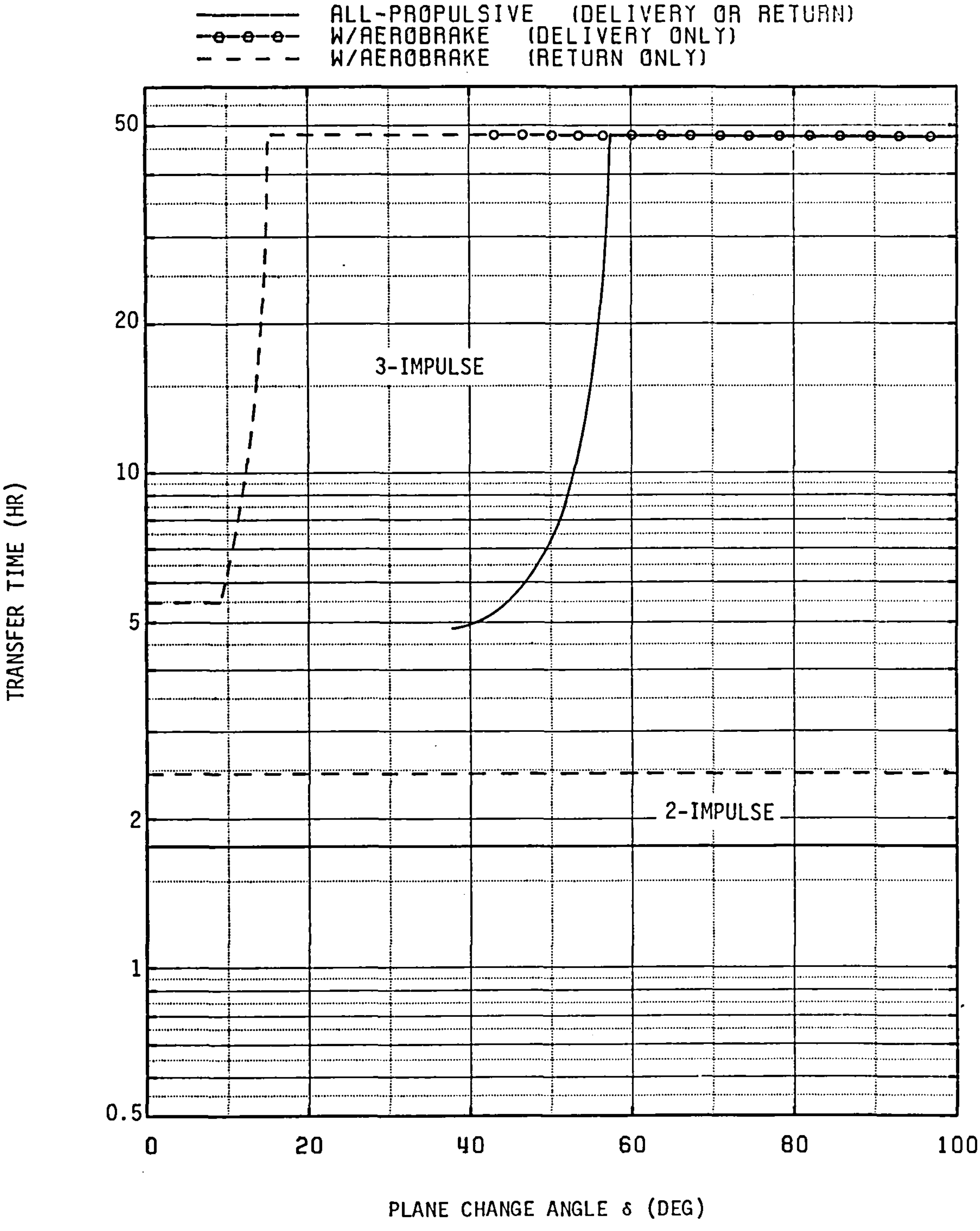
(b) Transfer time.

Figure 1-20.- Concluded.



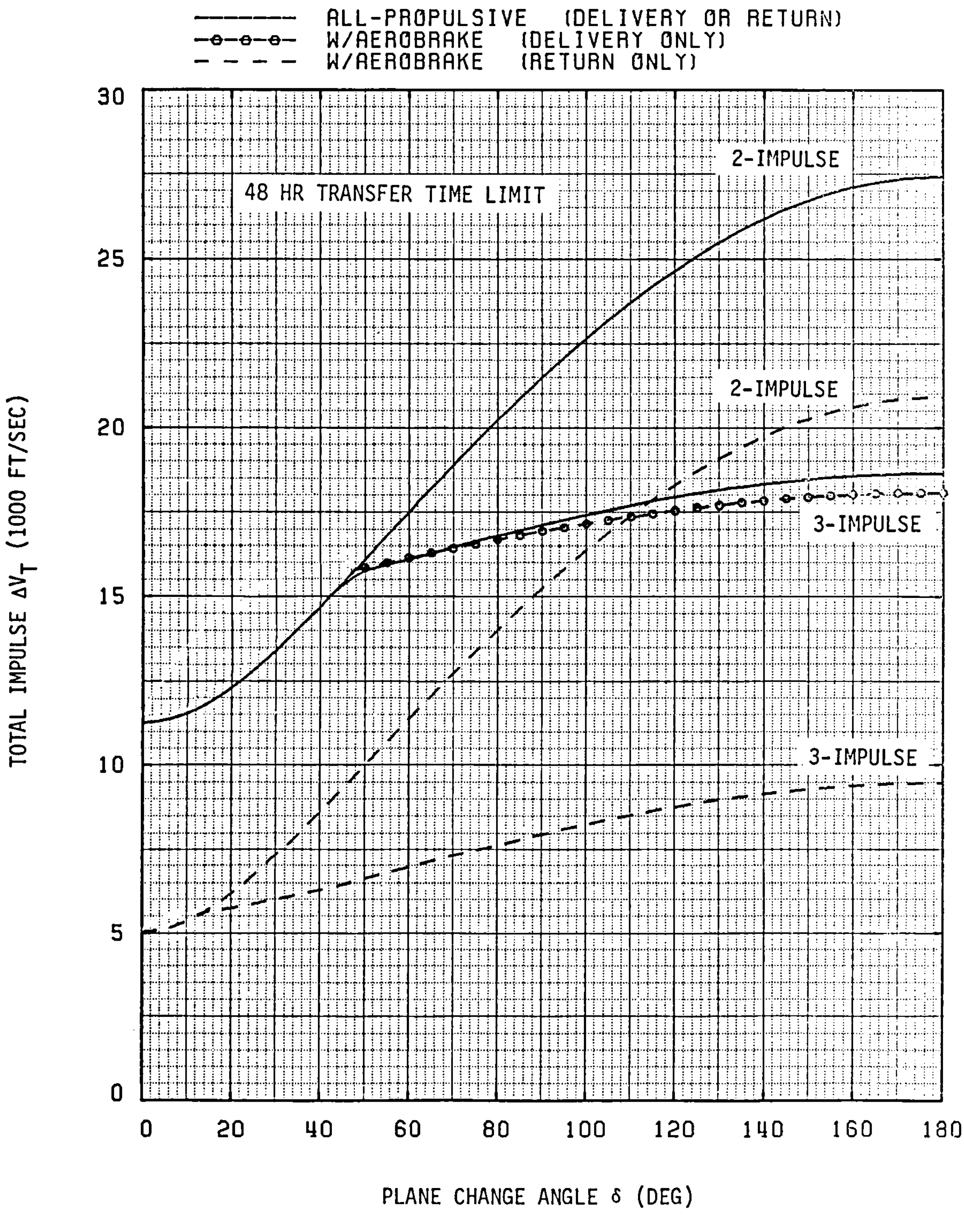
(a) Total impulse ΔV.

Figure 1-21.- Effect of maneuver strategy options for a 6-hour circular mission orbit.



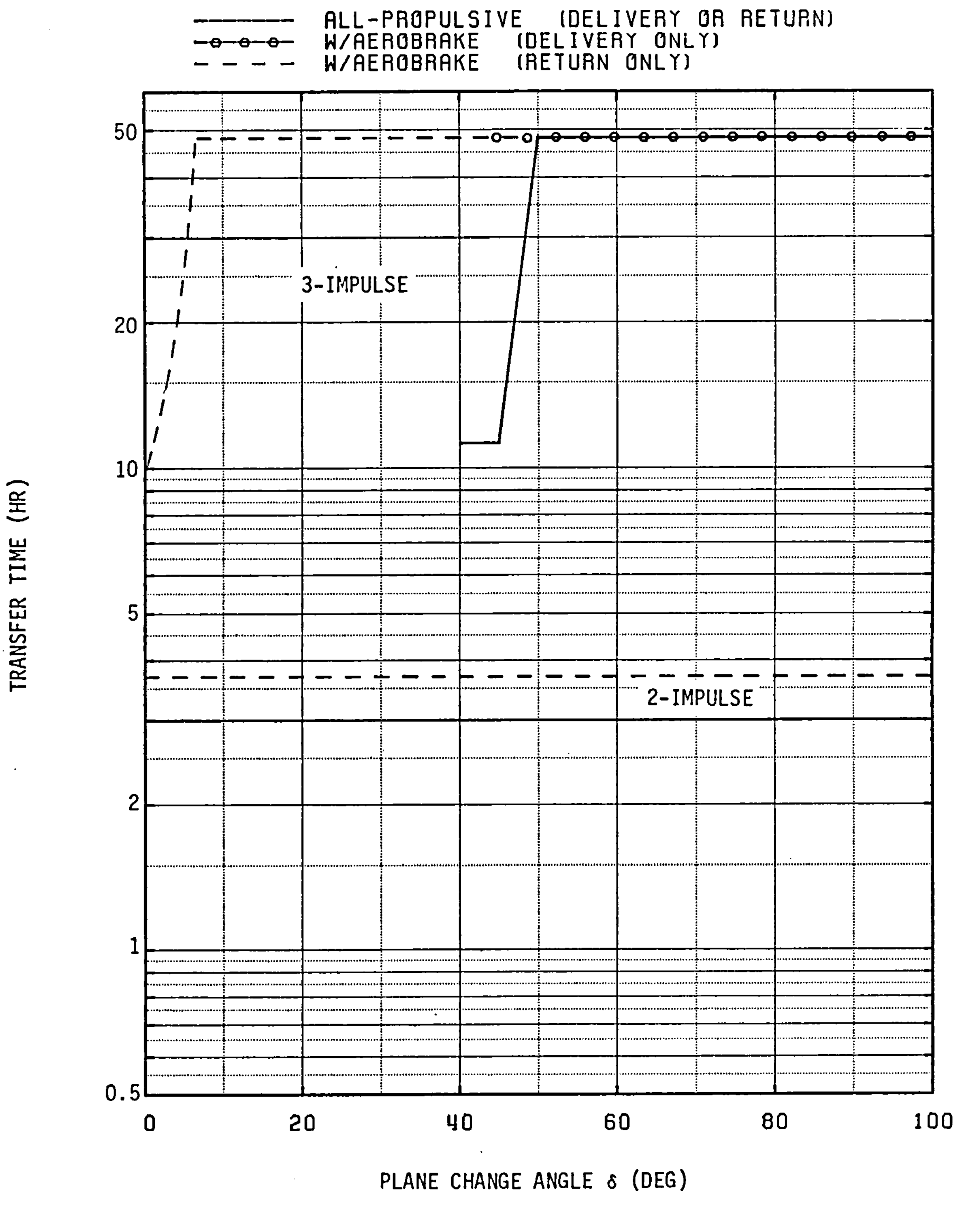
(b) Transfer time.

Figure 1-21.- Concluded.



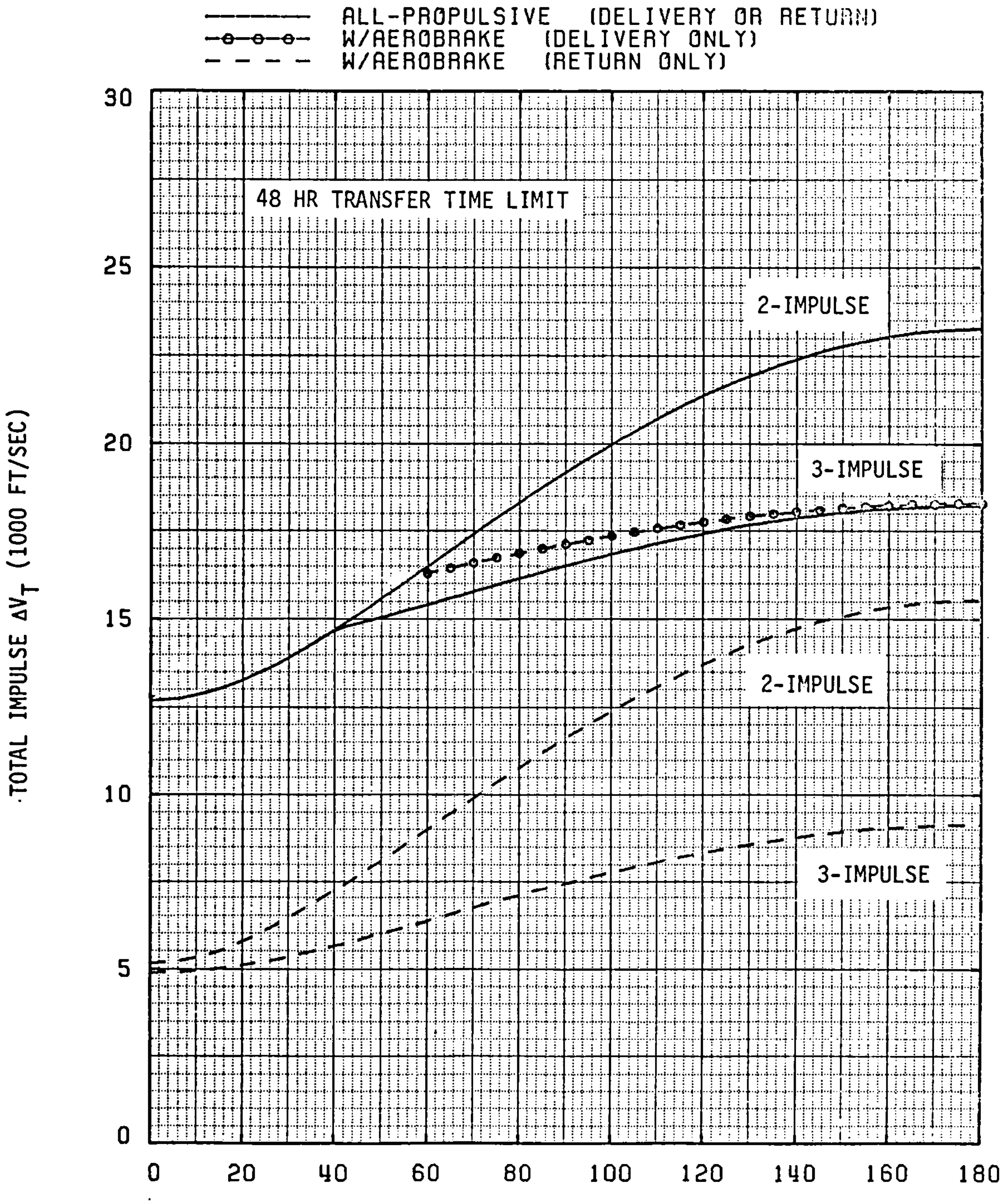
(a) Total impulse ΔV.

Figure 1-22.- Effect of maneuver strategy options for a 12-hour circular mission orbit.



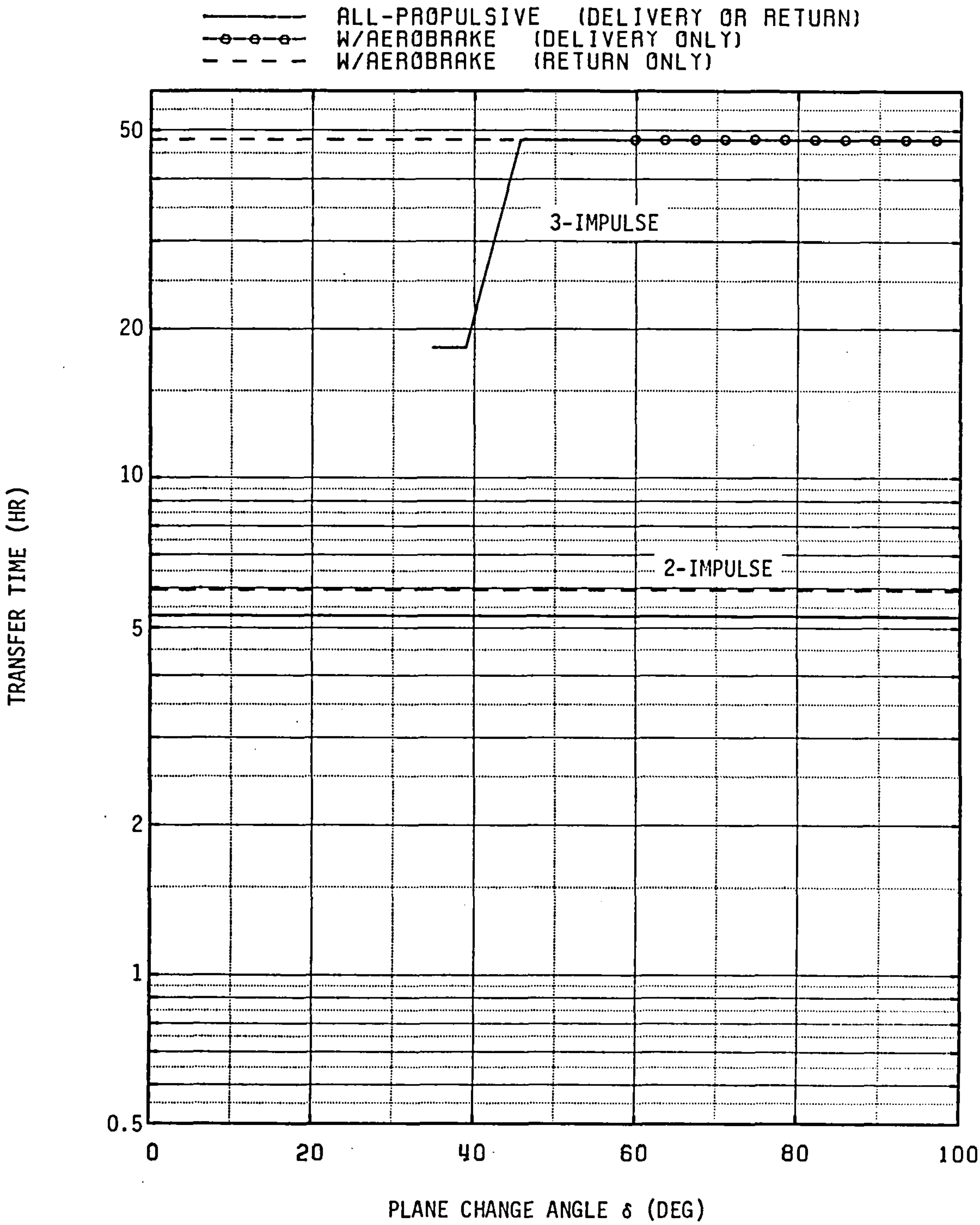
(b) Transfer time.

Figure 1-22.- Concluded.



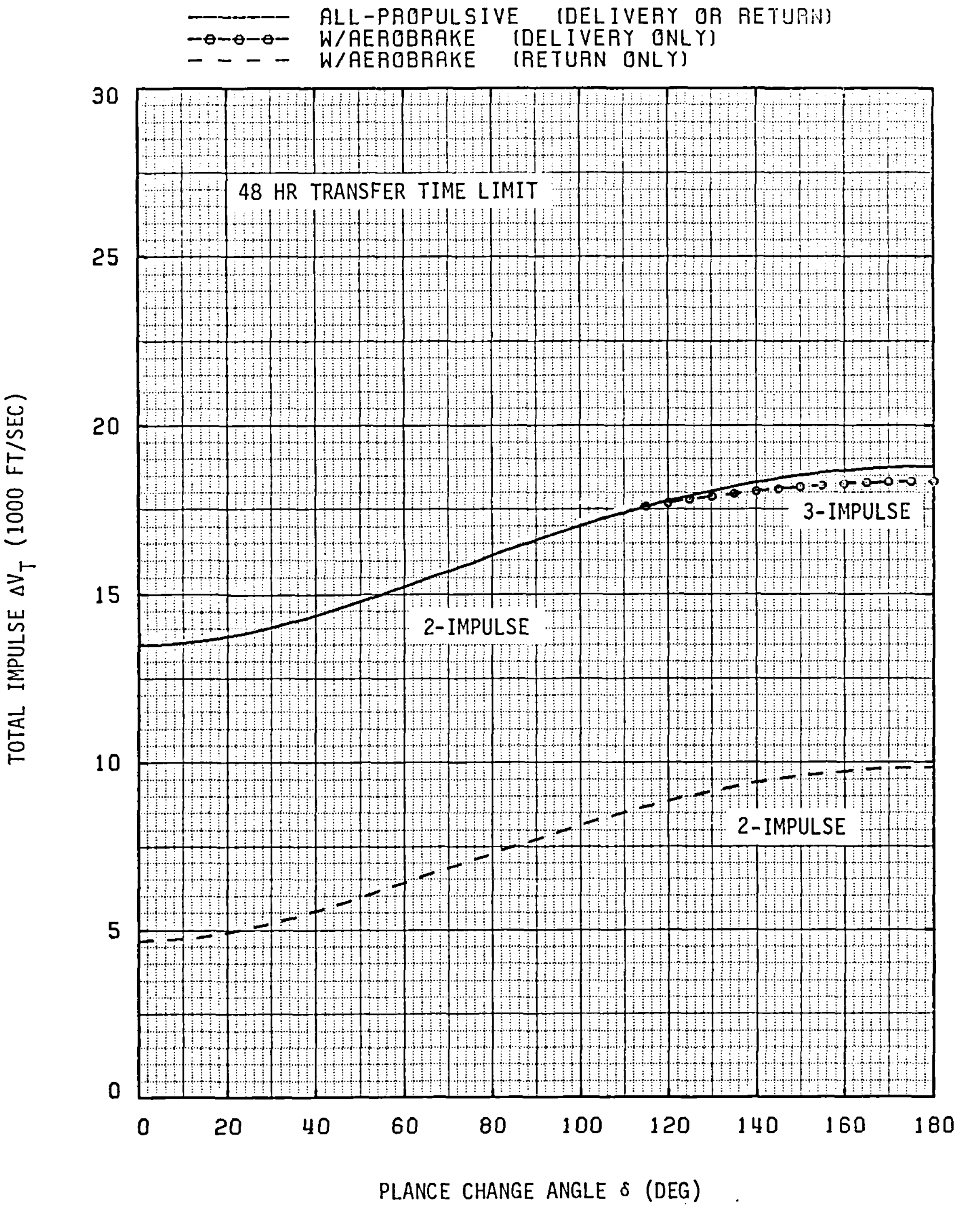
(a) Total impulse ΔV.

Figure 1-23.- Effect of maneuver strategy options for a 24-hour circular mission orbit.



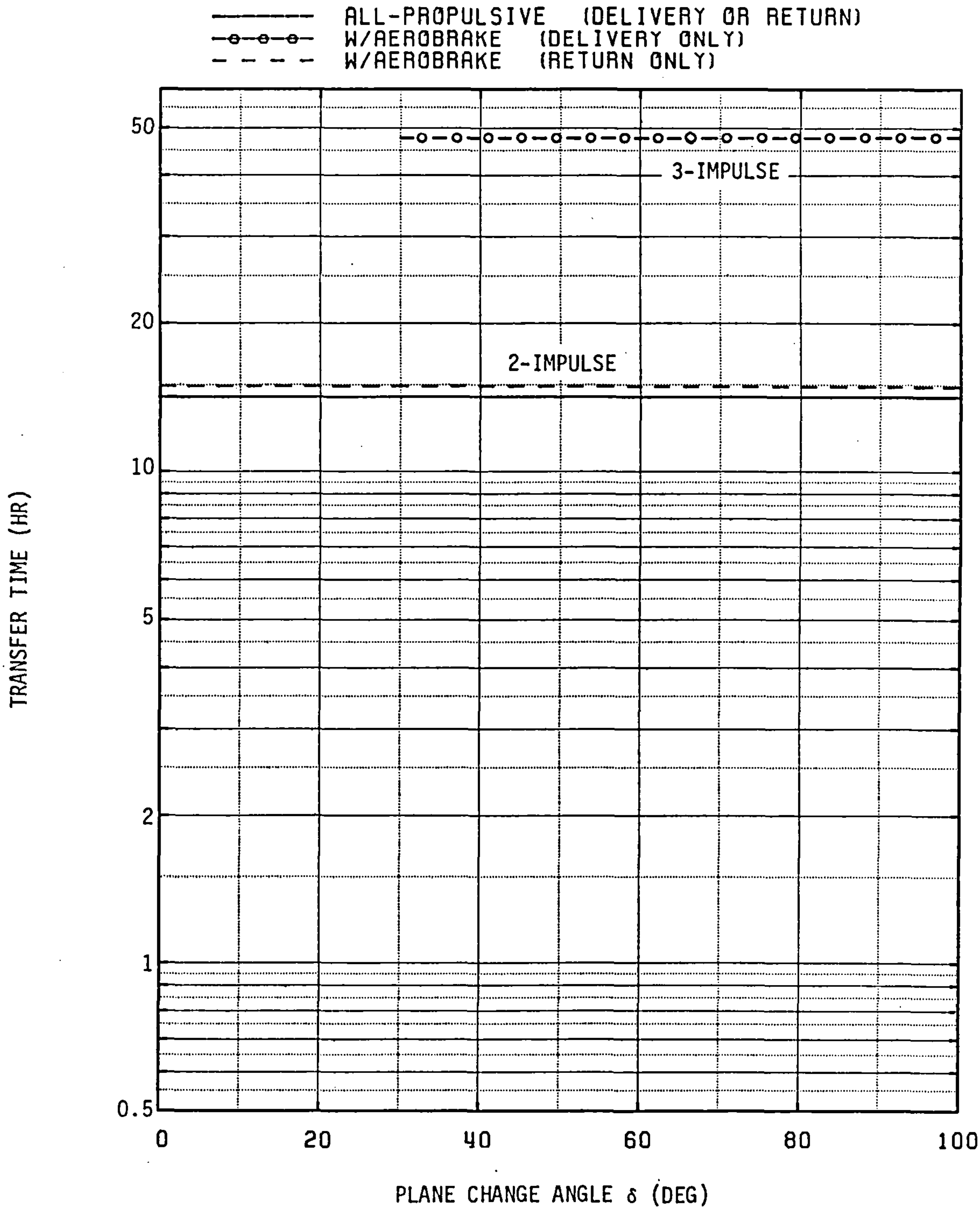
(b) Transfer time.

Figure 1-23.- Concluded.



(a) Total impulse ΔV.

Figure 1-24.- Effect of maneuver strategy options for a 72-hour circular mission orbit.



(b) Transfer time.

Figure 1-24.- Concluded.

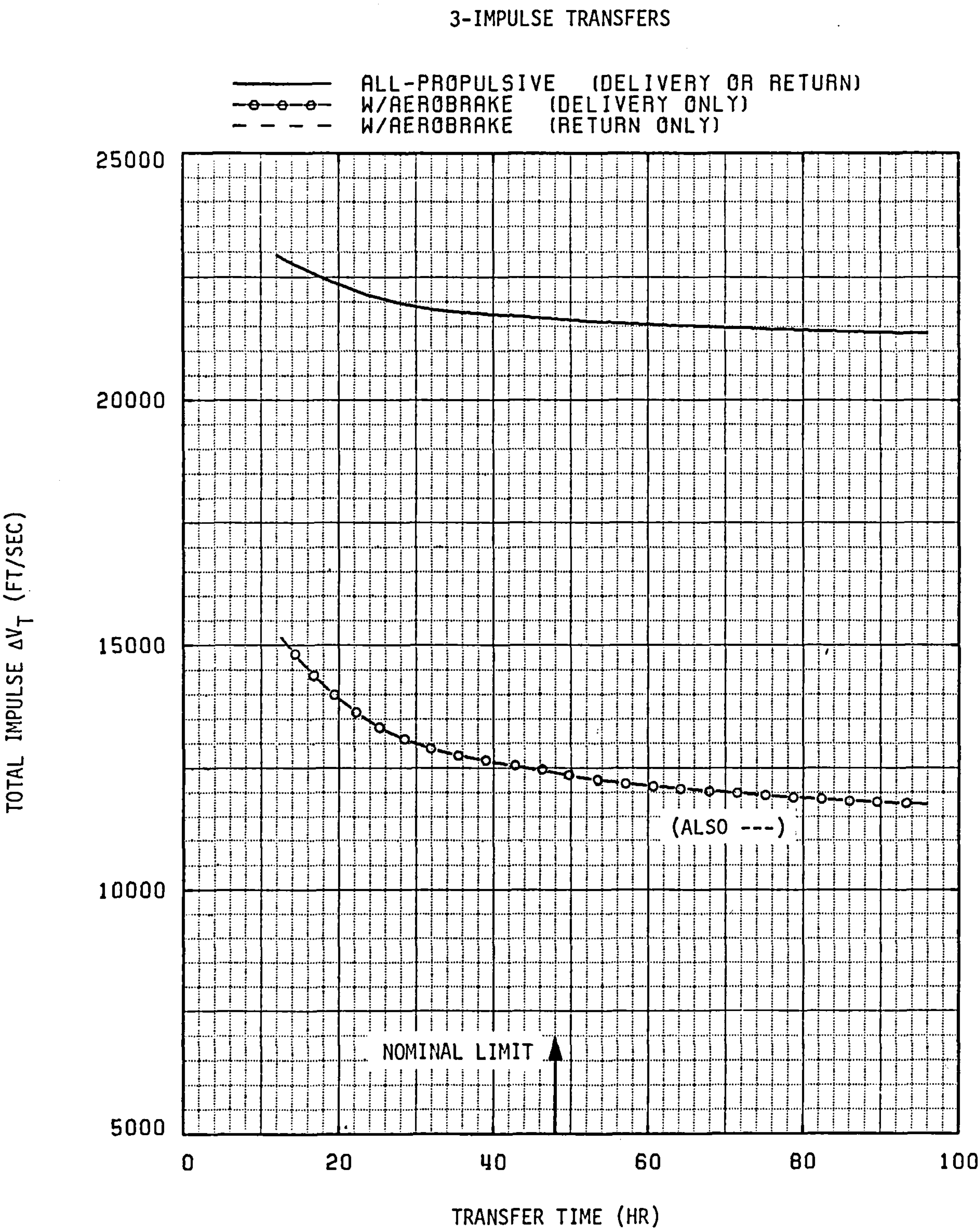


Figure 1-25.- Sensitivity to transfer time for 1.53-hour circular mission orbit with $\delta = 90^{\circ}$.

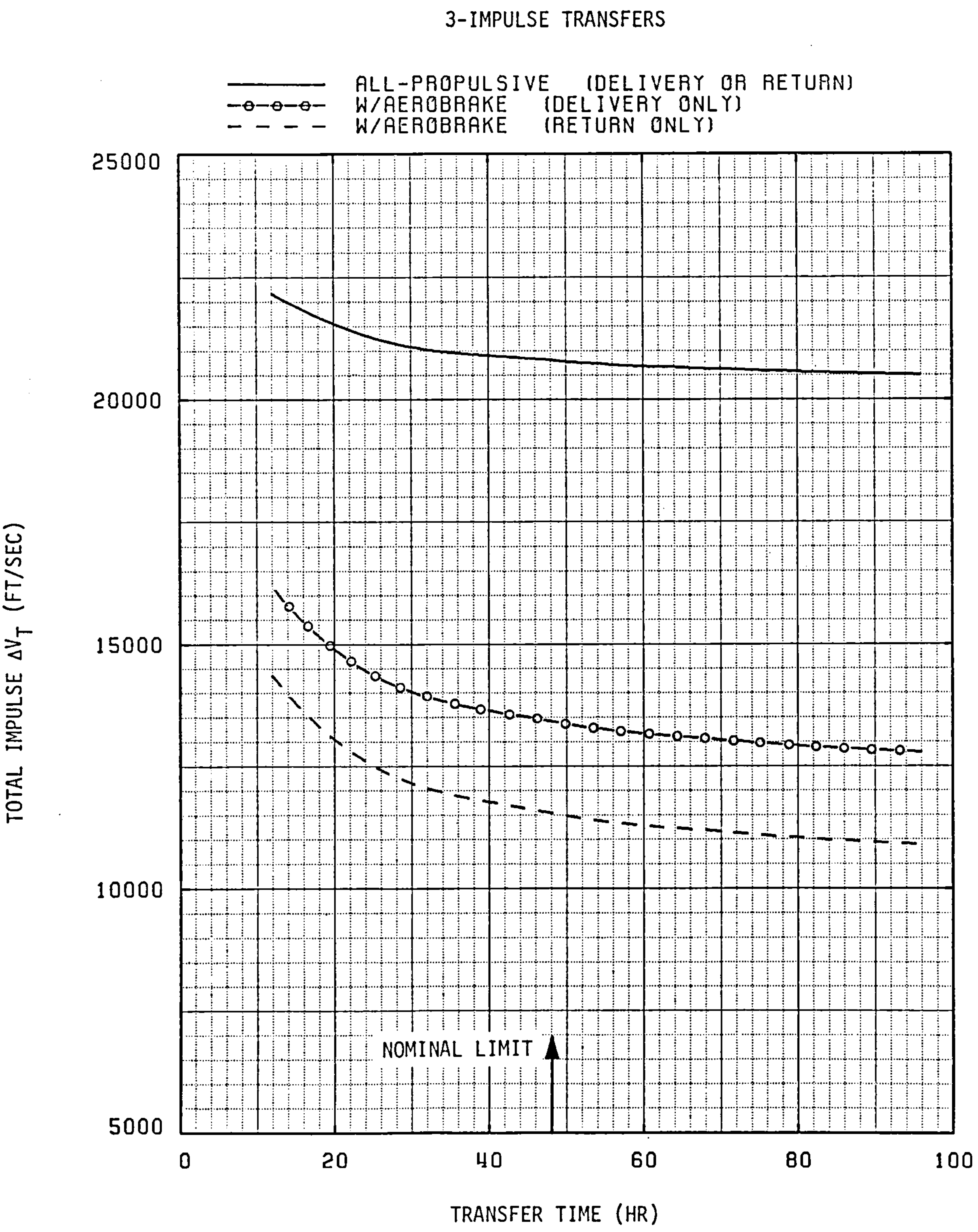


Figure 1-26.- Sensitivity to transfer time for 2-hour circular mission orbit with $\delta = 90^{\circ}$.

3-IMPULSE TRANSFERS

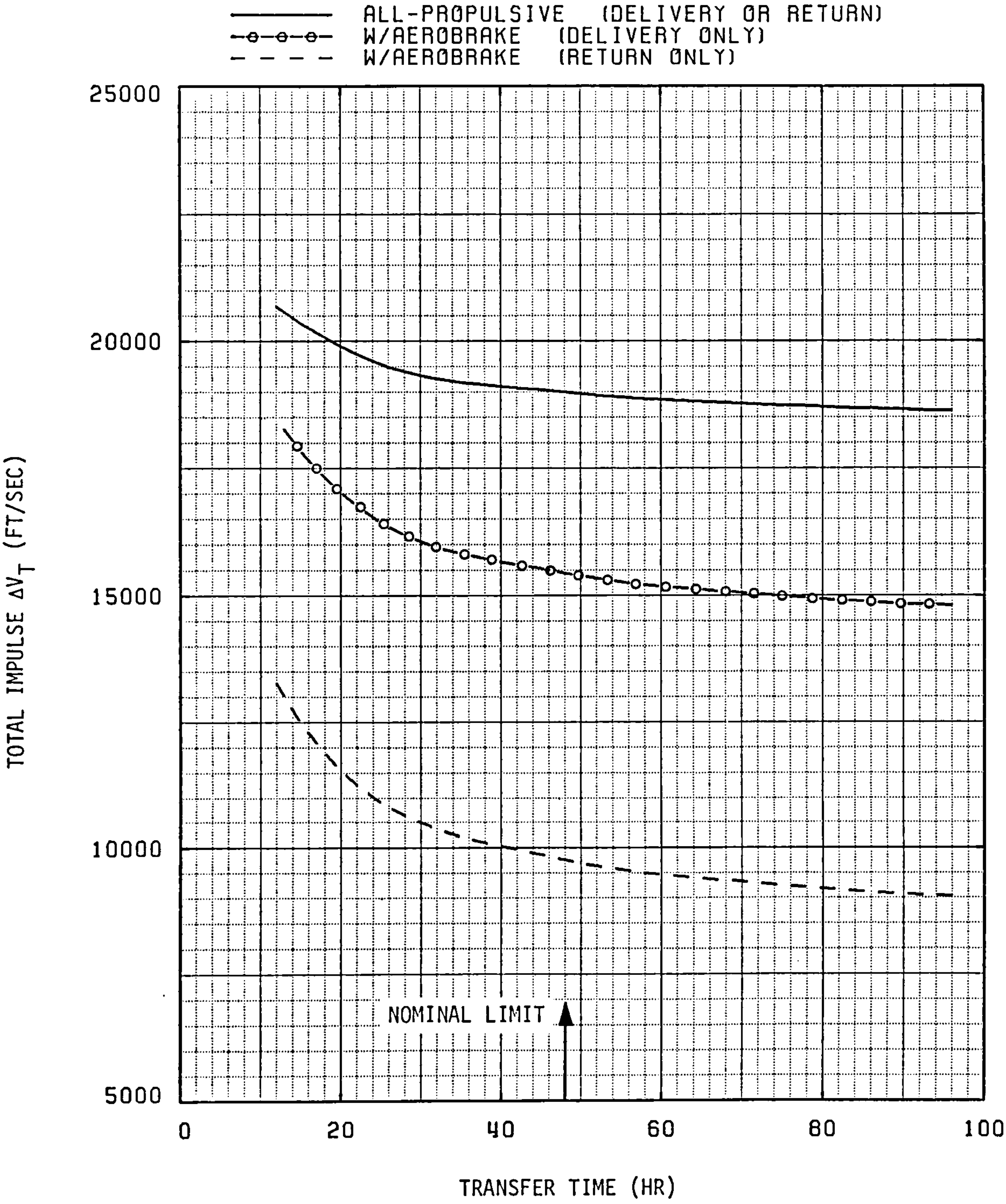


Figure 1-27.- Sensitivity to transfer time for 4-hour circular mission orbit with $\delta = 90^\circ$.

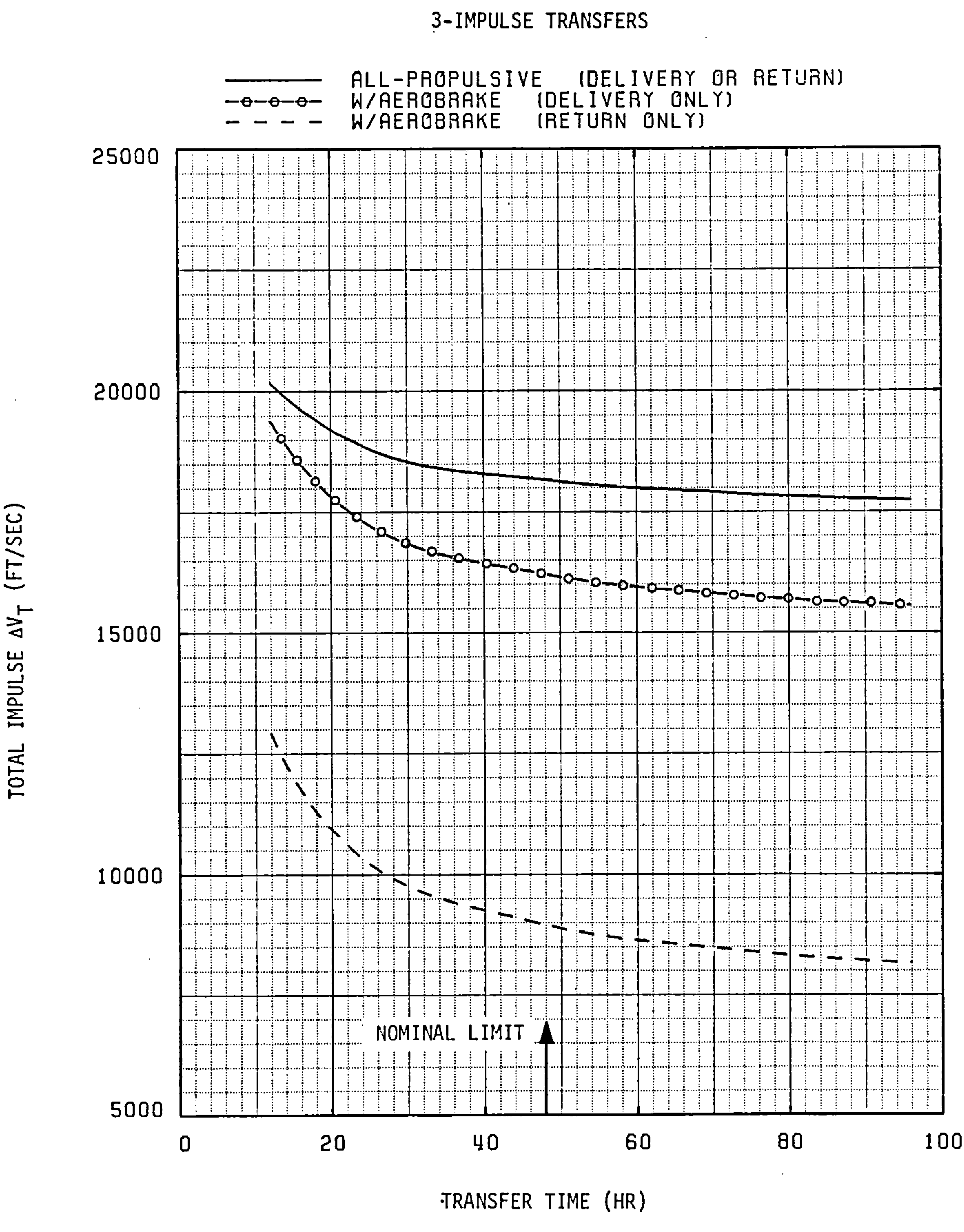


Figure 1-28.- Sensitivity to transfer time for 6-hour circular mission orbit with $\delta = 90^{\circ}$.

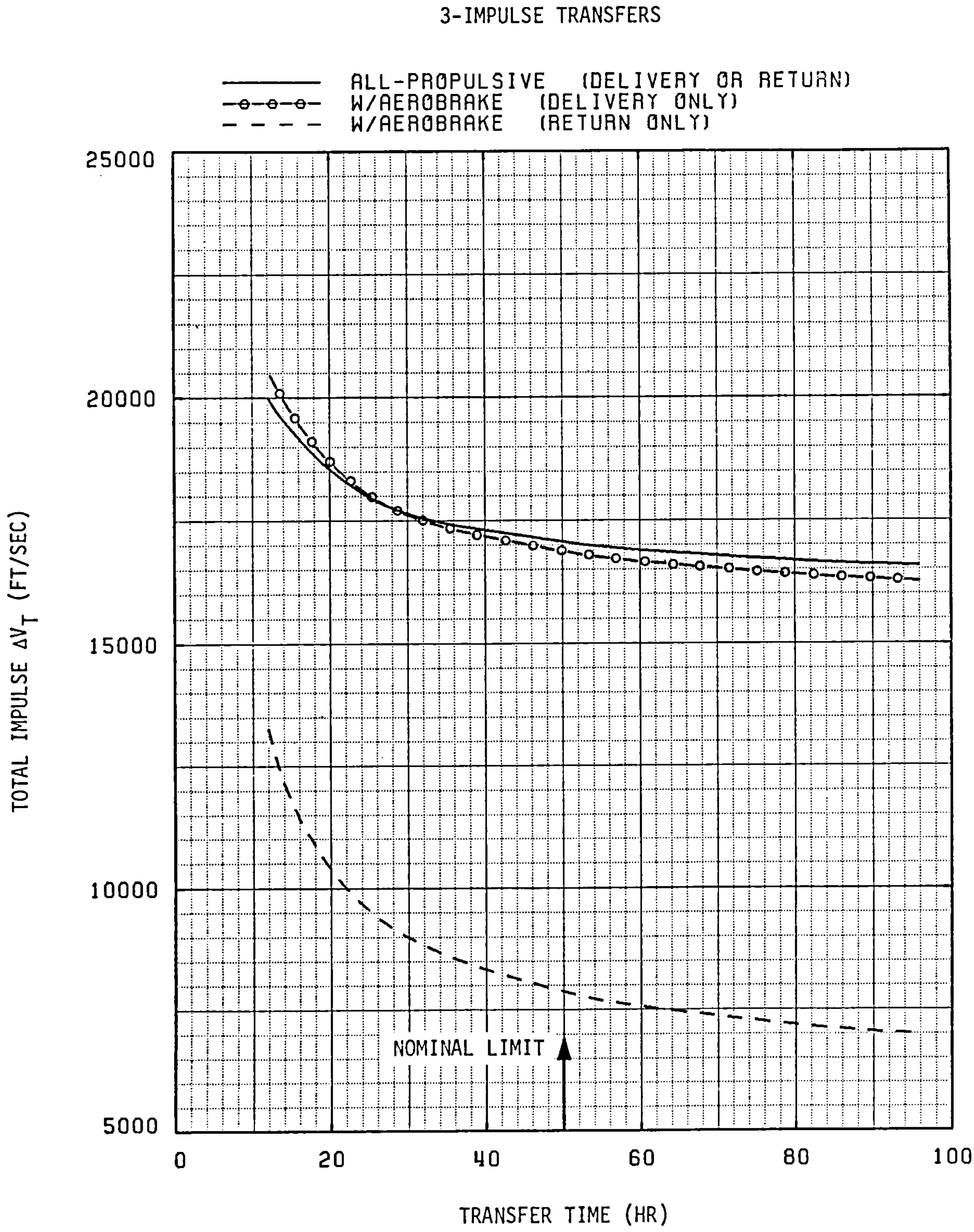


Figure 1-29.- Sensitivity to transfer time for 12-hour circular mission orbit with $\delta = 90^{\circ}$.

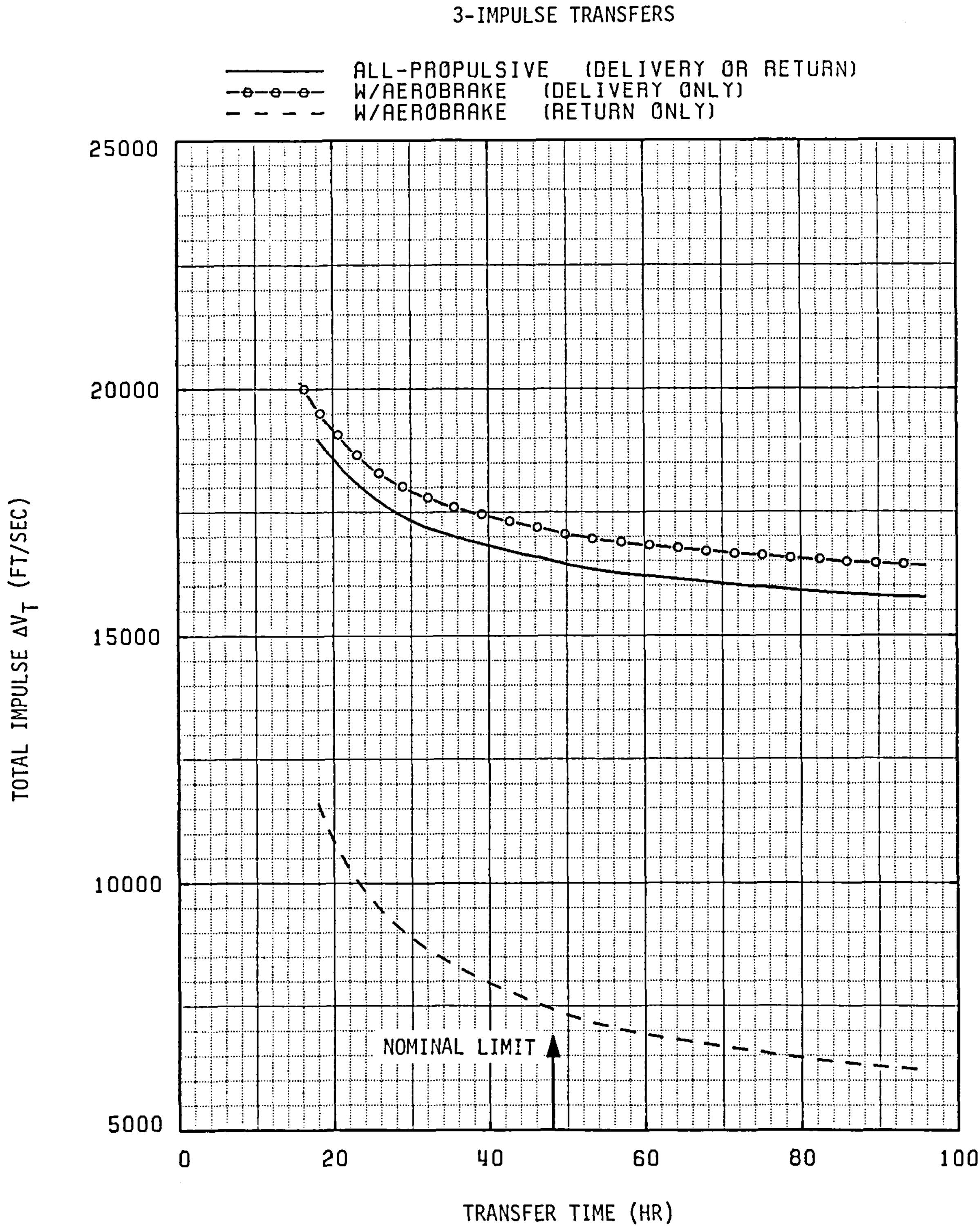


Figure 1-30.- Sensitivity to transfer time for 24-hour circular mission orbit with $\delta = 90^{o}$.

3-IMPULSE TRANSFERS

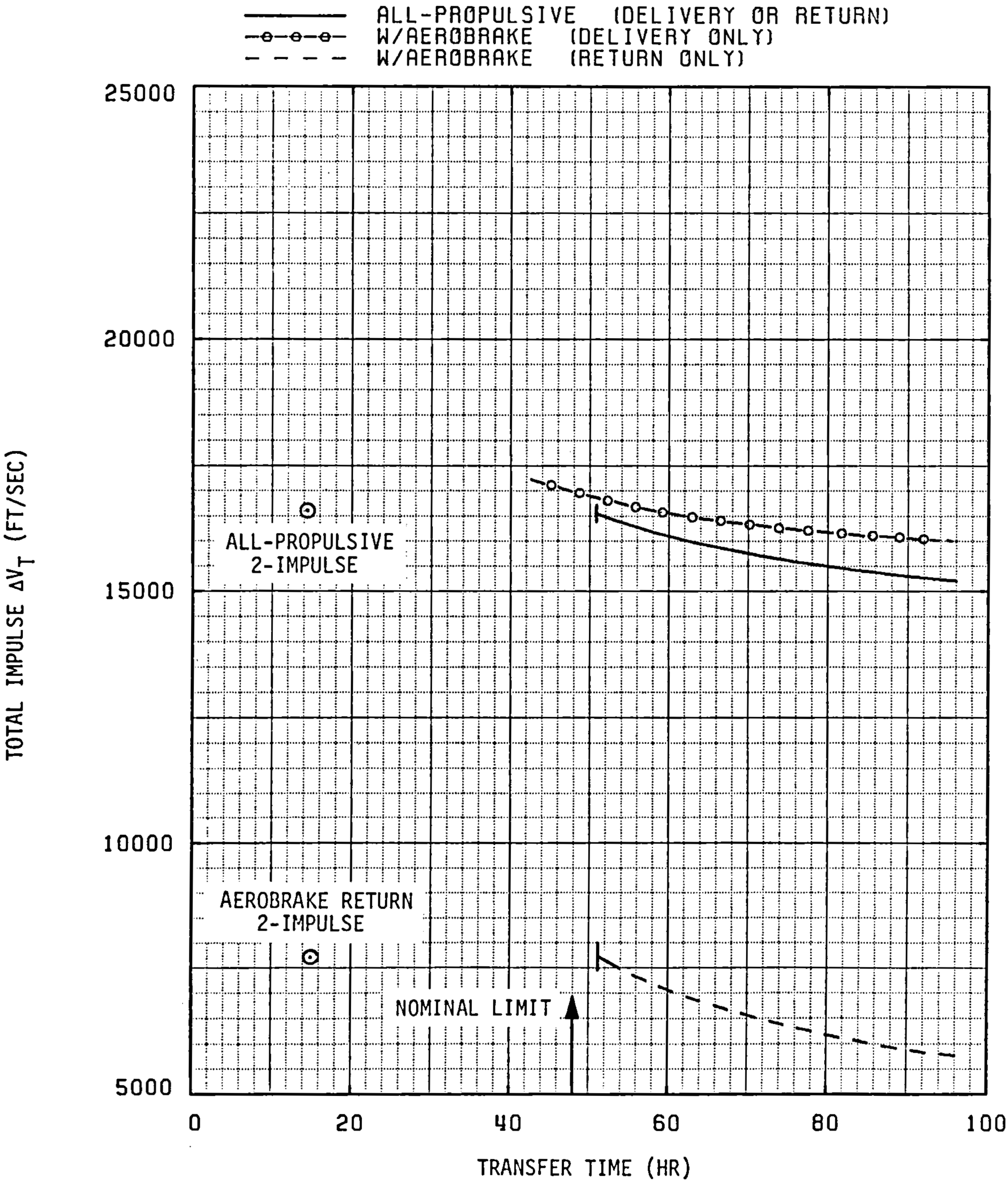


Figure 1-31.- Sensitivity to transfer time for 72-hour circular mission orbit with $\delta = 90^{\circ}$.

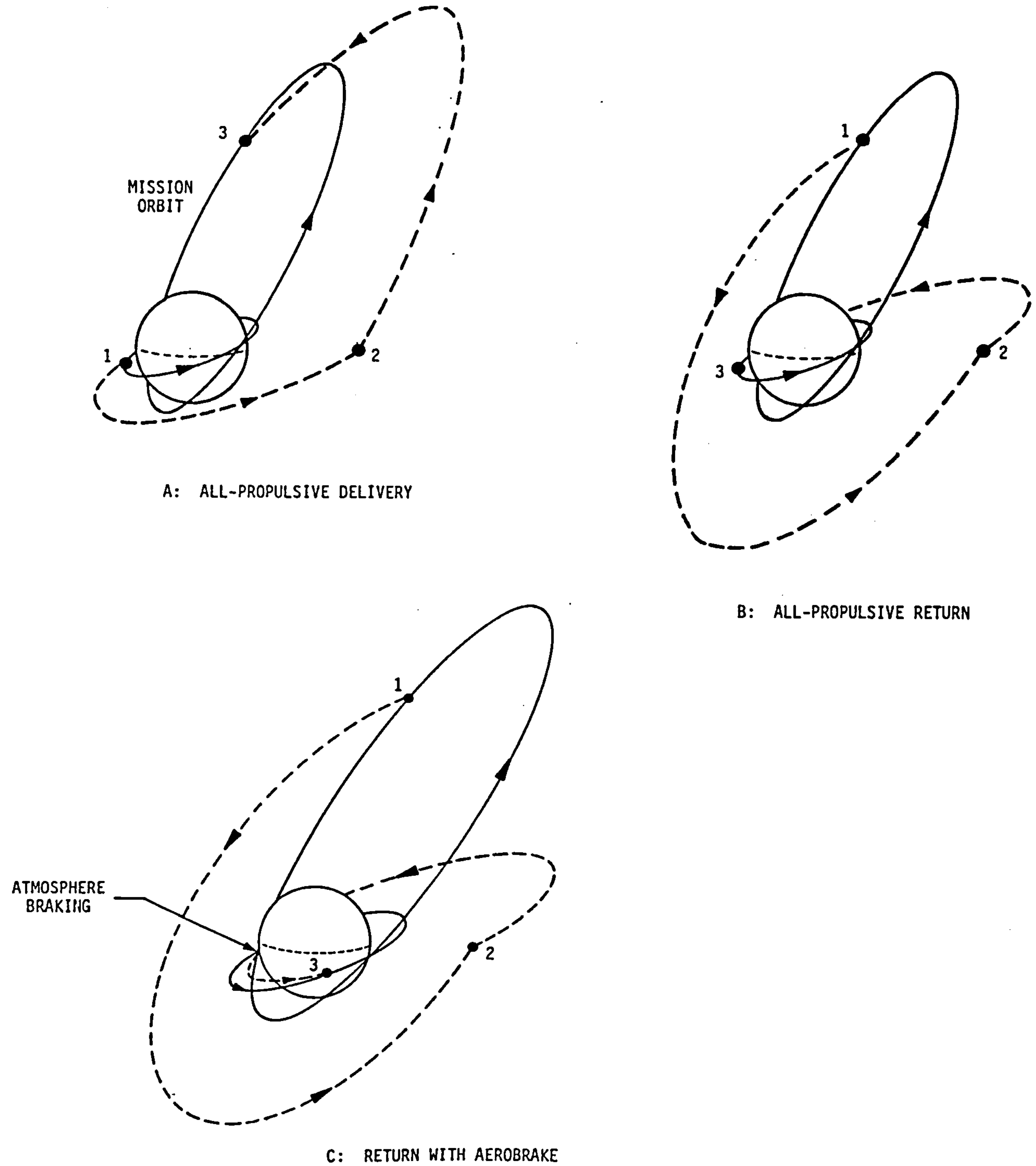


Figure 1-32.- Three-impulse transfer strategies for elliptical mission orbits.

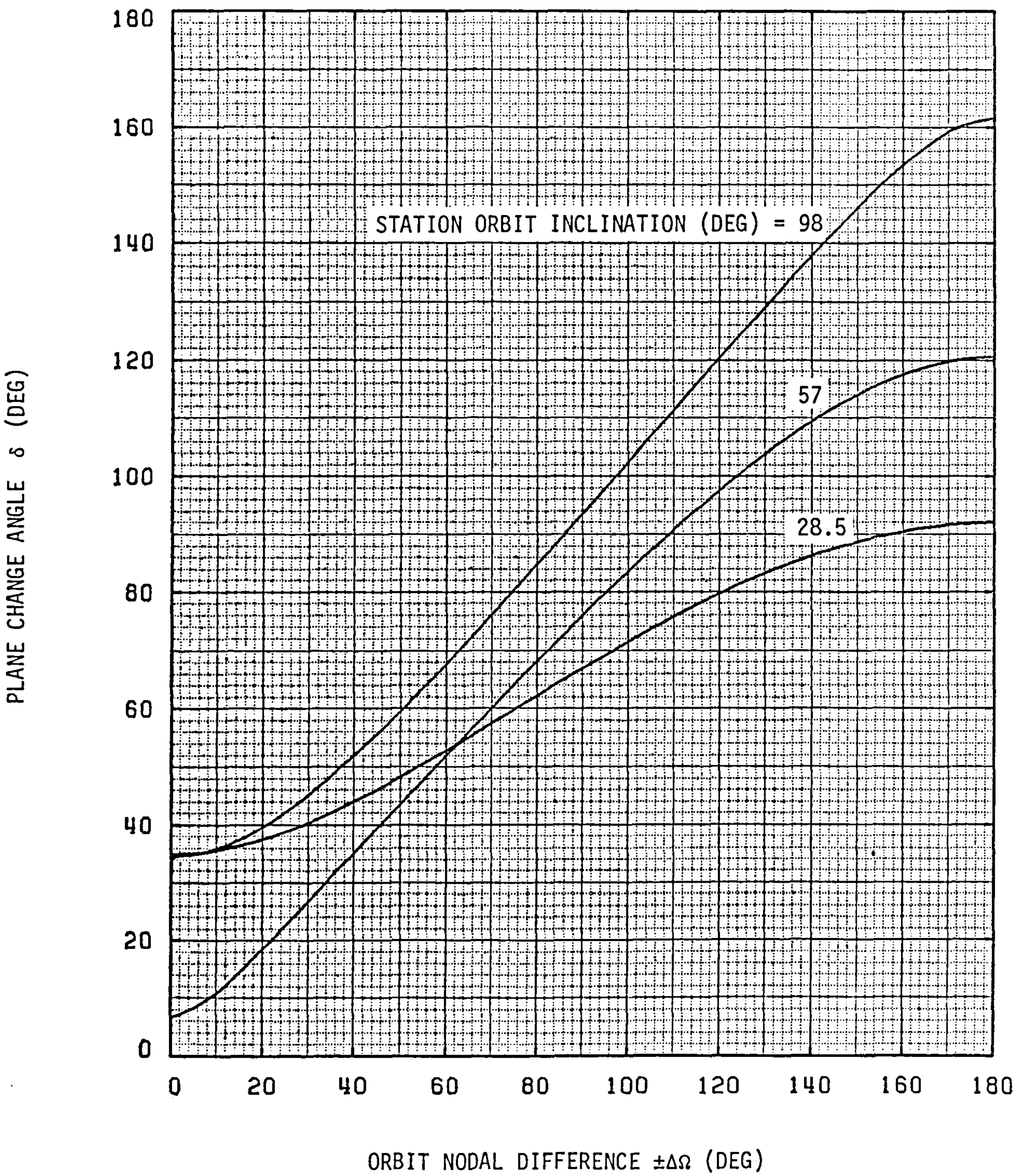


Figure 1-33.- Plane change requirements for mission orbit at 63.5 degrees inclination.

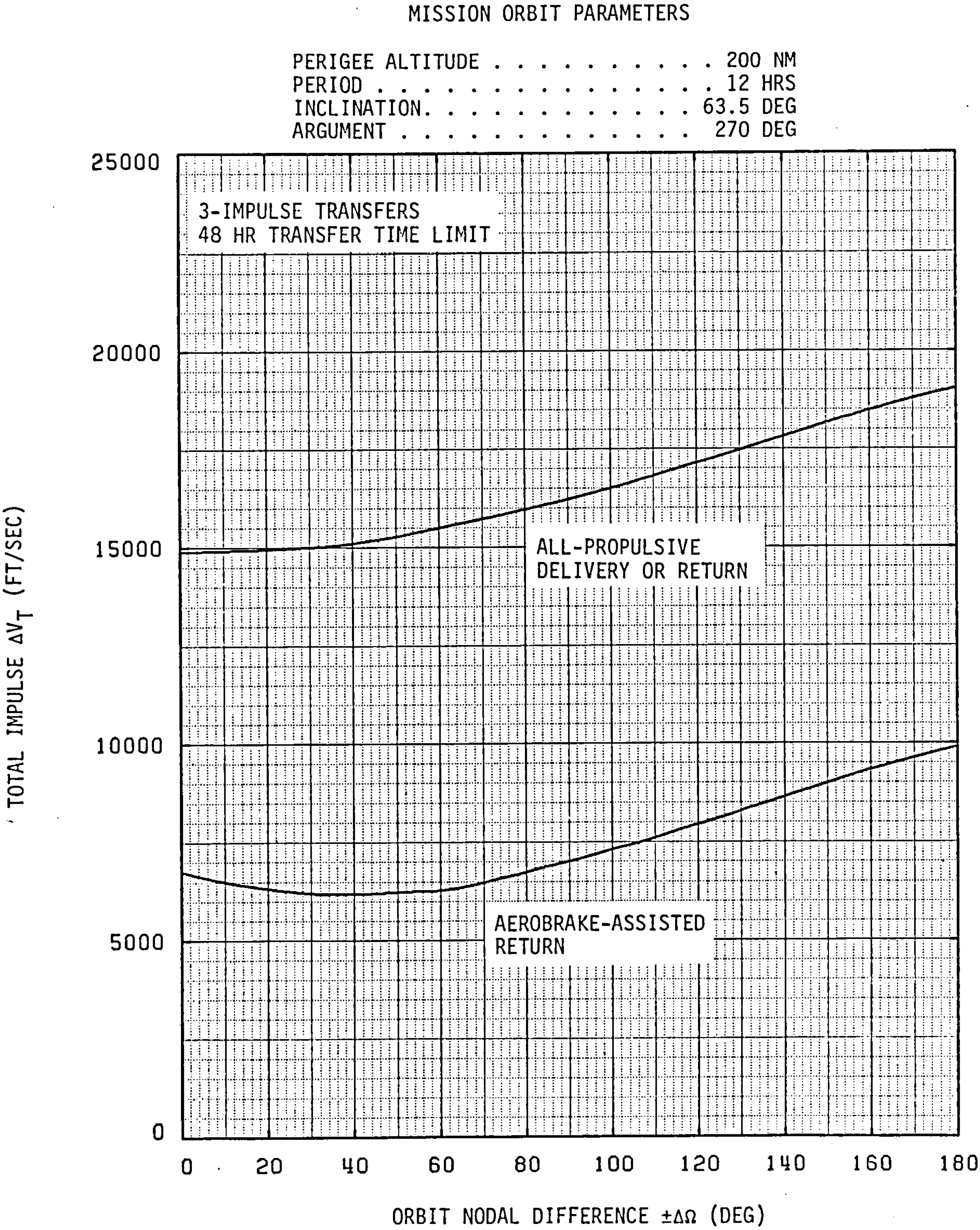


Figure 1-34.- Impulse requirements for semisynchronous (Molniya) elliptical orbit transfers with space station at 28.5 degrees inclination.

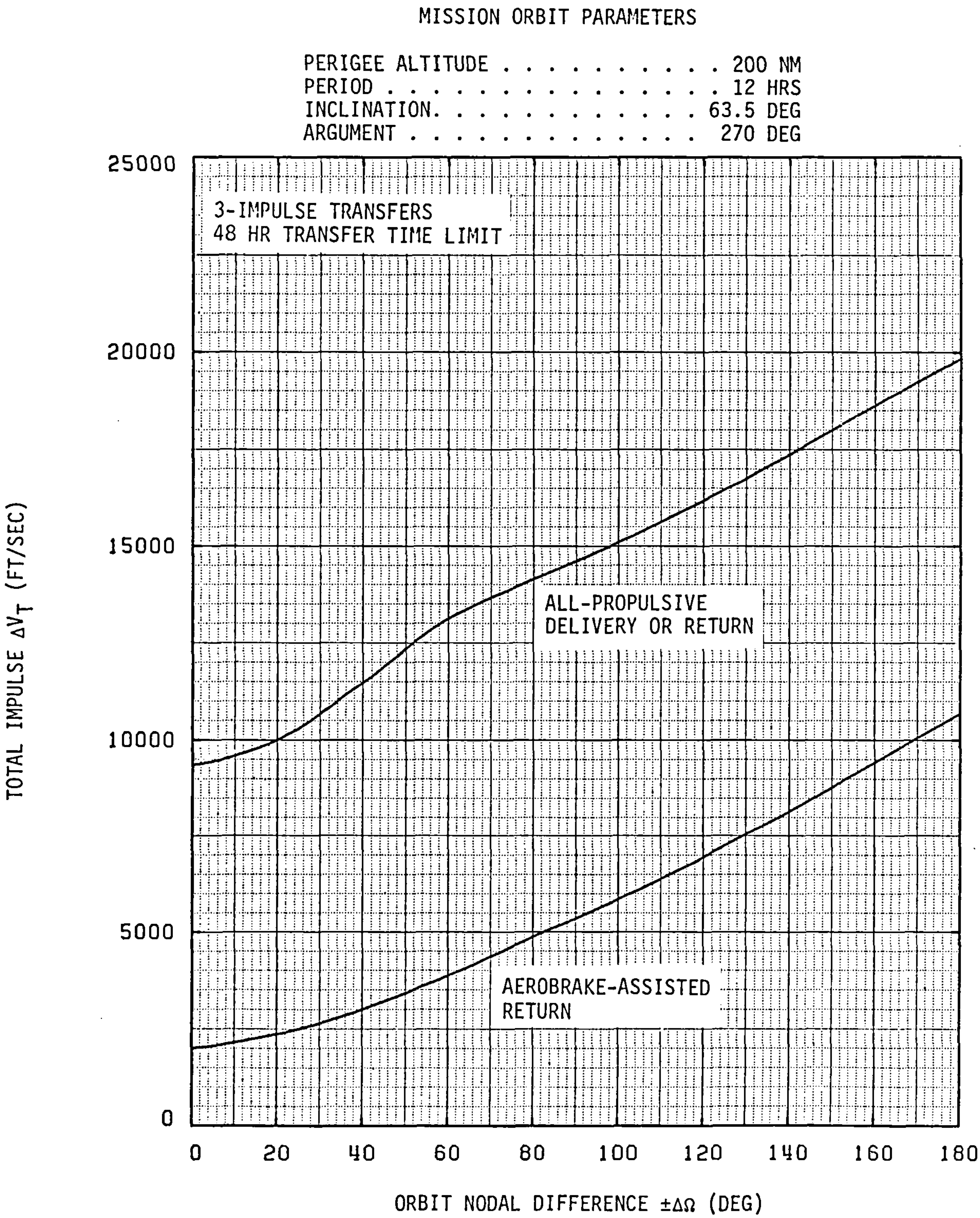


Figure 1-35.- Impulse requirements for semisynchronous (Molniya) elliptical orbit transfers with space station at 57 degrees inclinattion.

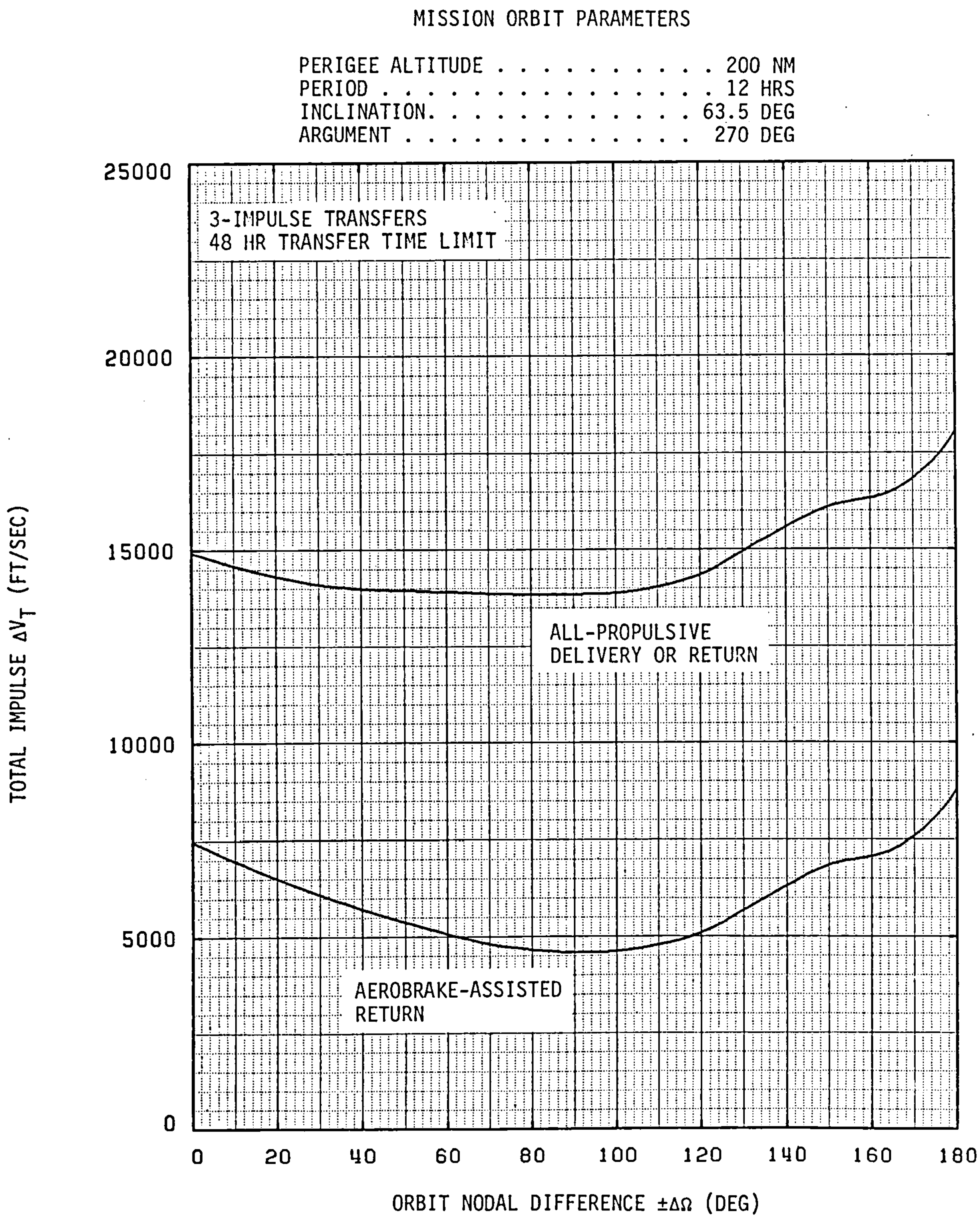
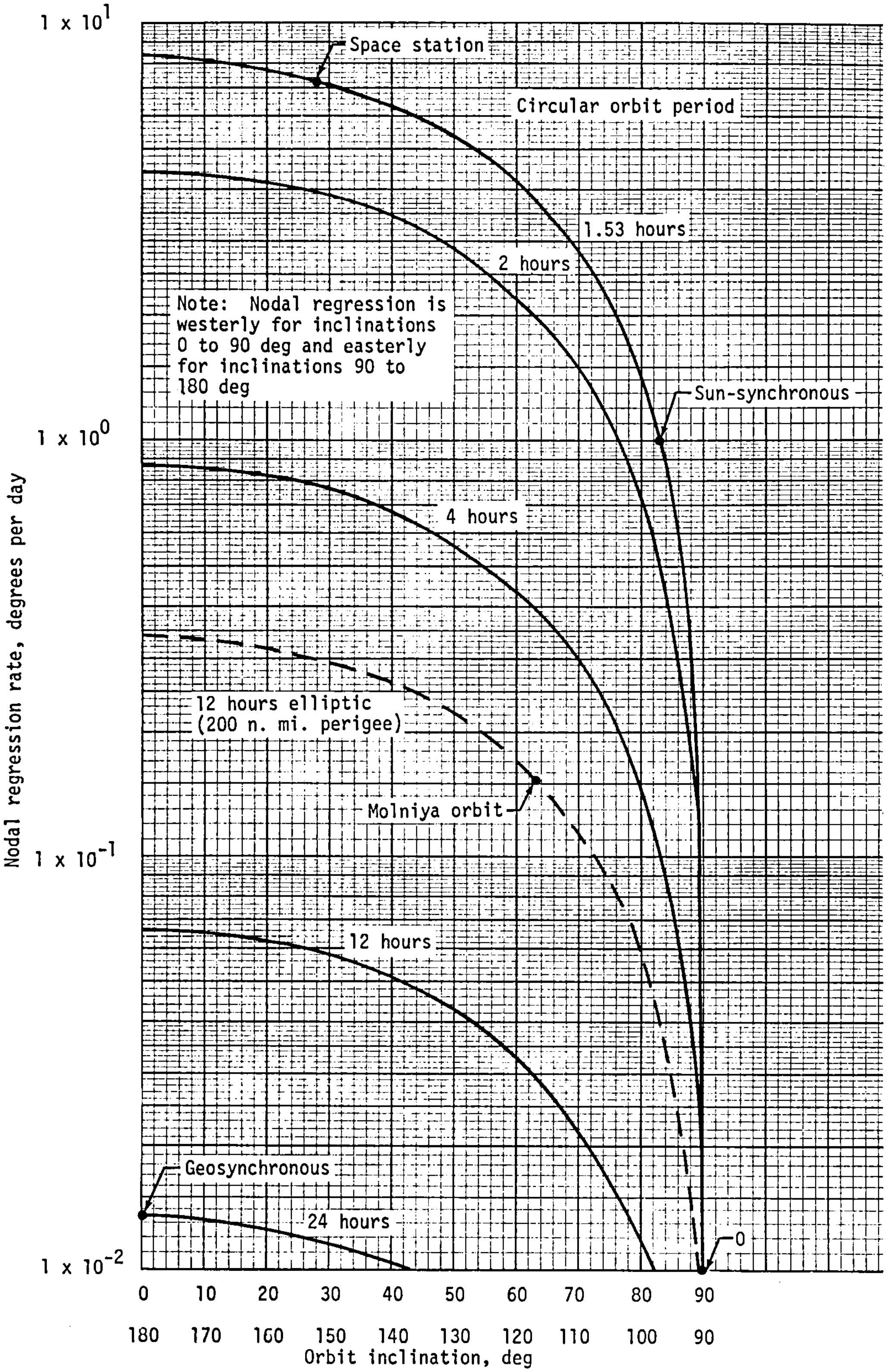
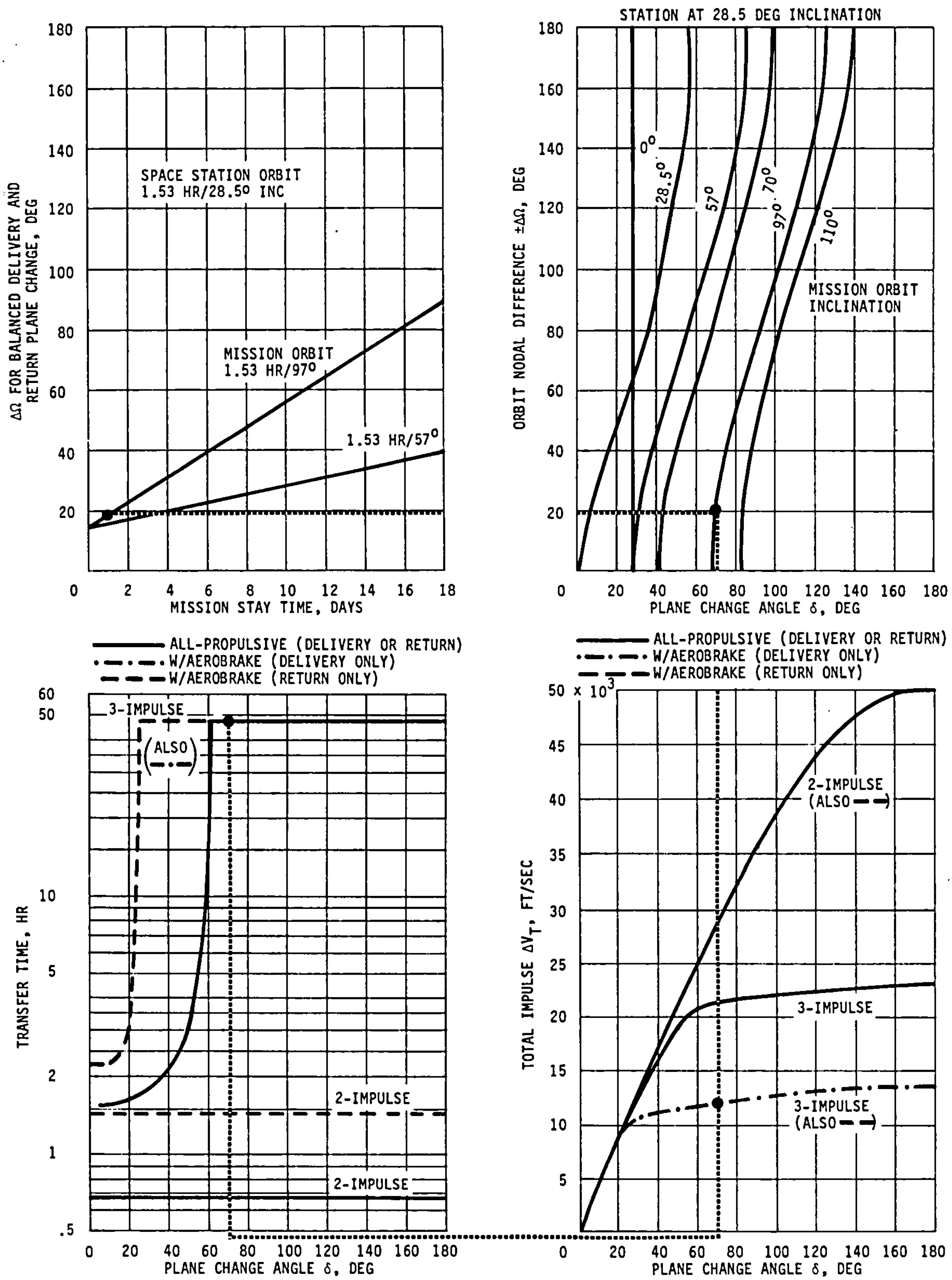


Figure 1-36.- Impulse requirements for semisynchronous (Molniya) elliptical orbit transfers with space station at 98 degrees inclination.



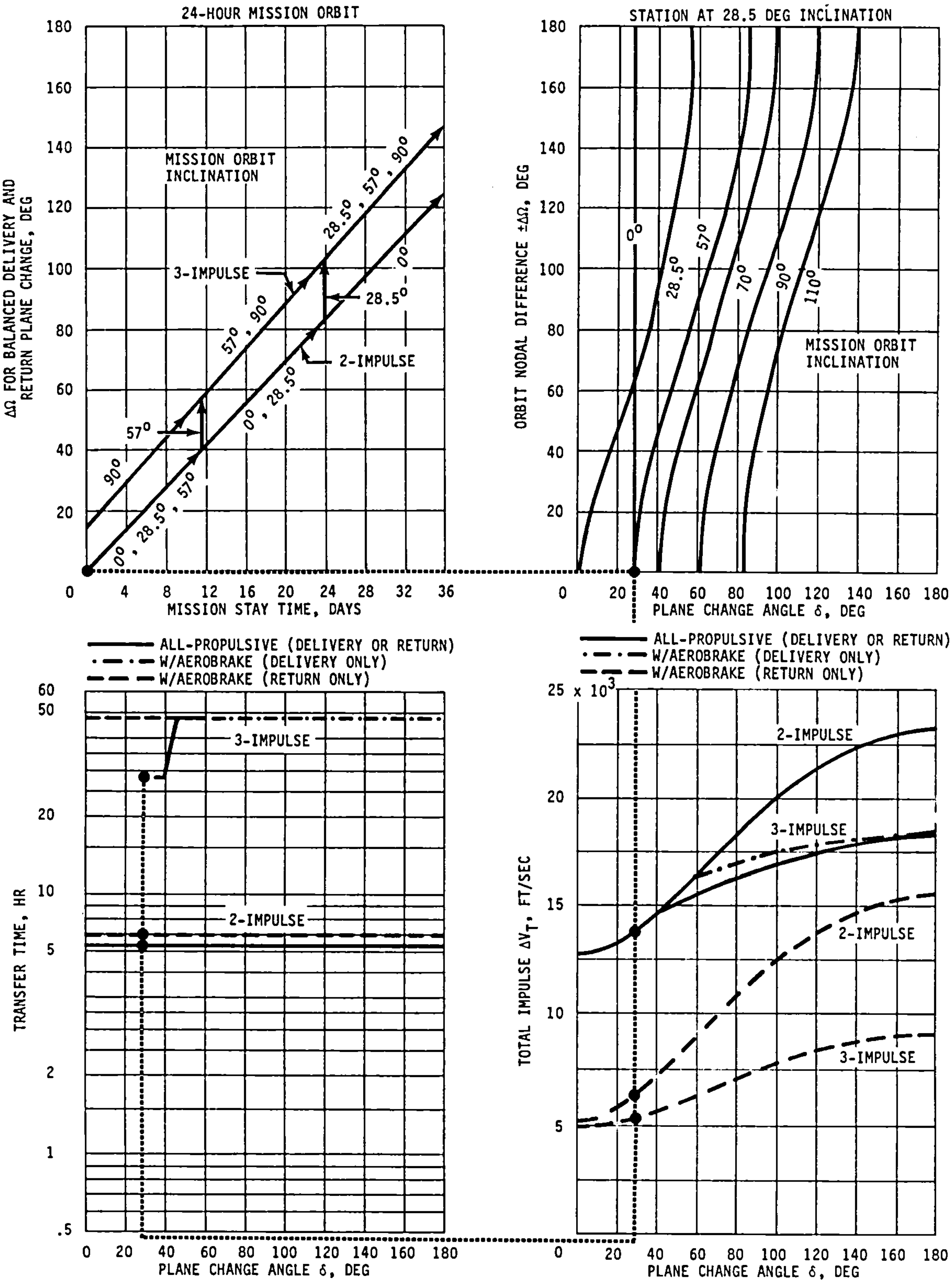
(a) Expanded resolution nodal regression rate.

Figure 1-37.- Mission nomograms.



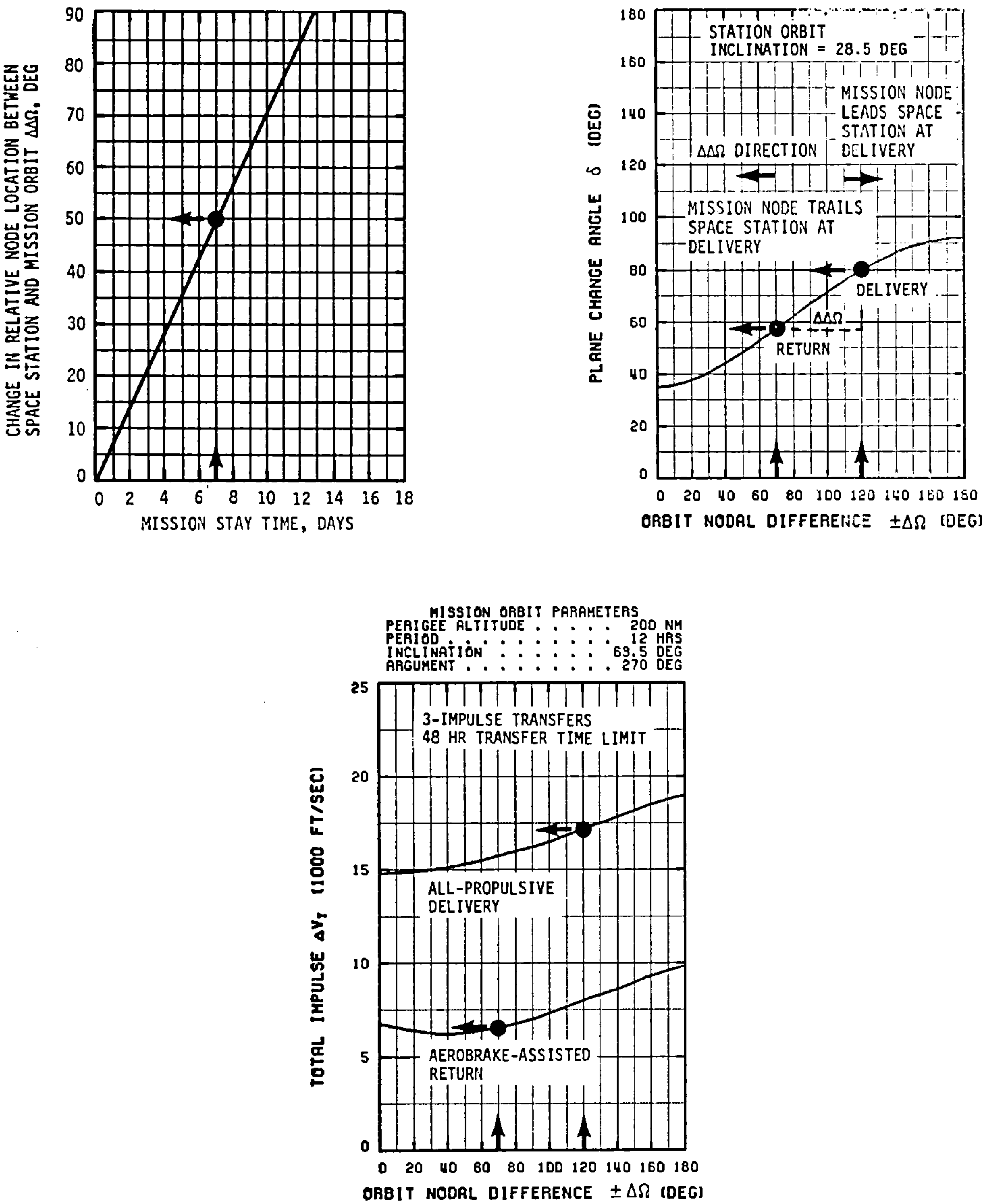
(b) Nomogram for 1.5 hour payload deploy orbit.

Figure 1-37.- Continued.



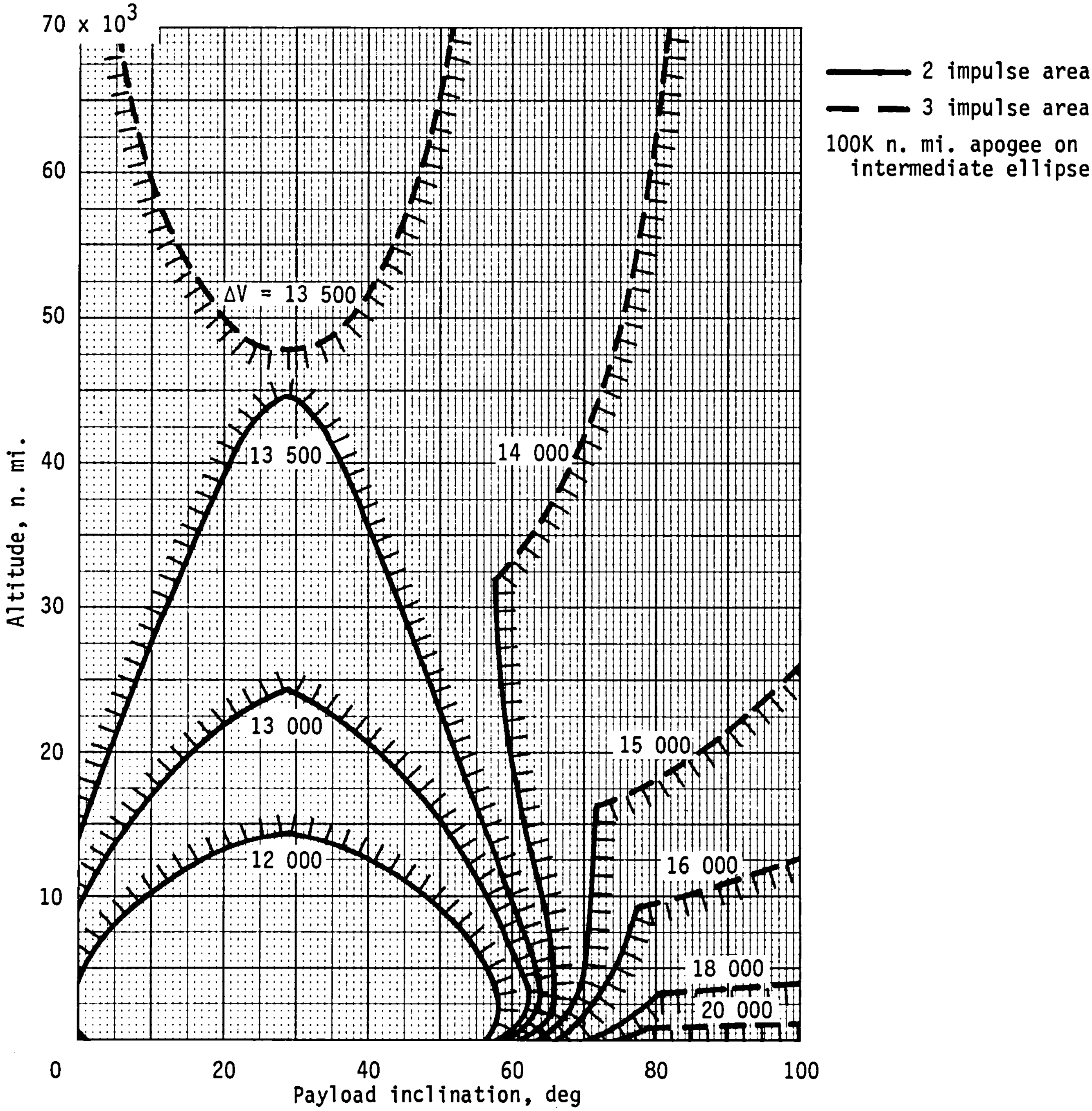
(c) Nomogram for 24-hour payload deploy orbit.

Figure 1-37.- Continued.



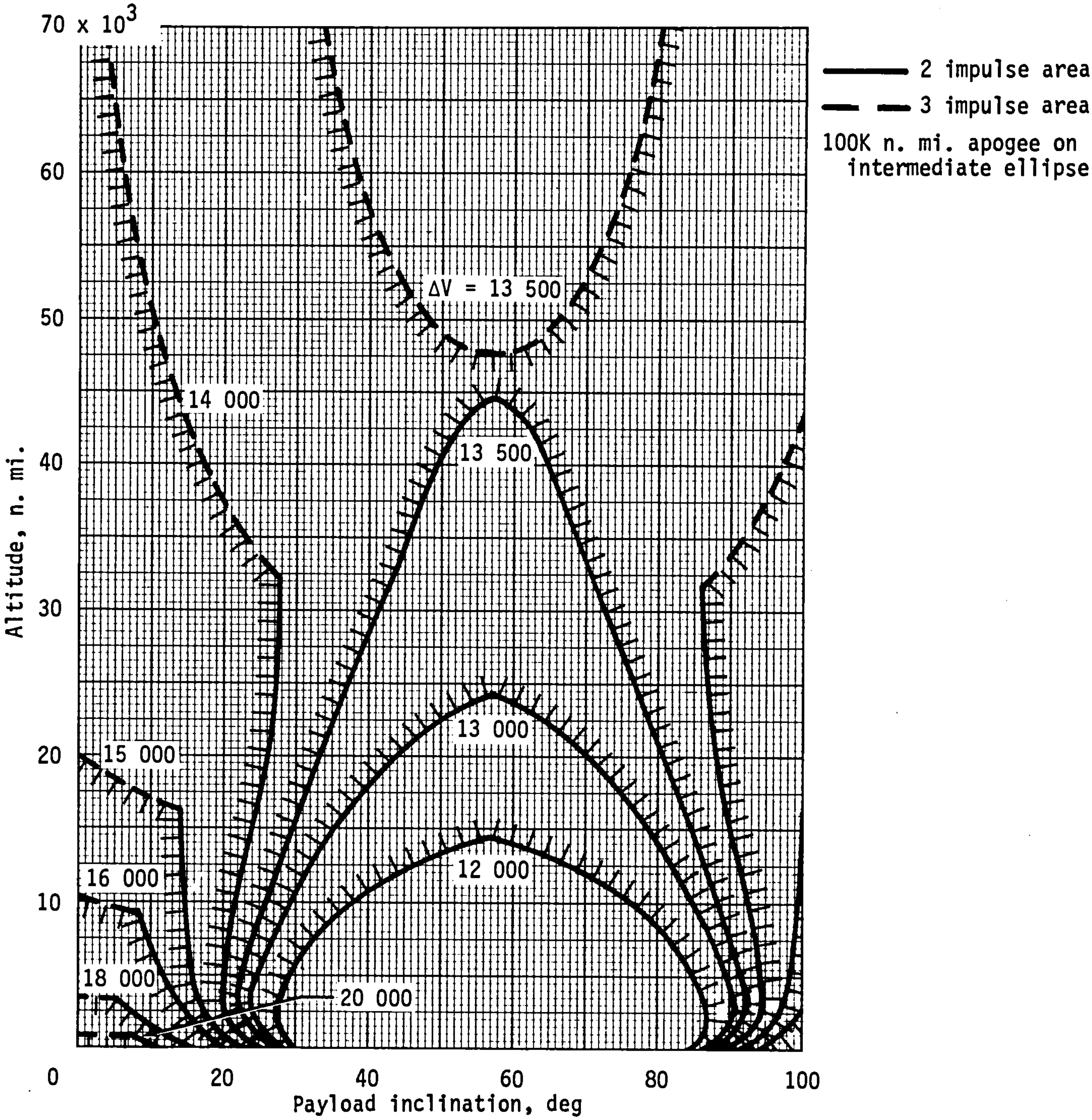
(d) Nomogram for 12-hour Molniya elliptic payload deploy orbit.

Figure 1-37.- Concluded.



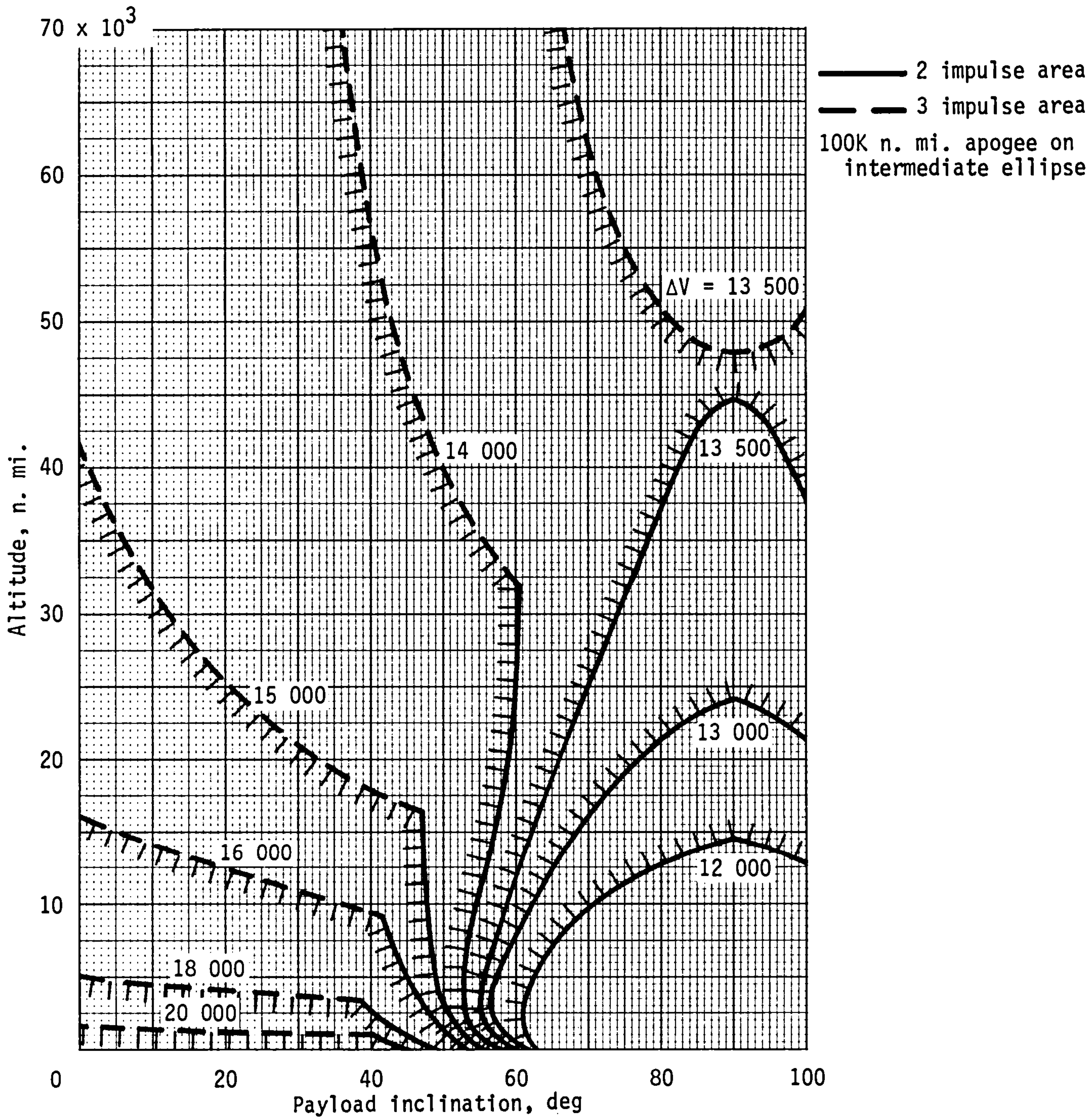
(a) Inclination 28.5°.

Figure 1-38.- Accessibility regions for fixed delta-V capability - outbound all propulsive.



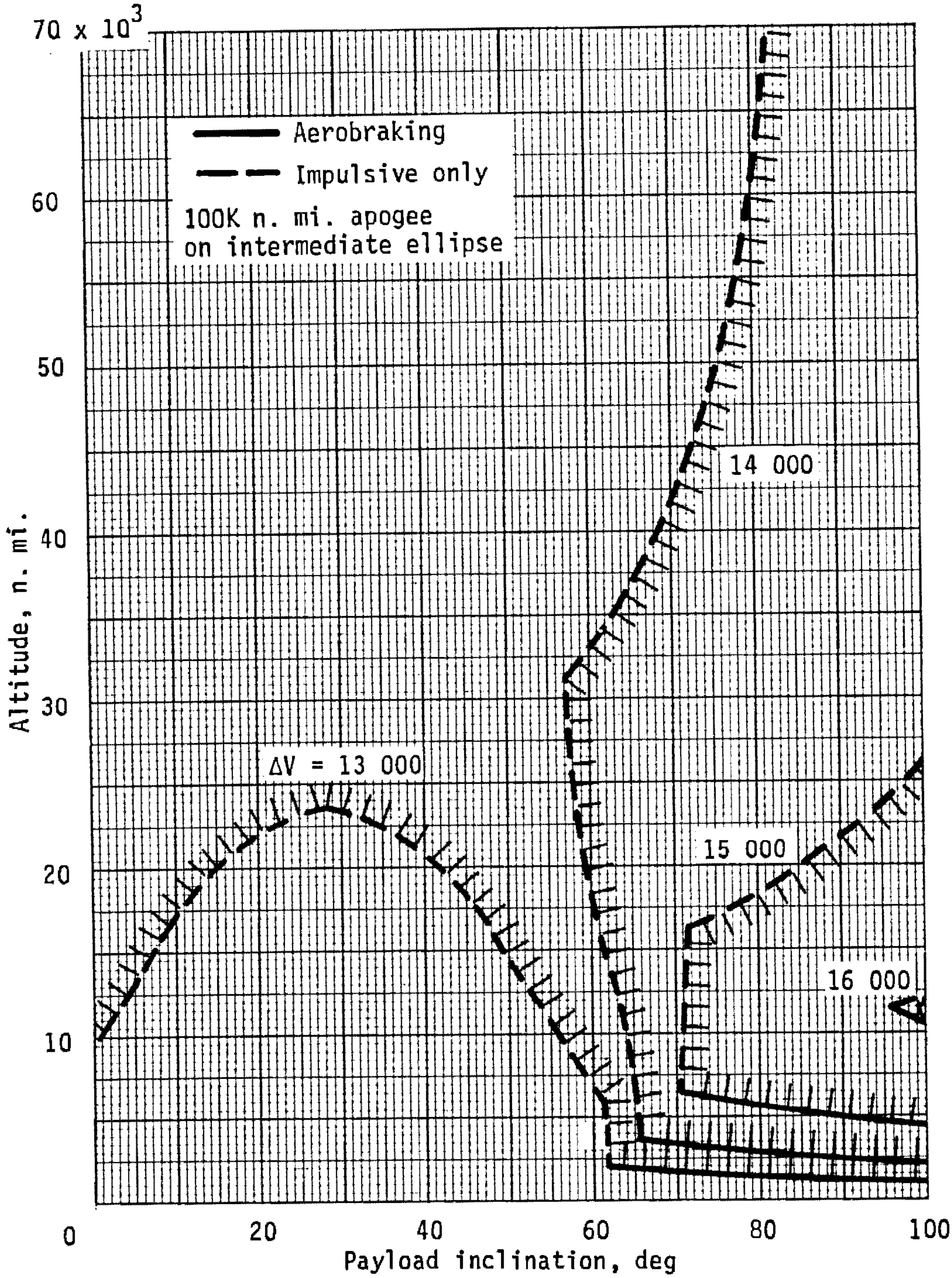
(b) Inclination 57°.

Figure 1-38.- Continued.



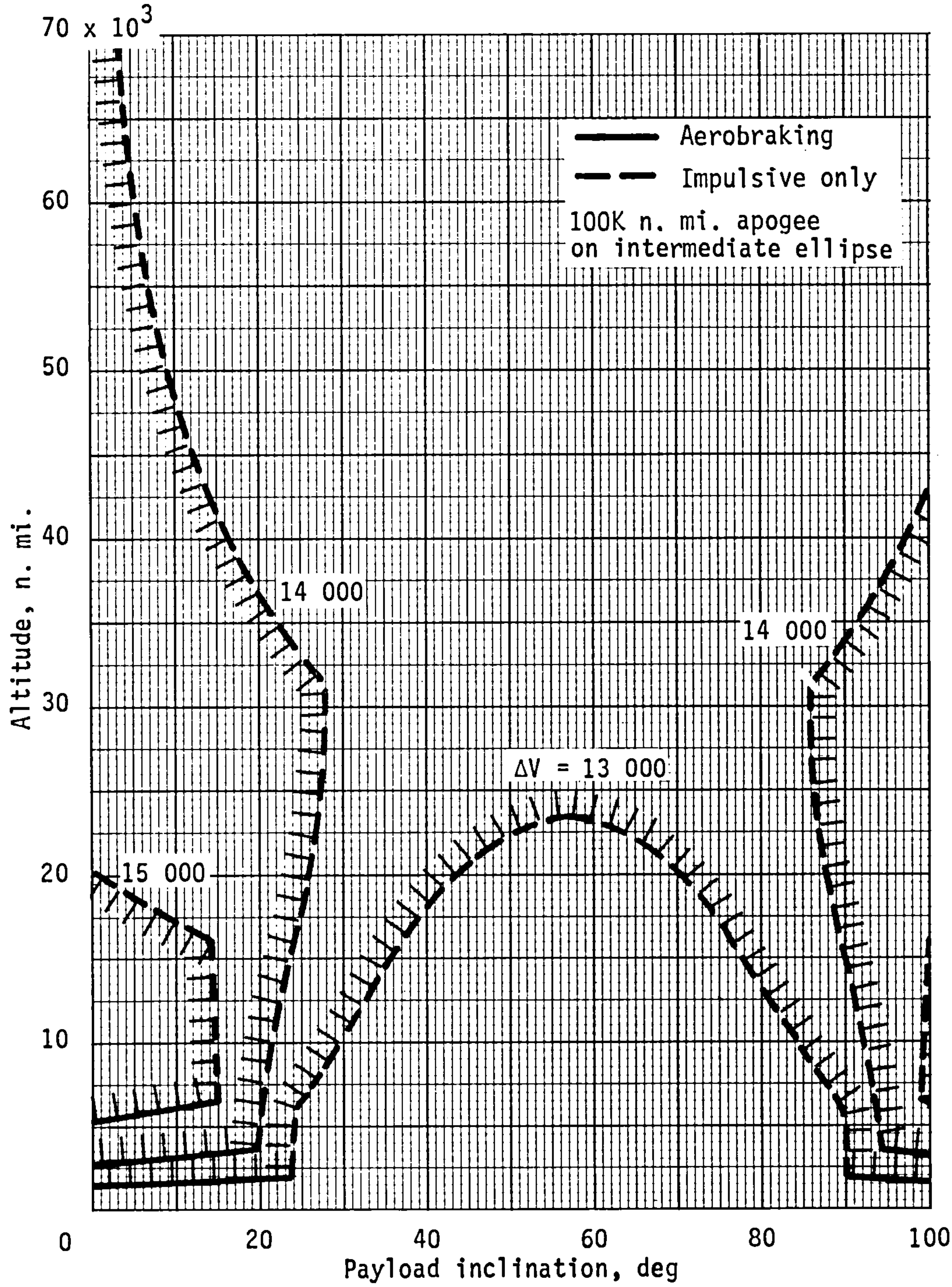
(c) Inclination 90°.

Figure 1-38.- Concluded.



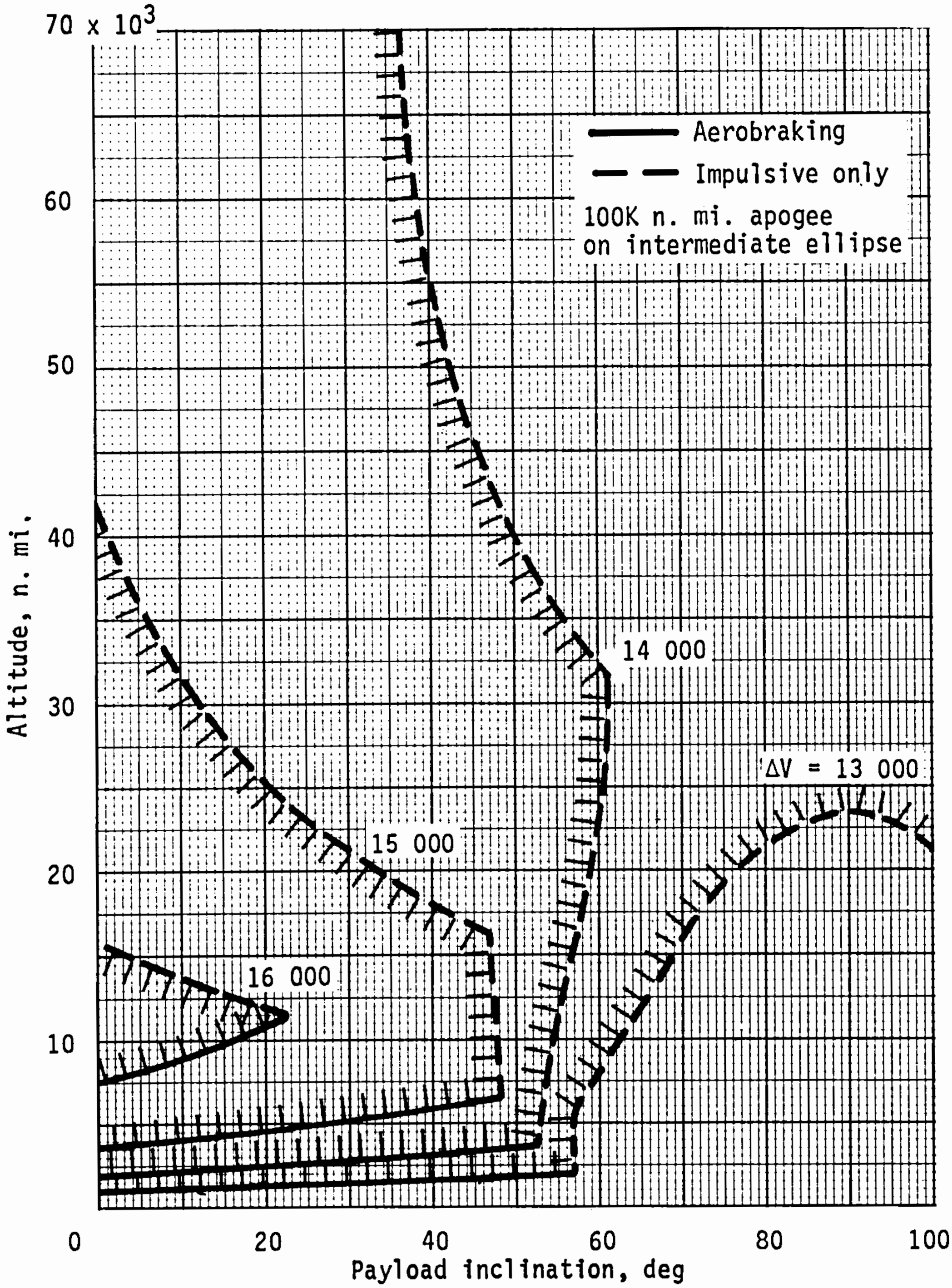
(a) Inclination 28.5°.

Figure 1-39.- Accessibility regions for fixed delta-V capability - outbound with aerobraking capability.



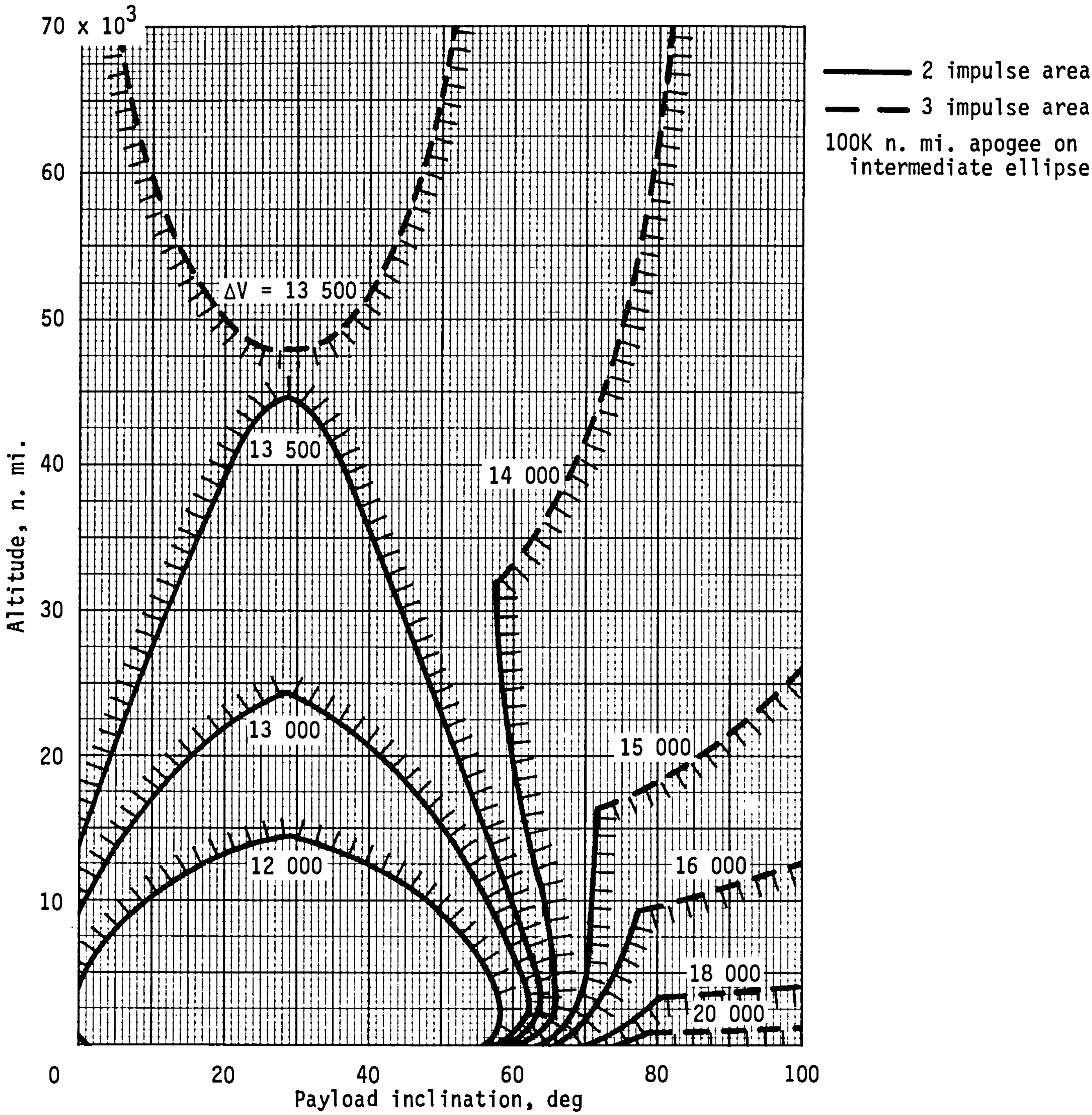
(b) Inclination 57°.

Figure 1-39.- Continued.



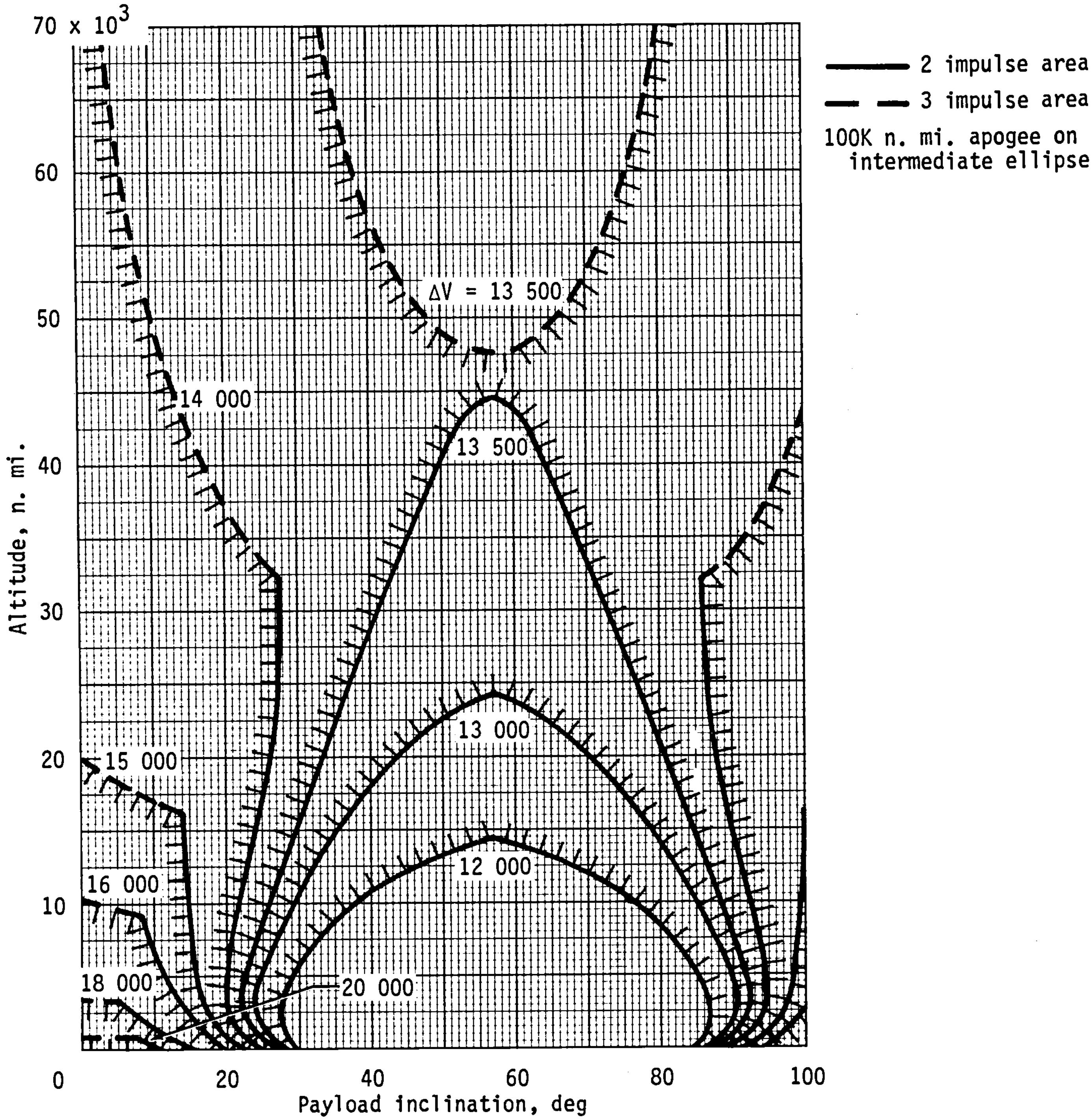
(c) Inclination 90°.

Figure 1-39.- Concluded.



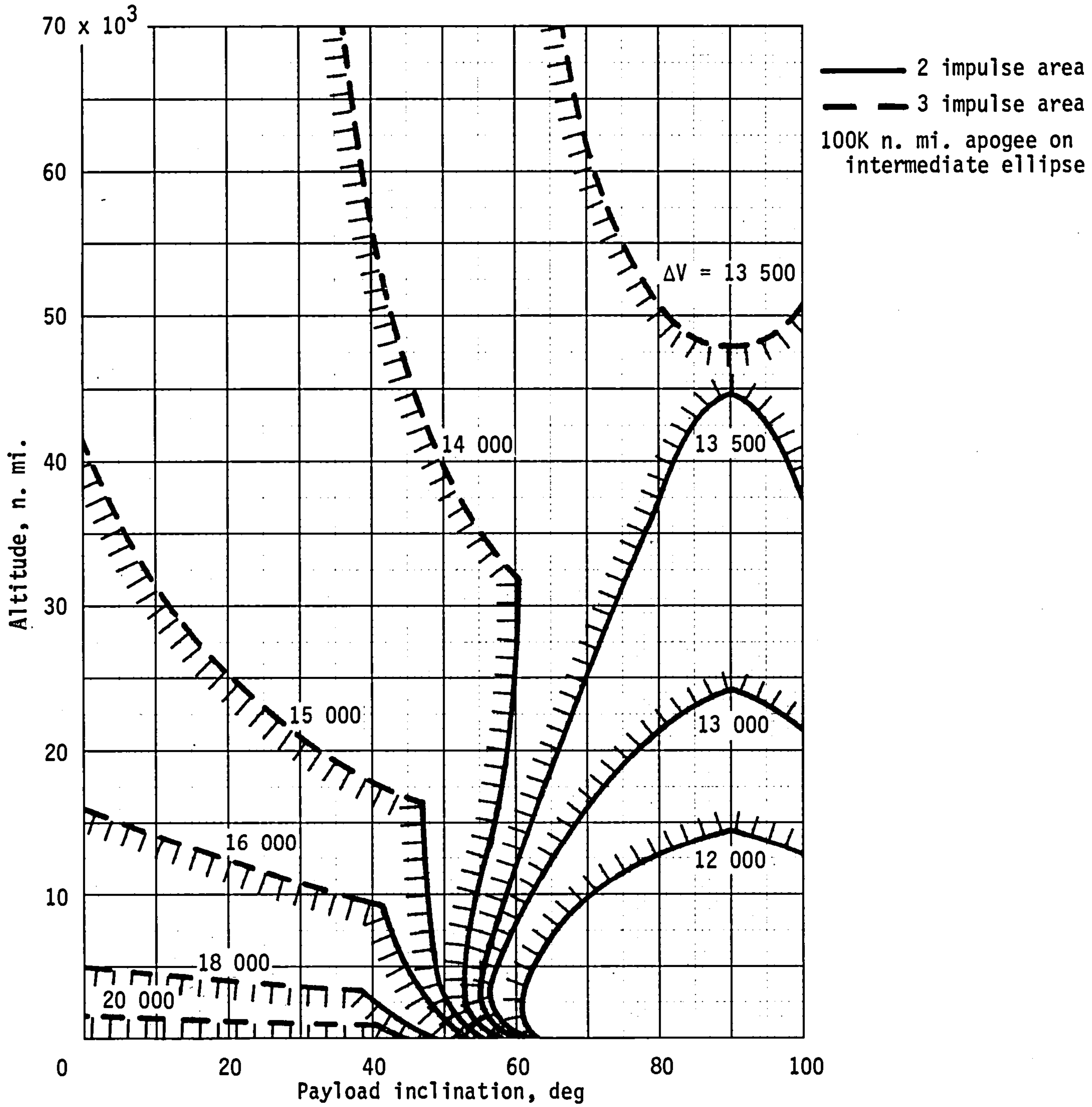
(a) Inclination 28.5°.

Figure 1-40.- Accessibility regions for fixed delta-V capability - return all propulsive.



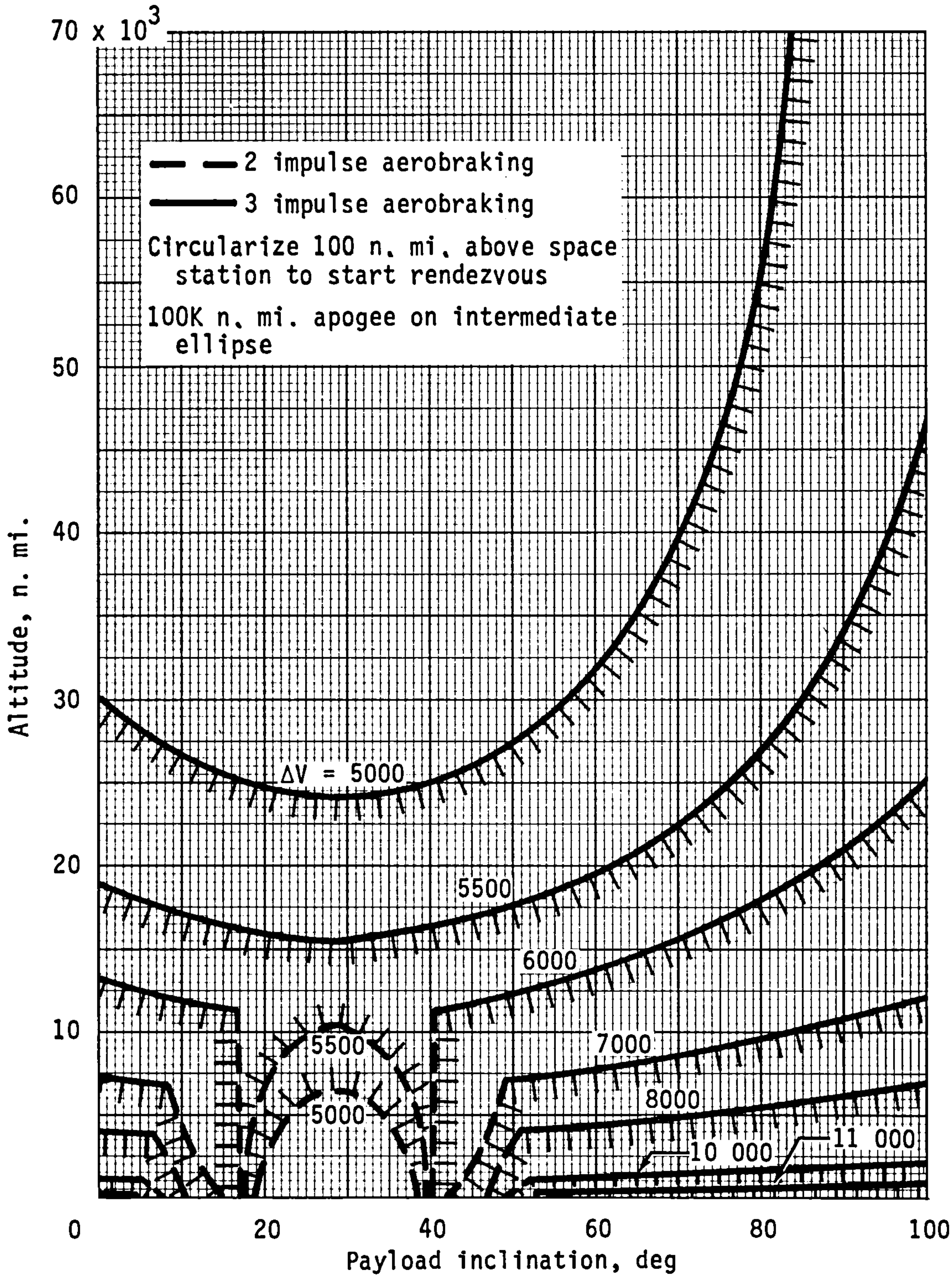
(b) Inclination 57°.

Figure 1-40.- Continued.



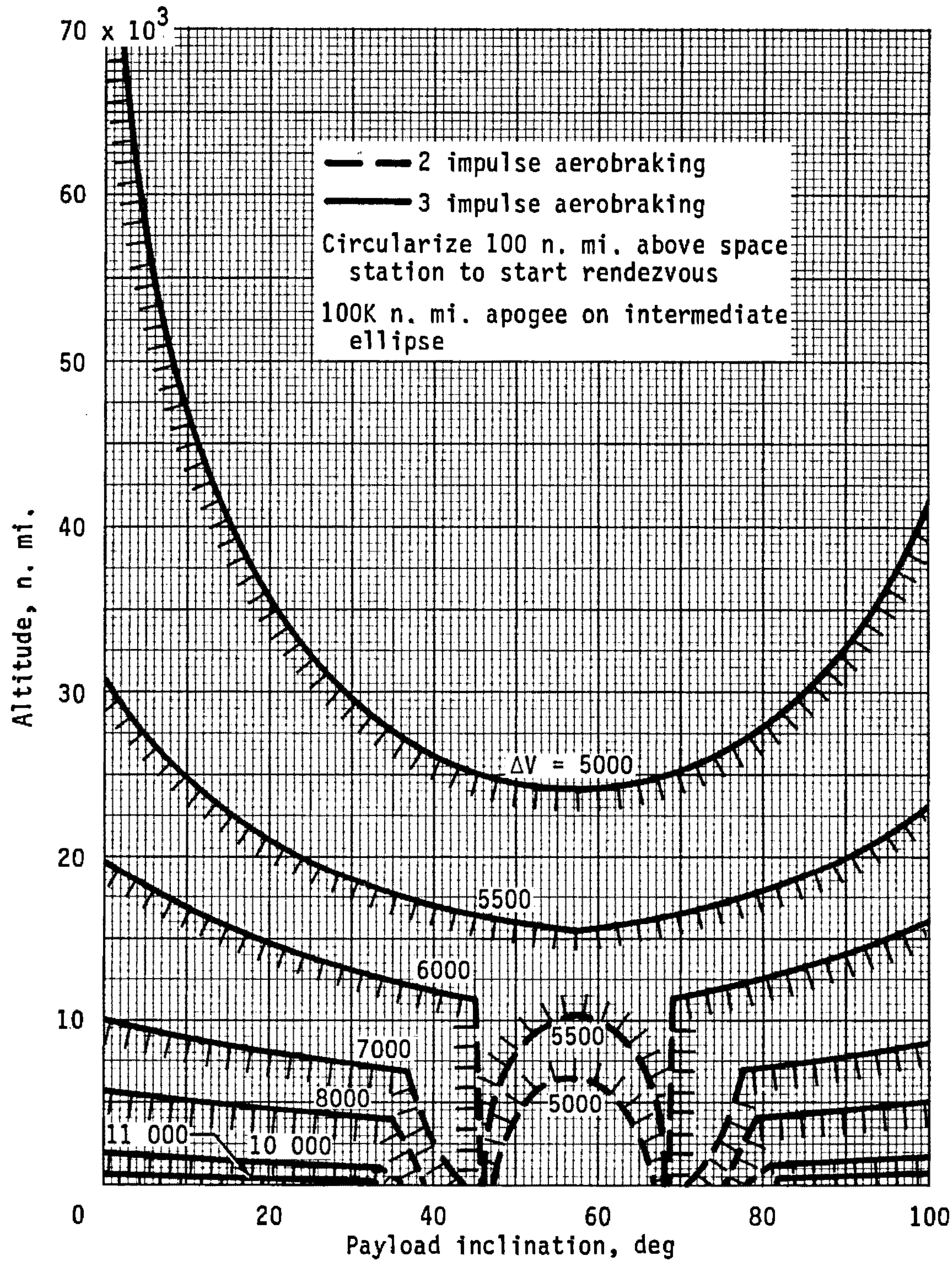
(c) Inclination 90°.

Figure 1-40.- Concluded.



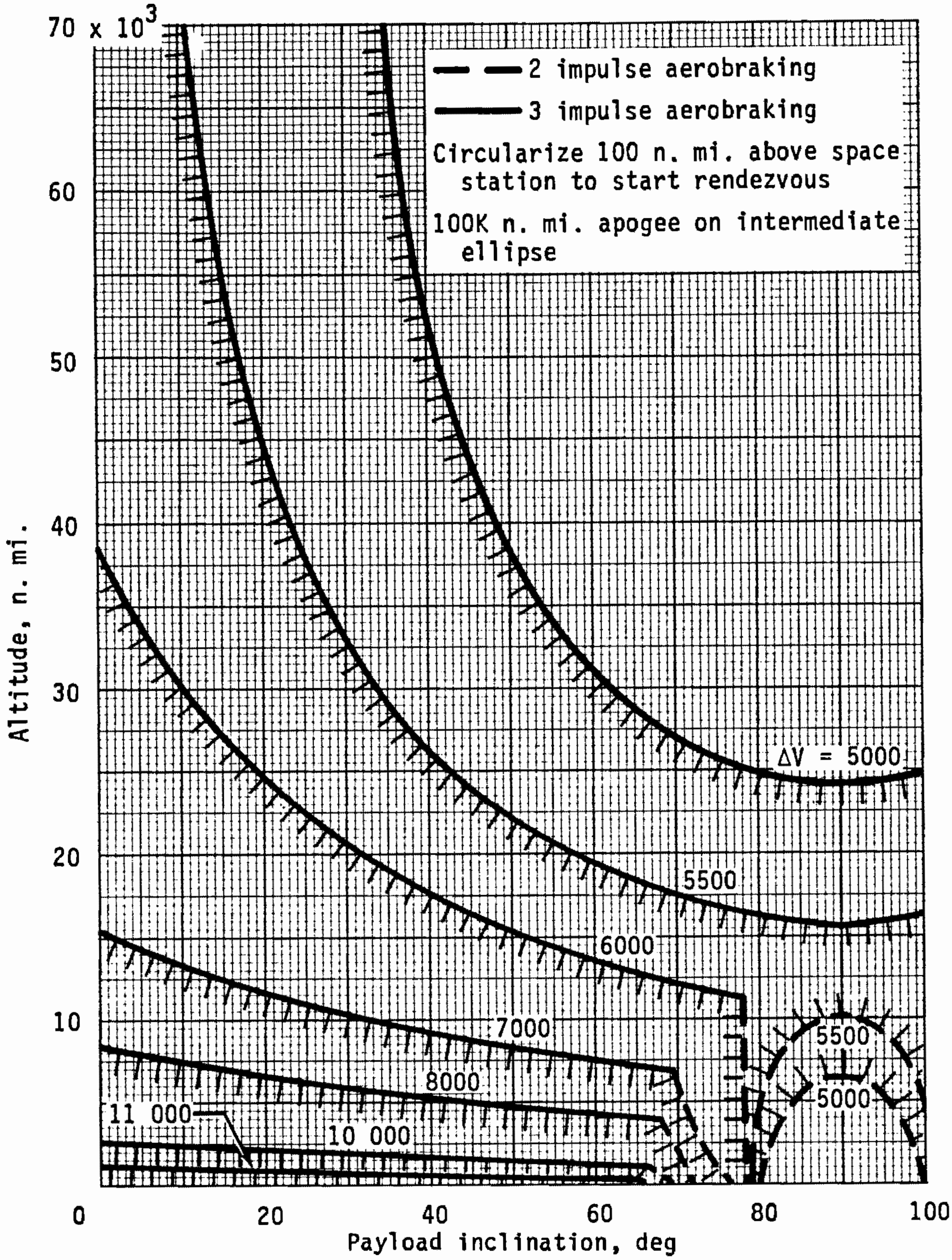
(a) Inclination 28.5°.

Figure 1-41.- Accessibility regions for fixed delta-V capability - return with aerobraking capability.



(b) Inclination 57°.

Figure 1-41.- Continued.



(c) Inclination 90°.

Figure 1-41.- Concluded.

2.0 SPACE STATION/ORBIT TRANSFER VEHICLE MISSIONS

2.1 INTRODUCTION

Section 2.0 presents descriptions of four classes of applications missions which typify those that are candidates for being serviced from a space station, OTV, and free-flyer system. The generic application missions that are addressed are: (a) OTV geosynchronous mission; (b) OTV planetary mission; (c) OTV Sun-synchronous mission; and (d) space station supported co-orbiting satellite operations. Where appropriate, the above mission classes were further subdivided within each class to reflect a potential range of requirement options; i.e., delivery, retrieval, servicing, etc. For all OTV missions, return of the OTV to the space station is assumed to be a standing requirement.

Each mission is presented in terms of a Conceptual Flight Profile, in which the key guidelines and assumptions, flight design considerations, and implementation rationale are highlighted. The profiles have been developed generically, independent of specific vehicle capability and constraints and independent of any specifically identified mission/payload set of requirements. These profiles, therefore, are intended to reflect typical operations requirements and to serve as a point of departure for the flight design associated with specific vehicle configurations and mission-unique requirements.

2.2 BASIC GUIDELINES AND ASSUMPTIONS

A. The space station is in a 200 n. mi. circular orbit at 28.5° inclination.

B. The OTV is space-based and is to be returned to the space station.

C. For OTV return, both all-propulsive and aerobraking/aeromaneuvering options will be considered.

D. For aerobraking the OTV is assumed to have the following characteristics:

   (1) .5 L/D

   (2) 5.12 W/C_DA lb/ft^2

E. The OTV configuration; i.e., number of stages, propellant loading, etc, was not considered.

F. The OTV returns to a 400 n. mi. low Earth parking orbit prior to rendezvous with the space station.

G. A cluster of co-orbiting satellites will be operating in an equiperiod, noncoplanar orbit ahead of the space station.

H. Flight duration was not considered as an OTV flight design constraint.

I. The coast time required from OTV deployment from the space station to OTV ignition is assumed to be a minimum of 45 minutes and a maximum of six hours (four orbits) to encompass safe distance requirements, maneuvering constraints, and OTV preignition detached operations.

J. A space station based maneuvering system separate from the OTV is used to support ferrying co-orbiting satellites to and from the space station.

K. To support OTV space station rendezvous operations, a minimum of two revolutions in the LEO parking orbit prior to OTV initiation of the rendezvous maneuvers is assumed.

2.3 SPACE STATION/OTV SEPARATION PHASE

At the proper time (as indicated by unique mission requirements) the OTV will separate from the space station with a ΔV-magnitude and direction sufficient to protect both the space station and any space station supported co-orbiting satellites from all hazards associated with OTV/payload post deployment operations. Hazards may constitute not only safety considerations, i.e., recontact, explosion and fragmentation, etc., but also mission degradation concerns such as contamination, etc. Under the assumption that a cluster of co-orbiting satellites will be operating in an equiperiod noncoplanar orbit ahead of the space station the strategy employed here is to place the OTV behind the space station at OTV ignition, providing adequate distance through a combination of separation ΔV and coast time prior to ignition.

Given a specified separation distance requirement at OTV ignition, the separation velocity will be minimized at the expense of maximizing the required coast time and vice versa. STS planning has established 45 minutes as a minimum time from deployment to an upper stage ignition; however, in addition to the fact that this coast time was principally associated with minimal postdeployment operations, the separation velocity, hence distance, was identified to be a function of the type of upper stage, engine size, and propellant loading. As significant, the distance specification took advantage of the STS maneuvering capability (translation and rotation) for self protection and the fact that the STS could be assured of always trailing the upper stage, thus negating a recontact concern.

As applied to space station, potential requirements are for the space station to be ahead of the OTV rather than trailing; furthermore, the space station is not assumed to have the maneuvering flexibility as did the STS. Assuming that there still remains a dependence upon upper stage configuration for "safe separation distance" determination, a more conservative philosophy may be warranted for space station than for STS. This, then, manifests itself into a coast time-separation maneuver trade-off suggesting a potential multirevolution postdeployment coast to minimize separation ΔV requirements.

Figure 2-1 illustrates typical relative motion between the OTV and space station during the postdeployment coast prior to OTV ignition. Noted on this figure are the candidate OTV nominal ignition options in terms of multiples of half orbit post separation coast; for each option, the relative position of ignition is biased to account for finite burn effects which then becomes a function of the OTV initial thrust-to-weight as indicated.

Figures 2-2(a) and 2-2(b) primarily illustrate the effects of the postdeployment coast on the relative motion between the OTV and space station during the OTV burn. In addition to the above, these figures also show the potential effect the OTV steering law can have upon the relative motion. These figures graphically highlight one significant fact: the deployment planning will need to extend beyond space station considerations themselves to also embrace considerations associated with the operations of a nest of co-orbiting satellites. Figures 2-3(a) and 2-3(b) illustrate the effect of the OTV initial acceleration (thrust-to-weight) on the relative motion during the OTV burn in a manner similar to figure 2-2.

Therefore, to summarize the above, from a traffic control point of view, the following parameters will influence to some degree the deployment planning: (a) separation ΔV; (b) postdeployment coast time; (c) OTV steering law; (d) OTV burn properties; (e) co-orbiting satellite nest location; and (f) allowable point of closest approach.

2.4 OTV MISSION PHASE

This section describes the Conceptual Flight Profile associated with four candidate space station/OTV/free-flyer application mission scenarios. The four mission classes are: (a) an OTV geosynchronous mission; (b) an OTV planetary mission; (c) an OTV sun-synchronous mission and (d) space station support of co-orbiting satellites.

For each of the above mission classes the associated profiles contain both the key guidelines and assumptions relating to mission requirements and the major flight phases that comprise the respective profile. The flight phases primarily address the appropriate flight design considerations and rationale which lead toward the specific conceptual implementation. The implementation details which are illustrative of a typical mission are presented as a sequence of events.

2.4.1 Geosynchronous Missions

2.4.1.1 Mission Guidelines and Assumptions

A. Three classes of geosynchronous mission requirements are assumed:

1. Delivery of a payload to a specific geographic longitude.

2. Retrieval and return to the space station of a geosynchronous payload.

3. Servicing of multiple geosynchronous payloads.

B. The geosynchronous payload is to be delivered to, retrieved from, and serviced at a zero degree inclined (equatorial) orbit.

C. Constraints on the inertial orientation of the trajectory prior to arrival at geosynchronous orbit were not considered.

2.4.1.2 Payload Delivery to a Specified Longitude

Typical flight profiles and ΔV requirements for this class of missions are summarized in the sequence of events and ΔV tables (tables 2-I and 2-II). Table 2-I(a) reflects the option for an all-propulsive OTV return to the space station while table 2-I(b) reflects the aerobraking/aeromaneuvering return option. These profiles are illustrated in the orbital diagrams shown in figures 2-4(a) through 2-4(d). Discussed below are the flight phases that comprise the mission class.

2.4.1.2.1 Deployment and separation.- The basic considerations and rationale that will dictate the deployment and separation strategy have been discussed in paragraph 2.3. The timing of this set of operations, however, will be driven by the specific OTV mission in terms of meeting OTV ignition requirements. Geosynchronous payloads will have a requirement for precise placement into the assigned Earth-fixed longitude slot in geosynchronous orbit. This requirement then dictates that a unified strategy of timing and maneuvering be implemented to minimize OTV performance penalties and operational complexities, while at the same time maximizing the opportunities to perform the mission.

The longitude at the time of arrival at geosynchronous altitude is dependent upon the Earth-fixed longitude at the time of OTV departure from LEO and the longitude change incurred during the geosynchronous transfer.

For a standard geosynchronous transfer orbit, the longitude change for the transfer is fixed; the OTV will have traversed an inertial angle of 180° while the Earth will have rotated approximately 80° during the 5-1/4 hour transfer. Thus, the Earth-fixed longitude for apogee arrival will be approximately 100° east of the injection point of the transfer ellipse. Or, more simply, to control payload placement to a specified Earth-fixed longitude, the longitude for the initial transfer maneuver should lie 100° west of that point.

For performance reasons (minimizing the plane change requirements) geosynchronous equatorial missions dictate that the OTV inject into a geosynchronous transfer orbit at or near the equator. For space station based operations, the time for the geosynchronous transfer orbit injection will be based on a nodal crossing opportunity that provides a longitude at first apogee of the transfer orbit within 25° west of the desired payload placement location. (The Earth-fixed longitude traversed between any two successive nodal crossings is a function of both orbit altitude and inclination; figure 2-5 illustrates this longitude shift for a fixed 28.5° inclination). For a 200 n. mi. orbit, the Earth-fixed longitude between two successive nodal crossings is approximately 23.5° which represents a theoretical worst case longitude correction capability that need be provided. However, a 25° longitude correction is assumed in the analysis in order to account for some initial space station altitude variations, transfer orbit dispersions, and flexibility to alter the ignition time by one orbit following OTV separation from the space station.

Two injection opportunities per day are available, one occuring at an ascending node and one occuring approximately 12 hours later at a descending node. The time of deployment and separation is then established by biasing the time of arrival at the ignition node to accommodate postdeployment (from the space station) coasting requirements.

The separation maneuver imparted to the OTV is timed to occur at a nodal crossing (that is, either ascending or descending) corresponding to the desired node for OTV injection into its geosynchronous transfer orbit. This then establishes the line-of-apsides of the parking orbit to be coincident with the line-of-nodes while simultaneously co-locating perigee of the post separation parking orbit at the same point. For this study, a 20 fps posigrade separation maneuver was applied to the OTV and a four-orbit coast from separation to OTV ignition was provided for any checkout, activation, etc., that may be required.

2.4.1.2.2 Geosynchronous transfer phase.- The strategy employed in the flight design of the geosynchronous transfer phase is quite similar to present geosynchronous transfer operation concepts; namely (a) the transfer from LEO is a standard trajectory to geosynchronous altitude independent of any specific longitude phasing requirements and (b) longitude corrections will be made in the near-geosynchronous regime.

The nominal transfer orbit injection maneuver is determined from the following targeting criteria:

A. The desired apogee altitude is 19 323 n. mi. (geosynchronous).

B. Apogee lies on the equator approximately one-half orbit after transfer orbit injection.

C. A 2.2° inclination is achieved, reflective of optimizing the plane change.

Thus, after inserting into the geosynchronous transfer orbit, the OTV/payload is in a 19 323 n. mi.-by-200 n. mi. orbit at 26.3° inclination.

The geosynchronous transfer phase terminates after a coast of approximately one-half revolution in the transfer orbit. Specifically, this phase is terminated when the OTV arrives at the equator, at which time the OTV performs the maneuver which establishes the longitude drift phase of the flight.

2.4.1.2.3 Geosynchronous longitude drift phase.- The maneuver to initiate the geosynchronous longitude drift phase occurs at the equator after one-half revolution in the transfer orbit. The out-of-plane component of this maneuver is targeted to place the OTV into an equatorial orbit removing the transfer orbit's 26.3° inclination. The horizontal component of this burn is determined from the orbital period requirement to phase for longitude in one revolution. Restricting the phasing requirement to a maximum "catchup" of 25°, the resulting geosynchronous drift orbit could vary between approximately 19 323 n. mi.-by-17 200 n. mi. and 19 323 n. mi. circular (for being at the proper longitude at the time of the manuever). Even without reoptimizing the plane change partitioning for the transfer orbit, the total additional ΔV required for longitude phasing is approximately 20 fps for this strategy. This is shown in figure 2-6(a). For information purposes, figure 2-6(b) shows the drift orbit requirements reflecting LEO departure one revolution earlier whereby the OTV longitude at geosynchronous arrival would be east of the desired location requiring the OTV to "fall back" by going to a drift orbit above geosynchronous. It seems to be substantially better for performance to plan LEO departure opportunities based on a 25° easterly longitude shift requirement than for maintaining a capability to limit the shift to 12.5°; this would imply both an easterly and westerly shift requirement.

One-half revolution after inserting into the equatorial geosynchronous phasing orbit, there is provision for correcting orbital dispersions. Planned to occur either (a) at the equator or (b) one-half revolution after insertion into the required phasing orbit (in that order of priority), this nominally zero fps maneuver will provide the capability to correct altitude dispersions resulting from the geosynchronous transfer orbit injection and also inclination dispersions. One revolution after insertion into the geosynchronous longitude drift orbit, the OTV performs a maneuver to circularize at 19 323 n. mi. The OTV should be in an equatorial orbit and, hence, no plane change should be required.

Shown in figure 2-7 is the relative motion of the OTV in the geosynchronous drift orbit with respect to the desired geosynchronous placement. The dependence of this motion upon the required longitude make-up is also illustrated.

2.4.1.2.4 Geosynchronous operations phase.- The specific activities and duration of this phase is a variable depending upon the requirements and operations associated with the deployment of the payload from the OTV. Included in these operations would be all OTV support to payload activation and checkout, physical release of the payload from the OTV, and separating the OTV away from the payload prior to its return to low Earth orbit.

The OTV return to the space station must be initiated in the vicinity of the node of the space station orbit plane to minimize the performance requirements for the plane change. Two return opportunities are available each day (similar to the geosynchronous delivery opportunities) -- one opportunity corresponding to the OTV arriving at the space station orbit plane ascending node and the other opportunity occurring at the space station orbit plane descending node. The time between successive nodal crossings is approximately 11.7 hours; therefore, while the actual deployment operations could be of any duration, the 11.7 hours should be considered the minimum time the OTV would remain in geosynchronous orbit prior to initiating a return to LEO and the space station.

After deployment of the payload, the OTV will perform a retrograde separation maneuver to move away from the payload. The targeting and timing criteria for this maneuver should provide sufficient separation distance between the OTV and the payload prior to the OTV LEO transfer orbit burn and not be performed so near the LEO transfer orbit injection point as to force the LEO transfer to be delayed until the next opportunity (11.7 hours later). For this study a 20 fps separation maneuver was assumed with the companion assumption it may occur anywhere with respect to the LEO injection point.

2.4.1.2.5 Low Earth orbit transfer phase.- Two options are presented here for the OTV return to the space station. They are: (1) an all-propulsive OTV return which requires that all maneuvers be performed utilizing the OTV propulsion system, and (2) an aerobraking/aeromaneuvering OTV return which substitutes for OTV propulsion in the dissipation of orbital energy of the transfer orbit. In either case, the approach taken is to bring the OTV to a nominal LEO parking orbit prior to initiation of maneuvers for effecting rendezvous with the space station. To accommodate any downrange angular separation (phasing) between the space station and OTV which can be induced by the specific flight variables; e.g., geosynchronous longitude requirements, duration in geosynchronous orbit, dispersion sensitivity, etc., a 400 n. mi. circular orbit is selected. This parking orbit (200 n. mi. above the space station) permits a rendezvous for the worst phasing situation to be accomplished within approximately one day after arriving at low Earth orbit.

It is to be noted that, while the phasing is not a targeting criteria prior to attainment of the LEO parking orbit, the same philosophy applied to the orientation of the OTV orbit plane in low Earth orbit relative to the space station orbit plane could result in prohibitive performance penalties created by the required plane change. While there is almost no movement of the inertial node in the geosynchronous and LEO transfer orbits, the space station orbit plane will regress approximately 7.2° per day. Therefore the planning philosophy is to target the OTV to establish a plane that the space station will regress into at the time of rendezvous. The concept, therefore, is to choose the geosynchronous departure time that provides the proper nodal bias to account for the total differential nodal regression experienced during the descent from geosynchronous orbit through rendezvous. This is illustrated in figures 2-8(a) and 2-8(b). Beyond this timing planning, which applies to both the all propulsive and aerobraking/aeromaneuvering options, it is assumed no other explicit targeting provision is made for controlling relative orientation between the OTV and space station orbit planes prior to initiation of LEO rendezvous maneuvers.

2.4.1.2.5.1 OTV all-propulsive transfer: Prior to OTV arrival at the common node of the space station orbit plane and the OTV's geosynchronous equatorial orbit, the OTV executes the maneuver which places it on a transfer trajectory from geosynchronous orbit to a prerendezvous LEO parking orbit. By properly timing this maneuver, the OTV can nominally be placed in a phantom space station orbit plane which, at the time of rendezvous will be coincident with the actual space station orbit plane through maneuvers which achieve the required change in inclination as opposed to maneuvers which are required to change both inclination and node. This strategy optimizes the performance requirements for the OTV to match the orbit plane of the space station.

A typical geosynchronous departure window for the all-propulsive return is shown in figure 2-8(a). This figure illustrates the maneuver location dependence upon the phasing between the OTV and space station referenced to OTV arrival at the space station node. (The figure also relates the maneuver location to a nominal impulse time). The major contributor to the maneuver location is the differential nodal regression between the OTV and space station orbit during the transfer to low Earth orbit. This is independent of phasing and represents a minimum location bias. To this is applied an additional correction to account for relative nodal effects during the LEO rendezvous operations. The maximum location bias, approximately 2.92^{o} prior to arrival at the instantaneous space station node (equivalent to 11.4 minutes) is based on the worst case phasing for rendezvous. (The approximate worst case phase angle between the space station and OTV at the time of OTV descent initiation is 72^{o}, the OTV ahead of the space station).

The low Earth orbit transfer maneuver is similar to the strategy employed in the transfer from LEO to geosynchronous orbit, namely the transfer from geosynchronous orbit is a standard trajectory to LEO independent of specific phasing and results from the following targeting criteria:

A. The desired perigee altitude be 400 n. mi.

B. Perigee to lie on the equator approximately one-half orbit after transfer orbit injection.

C. A 26.3^{o} inclination be achieved, reflective of optimizing the plane change. Thus, after insertion into the LEO transfer orbit, the OTV is in a transfer orbit with an apogee altitude that is a function of separation maneuver, but a maximum of 19 323 n. mi., a perigee altitude of 400 n. mi. and a 26.3^{o} inclination. Upon arriving at the equator, the LEO insertion maneuver is performed.

2.4.1.2.5.2 <u>OTV aerobraking/aeromaneuvering transfer</u>: The aerobraking/aeromaneuvering transfer option is fundamentally similar in targeting to the all-propulsive option; the primary difference between the two options during this phase is the targets for the maneuver to initiate the descent from geosynchronous orbit. Shown in figure 2-8(b) is the departure window for this OTV return option.

Preliminary studies have indicated that, based upon a set of assumed vehicle capabilities and constraints; e.g., lift-to-drag, ballistic coefficient, loads, etc., a flight path angle of about -4^{o} at entry interface (400 000 ft) will be required. For geosynchronous descent trajectories, this target set corresponds to a vacuum perigee of approximately 45.6 n. mi. altitude. These same studies have also shown that, for the vehicle flight characteristics assumed; e.g., 0.5 L/D, and 5.12 W/C_DA, there is sufficient maneuvering capability to attain a 2.2^{o} inclination change during the aerodynamic flight regime. This then permits the transfer maneuver to be targeted for a 26.3^{o} inclination change as opposed to the entire 28.5^{o} inclination; thereby, saving 165 fps for the maneuver.

As previously described for the OTV all propulsive option, OTV transfer from geosynchronous orbit is initiated prior to the OTV arrival at the common node between the OTV and space station orbits. This maneuver places the OTV in an orbit with a maximum apogee of 19 323 n. mi., a vacuum perigee of 45.6 n. mi. and an inclination of 26.3^{o}. After the initial maneuver is performed, one or more midcourse maneuvers will be required to control the flight path angle at 400 000 feet. The sensitivity of entry conditions at 400 000 feet are shown in figure 2-9. Precise navigation and adequate propellant for performing the necessary trajectory corrections will be a requirement.

Upon reaching 400 000 feet altitude, the OTV begins its aerodynamic braking/maneuvering to lower the apogee from 19 323 n. mi. (maximum) to 400 n. mi. and to increase the inclination of the orbit from 26.3° to 28.5°. Using as a selected set of targets, the inertial velocity, flight path angle, and inclination associated with a 28.5° inclined 400 n. mi./45.6 n. mi. orbit at 340 000 feet, the OTV atmospheric flight regime (time below 400 000 feet) is on the order of 450 second duration. The OTV then coasts to apogee and maneuvers to raise apogee to approximately 400 n. mi.

The details associated with aerobraking/aeromaneuvering phase are more fully described in paragraph 2.5.

2.4.1.2.6 <u>OTV return to space station</u>.- This phase encompasses all events from OTV attainment of a low Earth parking orbit through the rendezvous and space station recovery of the OTV. While the technique; e.g., all-propulsive or aerobraking/aeromaneuvering, produces inherent differences in the manner in which the OTV achieves its LEO parking orbit, the concept for the OTV final return phase is based on a strategy of maximizing profile similarity between the two options.

To effect this objective, rendezvous phasing operations are initiated from the LEO parking orbit at the first available OTV crossing of the common node between the space station and OTV orbit planes. While the nominal mission can theoretically be designed such that the proper relative alinement between the OTV and space station orbit plane can be achieved, operational planning dictates that viable off-nominal conditions be accommodated. Scheduling the phasing maneuver at the common node provides a capability to account for OTV dispersions with minimized effect on performance, in that any out-of-plane maneuvers can be combined with the rendezvous in-plane maneuvers. See section 2.7 for more discussion.

It has been assumed that the OTV dwell a minimum of two revolutions in the low Earth orbit prior to initiation of rendezvous to support the required data acquisition, processing, and commanding activities. This will result in the OTV, prior to initiation of the rendezvous sequence, being in the LEO parking orbit approximately one-half revolution longer for the all propulsive option than for the aerobraking/aeromaneuvering option. The difference in LEO coast time occurs from the fact that the circularization point is characteristically 180° out-of-phase between the two options; the number of orbits measured from departure from geosynchronous orbit into the LEO transfer orbit remains constant.

2.4.1.2.6.1 <u>LEO parking orbit</u>

A. All-propulsive option: At the equatorial crossing, the OTV performs a maneuver which both nominally circularizes the OTV orbit at 400 n. mi. and changes the inclination from 26.3° transfer orbit to the space station orbital inclination of 28.5°.

It should be recognized the above targeting strategy is shown as baseline; the precise maneuver strategy and associated targeting criteria for insertion into the LEO parking orbit should be derived from a performance analysis based upon OTV GN&C capability and supporting flight software. Typically, what must be considered are

1. Performance implications of controlling the relative orbital plane (both inclination and node) with this maneuver as opposed to controlling only the inclination and deferring the nodal adjustment for incorporation into the rendezvous sequence.

2. Resultant parking orbit orbital period (and impact on rendezvous time) from altitude dispersions.

3. Performance implications associated with the control of post-LEO insertion orbital period.

4. The assumed OTV operations scenario and targeting limitations.

Following a nominal coast in the 400 n. mi. circular LEO parking orbit the OTV arrives at the common node between the OTV and space station orbits and the rendezvous sequence associated with the OTV return to the space station is initiated.

B. Aerobraking/aeromaneuvering option: Upon arrival at apogee of the post-aerobraking/aeromaneuvering orbit the OTV then performs a maneuver to raise perigree from the reentry altitude of approximately 46 n. mi. to 400 n. mi. Although this maneuver is planned to theoretically result in a circular orbit, the inaccuracies in the control during the aerodynamic flight will result in apogee altitudes dispersed from the planned 400 n. mi.; therefore, the exact maneuver strategy and targeting criterion options that are available will need to be developed and subjected to future analysis. In the interim, the assumption for the present profile is that the OTV targets this maneuver to achieve a predetermined altitude; i.e., 400 n. mi.

2.4.1.2.6.2 <u>OTV/space station rendezvous</u>: The maneuvers associated with rendezvous are discussed in detail in paragraph 2.6. While the maneuver sequence and strategy is independent of the return option; i.e., all-propulsive aerobraking/aeromaneuvering, the specific profile data are not. This is due to two factors: (a) the profile time lines between the two options differ and (b) the all-propulsive simulation assumed perfect guidance, navigation, and control for the OTV with respect to being able to achieve the desired orbital targets while the aerobraking/aeromaneuvering simulation utilized a preliminary concept for a control algorithm and actually flew an aerobraking/ aeromaneuvering trajectory that resulted in actual dispersions of the desired orbital targets.

For the profile contained in this document the worst case phasing situation for the time line associated with the all-propulsive option was assumed. (This worst phasing is for the OTV to lead the space station by 72° at the time the OTV injects into the LEO transfer orbit). For the aerobraking/ aeromaneuvering return option, the space station and OTV geosynchronous state vectors used for the all-propulsive option remained the same. The differences in trajectory between the two return options resulted in different phasing conditions at the time the rendezvous sequence is initiated.

Due to a combination of: (a) trajectory difference between the all-propulsive option and aerobraking/aeromaneuvering option and (b) the difference in the location of the common node; e.g., time of the phasing maneuver between the above options, the rendezvous for the all-propulsive option (worst case) required 32 hours from GEO departure until achieving the stable orbit whereas the aerobraking/aeromaneuvering option required 13 hours.

2.4.1.3 Payload Retrieval From Geosynchronous Orbit

Typical flight profiles and ΔV requirements for this class of missions are summarized in the sequence of events and ΔV tables (tables 2-III and 2-IV). Table 2-III(a) reflects the option for an all-propulsive OTV return to the space station; the aerobraking/ aeromaneuvering return option is shown in table 2-III(b). The profile difference between this retrieval mission and the delivery mission described in paragraph 2.4.1.2 occurs in the events between the establishment of the geosynchronous drift (phasing) orbit and the final geosynchronous orbit. These two phases of the profile are illustrated in the orbital diagrams in figures 2-10(a) and 2-10(b). The remainder of the profiles are as previously illustrated in figures 2-4.

2.4.1.3.1 Deployment and separation phase.- The strategy, sequence of events and timeline for this phase of the payload retrieval mission are identical to those described for the payload delivery mission (paragraphs 2.3 and 2.4.1.2.1). It should be noted that the above is valid in its entirety only if the payload is performing precise north-south stationkeeping in geostationary orbit; e.g., equatorial orbit. If the payload is permitted to drift in a north-south direction and/or it was not precisely placed in an equatorial orbit the strategy must be slightly modified to accommodate the more general situation. The required modification is to deploy in such a manner that sets up the geosynchronous transfer point approximately at the common node between the space station orbit plane and the desired geosynchronous plane.

In order to provide a capability to retrieve a geosynchronous payload in an inclined orbit at anytime throughout the year the OTV performance should be sized for the maximum outbound plane change anticipated. In this instance, the 28.5° is not the maximum but rather an increment must be added equivalent to the design-to-inclination of the geosynchronous payload orbit.

2.4.1.3.2 Geosynchronous transfer phase.- This phase for the geosynchronous payload retrieval mission is identical to that described for the delivery mission in paragraph 2.4.1.2.2.

As noted in the preceding section, consideration of a more generalized payload orbit inclination (nonzero) would have the effect of locating the maneuvers at the common node rather than the equator.

2.4.1.3.3 <u>Geosynchronous phasing</u>.- In a manner very similar to drifting to the desired longitude for the delivery mission (paragraph 2.4.1.2.3) the OTV performs a maneuver at the equator (common node) that nominally establishes a geosynchronous drift (phasing) orbit coplanar with the orbit of the payload. Initiated one-half revolution after insertion into the geosynchronous transfer orbit, this drift orbit serves to provide the mechanism for establishing a catchup to the target payload within controllable bounds.

Considerations associated with rendezvous that must be accounted for in the method of flight implementation are

A. Variable phasing conditions.

B. The time constant associated with maneuvering in both the geosynchronous transfer and geosynchronous orbits.

C. Magnitude and direction of required maneuvers.

D. Partitioning of inertial navigation and relative navigation phases and associated onboard sensor requirements.

E. Performance optimization.

F. Trajectory sensitivities and correction opportunities.

The implementation strategy employed for delivery of a payload to a specified longitude (paragraph 2.4.1.2) appears to be a candidate option that also has potential for favorably satisfying the above requirements for a non-time critical rendezvous. Specifically

A. The capability is provided to limit the phasing regime that must be accomodated by controlling the phasing envelop to within 25° prior to committing the OTV to free-flight.

B. The time required in the phasing orbit can be limited to one revolution (24 hours) for the phase range considered within minimal performance impact.

C. By limiting the phasing range and providing one full orbit in the phasing orbit, the out-of-plane motion can effectively be removed; thus, permitting a bounded maneuver in terms of magnitude and direction.

D. Targeting for a position which is offset from the payload consistent with OTV onboard tracking can readily be accommodated without perturbation to the previous criteria.

E. The performance optimization strategy for a general geosynchronous transfer can be largely employed without encountering a performance dependence upon initial relative conditions.

F. The opportunity for phasing and plane change corrections is afforded approximately one-half orbit after insertion into the geosynchronous phasing orbit.

To accommodate a 25^{o} phasing (OTV trailing) the OTV inserts into a 19 323-by-17 200 n. mi. phasing orbit with zero degree inclination. This maneuver is a combined phasing and plane change and is designed to place the OTV 50 n. mi. ahead of the payload and at the same attitude in one orbit. Unlike the delivery mission, the payload retrieval requires additional catch-up of approximately 0.1^{o} in order that the OTV establish the leading relative position.

Approximately one-half orbit following insertion into the geosynchronous phasing orbit, the OTV performs a midcourse correction if required. Finally, after one revolution in the geosynchronous phasing orbit, the OTV initiates the payload rendezvous phase.

The relative motion between the OTV in the geosynchronous phasing orbit and the geosynchronous payload very closely parallels to figure 2-7 which shows the relative motion with respect to the desired geosynchronous placement. The slight difference involved is due to a small change in orbital period required to attain the 50 n. mi. offset position; graphically, for all practical purposes, the origin of figure 2-7 can be shifted 50 n. mi. to the right and the resulting motion profile be indicative of an initial phasing approximately 0.1^{o} less than indicated.

2.4.1.3.4 OTV/payload rendezvous phase.- Following one orbit coast in the geosynchronous phasing orbit, the OTV initiates the set of maneuvers and associated onboard navigation to effect rendezvous with the payload. These maneuvers consist primarily of the OTV establishing an intercept trajectory with that of the payload, maintenance of the intercept trajectory through the implementation of a midcourse correction strategy, and braking (nulling the relative velocity between the OTV and payload by the time the OTV arrives at the payload) according to a predetermined control strategy of the approach corridor relative conditions. Additional maneuvers for proximity operations such as stationkeeping, fly around, docking alinement, etc., following the OTV arrival at the payload will also be required.

To establish the intercept trajectory, the following relative conditions were the assumed nominal.

A. The OTV is 50 n. mi. ahead of the payload.

B. The OTV is at the same altitude as the payload.

C. The OTV orbit and payload orbit are coplanar.

D. The relative velocity between the OTV and payload (prior to intercept initiation) is a variable, depending upon the phasing that existed at the time the geosynchronous phasing orbit was established.

It should be noted that the preceding relative conditions not only influence the intercept trajectory during the rendezvous phase but they also reflect back on the geosynchronous phasing orbit in terms of the maneuver targeting requirements (including midcourse) for that phase. The 50 n. mi. distance is a present assumption which will be subject to modification as the detailed OTV systems and operations mature. Primary factors that will govern the selection of the nominal aim point will be the accuracy with which the OTV can attain the desired relative position; the capability of the OTV onboard tracking sensors; trajectory sensitivities to ground and onboard system performance; performance impact to recover from the expected trajectory dispersion envelop; operational support requirement; etc.

Targeting the OTV ahead of the payload prior to initiating the intercept may be more optimum in terms of performance than targeting for a trailing position if feasibility studies show that the OTV system can accommodate an intercept maneuver that is targeted and executed directly from a geosynchronous phasing orbit within imposed operational constraints. This strategy will nominally yield a savings in ΔV of approximately 165 fps for the worst case 25° phasing. If such strategy is not practical from operational considerations or OTV system constraints, the OTV would be required to attain what is tantamount to a stable orbit prior to initiation of the rendezvous intercept maneuver; thus any performance advantage would be lost. Figure 2-11 shows the total nominal ΔV requirement to rendezvous as a function of phasing and the approach quadrant. The targeting for the intercept maneuver was selected such that the time for OTV rendezvous with the payload was one hour (approximately 15° of central angle at geosynchronous altitude). Over the transfer time, the relative motion between the OTV and payload is essentially linear with respect to an inertial reference frame. This is shown in figure 2-12(a). Also, with respect to an inertial reference, a nadir pointing payload will rotate only 15°. The relative motion is also shown in a rotating local vertical reference frame in figure 2-12(b).

It appears reasonable to expect that rendezvous and docking in geosynchronous orbit would be an easier task than in low Earth orbit.

2.4.1.3.5 Geosynchronous operations.- Following retrieval of the payload by the OTV for return to the space station, the OTV is assumed to be in a state of readiness for LEO transfer orbit insertion. By virtue of the planning strategy employed, a wait of approximately 10 hours following rendezvous will be required for the first available nodal opportunity to initiate the descent. No geosynchronous operations, other than the rendezvous and docking are assumed to be required for the retrieval mission. Unlike the delivery mission described in paragraph 2.4.1.2 that required a pre-LEO transfer orbit separation maneuver, the LEO transfer orbit for the retrieval mission is initiated directly from the circular geosynchronous payload orbit.

2.4.1.3.6 Low Earth orbit return.- The return operations of the OTV from geosynchronous orbit for this mission is as described in paragraphs 2.4.1.2, 2.5, and 2.6 for the OTV return phase of the delivery mission. The specific profile has been designed to accommodate the specific phasing conditions.

2.4.1.4 Servicing Geosynchronous Satellite Cluster

This mission is fundamentally identical to the retrieval mission (paragraph 2.4.1.3) with the modification that: (a) subsequent to the initial rendezvous, additional geosynchronous phasing segments are inserted to permit rendezvous with multiple payloads and (b) prior to LEO transfer insertion an OTV separation maneuver will be required. This is also applicable following each servicing operation; however it is expected to be a reasonable assumption that this separation maneuver can be incorporated as part of the required phasing. Unlike the strategy emloyed for the initial rendezvous, the phasing cannot be bounded to the 25° for the following rendezvous in as much as that phasing is established by the relative spacing between the payloads in geosynchronous orbit; thus for a given relative phasing between payloads, the only tactic available is to trade off propellant with time spent in the phasing orbit. In lieu of orbital diagrams to depict this series of operations, a generic (typical) relative motion profile is presented in figure 2-13 to convey, pictorially, the profile events. Figure 2-14 shows the ΔV trade-off with time spent in the phasing orbit. For information purposes, should a constraint be imposed that the phasing orbit altitude not exceed the worst case for the initial rendezvous; e.g., 25° per orbit catch-up, 8 days (8 orbits) would then be required to be spent phasing to rendezvous with a payload 180° away.

2.4.2 OTV Planetary Mission Support

Utilization of a space station based OTV and a S/C kickstage module in support of a typical Mars mission profile is addressed. The OTV provides part of the impulse to achieve the desired escape trajectory by placing the S/C and kickstage on a high elliptic orbit and is then returned to the space station. The final impulse is provided by the S/C kickstage module after one revolution in the interim elliptic orbit.

2.4.2.1 Planetary Mission Support Requirements

The direction of the Earth escape trajectory for a planetary mission is defined by the velocity-at-infinity vector ($\vec{V}\infty$) while the required injection energy is specified in terms of C_3 - the magnitude of $\vec{V}\infty^2$. Optimum injection conditions occur when the plane from which the OTV would depart contains the $\vec{V}\infty$ resulting in a coplanar departure. Injection from any orbit plane which does not contain the $\vec{V}\infty$ will result in a performance penalty because a plane change will be necessary to satisfy the proper escape conditions.

The required injection energy, C_3, changes with time and generally, planetary missions are planned within a time frame of minimum C_3. This time frame (e.g., planetary launch window) is governed by the amount of vehicle performance available. Typically, these injection windows can be from 20 to 40 days (or more) with the minimum C_3 requirement often occurring near the middle of the window.

Previous Earth-originated planetary mission support provided the optimum departure conditions for each daily window by launch time selection. This made use of the Earth rotation to achieve the desired orbit plane orientation in inertial space at a specified time. This freedom of orbit orientation will not be afforded when supporting a planetary mission from a space station since the space station orbit orientation will not be actively controlled and will be determined principally by the natural and uncontrolled forces acting on the space station over time. (The space station orbital maneuvering capability will be very limited).

Some flexibility, however, does exist that makes utilization of a space station base feasible for planetary mission support. The direction of the Earth escape or $\vec{V}_\infty$ is relatively constant across the launch period while the space station orbit (200 n. mi. at 28.5° inclination) is processing at a rate of ≃7.2° per day in the opposite direction of its orbital motion. This means that the space station orbit plane will precess a total of 360° every 50 days.

Assuming that the absolute value of the declination of the $\vec{V}_\infty$ (DLA) is less than the inclination of the space station orbit (28.5°), the $\vec{V}_\infty$ will lie in the space station orbit plane twice during a given 50 day period. At these times an inplane injection burn is possible. If injection occurs at any other time a plane change and hence a performance penalty is required to satisfy the required escape conditions.

If the |DLA| is greater than 28.5°, the space station orbit plane will never contain the $\vec{V}_\infty$ and a minimum out-of-plane condition exists only once every 50 days.

From these considerations it becomes obvious that rarely will the space station be alined optimally (inplane injection with minimum C_3) to support a planetary mission and a trade off exists between the plane change required and the C_3 required for off optimum launch dates. Given a reasonable injection period (e.g., 20 to 40 days) however, and an OTV of related performance capability, conditions at times will be favorable to support some classes of planetary missions from a space station base.

2.4.2.2 Mission Guidelines and Assumptions

A. The assumed energy requirements represent a typical Mars mission.

B. The assumed enery (C_3) was 12 km^2/sec^2 which equates to an injection ΔV of 12 000 fps.

C. The OTV provides approximately 9000 fps of the required energy. A spacecraft kickstage module provides the remaining 3000 fps impulse following separation from the OTV.

D. The OTV separates from the spacecraft within the first half orbit following OTV burn termination; the spacecraft kickstage provides the final impulse after a one orbit coast.

E. The declination of the $\vec{V}_\infty$ (DLA) is +15° placing the optimum perigee location of the escape trajectory and hence the OTV and kickstage burn locations near the equator.

2.4.2.3 Deployment and Separation Phase

The nominal ignition time for the OTV injection burn is dictated by the $\vec{V}_\infty$ and associated C_3 required for the proper escape conditions. These escape conditions are achieved by a combination of the OTV burn and the final S/C kickstage module.

The OTV/payload deployment time from the space station should be planned prior to the selected OTV ignition time allowing sufficient time for systems tests, achieving safe separation from the station, and other functions that may be required.

Considering the aforementioned trade-off between plane change and off optimum injection energy requirements inherent in the utilization of a space station for planetary mission support, it would appear desirable to design the total system to accommodate several possible backup deployment opportunities and/or OTV injection opportunities. Since the space station orbit precession rate of 7.2°/day translates to a ≃.5°/revolution, the performance penalty for providing several consecutive orbital injection opportunities would be relatively small when combined with the total OTV ΔV required.

Details for a typical OTV/payload deployment and separation sequence from the space station is contained in section 2.4.

2.4.2.4 OTV/Payload Transfer to Interim Ellipse

The initial OTV burn ($\Delta V \approx 9000$ fps) places the OTV/payload on a high (apogee ≃36 450 n. mi.) interim ellipse. The orientation of this ellipse and hence the OTV burn location is dictated by the target parameters necessary to achieve the desired arrival conditions at Mars. The perigee location of the ellipse is chosen to optimize the S/C kickstage performance in achieving the final hyperbolic escape trajectory. A schematic of this portion of the mission is provided in figure 2-15(a).

2.4.2.5 OTV Separation From Payload/OTV LEO Return

Following injection into the high interim ellipse, the OTV trajectory support to the combined planetary spacecraft/kickstage module is terminated. The OTV will then be required to maneuver from this planetary injection assist orbit to low Earth orbit and rendezvous with the space station. The trajectory requirements to be satisfied by OTV maneuvering prior to reaching low Earth orbit are: (a) establishing the proper perigee altitude consistent with requirements for either the all propulsive or aerobraking return option and (b) establishing the proper orbital plane with respect to the space station.

2.4.2.5.1 <u>OTV nodal alinement/plane change correction maneuver</u>.- Prior to OTV rendezvous with the space station, any nodal misalinement and/or inclination difference between the OTV orbit plane and the space station plane must be corrected. The nodal misalinement is caused by the differential nodal precession rates between the two orbit planes while the inclination difference would appear if a plane change during the OTV injection burn were required.

Based on the assumption that a coplanar OTV injection burn was possible, this specific profile does not reflect a more generalized OTV plane change requirement. This requirement, however, should be given consideration for further flight design activities and systems analysis trade studies.

As mentioned earlier the ignition time and position (declination) of the OTV injection burn and hence the line of apsides of the transfer ellipse is dictated by the required V_∞ and C_3. This also defines the line of intersection of the two planes and therefore the two common nodal positions (180° apart) where the nodal correction maneuver must be performed.

The optimum place to perform this maneuver would be at the common node of the two planes that is closest to apogee of the transfer ellipse since the velocity of the OTV is less at that point and hence, the ΔV required to change direction is less. This optimum position could either be on the outbound leg or the return leg of the transfer ellipse depending on the declination of the OTV injection burn.

Assuming that OTV separation from the S/C/kickstage occurs shortly before the first apogee of the transfer ellipse, the OTV would be free to perform this alinement maneuver if the optimum maneuver time occurs on the return leg. If the separation time from the OTV is constrained by S/C and/or kickstage requirements on the outbound leg, however, the OTV may not be free to perform this maneuver at the optimum time on the outbound leg. This would probably result in a significant ΔV penalty and/or impact on the OTV stay time in the transfer ellipse. It appears feasible that a second revolution in the transfer ellipse might be required to accommodate this situation. Another alternative would be to perform this maneuver after the OTV has returned to low Earth orbit, however, this may also result in a performance penalty. It is not the intent of this document to address in detail all of these considerations but to indicate areas that need detailed attention in future trade studies.

As mentioned in the guidelines and assumptions, for development of the mission profile described herein, the OTV injection burn was located at the equator. This places the common node of the OTV transfer orbit plane and the space station orbit plane approximately $\pm 90^\circ$ from the perigee position (true anomaly). Assuming the OTV has the flexibility to perform this maneuver on either leg of the transfer ellipse, this represents a "worst" case in terms of the ΔV required to correct for the nodal misalinement. This also results in a very short time between this maneuver and the OTV circularization maneuver for the all propulsive case (see table 2-V(a); between this maneuver and the entry interface point for the aerobraking case (table 2-V(b)).

2.4.2.5.2 <u>OTV all-propulsive return</u>.- Prior to arrival at apogee of the interim elliptic orbit, the OTV separates from the spacecraft/kickstage. Separation is planned to occur at a time that insures the OTV is able to meet an apogee arrival ignition time while also providing adequate distance between the OTV and spacecraft. At apogee, the OTV adjusts perigee to the prerendezvous LEO parking orbit altitude of 400 n. mi.

Predicated on the assumption that OTV separation from the spacecraft occurs beyond the first opportunity to perform the return plane change (which would occur prior to apogee arrival for this assumed mission profile), the required plane change is made at the second opportunity (second crossing of the common node between the interim ellipse and space station parking orbit). This results in the plane change being performed after the perigee adjust maneuver. Upon reaching perigee, the OTV establishes a prerendezvous LEO parking orbit, after which the rendezvous sequence (see paragraph 2.6) will be initiated. Table 2-V(a) contains the sequence of events for this mission profile and figure 2-15(a) provides a graphic illustration.

2.4.2.5.3 OTV aerobraking return.- The strategy described in paragraph 2.4.2.5.2 (OTV all-propulsive return) applies to the aerobraking return option as well. The significant difference between the two options lies in the altitude adjustment maneuver targeting performed at apogee of the interim planetary injection ellipse. Rather than raising perigee to 400 n. mi. as was required for the all-propulsive option, perigee is targeted to provide the proper conditions for entry into the atmosphere (see paragraph 2.5). The result is that the OTV will be required to achieve a vacumn perigee altitude of approximately 45 n. mi.

During the aerodynamic flight regime, energy is dissipated to the level which lowers apogee to 400 n. mi. upon exiting the atmosphere. After coasting for approximately one-half orbit (apogee arrival) the OTV circularizes into the prerendezvous LEO parking orbit. The remainder of the rendezvous operations are described in paragraph 2.6. Table 2-V(b) contains the sequence of events for this mission profile while figure 2-15(b) provides a graphic illustration.

2.4.3 Sun-Synchronous Mission

Guidelines and assumptions used in developing the Sun-synchronous mission profile are as follows:

A. The Landsat orbit is assumed for the sun-synchronous satellite target; the orbit parameters are: 383 n. mi. altitude, 98.2° inclination, and a 0930 local time descending node crossing.

B. The aerobraking perigee altitude is 45 n. mi. allowing one-pass braking to desired apogee altitude.

C. The OTV separation sequence assumed is that described in section 2.3.

The flight design rationale for three different profile options are described: (1) an all propulsive maneuver sequence, (2) an all-propulsive outbound flight phase with use of aerobraking in the return segment, and (3) use of aerobraking in both the outbound and the return segments. The basic maneuver strategy applied in these mission profiles to minimize OTV energy requirements is to transfer the OTV into a very high apogee (approximately 100 000 n. mi.) ellipse, where, at apogee, the large plane change (28.5° to 98.2° inclination) is performed; this is followed by a transfer down to low orbit. Further propulsion requirement reduction can be realized by bringing the OTV down from the high ellipses by allowing a low-perigee (45 n. mi.) passage with aerobraking for energy dissipation. Comparison of energy requirements for the all propulsive versus one and two aerobraking segments can be made from table 2-VIII.

2.4.3.1 All-Propulsive Mode

A detailed sequence of events for this mode is given in tables 2-VII(a), 2-VII(b), and 2-VII(d). The profile is pictorially represented in figure 2-16(a). The separation sequence of the OTV from the space station is described in section 2.3.

The amount of plane change required to transfer from the space station orbit with an inclination of 28.5°, to the Sun-synchronous orbit with an inclination of 98.2°, can vary from 69.7° to 126.7° depending on the alinement of the ascending nodes of the two orbits. This profile assumes a transfer at the best (minimum ΔV cost) nodal alinement. This occurs when the ascending node of the OTV at departure coincides with the ascending node of the Sun-synchronous orbit at the time of arrival. This alinement occurs once every 43 days. In order to determine the precise location of the node between the initial OTV orbit and the Sun-synchronous orbit at the rendezvous point, the differential nodal regression between the orbits during the transfer phase must be applied. This differential nodal regression is applied to the node of the Sun-synchronous orbit at the time of final circularization to obtain a phantom target plane. The node between the OTV plane and the phantom plane determines the injection point. At this node a large Hohman transfer is performed by the OTV to raise apogee from 210 n. mi. to approximately 100 000 n. mi. The coast to apogee requires approximately 40 hours. At apogee the OTV performs a maneuver to rotate its plane into that of the phantom target plane. The out-of-plane component to effect this plane change is only 1262 fps. This apogee maneuver also has an inplane component to raise the transfer ellipse perigee to 583 n. mi., or 200 n. mi. above the target orbit. Approximately 41 hours later, the OTV arrives at perigee, where a Hohman retrograde transfer maneuver is performed to lower apogee from nearly 100 000 n. mi. to 583 n. mi. The OTV is now in a circular orbit 200 n. mi. above the target Sun-synchronous orbit.

As mentioned above, the transfer orbit plane change was executed such that the wedge angle (angle between the OTV and target orbit planes) is near zero at the time of the rendezvous plane change; therefore, the two orbits are essentially coplanar, with the OTV orbit approximately 200 n. mi. above that of the target spacecraft. A detailed description of the rendezvous phase is discussed in section 2.6. The outbound phase of the mission requires approximately four days and four hours.

For the return flight phase, it is necessary to find the node between the sun-synchronous orbit at departure and the space station orbit at the time of OTV rearrival. For this profile more than eight days have elapsed between the time of departure and the time of return to the space station. Thus the ascending nodes of the Sun-synchronous and space station orbits have changed to a non-optimum alinement. The resulting plane change for the return phase is 84.7° as opposed to 69.7° for the outbound phase. Targeting for the plane change for the return is similar to the outbound targeting. The differential nodal regression during this flight phase is applied to the node of the space station at the time of circularization to obtain a phantom target plane. The node between the Sun-synchronous orbit and the phantom plane determines the injection point. The return phase of the mission is initiated at this injection point to place the OTV into a 100 000 n. mi. by 383 n. mi. ellipse.

As in the outbound leg, the apogee of the high ellipse is utilized to perform the large plane change. Again, the maneuver is positioned such that the plane change will take out the wedge angle for the return phase. The apogee maneuver also raises the perigee to 400 n. mi., 200 n. mi. above the space station orbit. At perigee, the OTV circularizes its orbit at 400 n. mi., and the standard rendezvous phase described in section 2.6 is carried out. The return phase of the mission requires approximately 4 days 14 hours for a total round trip flight time of approximately 8-3/4 days. The total ΔV required for this all propulsive mission is roughly 42 700 fps.

2.4.3.2 All-Propulsive Outbound, Aerobraking Return Mode

A detailed sequence of events for this mode is given in tables 2-VII(a), 2-VII(b), and 2-VII(e). The profile is pictorially represented in the left portion of figure 2-16(a) for the outbound part of the mission and the right half of figure 2-16(b) for the return sequence. This mission type is identical to that of the all-propulsive mode except for the return leg, where aerodynamic drag is utilized during a low perigee pass for lowering the high apogee of the return transfer orbit. In this case, the apogee plane change is combined with an inplane component to lower perigee to 45 n. mi. The subsequent perigee pass effectively reduces the 100 000 n. mi. apogee to 400 n. mi. One-half orbit later, the OTV performs a circularization maneuver to produce a 400 n. mi. circular orbit. From this point the OTV/space station rendezvous phase (section 2.6) is carried out. The total ΔV requirement for this mission type is approximately 33 700 fps. Total mission duration is approximately one hour longer than the all-propulsive mission due to the extra half orbit required for the circularization at 400 n. mi.

2.4.3.3 Aerobraking Outbound, Aerobraking Return

A detailed sequence of events for this mode is given in tables 2-VII(a), 2-VII(c), and 2-VII(e). The profile is pictorially represented in figure 2-16(b). This mission type utilizes the aerobraking technique for both the outbound and the return phases of the mission. The outbound apogee plane change/perigee adjust maneuver lowers the OTV orbit perigee to 45 n. mi. A single perigee pass lowers the apogee to 583 n. mi. (200 n. mi. above the Sun-synchronous spacecraft orbit). The OTV orbit is circularized at the first apogee, and the rendezvous sequence outlined in all-propulsive mission description is executed. The aerobraking return leg was described previously. The total ΔV requirements for this mission type is approximately 24 400 fps.

2.4.4 Co-Orbiting Satellite Support Mission

2.4.4.1 Co-Orbiting Satellite Concept And Requirements

This mission class characterizes the application of a space station system to support payloads that are detached from the space station when in operation but: (a) require frequent revisits and/or return to the space station for recovery of experiments, resupply, etc; and (b) require redeployment and resumption of operations following a period of payload turn-around activity onboard the space station. Support to this class of missions infer certain operational capability (requirements) and implementation.

The need for frequent revisits mandates readily available access to the payload. This in turn, implies standardized operations. From these criteria, the following requirements can be extracted:

A. The payloads must be operated within line-of-sight of the space station for communications for normal operation.

B. The maximum (limiting) distance between the payload and space station for direct line-of-sight is approximately 2400 n. mi.

C. A space based vehicle capable of providing the required transportation services must be available.

D. Continuous communication for the required and control of the transport vehicle must be provided.

E. The payload position relative to the space station must be maintained within limits; the relative position between payloads must be maintained; the relative position of the centroid of the "nest" of co-orbiting satellites with respect to the space station must be maintained.

F. The co-orbiting satellite nest location must provide: an open corridor for vehicles that are arriving and departing the space station.

To satisfy the above, the concept is to place satellites in orbits which possess the following characteristics:

A. Have a period equal to that of the space station.

B. Result in a relative motion envelope with respect to the space station sufficient to accommodate spacing requirements for co-orbiting multiple coplanar payloads.

C. Are inclined with respect to the space station orbit plane (and also inclined relative to the orbit planes of other satellite nests) for maintaining an open approach and departure corridor to the space station.

D. Have the centroid of the relative motion envelope such that all points along the relative trajectory are within line-of-sight.

The geometry of the co-orbiting nest and space station is shown in figure 2-17. These out-of-plane satellites will "orbit" about the cluster centroid, leaving the approach and departure corridors open for incoming and outgoing rendezvous vehicle; e.g., space shuttle, OTV, etc. The clustering geometry allows for a large number of orbiting satellites within reasonable ranges of the space station for visits and retrievals.

2.4.4.2 Mission Guidelines and Assumptions

A. A space station based maneuvering system, or ferry vehicle (FV), separate from the OTV is used to transport co-orbiting satellites to and from the space station.

B. The co-orbiting satellite support mission consists of providing the following services:

   1. Rendezvous and dock with a specific satellite located within a "nest" or cluster of satellites.

   2. Return a satellite to the space station.

   3. Transport a satellite to a specific location within a specific cluster of satellites.

C. The nested satellites have stationkeeping capability to maintain the cluster.

D. The propulsion system to be used for co-orbiting satellite support mission maneuvers is assumed not to require large separation distances prior to engine ignition.

E. The assumed distance between the centroid of the satellite cluster and the space station is 320 n. mi.; the cluster envelope, with respect to its centroid, is assumed to be $\pm$ 20 n. mi. downrange and $\pm$ 10 n. mi. along the radius.

2.4.4.3 Co-Orbiting Satellite Rendezvous

The scenerio for the co-orbiting satellite mission is pictorially represented in figures 2-18(a) and 2-18(b). The ferry vehicle (FV) is deployed from the space station with an initial separation impulse; this impulse may be imparted by the deployment system or provided by the vehicle itself. When sufficient distance is gained to satisfy space station contamination constraints, the FV performs 22 fps retrograde maneuvers (PET of 0). This height maneuver lowers perigee to 190 n. mi. (Δh = 10 n. mi.). Approximately 45 minutes (1/2 orbit) later, a co-circular maneuver is performed to produce an FV orbit which is 10 n. mi. below that of the target nest centroid. The burn is retrograde and provides 24 fps ΔV. For the next five hours (approximately), the FV coasts closing the separation distance to the nest. At approximately 5 hours, the 45 minutes PET, the FV performs a horizontal, posigrade maneuver of 36 fps. This maneuver, performed in the plane of the space station and nest orbits, is at a point below the nest centroid and raises the FV apogee to 210 n. mi., which is also the apogee of the target satellite "orbiting" the nest centroid.

The assumed orbit for the specific target satellite is slightly out of the space station/nest plane, as described pictorially in figure 2-17. At 6 hours, 8 minutes PET, the FV crosses ahead of the cluster centroid orbit. At this point a combined plane change and terminal phase initiation (TPI) maneuver of 32 fps is performed to (1) rotate the FV plane into that of the target satellite and (2) initiate an intercept with the satellite approximately 33 minutes later. During the terminal phase approach, the FV performs midcourse corrections as necessary to improve the intercept. At 6 hours, 41 minutes PET, the FV performs a 10 fps braking maneuver to attain a stable orbit offset to the target satellite. The FV then approaches and docks with the target.

2.4.4.4 Space Station Return

The return of the FV (with the docked satellite) to the space station is assumed to be initiated at the next crossing of the space station altitude (see figure 2-18(b)) ahead of the centroid. At this point, at a PET of 7 hours, 38 minutes, a plane change (NPC) maneuver is performed (ΔV = 32 fps) to rotate the plane of the FV from the target satellite orbit back into the space station orbit plane. At 8 hours PET, the FV at a Δh of 10 n. mi. above the cluster centroid and space station, executes a horizontal, posigrade maneuver (ΔV + 24 fps) to produce a 210 n. mi. cocircular orbit. This results in a constant-height closing rate toward the space station. At 13 hours PET, the FV executes an intercept maneuver (TPI) of 22 fps. Following midcourse maneuvers (as required) during the intercept transfer, the FV arrives at an offset point, ahead of the space station. At 13 hours, 32 minutes PET, the FV performs a 24 fps braking maneuver, which results in an offset, stable orbit position several thousand feet ahead of the space station. The FV then initiates final closure with the space station.

2.4.4.5 Co-Orbiting Satellite Return To The Nested Location

Fundamentally, the operations associated with the return and redeployment of the co-orbiting satellite to its original nested location and the subsequent return of the FV to the space station are virtually identical to the concept for the retrieval and the space station return as discussed in paragraphs 2.4.4.3 and 2.4.4.4.

While this mission does not require a rendezvous and docking with a co-orbiting satellite, it does require the FV to rendezvous with a point in space; i.e., a "phantom" rendezvous, in order to preserve its nested relationship to other co-orbiting satellites. Therefore, in order to redeploy the co-orbiting satellite, the FV will be required to satisfy the classical rendezvous constraints; i.e., achieving a specified orbit plane, and arriving at a specified position in that plane at a specified time.

Although a specific sequence of events for this mission requirement is not specifically shown, figure 2-18 effectively "mirrors" the operations to be performed.

2.5 OTV AEROBRAKING/AEROMANEUVERING PHASE

The OTV maneuver requirements for the aerobraking/aeromaneuvering phase have been described in paragraph 2.4. The following paragraphs describe the flight segments associated with aerobraking/aeromaneuvering, the guidance logic used for the aerobraking/aeromaneuvering task, and a flight phase analysis employing this logic.

2.5.1 Aerobraking/Aeromaneuvering Flight Segments

The aerobraking/aeromaneuvering phase consists of the three following flight segments: the initial atmospheric penetration segment; the high-speed equilibrium glide segment; and the exit control segment.

2.5.1.1 Initial Atmospheric Penetration Segment

The function of the initial atmospheric penetration segment is to guide the OTV to a trajectory that will cross or acquire the desired value of reference acceleration in a minimum time and with a minimum overshoot of the reference acceleration. A trajectory that overshoots the reference value of acceleration during the initial penetration segment results in a higher peak load factor on the OTV.

2.5.1.2 High-Speed Equilibrium Glide Segment

The high-speed equilibrium glide segment is initiated on the basis of the drag acceleration of the OTV being equal to or greater than 95 percent of the reference value of drag acceleration. This occurs at the minimum altitude of the trajectory corresponding to maximum dynamic pressure and maximum surface temperature. (The reference drag acceleration which contains an added 1-g bias for controllability is based on a lift vector down attitude and is updated on a predetermined computational frequency). The equilibrium glide trajectory provides velocity dissipation in a regime where the aerodynamic loads and surface temperatures are decreasing and altitude is increasing.

2.5.1.3 Exit Control Segment

The exit control segment controls to the desired exit velocity and flight path angle. The reference drag acceleration for an equilibrium glide trajectory is abandoned for this segment. The reference drag acceleration now assumes a linear reduction in acceleration between the last value of reference equilibrium glide acceleration and an assumed value of zero at 340 000 ft altitude.

(NOTE: Several profiles were considered for the exit phase of the trajectory. To date, the linear profile has produced the most consistent results.)

2.5.2 OTV Atmospheric Guidance Logic

In order to achieve the aerobraking maneuvers described in paragraph 2.4, a guidance system is required which will control the OTV during its flight through the Earth's atmosphere. The function of the guidance logic is to control the reduction of the OTV's velocity and hence a lowering of apogee as the vehicle passes through the Earth's atmosphere. The OTV exits the Earth's atmosphere at a prescribed velocity, flightpath angle and orbital plane inclination. Vehicle control is obtained by lift vector modulation about the velocity vector.

Control during the initial atmospheric penetration segment is accomplished primarily by maintaining a lift vector down attitude until 0.05 g's. Lift vector modulation then is allowed to control the drag acceleration error and the altitude rate error during the initial penetration segment. Once the equilibrium glide condition has been established, the trajectory is controlled to a reference value of acceleration, DO, which is defined as:

$$DO = ETA1 + g$$

where g is an acceleration BIAS that is required to maintain vehicle control and is 32.2 f/sec^2.

The exit maneuver is initiated at a velocity of $VI_2 + 3000$ where VI_2 is the target exit velocity at 340 000 ft altitude. The reference drag acceleration profile, DO, for the exit phase assumes a linear reduction with velocity. This is defined as:

$$DO = DO_1 \quad (VI - VI_2)\ /3000$$

where DO_1 is the value of DO at the initiation of the exit phase.

The reference altitude rate, $\dot{R}_{n+1}$, during the exit maneuver is based on a linear variation as a function velocity; i.e., $(\dot{R}_2 - \dot{R}_1)/3000.$ where $\dot{R}_2$ is the target altitude rate at 340 000 ft and $\dot{R}_1$ is the reference value of altitude rate at the initiation of the exit phase.

The reference altitude rate during the exit maneuver is defined as:

$$\dot{R}_{n+1} = \dot{R}_n + (\dot{R}_2 - \dot{R}_1/3000.) \quad *CD1*\Delta t.$$

$\dot{R}_n$ is initialized to the reference altitude rate at the start of the exit phase. CD1 is the current value of actual drag acceleration and Δt is the integration time step used in the simulation programs.

Trajectory control is based on a reference value of L/D. The lift vector is modulated to control the desired vertical component of L/D. The resultant guidance logic is defined as:

$$\frac{(L)}{(D)}\ \text{command} = \frac{(L)}{(D)}\ \text{ref} + \frac{\delta(L)}{(D)}$$

where

$$\frac{\delta(L)}{(D)} = C16(CD1-DO) - C17(\dot{R} - \dot{R}_{n+1})$$

C16 and C17 are variable coefficients which are updated every guidance integration cycle. $\dot{R}$ is the current altitude rate.

2.5.3 Flight Phase Analysis

The entry trajectory is targeted to a velocity and flightpath angle at 400 000 ft altitude. The outbound state vector is targeted to a velocity, a flightpath angle, and an orbital plane inclination at 340 000 ft altitude.

This investigation assumed an L/D of 0.5 and a W/C_DA of 5.12 lb/ft^2. The guidance gains were adjusted for this vehicle configuration. Figures 2-19(a) through 2-19(k) present typical plots for an OTV atmospheric trajectory. Table 2-IX(a) presents the tabulated conditions for a 19 323 n. mi. x 45.6 n. mi. orbital ellipse. Table 2-IX(b) presents the tabulated conditions for 95 982 n. mi. x 41.0 n. mi. orbital ellipse and table 2-IX(c) presents the tablulated conditions for a 9662 n. mi. x 45.6 n. mi. orbital ellipse.

The results in table I are very good and represent the results of an atmospheric trajectory from a Sun-synchronous orbit to a low Earth orbit. The data in table II represent the results of an atmospheric trajectory from a five-geo type (95 982 n. mi.) orbit to a low Earth orbit. The atmospheric guidance functions over a wide range of L/D and W/C_DA. Satisfactory performance can be achieved by adjustment of guidance gains.

2.6 OTV/SPACE STATION RENDEZVOUS PHASE

The planning consideration, strategy, and maneuvers for OTV rendezvous with the space station are discussed in the paragraphs below. The philosophy is for the OTV to return to a nominal 400 n. mi. LEO parking orbit prior to initiation of the rendezvous sequence; therefore, this rendezvous phase will be common to all OTV missions described in paragraph 2.4.

2.6.1 Rendezvous Profile Planning Objectives

Complementing the fundamental requirement of bringing together the OTV and space station are operational objectives relating to the manner in which the requirement is to be satisfied. Those that were assumed are as follows:

A. Provide capability to accommodate any phasing between OTV and space station without incurring a performance penalty.

B. Minimize the impact upon OTV performance of orbit plane inclination and node errors.

C. Provide a minimum altitude separation between the OTV and space station orbits of 25 n. mi. during the phasing periods.

D. Limit the maximum rendezvous time to one day.

E. Attain and maintain (if required) an OTV 15 n. mi. standoff position trailing the space station prior to space station/OTV docking.

F. Provide capability to control the OTV approach corridor and accommodate the presence of co-orbiting satellite clusters ahead of the space station.

The standardization of the relative trajectory for the terminal rendezvous is a significant objective to be met. This is necessary to insure trajectory and timeline compatibility with vehicle navigation systems, communications, performance, etc.

Typical relative motion between the space station and OTV denoting the extremes in the OTV phasing orbit is shown in figures 2-20(a) and 2-20(b). These figures reflect, in time, the relative trajectory events occurring from the last revolution in the phasing orbit through rendezvous.

2.6.2 Rendezvous Sequence Maneuvers

2.6.2.1 Phasing Maneuver

Upon arrival at the common node designated for initiation of the rendezvous activities, the OTV performs the first in a sequence of maneuvers designed to establish a position at the same altitude as the space station but trailing the station by approximately 15 n. mi. (stable orbit). The initial maneuver, the primary purpose of which is to establish a trajectory segment that accomplishes the majority of phasing (catch-up) between the OTV and space station, is targeted within the following bounds:

A. All OTV maneuvers are to result in an OTV descent from 400 n. mi. (retrograde).

B. The minimum altitude for the phasing orbit (the maximum in-plane ΔV component) is 225 n. mi., 25 n. mi. above the space station orbit.

C. A plane change maneuver component, if required, is to be combined with the phasing maneuver such that the OTV is in-plane with the space station at the stable orbit position. The amount of plane change to be combined with the phasing maneuver -- in relation to the overall plane change requirement -- will be in proportion to the required in-plane maneuver requirement (the criteria to be determined).

The strategy underlying the above guidelines in general provides:

A. A propellant optimum set of maneuvers independent of the initial relative state between the OTV and the space station.

B. An altitude buffer between the OTV/space station/co-orbiting satellites prior to the OTV commitment and attainment of the stable orbit position to alleviate recontact concerns and the associated traffic control problems.

C. An optimum/near optimum method of reducing the performance impact due to misalinement of the space station/OTV orbit plane when properly apportioned among all the maneuvers in the rendezvous sequence.

The result of this strategy is that the minimum rendezvous time must be commensurate with both the phasing condition existing at the time the rendezvous sequence is initiated and also the relative catch-up rate that is available in an OTV phasing orbit ranging between 400 n. mi. circular and 400 n. mi. - by 225 n. mi. Figure 2-21 illustrates the relationship between the phase angle and the required phasing duration for phasing orbits to be limited to the above constraints.

2.6.2.2 Height Maneuver

The primary objective of the height maneuver, the second planned rendezvous sequence maneuver (excluding course corrections during the coasting phase) is to lower perigee of the OTV orbit to that of the space station altitude, 200 n. mi.

Applied at a point which is located inertially approximately 180° from the phasing maneuver, the height maneuver affords an opportunity to remove the remaining out-of-plane errors in an optimum/near optimum manner prior to arrival at the desired target point with respect to the space station. To accomplish this, the maneuver would be planned such that it occurs at the appropriate common node with the magnitude and in the direction consistent with attaining the desired orbital parameters.

Following this maneuver the OTV coasts approximately one-half revolution. After the coast, the OTV is then at the same altitude as that of the space station, in-plane, and nominally trailing the space station by 15 n. mi.

2.6.2.3 OTV Terminal Rendezvous Maneuvers

This series of maneuvers embraces all trajectory control requirements for recovery of the OTV from the point the OTV arrives at the desired stable orbit position. Such requirements evolve from: the deactivation and safing considerations of the OTV and/or payload prior to committing to recovery of the system; the operations considerations associated with the necessary reconfiguration and transfer of control of the OTV; traffic management around the space station, etc.

The third maneuver of the rendezvous sequence places the OTV into the same orbit as that of the space station, trailing by the 15 n. mi. This provides a nominally stable condition to perform the necessary reconfiguration functions prior to committing the OTV to a space station approach trajectory. A two-orbit stay at the stable orbit position has been arbitrarily assumed for purposes of this document.

Once the OTV has been "cleared" for recovery operations, a set of maneuvers are then performed to bring the OTV from its stable orbit position to the space station. The OTV initiates a maneuver to transition from the 15 n.mi. trailing displacement to a position approximately 1 n.mi. ahead from which standard terminal approach operations including OTV proximity operations would be performed. During this phase, midcourse maneuvers will be performed to control the trajectory to within prescribed limits. One orbit is allocated for the initial transition from a trailing to leading position and an additional one-half to one full orbit (or more) will be required for the final rendezvous operations.

TABLE 2-I.- SEQUENCE OF EVENTS FOR THE ORBITAL TRANSFER VEHICLE GEOSYNCHRONOUS DELIVERY MISSION

(a) All-Propulsive Return Option

Event no.	Event	PET d:h:m:s	Duration d:h:m:s	HA / HP (n.mi.)	Incl (deg)	Wedge[1] angle (deg)	Phase[2] angle (deg)	Pitch angle (deg)	Yaw angle (deg)	Mnvr ΔV (fps)	Comments
				*** Outbound Phase ***							
0	Initial space station orbit			200.0 / 200.0	28.50						
1	OTV performs separation mnvr	0:00:00:00		211.0 / 200.0	28.50	0.00	0.00	0.0	0.0	20.0	OTV performs a separation mnvr from the space station. DV should be sufficient to protect the station from all hazards associated with OTV deployment operations. Sep mnvr occurs at a nodal crossing.
2	Coast for four orbital periods		0:06:08:05	211.0 / 200.0	28.50						Coast duration was assumed for any postdeployment checkout, activation, etc. which may be required.
3	OTV insertion into geosynchronous transfer orbit	0:06:08:05		19323.0 / 200.0	26.30	2.2		0.0	-9.2	7944.8	Optimum height and plane change Mnvr.
4	Coast to equator		0:05:16:36								Coast approximately 1/2 Rev
5	Insert into longitude drift orbit	0:11:24:41		19323.0 / 17174.0	0.00	28.5		0.0	51.1	5598.2	Mnvr DV depends upon amount of longitude phasing. One rev. coast for correct longitude placement is assumed.
6	Coast to midcourse mnvr		0:11:07:53								Coast to the equator or 1/2 rev. after inserting into longitude drift orbit.
7	Perform midcourse mnvr if required	0:22:32:34		19323.0 / 17174.0	0.00						Midcourse performed if orbital dispersions require it.
8	Coast to geosynch altitude		0:11:07:53								

[1] Wedge angle: Angle between OTV and space station orbit planes.
[2] Phase angle: Measured from the OTV to the target (space station), positive in the direction of motion.

TABLE 2-I.- Continued

(a) - Continued

Event no.	Event	PET d:h:m:s	Duration d:h:m:s	HA / HP (n.mi.)	Incl (deg)	Wedge[1] angle (deg)	Phase[2] angle (deg)	Pitch angle (deg)	Yaw angle (deg)	Mnvr ΔV (fps)	Comments
9	Circularize in geosynchronous orbit	1:09:40:27		19323.0 / 19323.0	0.00			0.0	0.0	262.3	OTV attains the required longitude for deployment of the payload.
				**** Return Phase ****							
10	Initial geosynchronous orbit			19323.0 / 19323.0	0.0	28.50					OTV in geosynchronous orbit prior to low Earth orbit transfer.
11	Perform payload SEP mnvr			19323.0 / 19141.0	0.0	28.50		0.0	180.0	20.0	Retrograde separation mnvr from P/L. May occur anywhere WRT the LEO injection.
12	Initiate transfer to low Earth parking orbit	0:00:00:00		19141.0 / 400.0	26.30	2.55	-71.0	0.0	155.2	5781.4	TIG occurs 11.3 minutes prior to OTV's crossing the space station's orbital line of nodes. This time is dependent upon the rendezvous phasing.
13	Coast to the LEPO		0:05:17:32	19141.0 / 400.0	26.30						Perform a midcourse mnvr if required during coast.
14	Insert into the low Earth parking orbit	0:05:17:32		400.0 / 400.0	28.50	0.62	-84.5	0.2	-172.9	7614.8	TIG occurs at equatorial crossing.
15	Coast to phasing mnvr		0:04:34:12	400.0 / 400.0	28.50						
16	Perform rendezvous phasing mnvr	0:09:51:44		400.0 / 376.0	28.50	0.51	-1.4	0.0	180.0	41.2	TIG occurs at the first common node 3 revs. after LEO transfer injection.
17	Coast to height mnvr		0:20:45:06	400.0 / 376	28.50						
18	Perform height mnvr	1:06:36:50		376.0 / 200.0	28.50	0.03	-6.5	0.0	180.0	331.1	TIG occurs at the common node.
19	Coast to stable orbit insertion mnvr		0:00:48:20	376.0 / 200.0	28.50						Coast for a 180° transfer.

[1] Wedge angle: Angle between OTV and space station orbit planes.
[2] Phase angle: Measured from the OTV to the target (space station), positive in the direction of motion.

TABLE 2-I.- Concluded

(a) - Concluded

Event no.	Event	PET d:h:m:s	Duration d:h:m:s	HA / HP (n.mi.)	Incl (deg)	Wedge[1] angle (deg)	Phase[2] angle (deg)	Pitch angle (deg)	Yaw angle (deg)	Mnvr ΔV (fps)	Comments
20	Perform stable orbit insertion mnvr	1:07:25:10		200.0 / 200.0	28.50	0.0	0.25	0.1	180.0	295.0	Stable orbit insertion completed with the OTV trailing the space station by 15 n. mi.
21	Maintain stable orbit point		0:03:03:59								A two rev. coast assumed for required prerecovery operations.
22	Initiate terminal rendezvous mnvr sequence	1:10:29:09	0:01:31:56	200.0 / 197.0	28.50	0.00	0.25	0.0	-180.0	5.9	Begin final close in rendezvous sequence with the space station. Target for a point 1 n. mi. in front of the station.
23	Rendezvous operations	1:12:01:05	0:00:46:00		28.50	0.00	-0.02				Perform midcourse mnvrs as required.
24	PROX OPS, braking, and docking mnvrs	1:12:47:05		200.0 / 200.0	28.50	0.00	0.00			11.3	Mnvr DV is an accumulated DV for the rendezvous operations, PROX OPS, braking, and docking.

[1] Wedge angle: Angle between OTV and space station orbit planes.
[2] Phase angle: Measured from the OTV to the target (space station), positive in the direction of motion.

TABLE 2-I.- Continued

(b) Aerobraking/Aeromaneuvering Return Option

Event no.	Event	PET d:h:m:s	Duration d:h:m:s	HA / HP (n.mi.)	Incl (deg)	Wedge[1] angle (deg)	Phase[2] angle (deg)	Pitch angle (deg)	Yaw angle (deg)	Mnvr ΔV (fps)	Comments
				*** Outbound Phase ***							
0	Initial space station orbit			200.0 / 200.0	28.50						
1	OTV performs separation mnvr	0:00:00:00		211.0 / 200.0	28.50	0.00	0.00	0.0	0.0	20.0	OTV performs a separation mnvr from the space station. DV should be sufficient to protect the station from all hazards associated with OTV deployment operations. Sep mnvr occurs at a nodal crossing.
2	Coast for 4 orbital periods		0:06:08:05	211.0 / 200.0	28.50						Coast duration was assumed for any postdeployment checkout, activation, etc., which may be required.
3	OTV insertion into geosynchronous transfer orbit	0:06:08:05		19323.0 / 200.0	26.30	2.2		0.0	-9.2	7944.8	Optimum height and plane change mnvr.
4	Coast to equator		0:05:16:36								Coast approximately 1/2 rev.
5	Insert into longitude drift orbit	0:11:24:41		1923.0 / 17174.0	0.00	28.5		0.0	51.1	5598.2	Mnvr DV depends upon amount of longitude phasing. One rev. coast for correct longitude placement is assumed.
6	Coast to midcourse mnvr		0:11:07:53								Coast to the equator or 1/2 rev. after inserting into longitude drift orbit.
7	Perform midcourse mnvr if required	0:22:32:34		19323.0 / 17174.0	0.00						Midcourse performed if orbital dispersions require it.
8	Coast to geosynch altitude		0:11:07:53								

[1] Wedge angle: Angle between OTV and space station orbit planes.
[2] Phase angle: Measured from the OTV to the target (space station), positive in the direction of motion.

TABLE 2-I.- Continued

(b) - Continued

Event no.	Event	PET d:h:m:s	Duration d:h:m:s	HA / HP (n.mi.)	Incl (deg)	Wedge[1] angle (deg)	Phase[2] angle (deg)	Pitch angle (deg)	Yaw angle (deg)	Mnvr ΔV (fps)	Comments
9	Circularize in geosynchronous orbit	1:09:40:27		19323.0 / 19323.0	0.00			0.0	0.0	262.3	OTV attains the required longitude for deployment of the payload.
				**** Return Phase ****							
10	Initial geosynchronous orbit			19323.0 / 19323.0	0.0	28.50					OTV in geosynchronous orbit prior to low Earth orbit transfer.
11	Perform payload SEP mnvr			19323.0 / 19141.0	0.0	28.50		0.0	180.0	20.0	Retrograde separation mnvr from P/L. May occur anywhere WRT the LEO injection.
12	Initiate transfer to low Earth parking orbit	0:00:00:00		19141.0 / 45.5	26.30	2.34	-55.0	0.0	155.2	5924.1	TIG occurs 7.2 minutes prior to OTV's crossing the space station's orbital line of nodes. This time is dependent upon the rendezvous phasing.
13	Coast to EI at 400K feet		0:05:09:27	19141.0 / 45.5	26.30	2.21					Perform a midcourse mnvr if required during coast.
14	Aerodynamic flight regime	0:05:09:27	0:00:07:33								OTV begins aerodynamic braking and maneuvering.
15	Exit 400K feet and coast to apogee	0:05:17:00	0:00:38:56	377.8 / 44.8	28.50						
16	Insert into the low Earth parking orbit	0:05:55:56		400.0 / 377.8	28.50	0.32	-94.3	0.0	0.0	603.1	TIG occurs at apogee.
17	Coast to phasing mnvr		0:03:44:44	400.0 / 377.8	28.50						
18	Perform rendezvous phasing mnvr	0:09:40:40		391.4 / 258.0	28.50	0.15	-30.3	0.0	169.3	216.1	TIG occurs at the 1st common node 3 revs. after LEO transfer injection.

[1] Wedge angle: Angle between OTV and space station orbit planes.
[2] Phase angle: Measured from the OTV to the target (space station), positive in the direction of motion.

TABLE 2-I.- Concluded

(b) - Concluded

Event no.	Event	PET d:h:m:s	Duration d:h:m:s	HA / HP (n.mi.)	Incl (deg)	Wedge[1] angle (deg)	Phase[2] angle (deg)	Pitch angle (deg)	Yaw angle (deg)	Mnvr ΔV (fps)	Comments
19	Coast to height mnvr		0:02:26:23	391.4 / 258.0	28.50						
20	Perform height mnvr	0:12:07:03		258.8 / 200.0	28.50	0.00	-1.6	0.0	-172.3	319.5	TIG occurs at the common node.
21	Coast to stable orbit insertion mnvr		0:00:43:42	258.8 / 200.0	28.50						Coast for a 180° transfer.
22	Perform stable orbit insertion mnvr	0:12:50:45		200.0 / 200.0	28.50	0.00	0.25	-1.2	180.0	99.9	Stable orbit insertion completed with the OTV trailing the space station by 15 n. mi.
23	Maintain stable orbit point		0:03:03:59								A two rev. coast assumed for required prerecovery operations.
24	Initiate terminal rendezvous mnvr sequence	0:15:54:44	0:01:31:56	200.0 / 197.0	28.50	0.00	0.25	0.0	180.0	5.9	Begin final close in rendezvous sequence with the space station. Target for a point 1 n. mi. in front of the station.
25	Rendezvous operations	0:17:26:40	0:00:46:00		28.50	0.00	-0.02				Perform midcourse mnvrs as required.
26	Prox OPS, braking, and docking mnvrs.	0:18:12:40		200.0 / 200.0	28.50	0.00	0.00			11.3	Mnvr DV is an accumulated DV for the rendezvous operations, PROX OPS, braking, and docking.

[1] Wedge angle: Angle between OTV and space station orbit planes.
[2] Phase angle: Measured from the OTV to the target (space station), positive in the direction of motion.

TABLE 2-II.- ΔV SUMMARY FOR GEOSYNCHRONOUS PAYLOAD DELIVERY MISSION

Flight segment	OTV all-propulsive return	OTV aerobraking/ aeromaneuvering return
Outbound phase		
OTV/space station separation	20.0 fps	20.0 fps
Geosynchronous transfer orbit insertion	7944.8	7944.8
Geosynchronous longitude drift orbit insertion	5589.7	5589.7
Circularization at geosynchronous altitude	262.3	262.3
Subtotal	13816.8 fps	13816.8 fps
Return phase		
OTV/payload separation	20.0 fps	20.0 fps
LEO transfer orbit insertion	5781.4	5924.4
LEO parking orbit insertion	7614.8	603.1
OTV rendezvous phasing maneuver	41.2	216.1
OTV height maneuver	331.1	319.5
OTV stable orbit maneuver	295.0	99.9
OTV terminal rendezvous maneuvers	17.2	17.2
Subtotal	14100.7 fps	7199.9 fps
Total mission requirements	27917.5 fps	21016.7 fps

TABLE 2-III.- SEQUENCE OF EVENTS FOR THE ORBITAL TRANSFER VEHICLE GEOSYNCHRONOUS RETRIEVAL MISSION

(a) All-Propulsive Return Option

Event no.	Event	PET d:h:m:s	Duration d:h:m:s	HA / HP (n.mi.)	Incl (deg)	Wedge[1] angle (deg)	Phase[2] angle (deg)	Pitch angle (deg)	Yaw angle (deg)	Mnvr ΔV (fps)	Comments
				*** Outbound Phase ***							
0	Initial space station orbit			200.0 / 200.0	28.50	0.0	0.0				
1	OTV performs separation mnvr	0:00:00:00		211.0 / 200.0	28.50	0.0	0.0	0.0	0.0	20.0	OTV performs a separation mnvr from the space station. DV should be sufficient to protect the space station from all hazards associated with OTV deploy operations. Sep mnvr occurs at a nodal crossing.
2	Coast for four orbital periods		0:06:08:05	211.0 / 200.0	28.50						Coast duration was assumed for any postdeployment checkout, activation, etc. which may be required.
3	OTV insertion into geosynchronous transfer orbit	0:06:08:05		19323.0 / 200.0	26.30	2.2	3.0	0.0	-9.2	7944.8	Optimum height and plane change mnvr.
4	Coast to first nodal crossing		0:05:16:36	19323.0 / 200.0	26.30						Coast approximately 1/2 rev.
5	Insert into geosynchronous phasing orbit	0:11:24:41		19323.0 / 17174.0	0.0	28.50	-14.8	0.0	51.1	5598.2	Mnvr DV depends upon phasing requirement. Phasing mnvr to achieve OTV placement 50 n. mi. in front of payload and at the same altitude. One rev. coast in phasing orbit is assumed.
6	Coast To midcourse mnvr		0:11:07:53	19323.0 / 17174.0	0.0	28.50					Coast to the equator or 1/2 rev. after inserting into phasing orbit.
7	Perform midcourse mnvr if required	0:22:32:34		19323.0 / 17174.0	0.0	28.50					Midcourse performed if orbital dispersions require it.

[1] Wedge angle: Angle between OTV and space station orbit planes.
[2] Phase angle: Measured from the OTV to the target (space station), positive in the direction of motion.

TABLE 2-III.- Continued

(a) - Continued

Event no.	Event	PET d:h:m:s	Duration d:h:m:s	HA / HP (n.mi.)	Incl (deg)	Wedge[1] angle (deg)	Phase[2] angle (deg)	Pitch angle (deg)	Yaw angle (deg)	Mnvr ΔV (fps)	Comments
8	Coast to geosynch altitude		0:11:07:53	19323.0 / 17174.0	0.0	28.50					
9	Initiate rendezvous with payload	1:09:40:27	0:01:00:00	19325.0 / 18600.0	0.0	28.50		-1.5	0.0	171.4	OTV initiates intercept trajectory from the leading 50 n. mi. offset position.
10	PROX OPS, braking and docking with P/L	1:10:40:27		19323.0 / 19323.0	0.0	28.50		18.4	0.0	85.6	OTV retrieves payload after Prox OPS, braking and docking operations.
				*** RETURN Phase ***							
11	Coast to first nodal opportunity to initiate LEO descent		0:10:03:14	19323.0 / 19323.0	0.0	28.50					
12	Initiate transfer to low Earth parking orbit	1:20:43:41		19323.0 / 400.0	26.30	2.44	84.3	0.0	155.3	5751.6	TIG occurs 9.5 minutes prior to OTV's crossing the space station's orbital line of nodes. This time is dependent upon the rendezvous phasing.
13	Coast to the LEPO		0:05:20:48	19323.0 / 400.0	26.30						Perform a midcourse mnvr if required during coast.
14	Insert into the low Earth parking orbit	2:02:04:29		400.0 / 400.0	28.50	0.36	83.1	0.2	-172.9	7633.3	TIG occurs at equatorial crossing.
15	Coast to phasing mnvr		0:04:34:32	400.0 / 400.0	28.50						
16	Perform rendezvous phasing mnvr	2:06:39:01		400.0 / 340.9	28.50	0.25	166.3	0.0	180.0	97.5	TIG occurs at the first common node 3 revs. after LEO transfer injection.
17	Coast to height mnvr		0:12:07:17	400.0 / 340.9	28.50						

[1] Wedge angle: Angle between OTV and space station orbit planes.
[2] Phase angle: Measured from the OTV to the target (space station), positive in the direction of motion.

TABLE 2-III.- Continued

(a) - Concluded

Event no.	Event	PET d:h:m:s	Duration d:h:m:s	HA / HP (n.mi.)	Incl (deg)	Wedge[1] angle (deg)	Phase[2] angle (deg)	Pitch angle (deg)	Yaw angle (deg)	Mnvr ΔV (fps)	Comments
18	Perform height mnvr	2:18:46:18		351.0 / 195.6	28.50	0.01	-3.9	0.0	180.0	321.8	TIG occurs at the common node.
19	Coast to stable orbit insertion mnvr		0:00:48:30	351.0 / 195.6	28.50						Coast for a 180° transfer.
20	Perform stable orbit insertion mnvr	2:19:34:48		200.0 / 200.0	28.50	0.0	0.25	-37.0	180.0	309.4	Stable orbit insertion completed with the OTV trailing the space station by 15 n. mi.
21	Maintain stable orbit point		0:03:03:59								A two rev. coast assumed for required prerecovery operations.
22	Initiate terminal rendezvous mnvr sequence	2:22:38:47	0:01:31:56	200.0 / 197.0	28.50	0.00	0.25	0.0	180.0	5.9	Begin final close in rendezvous sequence with the space station. Target for a point 1 n. mi. in front of the station.
23	Rendezvous operations	3:00:10:43	0:00:46:00		28.50	0.00	-0.02				Perform midcourse mnvrs as required.
24	PROX OPS, braking, and docking mnvrs.	3:00:56:43		200.0 / 200.0	28.50	0.00	0.00			11.3	Mnvr DV is an accumulated DV for the rendezvous operations, PROX OPS, braking, and docking.

[1] Wedge angle: Angle between OTV and space station orbit planes.
[2] Phase angle: Measured from the OTV to the target (space station), positive in the direction of motion.

TABLE 2-III.- Continued

(b) Aerobraking/Aeromaneuvering Option

Event no.	Event	PET d:h:m:s	Duration d:h:m:s	HA / HP (n.mi.)	Incl (deg)	Wedge[1] angle (deg)	Phase[2] angle (deg)	Pitch angle (deg)	Yaw angle (deg)	Mnvr ΔV (fps)	Comments
				*** Outbound Phase ***							
0	Initial space station orbit			200.0 / 200.0	28.50	0.0	0.0				
1	OTV performs separation mnvr away from space station	0:00:00:00		211.0 / 200.0	28.50	0.0	0.0	0.0	0.0	20.0	OTV performs a separation mnvr from the space station. DV should be sufficient to protect the space station from all hazards associated with OTV deploy operations. SEP mnvr occurs at a nodal crossing.
2	Coast for four orbital periods		0:06:08:05	211.0 / 200.0	28.50						Coast duration was assumed for any postdeployment checkout, activation, etc., which may be required.
3	OTV transfer to geosynchronous orbit	0:06:08:05		19323.0 / 200.0	26.30	2.2	3.0	0.0	-9.2	7944.8	Optimum height and plane change mnvr.
4	Coast to first nodal crossing		0:05:16:36	19323.0 / 200.0	26.30						Coast approximately 1/2 rev.
5	Insert into longitude drift orbit	0:11:24:41		19323.0 / 17174.0	0.0	28.50	-14.8	0.0	51.1	5598.2	Mnvr DV depends upon phasing requirement. Phasing mnvr to achieve rev for OTV placement 50 n. mi. in front of payload and at the same altitude. One rev. coast in phasing orbit is assumed.
6	Coast to midcourse mnvr		0:11:07:53	19323.0 / 17174.0	0.0	28.50					Coast to the equator or 1/2 rev. after inserting into phasing orbit.
7	Perform midcourse mnvr if required	0:22:32:34		19323.0 / 17174.0	0.0	28.50					Midcourse performed if orbital dispersions require it.

[1] Wedge angle: Angle between OTV and space station orbit planes.
[2] Phase angle: Measured from the OTV to the target (space station), positive in the direction of motion.

TABLE 2-III.- Continued

(b) - Continued

Event no.	Event	PET d:h:m:s	Duration d:h:m:s	HA / HP (n.mi.)	Incl (deg)	Wedge[1] angle (deg)	Phase[2] angle (deg)	Pitch angle (deg)	Yaw angle (deg)	Mnvr ΔV (fps)	Comments
8	Coast to geosynch altitude		0:11:07:53	19323.0 / 17174.0	0.0	28.50					
9	Initiate rendezvous with payload	1:09:40:27	0:01:00:00	19325.0 / 18600.0	0.0	28.50		-1.5	0.0	171.4	OTV initiates intercept trajectory from the leading 50 n. mi. offset position.
10	Prox OPS, braking and docking with P/L	1:10:40:27		19323.0 / 19323.0	0.0	28.50		18.4	0.0	85.6	OTV retrieves payload after PROX OPS, braking and docking operations.
11	Coast to first nodal opportunity to initiate LEO descent		0:10:03:07	19323.0 / 19323.0	0.0	28.50					
12	Initiate transfer to low Earth parking orbit	1:20:43:34		19323.0 / 45.5	26.30	2.44	83.9	0.0	157.0	5893.5	TIG occurs 9.6 minutes prior to OTV's crossing the space station's orbital line of nodes. This time is dependent upon the rendezvous phasing.
13	Coast to EI at 400K feet		0:05:12:43	19323.0 / 45.5	26.30	2.21					Perform a midcourse mnvr if required during coast.
14	Aerodynamic flight regime	2:01:56:17	0:00:07:33								OTV begins aerodynamic braking and maneuvering
15	Exit 400K feet and coast to apogee	2:02:03:50	0:00:39:27	398.5 / 44.8	28.50						
16	Insert into the low Earth parking orbit	2:02:43:17		400.0 / 398.5	28.50	0.54	58.7	0.0	0.0	601.0	TIG occurs at apogee
17	Coast to phasing mnvr		0:03:42:49	400.0 / 398.5	28.50						
18	Perform rendezvous phasing mnvr	2:06:26:06		400.0 / 361.6	28.50	0.37	125.7	0.0	153.3	68.9	TIG occurs at the first common node 3 revs. after LEO transfer injection.

[1] Wedge angle: Angle between OTV and space station orbit planes.
[2] Phase angle: Measured from the OTV to the target (space station), positive in the direction of motion.

TABLE 2-III.- Concluded

(b) - Concluded

Event no.	Event	PET d:h:m:s	Duration d:h:m:s	HA / HP (n.mi.)	Incl (deg)	Wedge[1] angle (deg)	Phase[2] angle (deg)	Pitch angle (deg)	Yaw angle (deg)	Mnvr ΔV (fps)	Comments
19	Coast to height mnvr		0:13:54:43	400.0 / 361.6	28.50						
20	Perform height mnvr	2:20:20:49		363.3 / 200.0	28.50	0.00	-5.4	0.0	-175.6	326.2	TIG occurs at the common node.
21	Coast to stable orbit insertion mnvr		0:00:46:10	363.3 / 200.0	28.50						Coast for a 180° transfer.
22	Perform stable orbit insertion mnvr	2:21:06:59		200.0 / 200.0	28.50	0.00	0.25	-1.2	180.0	275.3	Stable orbit insertion completed with the OTV trailing the space station by 15 n. mi.
23	Maintain stable orbit point		0:03:03:59								A two rev. coast assumed for required prerecovery operations.
24	Initiate terminal rendezvous mnvr sequence	3:00:10:58	0:01:31:56	200.0 / 197.0	28.50	0.00	0.25	0.0	180.0	5.9	Begin final close in rendezvous sequence with the space station. Target for a point 1 n. mi. in front of the station.
25	Rendezvous operations	3:01:42:54	0:00:46:00		28.50	0.00	-0.02				Perform midcourse mnvrs as required.
26	PROX OPS, braking, and docking mnvrs	3:02:28:54		200.0 / 200.0	28.50	0.00	0.00			11.3	Mnvr DV is an accumulated DV for the rendezvous operations, PROX OPS, braking and docking.

[1] Wedge angle: Angle between OTV and space station orbit planes.
[2] Phase angle: Measured from the OTV to the target (space station), positive in the direction of motion.

TABLE 2-IV.- ΔV SUMMARY FOR GEOSYNCHRONOUS PAYLOAD RETRIEVAL MISSION

Flight segment	OTV all-propulsive return	OTV aerobraking/ aeromaneuvering return
Outbound phase		
OTV/space station separation	20.0 fps	20.0 fps
Geosynchronous transfer orbit insertion	7944.8	7944.8
Geosynchronous phasing orbit insertion (25 deg)	5598.2	5598.2
Geosynchronous rendezvous intercept maneuver	171.4	171.4
Geosynchronous rendezvous braking maneuver	85.6	85.6
Subtotal	13820.0 fps	13820.0 fps
Return phase		
LEO transfer orbit insertion	5751.6	5893.5
LEO parking orbit insertion	7633.3	601.0
OTV rendezvous phasing maneuver	97.5	68.9
OTV height maneuver	321.8	326.2
OTV stable orbit maneuver	309.4	275.3
OTV terminal rendezvous maneuvers	17.2	17.2
Subtotal	14130.8 fps	7182.1 fps
Total mission requirements	27950.8 fps	21002.1 fps

TABLE 2-V.- ORBITAL TRANSFER VEHICLE PLANETARY FLIGHT SCENARIO

(a) All-Propulsive

Event no.	Event	PET d:h:m:s	Duration d:h:m:s	HA / HP (n.mi.)	Incl (deg)	Wedge[1] angle (deg)	Phase[2] angle (deg)	Pitch angle (deg)	Yaw angle (deg)	Mnvr ΔV (fps)	Comments
0	Initial space station orbit			200 / 200	28.5						
1	OTV performs separation mnvr			210 / 200	28.5	0	0	0	0	20	SEP ΔV should be sufficient to protect the station from all hazards associated with OTV deployment operations.
2	OTV transfer to high ellipse	00:00		36450 / 210	28.5					8930	Mnvr places perigee in optimum position for final S/C kick stage burn to satisfy V_∞ target.
3	OTV separation from payload	10:41		36450 / 210	28.5						
4	OTV mnvr to adjust perigee altitude	11:11		36450 / 400	28.5					77	Raises perigee altitude to set up OTV return to space station
5	OTV midcourse mnvr	19:55									Midcourse performed if orbital dispersions require it.
6	OTV node adjustment mnvr	21:55								982	Mnvr performed at common node of OTV orbit plane and space station orbit plane.
7	S/C kick stage injection burn	22:22		00 / 210						(3000)	Final mnvr to place S/C on hyperbolic escape trajectory.

[1] Wedge angle: Angle between OTV and space station orbit planes.
[2] Phase angle: Measured from the OTV to the target (space station), positive in the direction of motion.

TABLE 2-V.- Continued

(a) - Concluded

Event no.	Event	PET d:h:m:s	Duration d:h:m:s	HA / HP (n.mi.)	Incl (deg)	Wedge[1] angle (deg)	Phase[2] angle (deg)	Pitch angle (deg)	Yaw angle (deg)	Mnvr ΔV (fps)	Comments
8	OTV circularization mnvr	22:26		400 / 400	28.5		-116			8602	
9	Coast to phasing mnvr		4:09								
10	Perform rendezvous phasing mnvr	26:35		400 / 230			-50			280	
11	Coast to height mnvr		2:24								
12	Perform height mnvr	28:59		230 / 200			-25			333	
13	Coast to stable orbit insertion mnvr		17:45								
14	Perform stable orbit insertion mnvr	46:44		200 / 200			.25			51	
15	Coast at stable orbit point		TBD								
16	Perform phasing mnvr	TBD		200 / TBD							
17	Perform midcourse mnvr if required	TBD									
18	Perform final approach mnvr to space station										

[1] Wedge angle: Angle between OTV and space station orbit planes.
[2] Phase angle: Measured from the OTV to the target (space station), positive in the direction of motion.

TABLE 2-V.- Continued

(b) Aerobraking

Event no.	Event	PET d:h:m:s	Duration d:h:m:s	HA / HP (n.mi.)	Incl (deg)	Wedge[1] angle (deg)	Phase[2] angle (deg)	Pitch angle (deg)	Yaw angle (deg)	Mnvr ΔV (fps)	Comments
0	Initial space station orbit			200 / 200	28.5						
1	OTV performs separation mnvr			210 / 200	28.5	0	0	0	0	20	SEP ΔV should be sufficient to protect the space station from all hazards associated with OTV deployment operations.
2	OTV transfer to high ellipse	00:00		36450 / 210	28.5					8930	Mnvr places perigee in optimum position for final S/C kick stage burn to satisfy V_∞ target.
3	OTV separation from payload	10:41		36450 / 210	28.5						
4	OTV mnvr to adjust perigee altitude	11:11		36540 / 45	28.5					62	Raises perigee altitude to set up OTV return to SOC
5	OTV midcourse mnvr	19:55								TBD	Midcourse performed if orbital dispersions require it.
6	OTV node adjustment mnvr	21:51		36450 / 45	28.5					1025	
7	OTV entry interface	22:17		36450 / 45	28.5						Entry interface defined
8	Aerobraking mnvr		00:07								
9	S/C kick stage injection burn	22:22		∞ / 210						(3000)	
10	OTV exit atmosphere	22:24		407 / 45							

[1] Wedge angle: Angle between OTV and space station orbit planes.
[2] Phase angle: Measured from the OTV to the target (space station), positive in the direction of motion.

TABLE 2-V.- Concluded

(b) - Concluded

Event no.	Event	PET d:h:m:s	Duration d:h:m:s	HA / HP (n.mi.)	Incl (deg)	Wedge[1] angle (deg)	Phase[2] angle (deg)	Pitch angle (deg)	Yaw angle (deg)	Mnvr ΔV (fps)	Comments
11	OTV circular-ization mnvr	23:04		407 / 407			-148			612	
12	Coast to phasing mnvr		3:14								
13	Perform rendezvous phasing mnvr	26:18		407 / 260			-97			240	
14	Coast to height mnvr		4:03								
15	Perform height mnvr	30:21		260 / 200			-47			344	
16	Coast to stable orbit insertion mnvr		16:02								
17	Perform stable orbit insertion mnvr	46:23		200 / 200			.25			104	
18	Coast at stable orbit point		TBD								
19	Perform phasing mnvr	TBD		200 / TBD							
20	Perform midcourse mnvr if required	TBD									
21	Perform final approach mnvr to space station										

[1] Wedge angle: Angle between OTV and space station orbit planes.
[2] Phase angle: Measured from the OTV to the target (space station), positive in the direction of motion.

TABLE 2-VI.- OTV ENERGY REQUIREMENTS FOR PLANETARY MISSION SUPPORT

	ΔV fps	Total
SEP maneuver	20	
Transfer to high ellipse	8 930	
	8 950	
All propulsive return		
Perigee adjustment maneuver	77	
Nodal alinement maneuver	982	
LEO circularization	8 602	
Phasing maneuver	280	
Height/plane change maneuver	333	
Stable orbit insertion	51	
	10 325	
		19 275
Aerobraking return		
Perigee adjustment maneuver	62	
Nodal alinement maneuver	1 025	
LEO circularization	612	
Phasing maneuver	240	
Height/plane change	344	
Stable orbit insertion	104	
	2 387	
		11 337

TABLE 2-VII.- ORBITAL TRANSFER VEHICLE RENDEZVOUS WITH LANDSAT

(a) Transfer To High Altitude Apogee

Event no.	Event	MET d:h:m:s	Duration d:h:m:s	HA / HP (n.mi.)	Incl (deg)	Wedge[1] angle (deg)	Phase[2] angle (deg)	Pitch angle (deg)	Yaw angle (deg)	Mnvr ΔV (fps)	Comments
0	Initial space station orbit			200.0 / 200.0	28.5						
1	OTV performs SEP mnvr	0:0:0:0		210.0 / 200.0	28.5					20.0	OTV performs a separation mnvr from the space station. SEP ΔV should be sufficient to protect the space station from all hazards associated with OTV deployment operations. SEP mnvr occurs at a nodal crossing.
2	Coast to ascending node		0:4:55:35	210.0 / 200.0	28.5						Coast duration was assumed for any postdeployment checkout, activation, etc. which may be required.
3	OTV insertion into high ellip. transfer orbit	0:4:55:35		100 000 / 210.0	28.5	69.7				9798	Optimum height mnvr occurs near the equator.
4	Coast to apogee		1:16:40:51	100 000 / 210.0	28.5	69.7					Coast approximately 1/2 rev. Midcourse mnvrs will be made if needed.

[1] Wedge angle: Angle between OTV and space station orbit planes.
[2] Phase angle: Measured from the OTV to the target (space station), positive in the direction of motion.

TABLE 2-VII.- Continued

(b) Outbound Propulsive

Event no.	Event	MET d:h:m:s	Duration d:h:m:s	HA / HP (n.mi.)	Incl (deg)	Wedge[1] angle (deg)	Phase[2] angle (deg)	Pitch angle (deg)	Yaw angle (deg)	Mnvr ΔV (fps)	Comments
1	Height/plane change mnvr	1:21:36:26		100 000 / 583	98.2	0.		0.	122.9	1504	Mnvr (made at apogee) necessary to achieve 98.2° inclination at time of rendezvous with Landsat.
2	Coast to low Earth parking orbit		1:17:00:19	100 000 / 583	98.2						Midcourse mnvrs will be made if necessary.
3	Insert into low Earth parking orbit	3:14:36:45		583 / 583	98.2	0.		0.	180.0	9270	TIG occurs near nodal crossing.
4	Coast to phasing mnvr		0:00:15:0	583 / 583	98.2						
5	Perform rendezvous phasing mnvr	3:14:51:45		583 / 467	98.2		-74.8	0.	-180.0	178	This phasing mnvr was done at the first common node of the OTV and Landsat. In reality some time should be alloted for orbit verification (waiting for the third or fourth common node should be sufficient).
6	Coast to height/plane change at the common node mnvr		0:00:53:17	583 / 467	98.2						

[1] Wedge angle: Angle between OTV and space station orbit planes.
[2] Phase angle: Measured from the OTV to the target (space station), positive in the direction of motion.

TABLE 2-VII.- Continued

(b) - Concluded

Event no.	Event	MET d:h:m:s	Duration d:h:m:s	HA / HP (n.mi.)	Incl (deg)	Wedge[1] angle (deg)	Phase[2] angle (deg)	Pitch angle (deg)	Yaw angle (deg)	Mnvr ΔV (fps)	Comments
7	Perform height/ plane change mnvr	3:15:44:0		467 / 383	98.2		-64.5	0.	-180.0	312	For the ideal rendezvous situation no ΔV would be required to further aline the nodes. Approximately 20 fps in the out-of-plane direction is included in this ΔV to account for dispersions. Since the out-of-plane component is root-sum-squared with the in-plane component of 307 fps, the ΔV needed for plane change is neglible.
8	Coast to stable orbit insertion mnvr		0 0:17:35:39	467 / 383	98.2						
9	Perform stable orbit insertion mnvr	4:09:19:39		383/383	98.2		0.25	0.	-180.	140	Stable orbit insertion completed with the OTV trailing the Landsat by 15 n. mi. The mnvrs required to move the OTV to a point 1 n. mi. in front of the Landsat prior to final approach is not included.

[1] Wedge angle: Angle between OTV and space station orbit planes.
[2] Phase angle: Measured from the OTV to the target (space station), positive in the direction of motion.

TABLE 2-VII.- Continued

(c) Outbound Aerobraking

Event no.	Event	MET d:h:m:s	Duration d:h:m:s	HA / HP (n.mi.)	Incl (deg)	Wedge[1] angle (deg)	Phase[2] angle (deg)	Pitch angle (deg)	Yaw angle (deg)	Mnvr ΔV (fps)	Comments
1	Height/plane change mnvr	1:21:36:26		100 000 / 45	98.2			0.	125.7	1453	Mnvr is performed to take advantage of aerobraking in Earth's atmosphere.
2	Coast to perigee		1:16:41:0	100 000 / 45	98.2						
3	Coast to 583 n. mi. altitude		0:0:47:39	583 / 45	98.2						
4	Circularize in low Earth parking	3:15:5:4.55		583/583	98.2			0.	0.	868	
5	Coast to phasing mnvr		0:0:57:13	583/583	98.2						
6	Perform rendezvous phasing mnvr	3:16:02:18		583/542	98.2		-136.9	0.	-180.	56	Done at the first common node; could be accomplished at later common node if additional time is required.
7	Coast to height/ plane change mnvr		0:0:53:1	583/542	98.2						
8	Perform height/ plane change mnvr	3:16:55:19		542/383	98.2		-123.9	0.	-180.	308	Approximately 20 fps in the out-of-plane direction is included in this ΔV to account for common node dispersions.
9	Coast to stable orbit insertion mnvr		0:17:51:55	542 / 383	98.2						

[1] Wedge angle: Angle between OTV and space station orbit planes.
[2] Phase angle: Measured from the OTV to the target (space station), positive in the direction of motion.

TABLE 2-VII.- Continued

(c) - Concluded

Event no.	Event	MET d:h:m:s	Duration d:h:m:s	HA / HP (n.mi.)	Incl (deg)	Wedge[1] angle (deg)	Phase[2] angle (deg)	Pitch angle (deg)	Yaw angle (deg)	Mnvr ΔV (fps)	Comments
10	Perform stable orbit insertion mnvr	4:10:47:14		383 / 383	98.2		0.25	0.	-180.0	262	Stable orbit insertion completed with the OTV trailing the Landsat by 15 n. mi. The Mnvrs required to move the OTV to a point 1 n. mi. in front of the Landsat prior to final approach are not included.

[1] Wedge angle: Angle between OTV and space station orbit planes.
[2] Phase angle: Measured from the OTV to the target (space station), positive in the direction of motion.

TABLE 2-VII.- Continued

(d) Return Propulsive

Event no.	Event	MET d:h:m:s	Duration d:h:m:s	HA / HP (n.mi.)	Incl (deg)	Wedge angle (deg) [1]	Phase angle (deg) [2]	Pitch angle (deg)	Yaw angle (deg)	Mnvr ΔV (fps)	Comments
1	OTV insertion into high elliptic transfer orbit	4:16:38:01		100 000 / 383	98.2	84.72				95.82	The mnvr is positioned such that the plane change at apogee will take out the wedge angle for the rendezvous phase of the return leg.
2	Coast to apogee		1:20:48:42	100 000 / 383	98.2	84.72					
3	Height/plane change mnvr	6:13:26:43		100 000 / 400	28.5			0.	132.3	1665	Mnvr made at apogee necessary to achieve 28.5° inclination at time of rendezvous with space station.
4	Coast to low Earth parking orbit		1:20:49:29	100 000 / 400	28.5						
5	Insert into low Earth parking orbit	8:10:16:12		400 / 400	28.5			0.	180.0	9555	
6	Coast to phasing mnvr		0:3:18:18	400 / 400	28.5						
7	Perform rendezvous phasing mnvr	8:13:34:30		400 / 309	28.5		-99.5	0.	180.0	145	Mnvr done at second common node of OTV and space station.
8	Coast to height/ plane change at the common node mnvr		0:0:48:55	400 / 309	28.5						

[1] Wedge angle: Angle between OTV and space station orbit planes.
[2] Phase angle: Measured from the OTV to the target (space station), positive in the direction of motion.

TABLE 2-VII.- Continued

(d) - Concluded

Event no.	Event	MET d:h:m:s	Duration d:h:m:s	HA / HP (n.mi.)	Incl (deg)	Wedge[1] angle (deg)	Phase[2] angle (deg)	Pitch angle (deg)	Yaw angle (deg)	Mnvr ΔV (fps)	Comments
9	Perform height/ plane change at the common node mnvr	8:14:23:25		300 / 200	28.5		-87.6	0.	180.0	339	21 fps in the yaw direction is included in this ΔV to account for dispersions.
10	Coast to stable orbit insertion mnvr		0:10:26:29	309 / 200	28.5						
11	Perform stable orbit insertion mnvr	9:06:49:54		200 / 200	28.5		.25	0.	180.0	193	Stable orbit insertion completed with the OTV trailing the space station by 15 n. mi. The mnvrs required to move the OTV to a position 1 n. mi. in front of the space station prior to final approach are not included.

[1] Wedge angle: Angle between OTV and space station orbit planes.
[2] Phase angle: Measured from the OTV to the target (space station), positive in the direction of motion.

TABLE 2-VII.- Continued

(e) Return Aerobraking

Event no.	Event	MET d:h:m:s	Duration d:h:m:s	HA / HP (n.mi.)	Incl (deg)	Wedge[1] angle (deg)	Phase[2] angle (deg)	Pitch angle (deg)	Yaw angle (deg)	Mnvr ΔV (fps)	Comments
1	OTV insertion into high elliptical transfer orbit	4:16:38:1		100 000 / 383	98.2	84.72				9582	The mnvr is positioned such that the plane change at apogee will take out the wedge angle for the rendezvous phase of the return leg.
2	Coast to apogee		1:20:48:42	100 000 / 383	98.2	84.72					
3	Height/plane change mnvr	6:13:26:43		100 000 / 45	28.5			0.	133.7	1628	Mnvr is performed to take advantage of aerobraking in Earth's atmosphere.
4	Coast to perigee		1:20:36:34	100 000 / 45	28.5						
5	Coast to 400 n. mi.		0:02:18:09	400 / 45	28.5						
6	Circularize in low Earth parking orbit	8:12:21:26		400 / 400	28.5			0.	0.	605	
7	Coast to phasing		0:02:35:27	400 / 400	28.5						
8	Perform rendezvous phasing mnvr	8:14:56:53		400 / 380	28.5		-157.1	0.	-180.0	33	Done at the second common node; could be accomplished at later common node alinements if additional time is required.
9	Coast to height/ plane change mnvr		0:0:49:37	400 / 380	28.5						

[1] Wedge angle: Angle between OTV and space station orbit planes.
[2] Phase angle: Measured from the OTV to the target (space station), positive in the direction of motion.

TABLE 2-VII.- Concluded

(e) - Concluded

Event no.	Event	MET d:h:m:s	Duration d:h:m:s	HA / HP (n.mi.)	Incl (deg)	Wedge[1] angle (deg)	Phase[2] angle (deg)	Pitch angle (deg)	Yaw angle (deg)	Mnvr ΔV (fps)	Comments
10	Perform height/ plane change mnvr	8:15:46:30		380 / 200	28.5		-142.6	0.	180.	344	Approximately 20 fps in the out-of-plane direction is included in this ΔV to account for common node dispersions.
11	Coast to stable orbit insertion mnvr		0:16:40:28	380 / 200	28.5						
12	Perform stable orbit insertion mnvr	9:08:26:58		200 / 200	28.5		0.25	0.	-180.	306	Stable orbit insertion completed with the OTV trailing the space station by 15 n. mi. The mnvrs required to move the OTV to a position 1 n. mi. in front of the space station prior to final approach are not included.

[1] Wedge angle: Angle between OTV and space station orbit planes.
[2] Phase angle: Measured from the OTV to the target (space station), positive in the direction of motion.

TABLE 2-VIII.- ENERGY REQUIREMENTS FOR PROPULSIVE AND AEROBRAKING METHODS

Transfer to High Altitude Apogee

Maneuver	ΔV, fps
SEP maneuver	20
High elliptic orbit insertion	9 798
	9 818

Outbound Propulsive

Maneuver	ΔV, fps
Height/plane change maneuver	1 504
Low Earth orbit circularization	9 270
Phasing maneuver	178
Height/plane change maneuver	312
Stable orbit insertion	140
	11 404

Outbound Aerobraking

Maneuver	ΔV, fps
Height/plane change maneuver	1 453
Low Earth orbit circularization	868
Phasing maneuver	56
Height/plane change maneuver	308
Stable orbit insertion	262
	2 947

Return Propulsive

Maneuver	ΔV, fps
High elliptic orbit insertion	9 582
Height/plane change maneuver	1 665
Low Earth orbit circularization	9 555
Phasing maneuver	145
Height/plane change maneuver	339
Stable orbit insertion	193
	21 479

TABLE 2-VIII.- Concluded

Return Aerobraking	
Maneuver	ΔV, FPS
High elliptic orbit insertion	9 582
Height/plane change maneuver	1 628
Low Earth orbit circularization	605
Phasing maneuver	33
Height/plane change maneuver	344
Stable orbit insertion	306
	12 498

TABLE 2-IX.- POSTAEROBRAKING EXIT CONDITIONS

(a) Entry from 19 323 x 45.6 n. mi. Elliptic Orbit

Entry conditions at 400 000 ft. altitude:

Inclination = 26.3 deg
Inertial velocity = 33828.60 fps
Inertial flightpath angle = -3.9918 deg

	Target	Actual	Target	Actual
Altitude - ft	340 000	340 108	400 000	400 173
Inertial velocity - fps	26 322	26 347	26 250	26 237
Inertial flightpath angle - deg	0.93266	0.9642	1.2839	1.2612
Inclination - deg	28.5	28.5024	28.5	28.499

$\frac{W}{CDA}$ = 5.12 lb/ft2 $\frac{L}{D}$ = 0.50 maximum load factor = 2.0 g's

TABLE 2-IX.- Continued

(b) Entry from 95 982 x 41.0 n. mi. Elliptic Orbit

Entry conditions at 400 000 ft. altitude:

Inclination = 97.964 deg
Inertial velocity = 35718 fps
Inertial flightpath angle = -4.7 deg

	Target	Actual	Target	Actual
Altitude - ft	340 000	341 261	400 000	400 670
Inertial velocity - fps	26 594	26 611	26 521	26 514
Inertial flightpath angle - deg	1.33	1.401	1.71	1.707
Inclination - deg	97.965	97.932	97.966	97.937

$\frac{W}{CDA}$ = 5.12 lb/ft2 $\frac{L}{D}$ = 0.50 maximum load factor = 2.4 g's

TABLE 2-IX.- Concluded

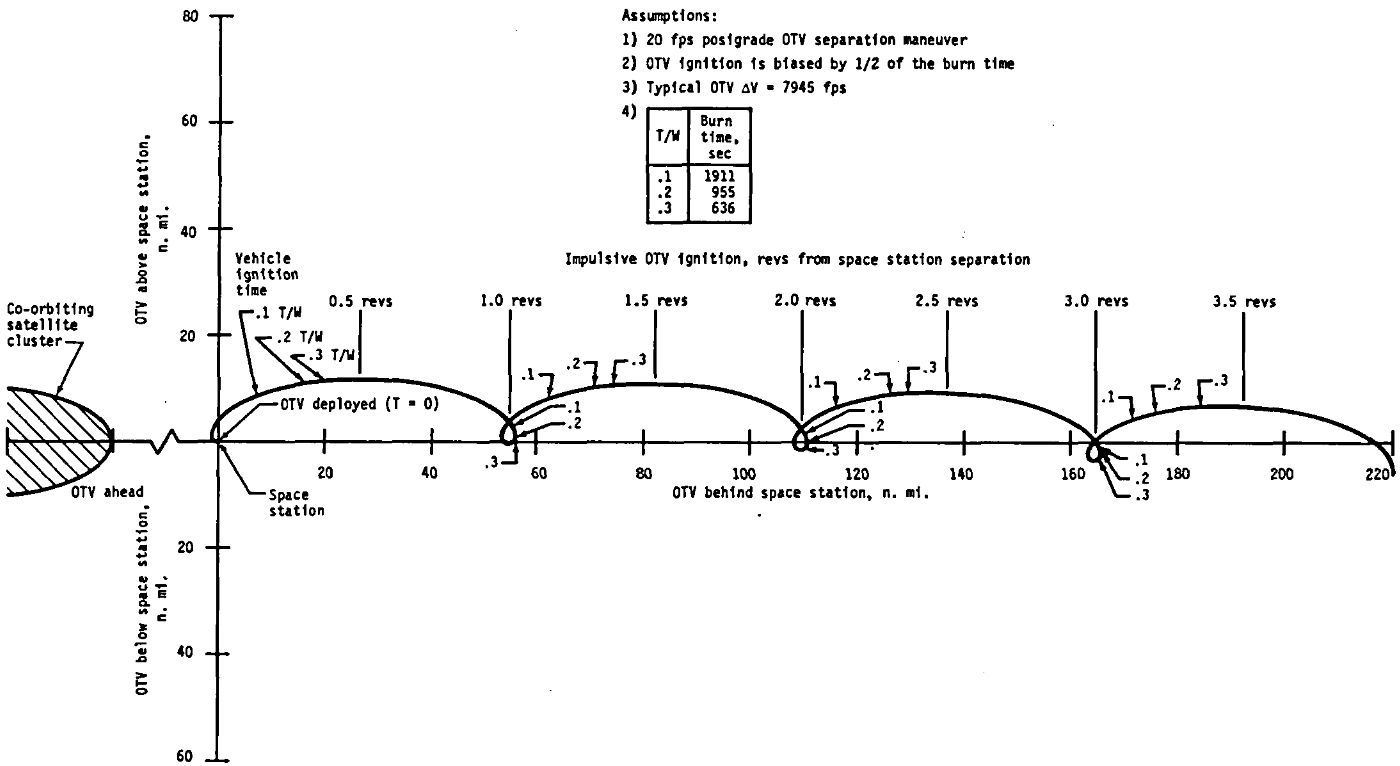
(c) Entry from 9662 x 45.6 n. mi. Elliptic Orbit

Entry conditions at 400 000 ft. altitude:

Inclination	= 26.3	deg
Inertial velocity	= 32578.0	fps
Inertial flightpath angle	= -3.99	deg

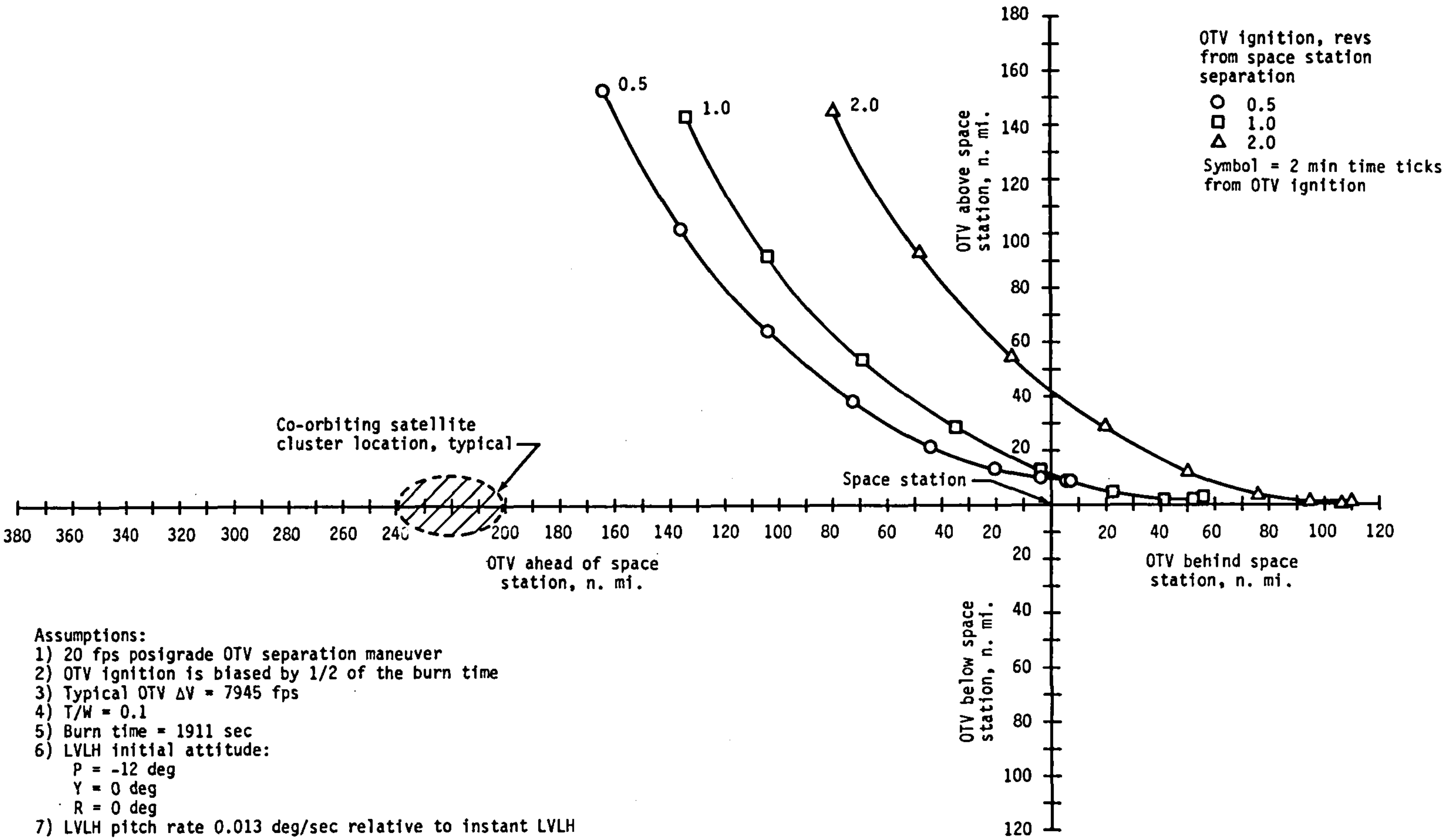
	Target	Actual	Target	Actual
Altitude - ft	340 000	340 723	400 000	400 998
Inertial velocity - fps	26 322	26 349	26 250	26 240
Inertial flightpath angle - deg	0.93266	1.0073	1.2839	1.2973
Inclination - deg	28.5	28.468	28.5	28.472

$\frac{W}{CDA}$ = 5.12 lb/ft2 $\frac{L}{D}$ = 0.50 maximum load factor = 1.83 g's



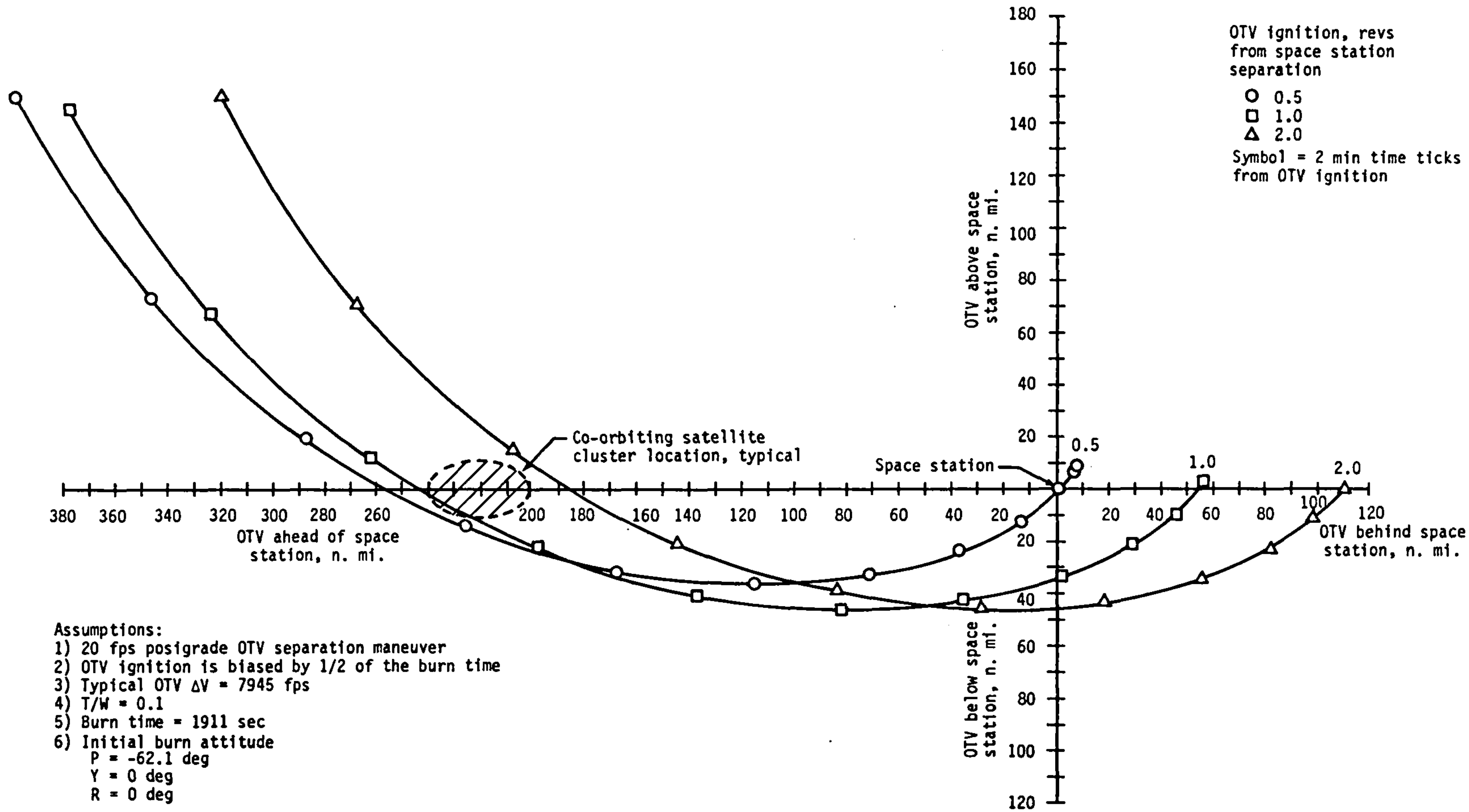
T/W	Burn time, sec
.1	1911
.2	955
.3	636

Figure 2-1.- Typical relative motion between the OTV and space station during the postdeployment coast prior to OTV ignition.



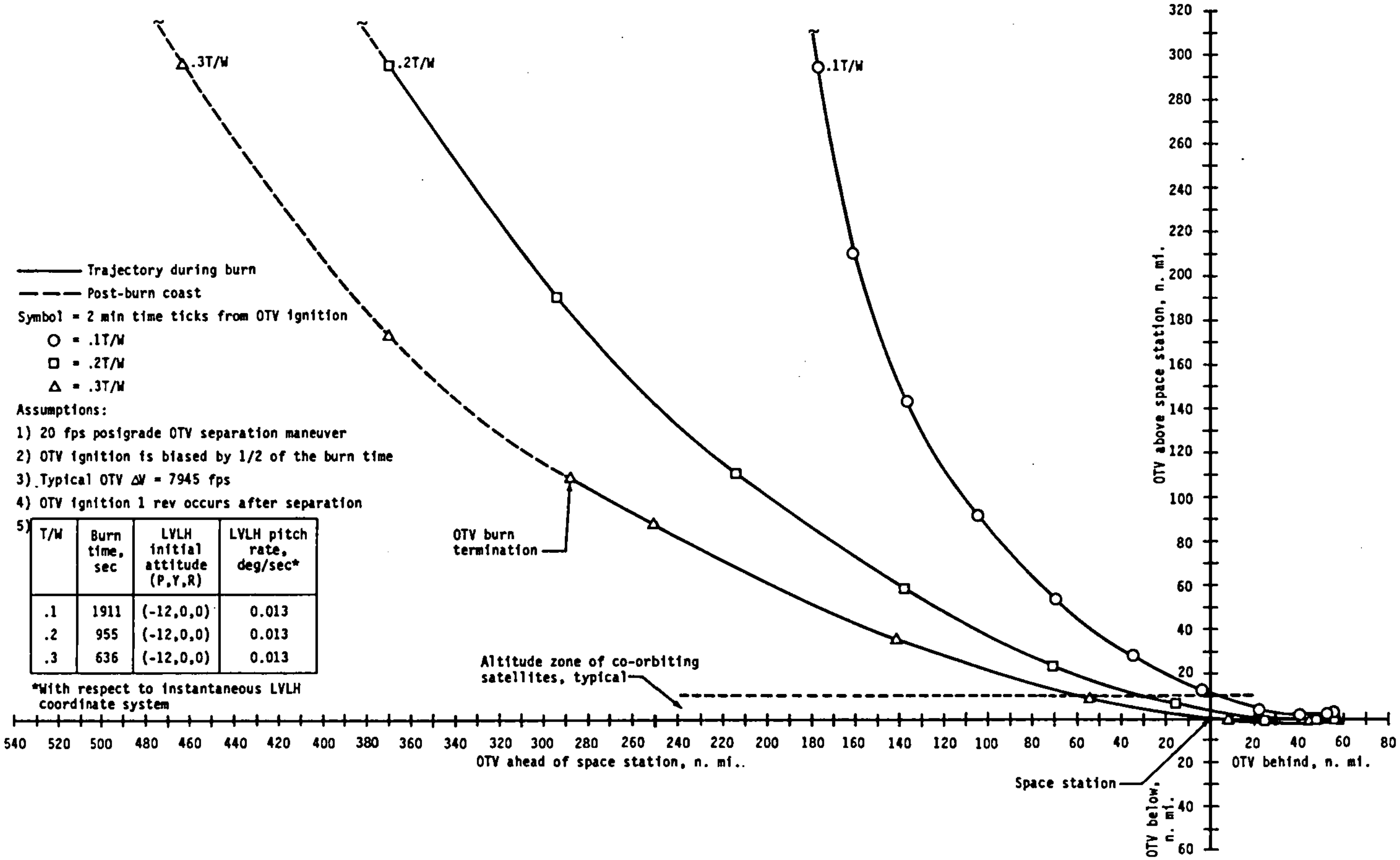
(a) OTV pitching during burn.

Figure 2-2.- Effect of postdeployment coast time on relative motion during OTV burn.



(b) OTV in inertial hold during burn.

Figure 2-2.- Concluded.

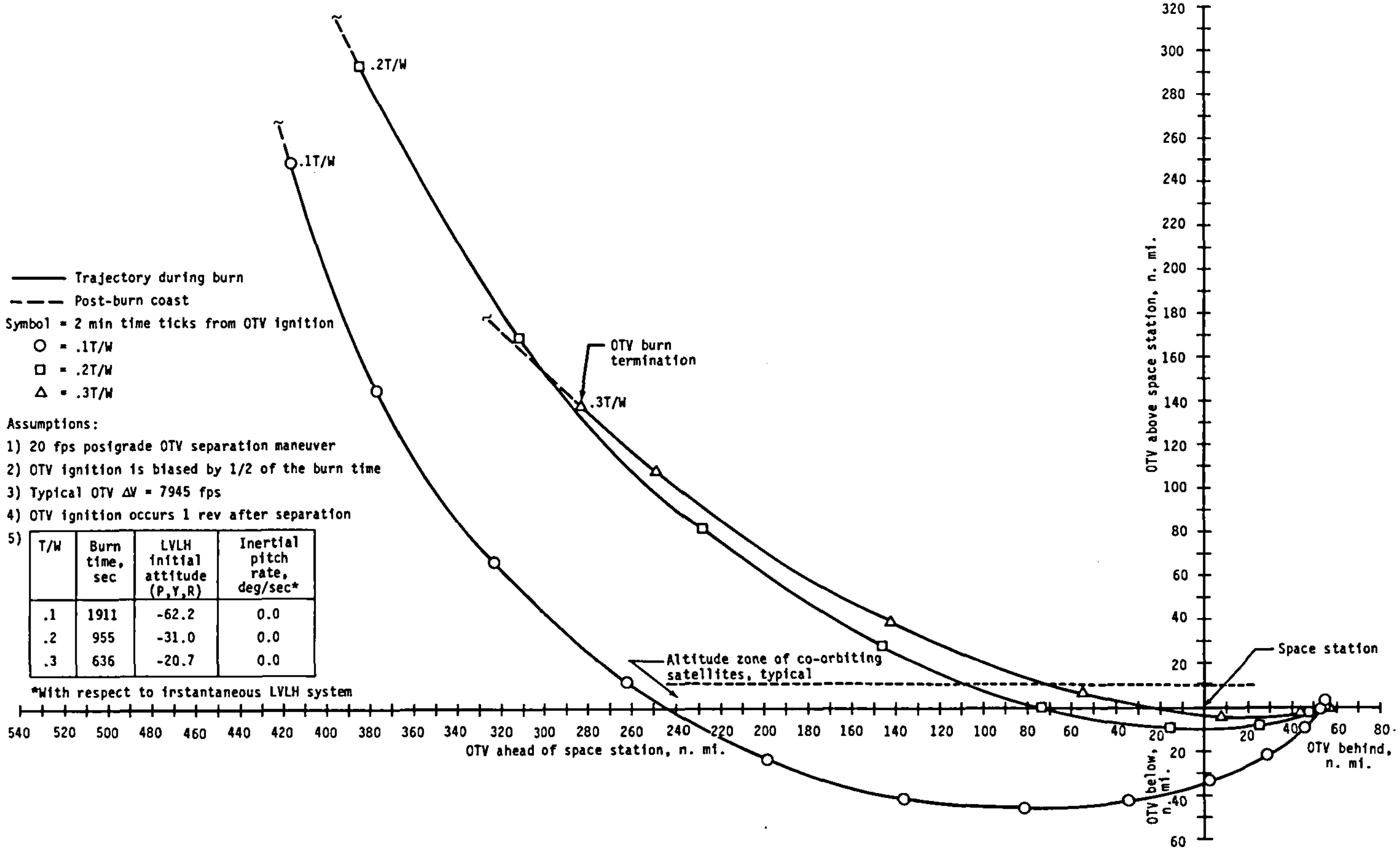


T/W	Burn time, sec	LVLH initial attitude (P,Y,R)	LVLH pitch rate, deg/sec*
.1	1911	(-12,0,0)	0.013
.2	955	(-12,0,0)	0.013
.3	636	(-12,0,0)	0.013

*With respect to instantaneous LVLH coordinate system

(a) OTV pitching during burn.

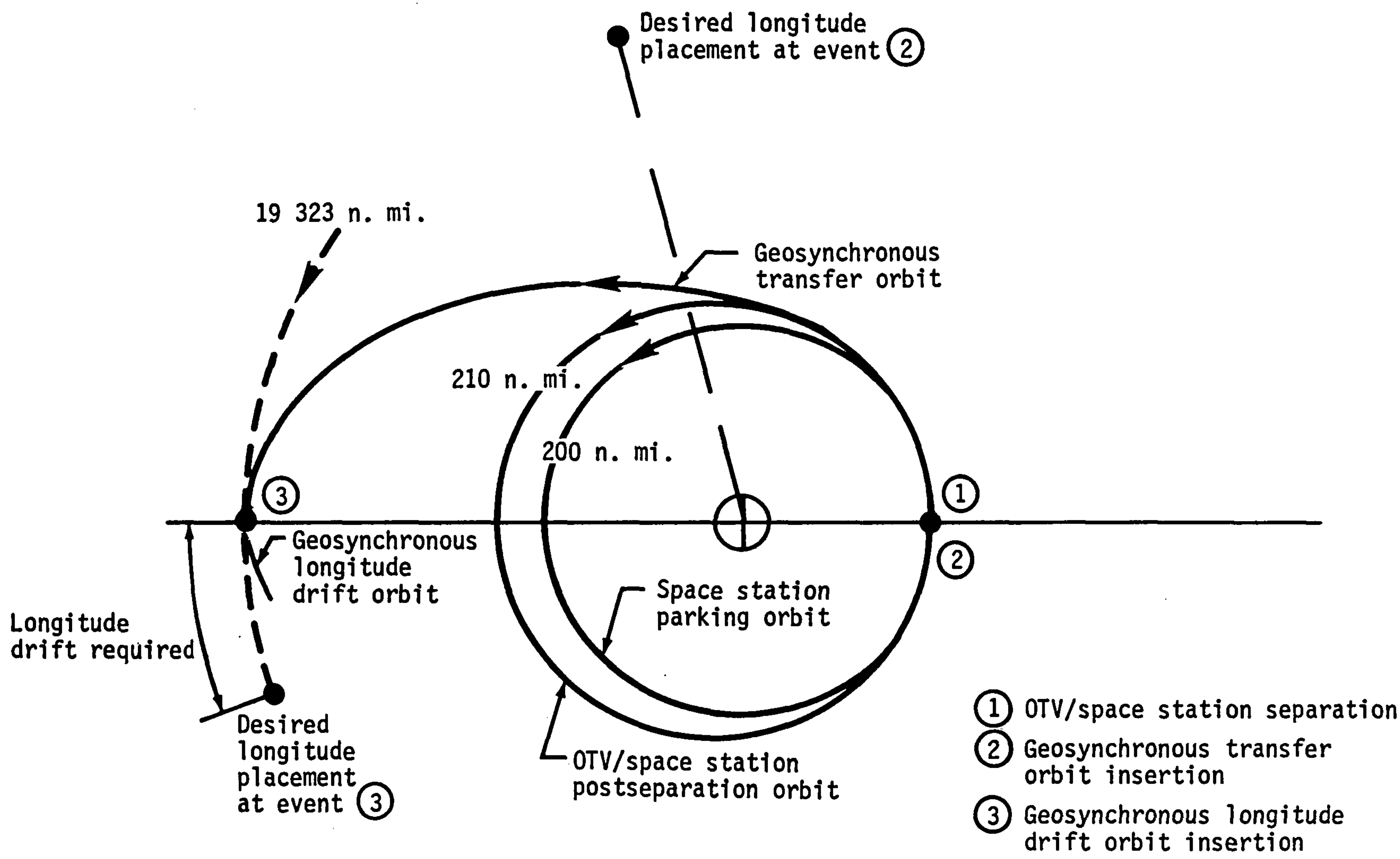
Figure 2-3.- Effect of OTV thrust-to-weight on the relative motion between the OTV and the space station.



T/W	Burn time, sec	LVLH initial attitude (P,Y,R)	Inertial pitch rate, deg/sec*
.1	1911	-62.2	0.0
.2	955	-31.0	0.0
.3	636	-20.7	0.0

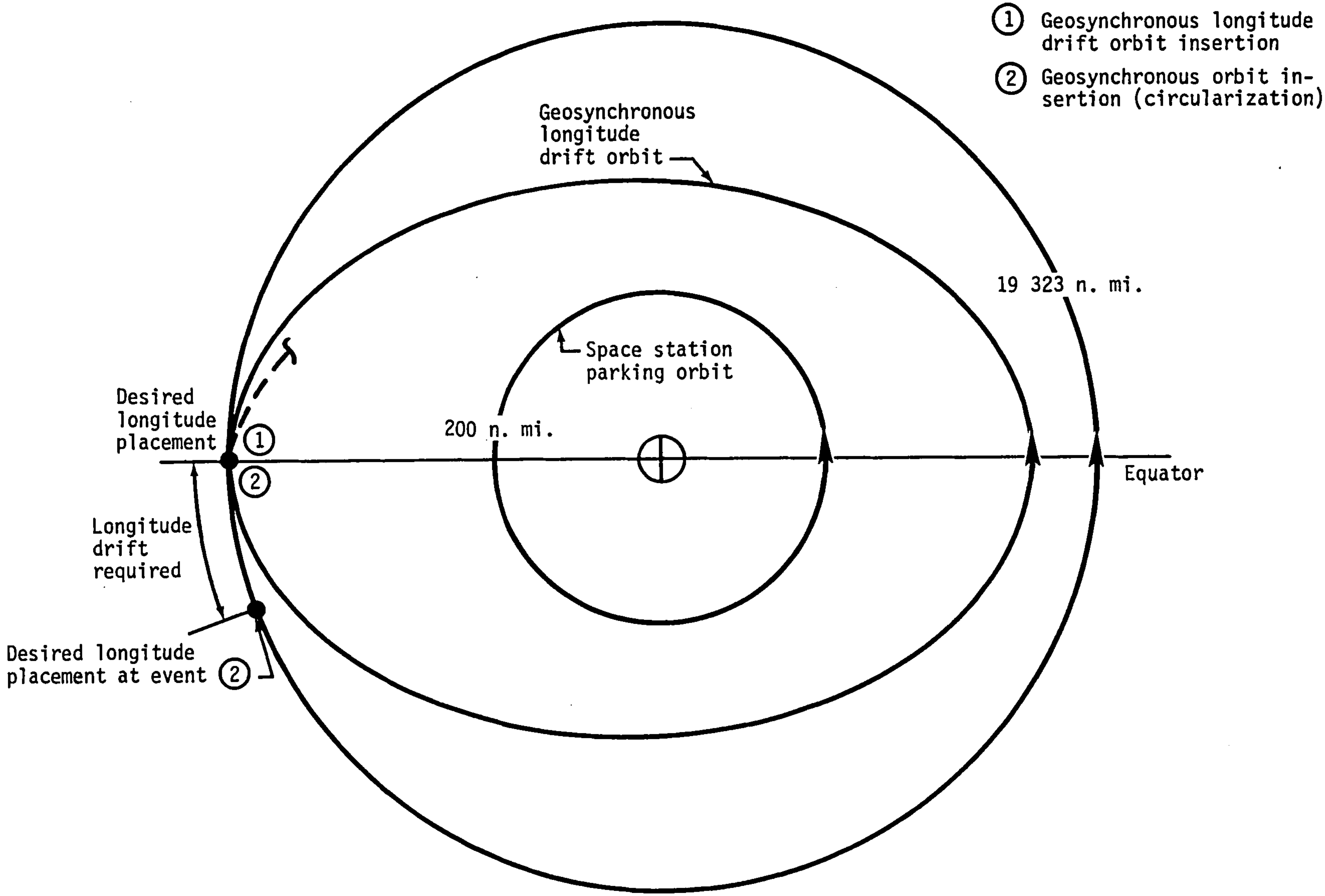
(b) OTV in inertial hold during burn.

Figure 2-3.- Concluded.



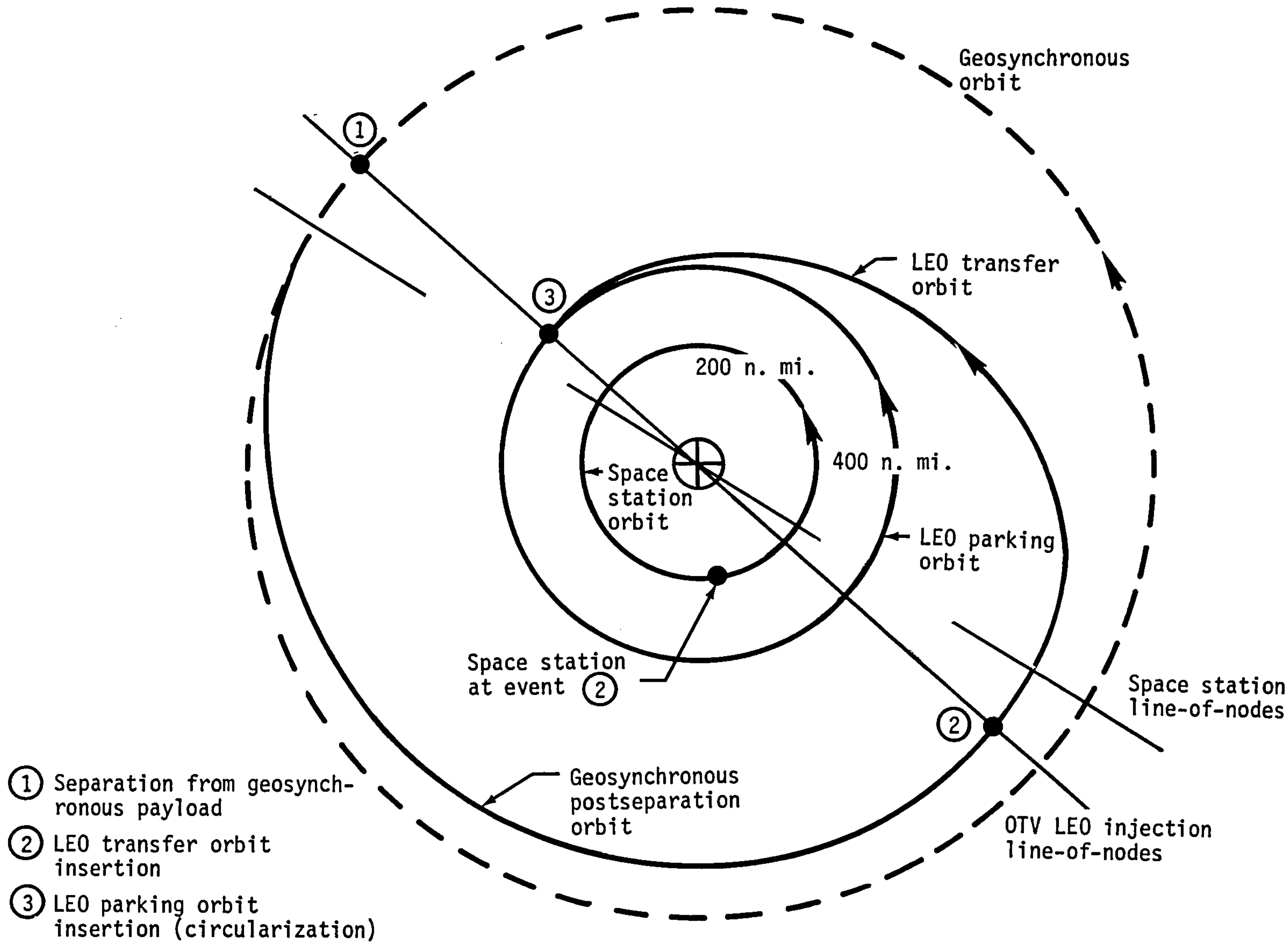
(a) Space station deployment phase through the geosynchronous transfer orbit phase.

Figure 2-4.- Orbital diagram for a geosynchronous payload delivery mission employing a reusable OTV.



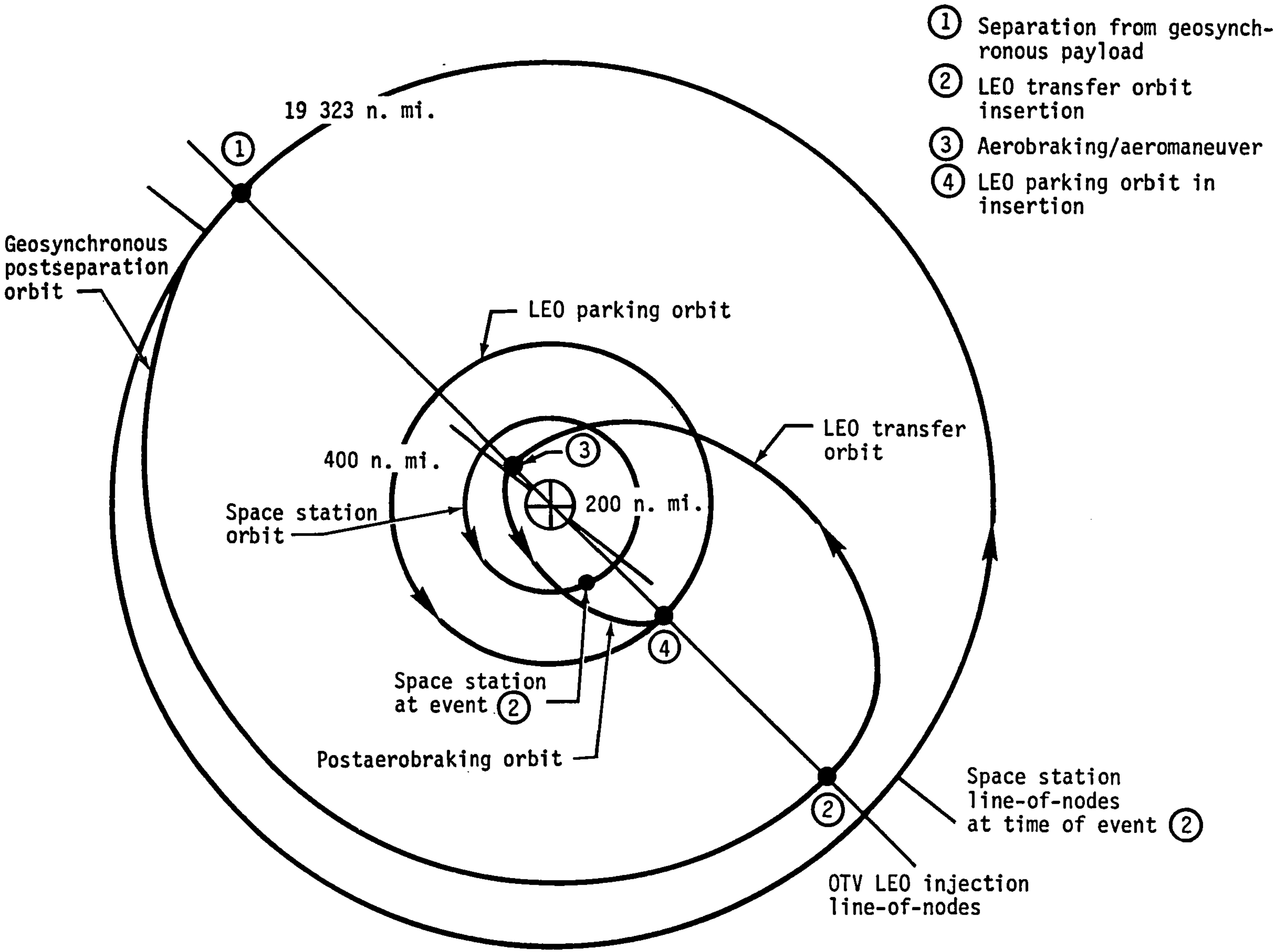
(b) Geosynchronous longitude drift phase.

Figure 2-4.- Continued.



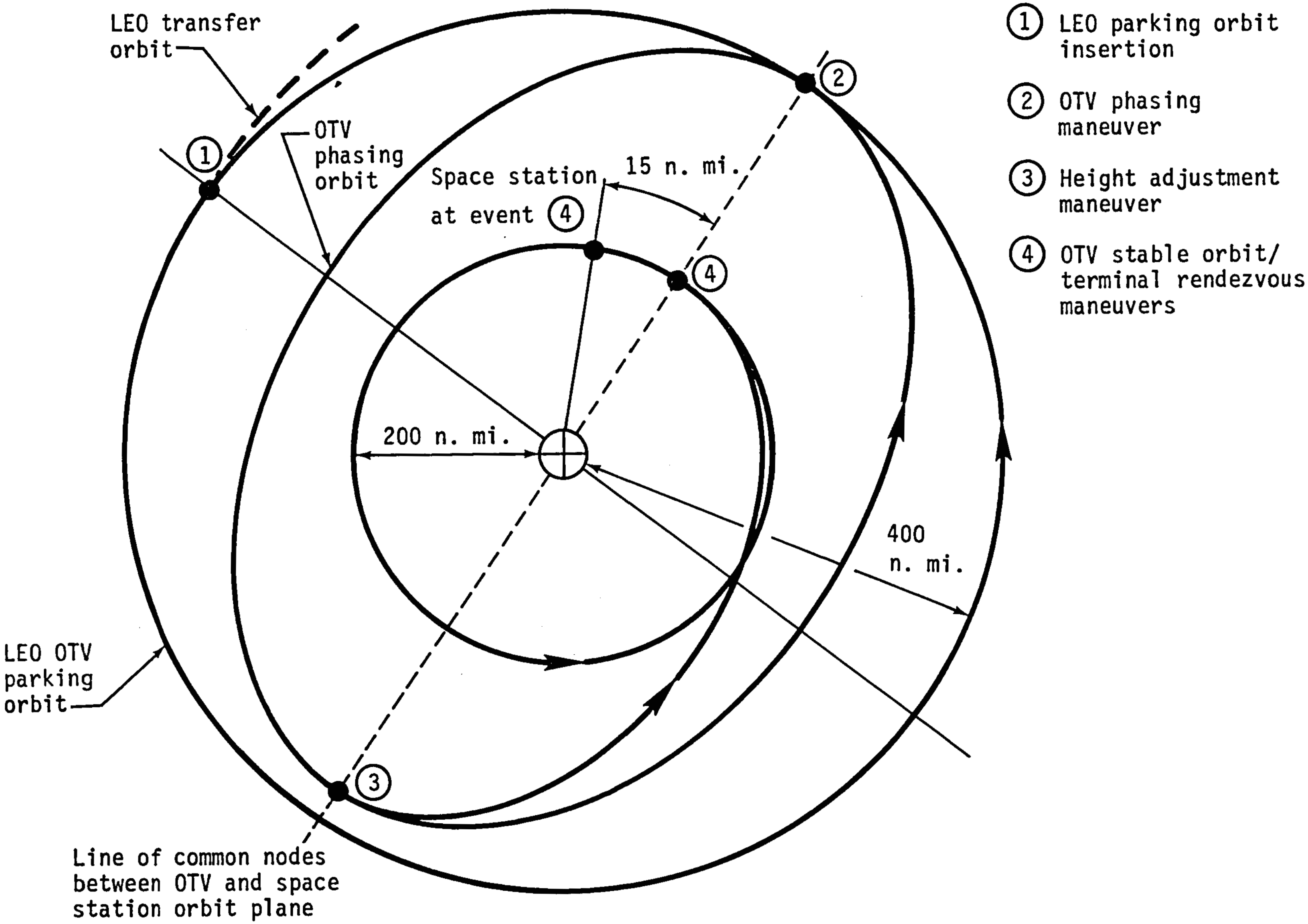
(c) Low Earth orbit transfer phase - OTV all-propulsive option.

Figure 2-4.- Continued.



(d) Low Earth orbit transfer phase - OTV aerobraking option.

Figure 2-4.- Continued.



(e) OTV/space station rendezvous phase.

Figure 2-4.- Concluded.

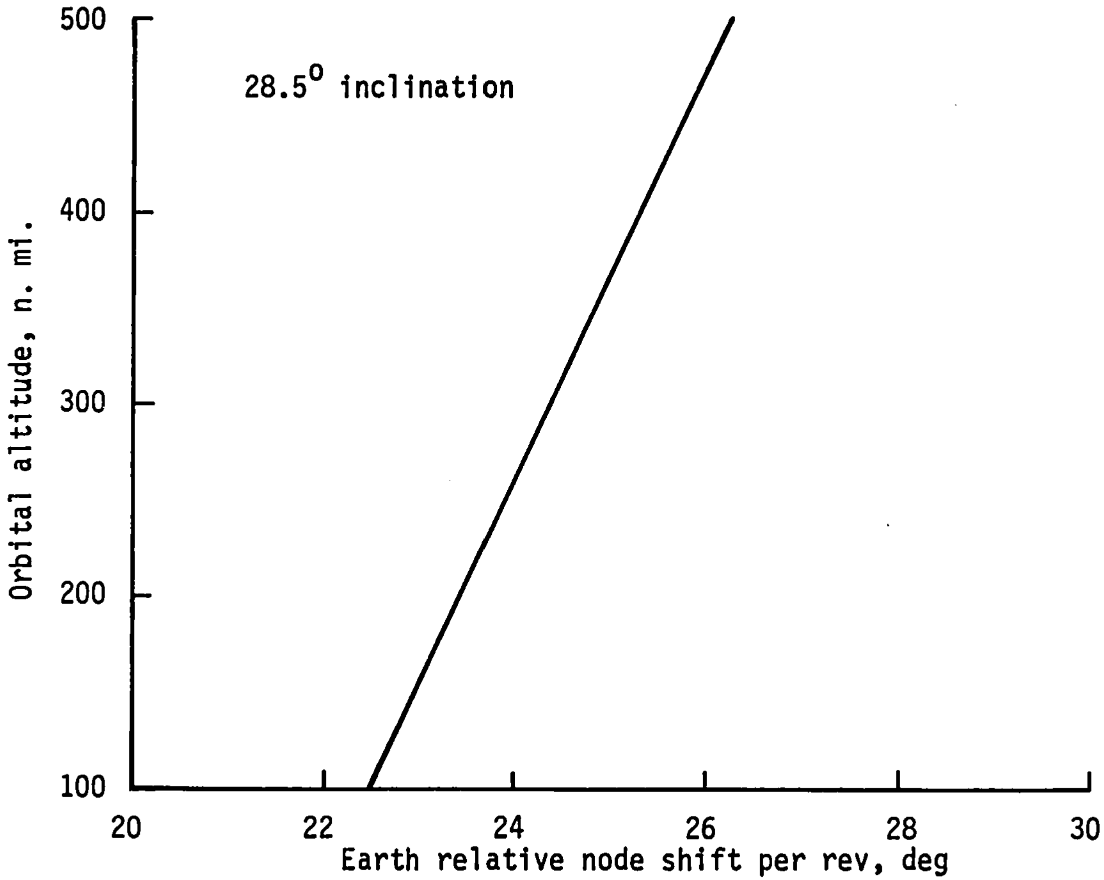
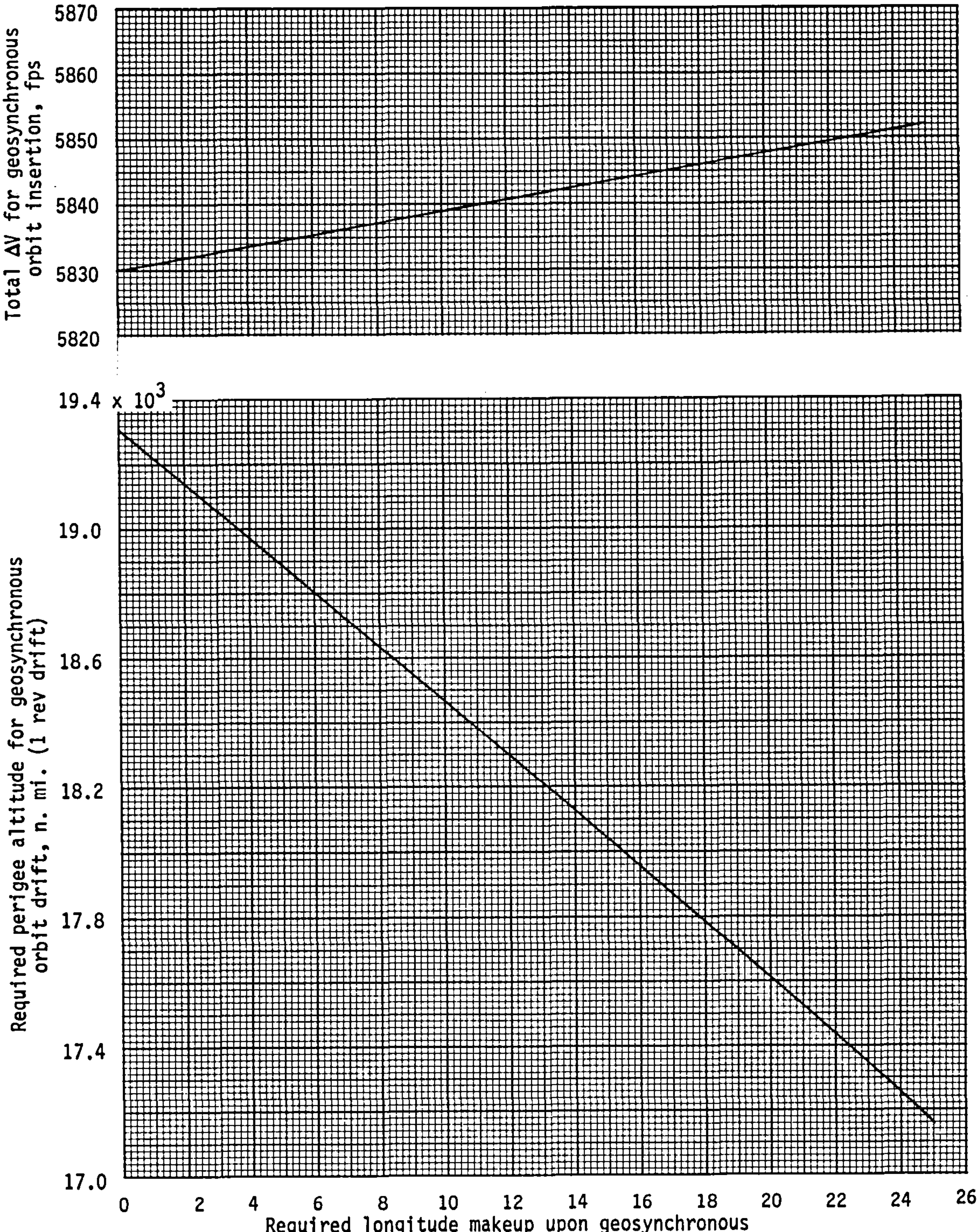
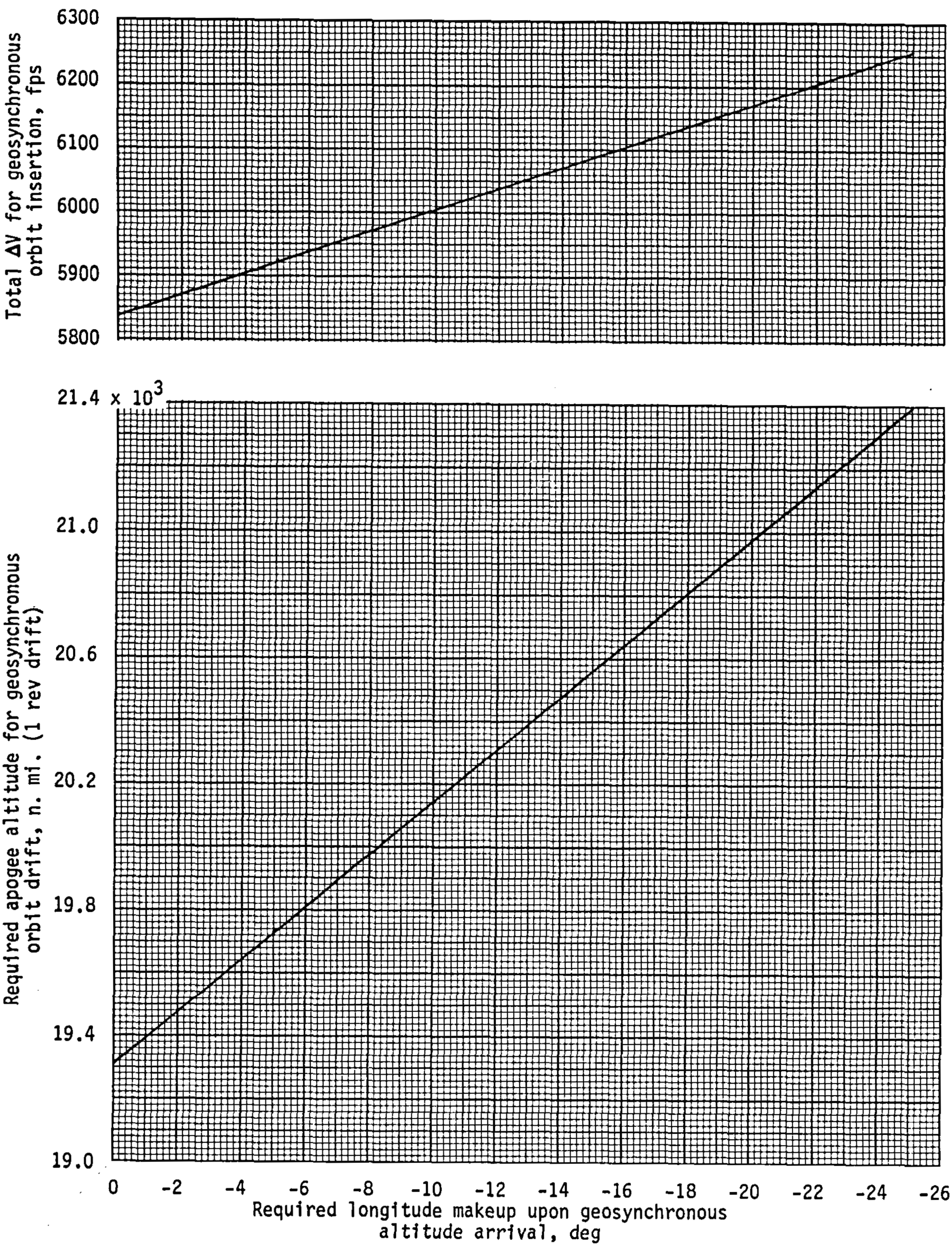


Figure 2-5.- Earth relative node shift per orbit rev.



(a) Easterly shift required.

Figure 2-6.- Geosynchronous phasing requirements for longitude placement.



(b) Westerly shift required.

Figure 2-6.- Concluded.

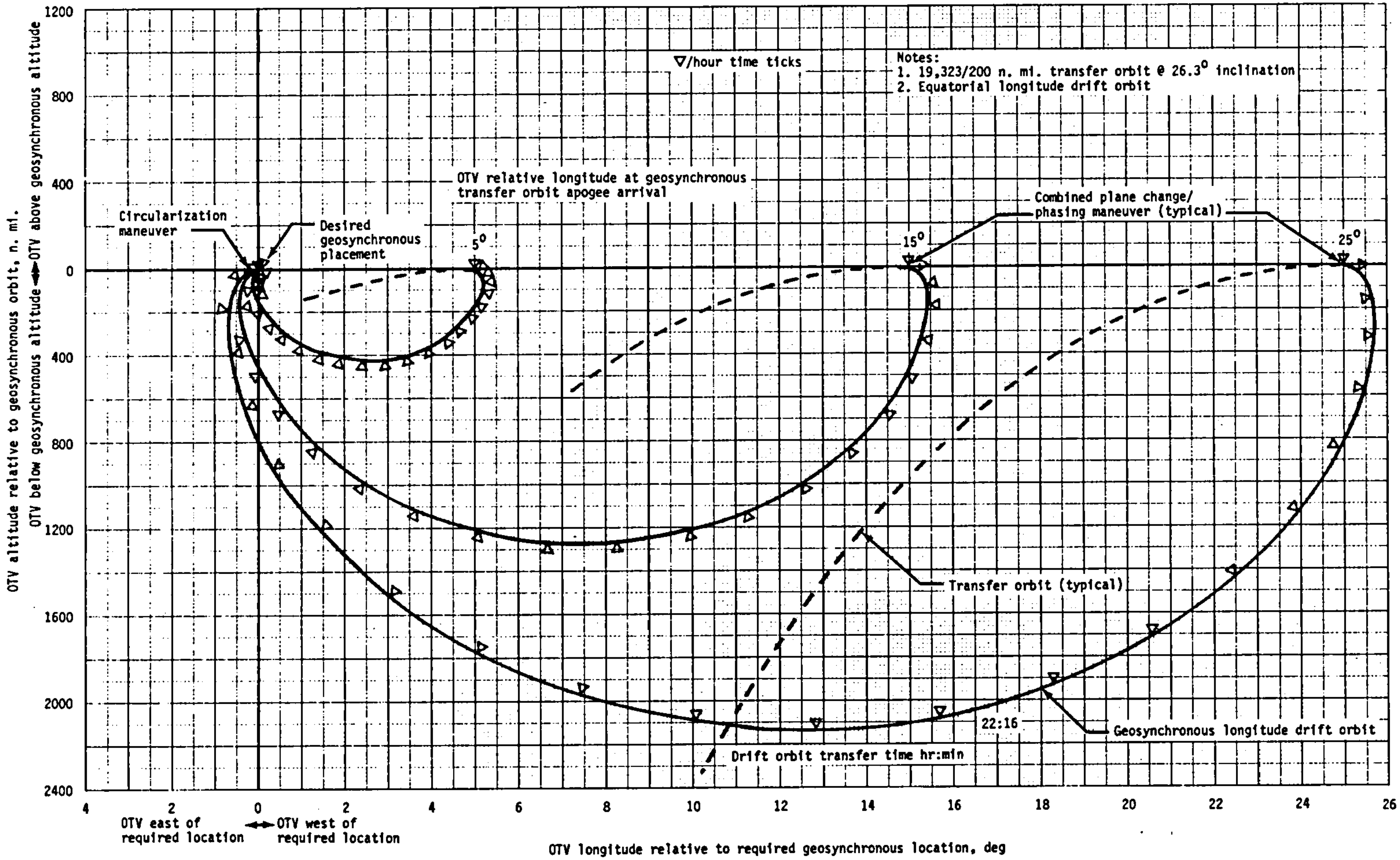
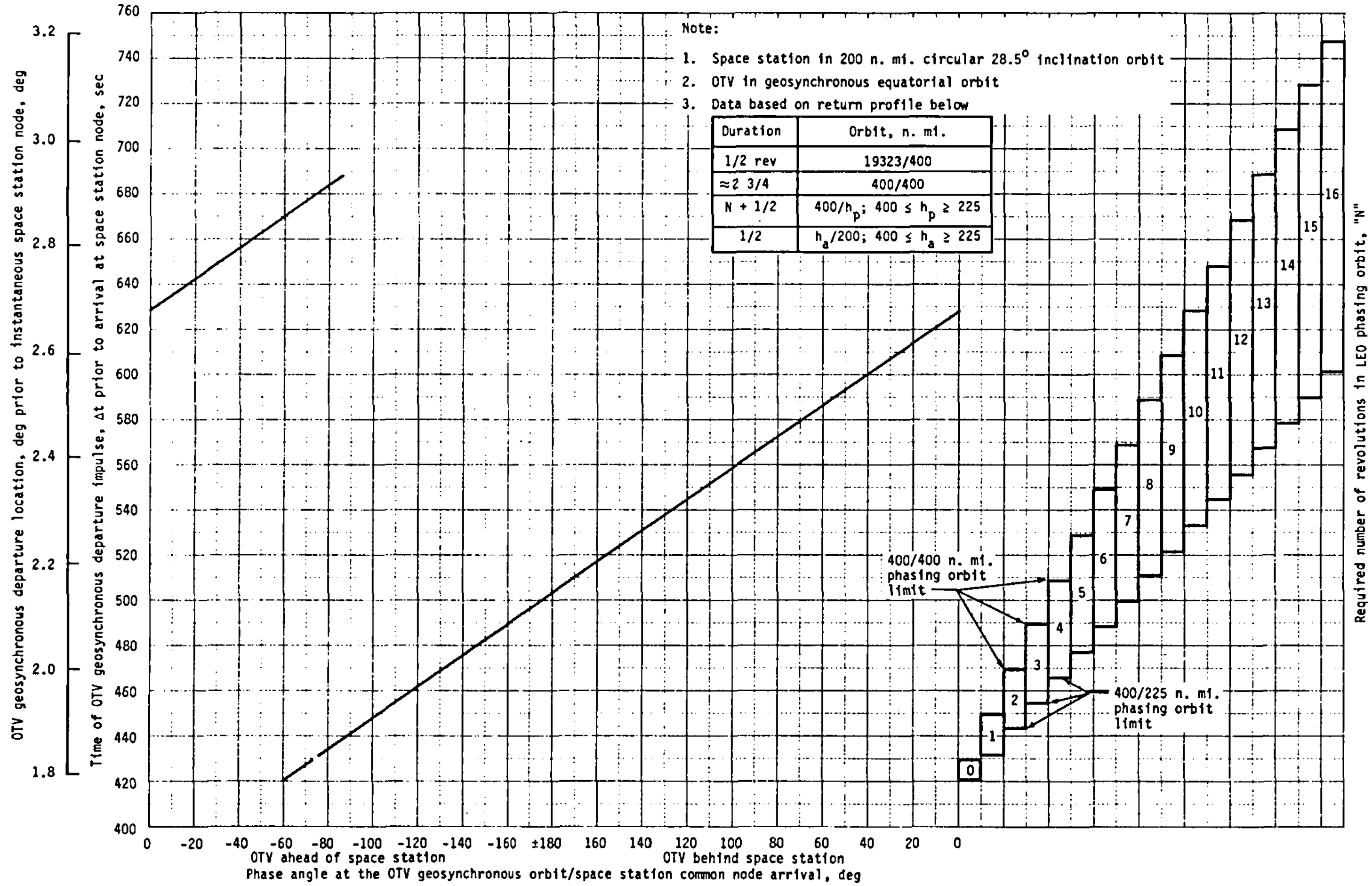
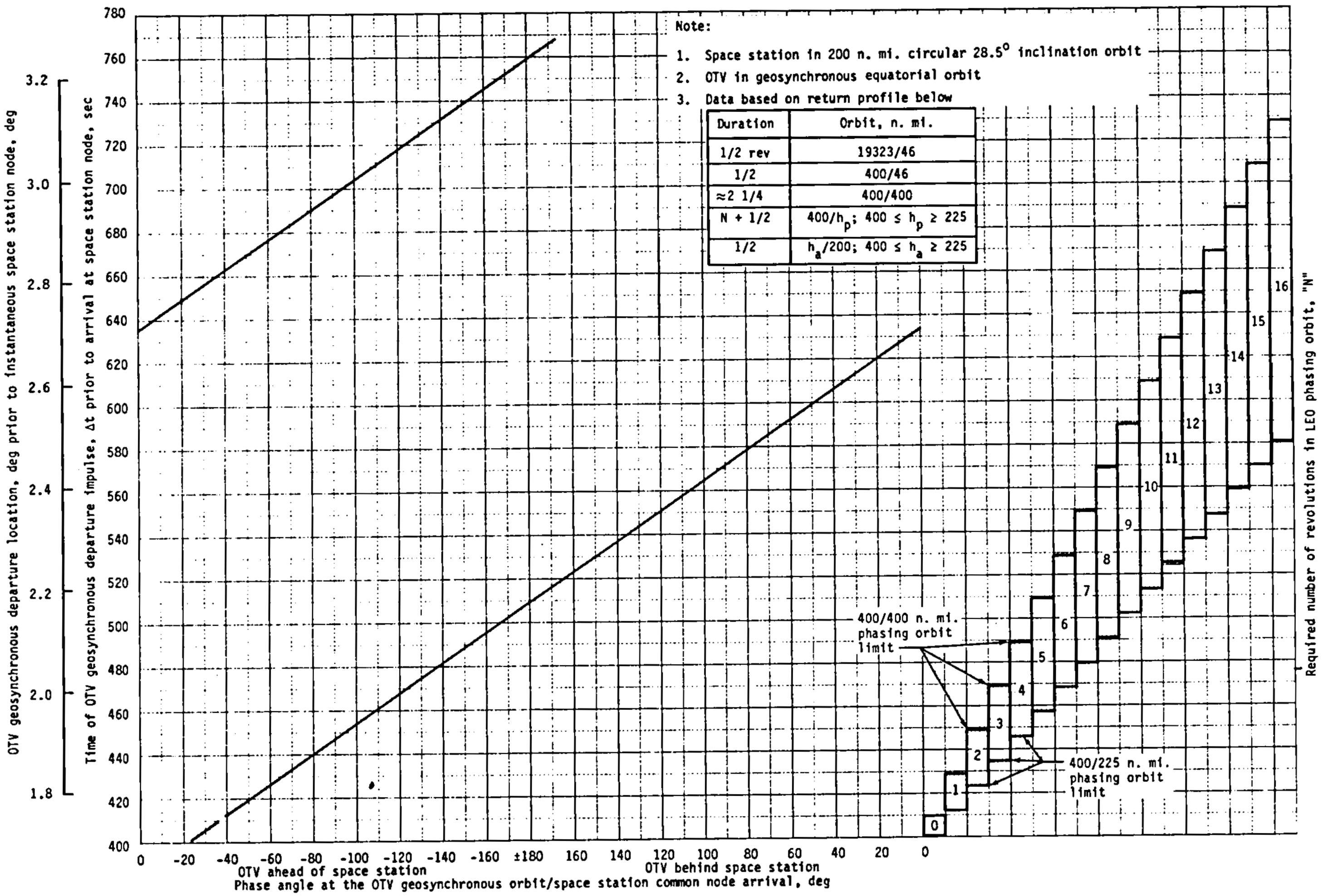


Figure 2-7.- Typical OTV relative motion during geosynchronous longitude drift phase.



(a) OTV all-propulsive return option.

Figure 2-8.- Impulsive geosynchronous departure time for optimum OTV return to space station.



(b) OTV aerobraking/aeromaneuvering option.

Figure 2-8.- Concluded.

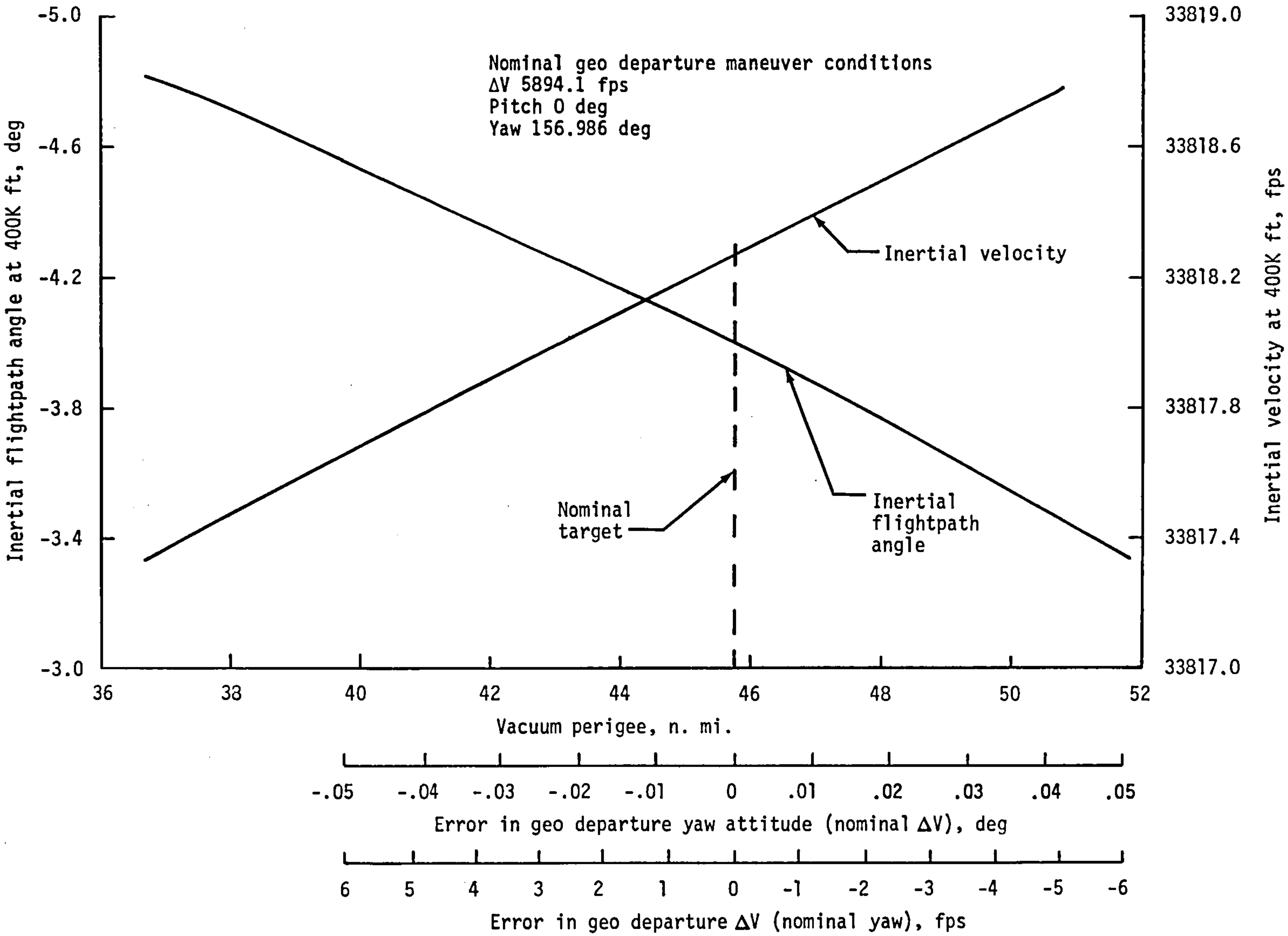
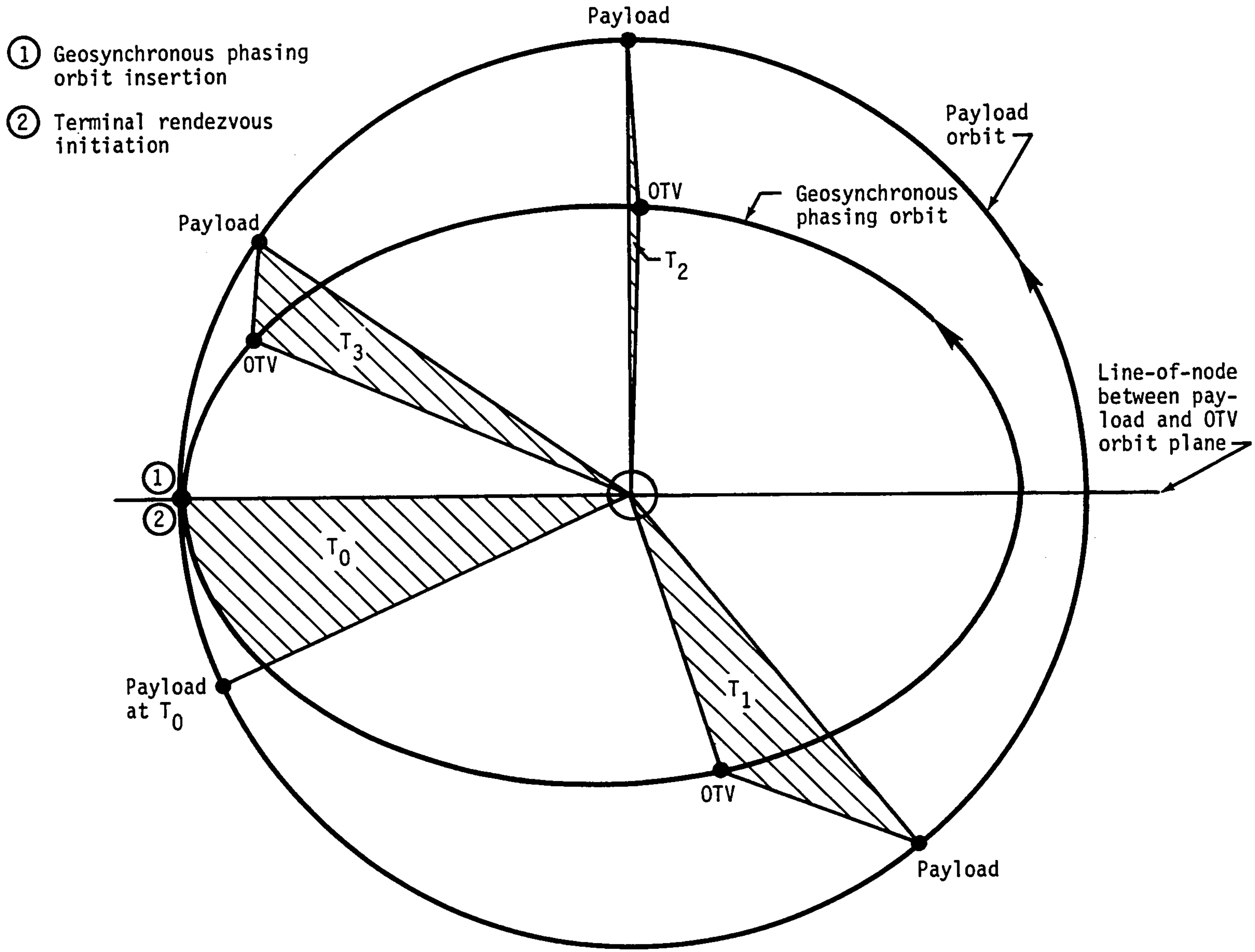
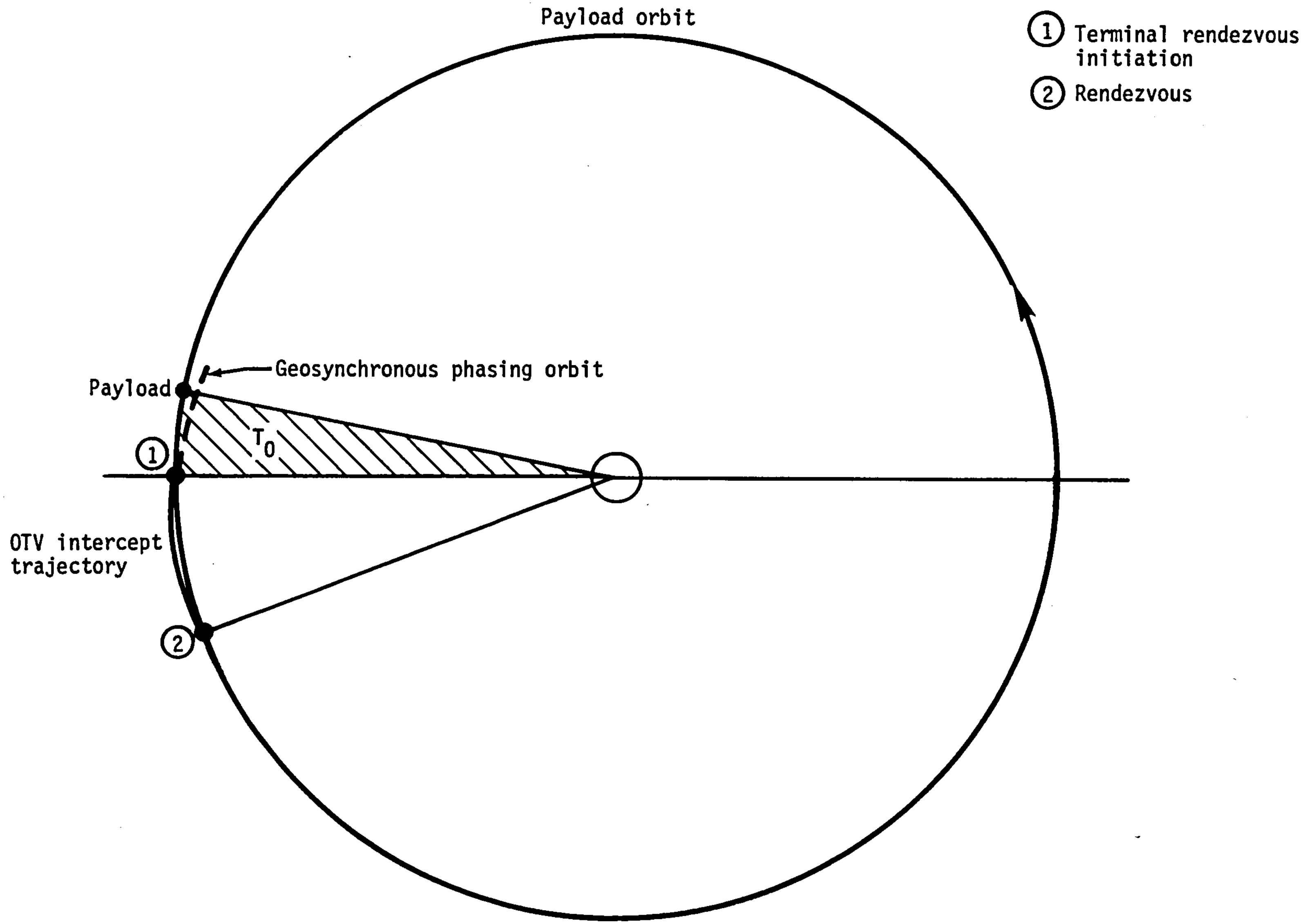


Figure 2-9.- Typical sensitivity of aerobraking initial entry conditions to geosynchronous orbit departure accuracy.



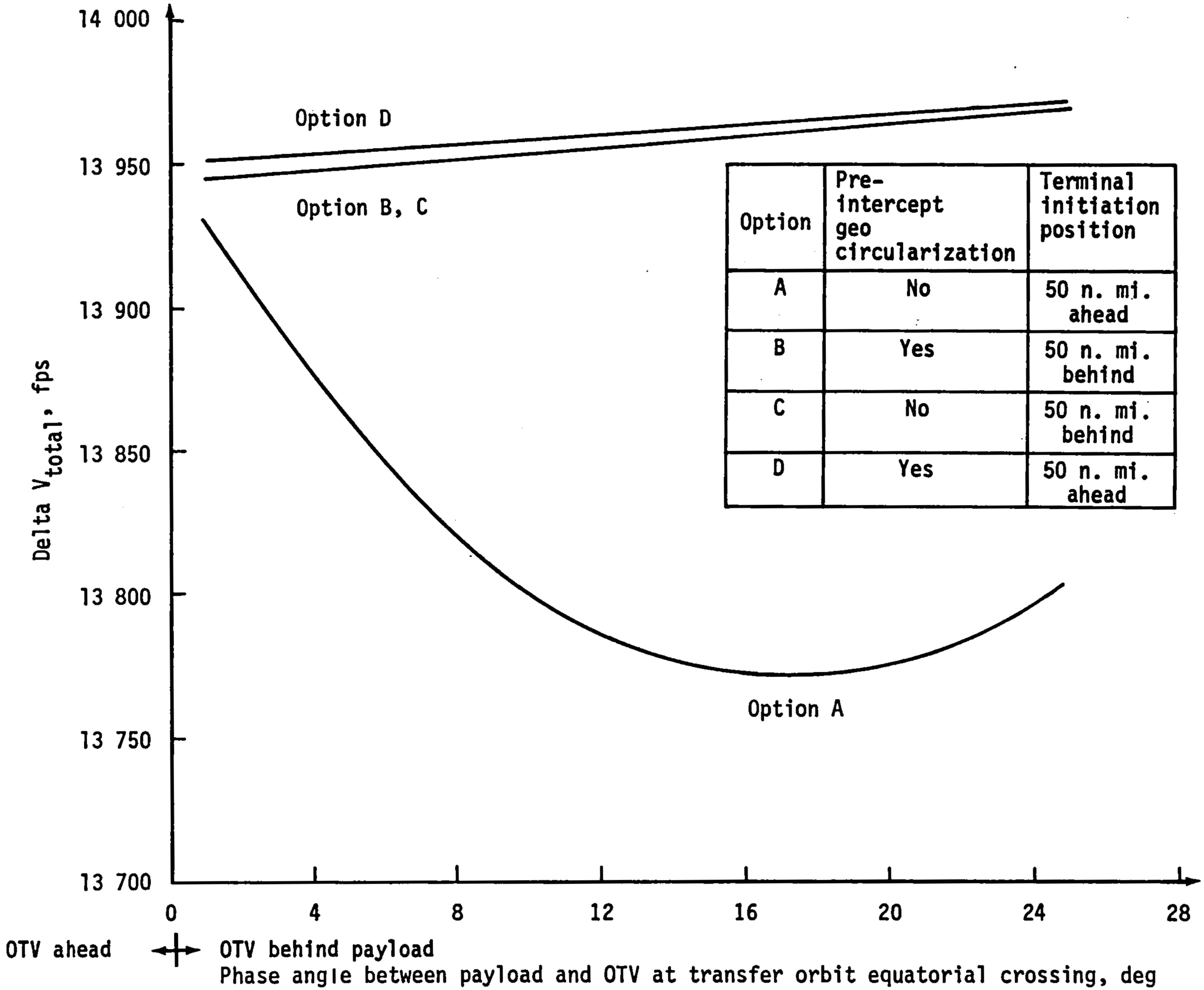
(a) Geosynchronous phasing orbit.

Figure 2-10.- Orbital diagram for a geosynchronous rendezvous.



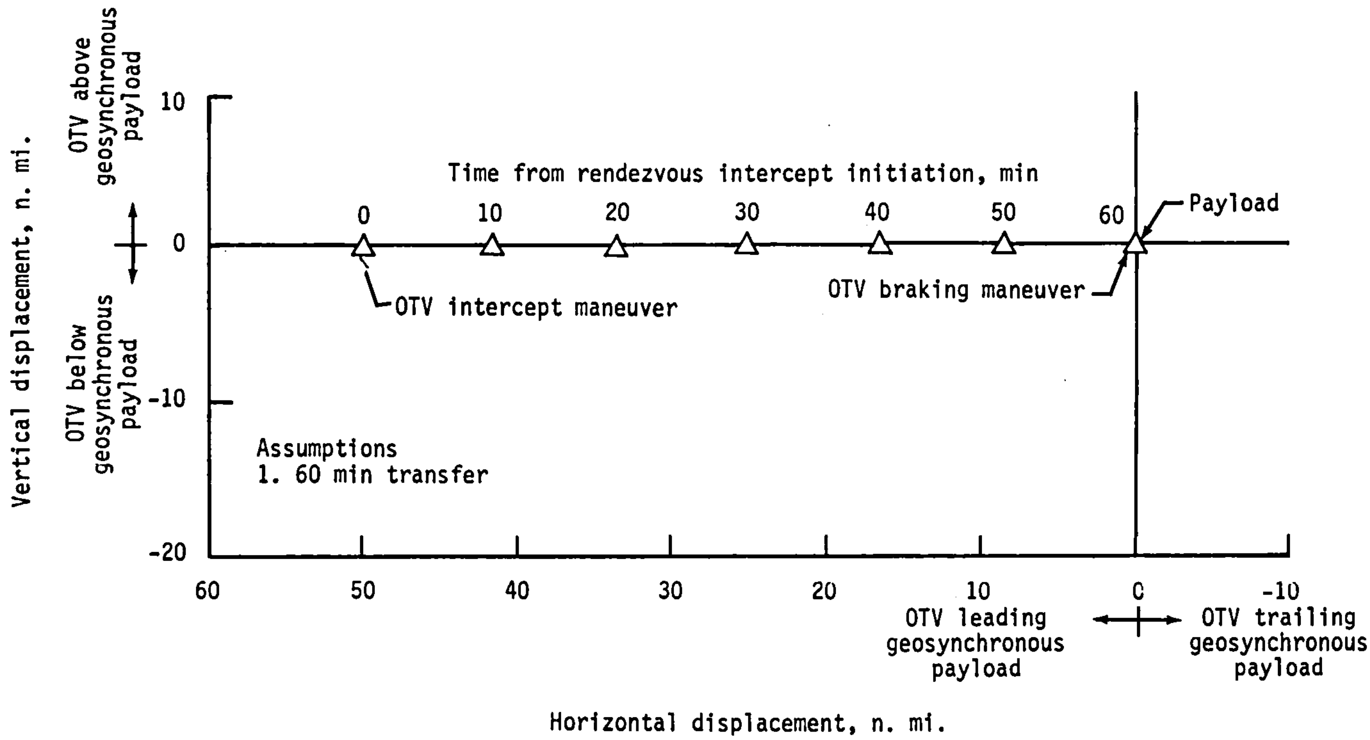
(b) Geosynchronous intercept.

Figure 2-10.- Concluded.

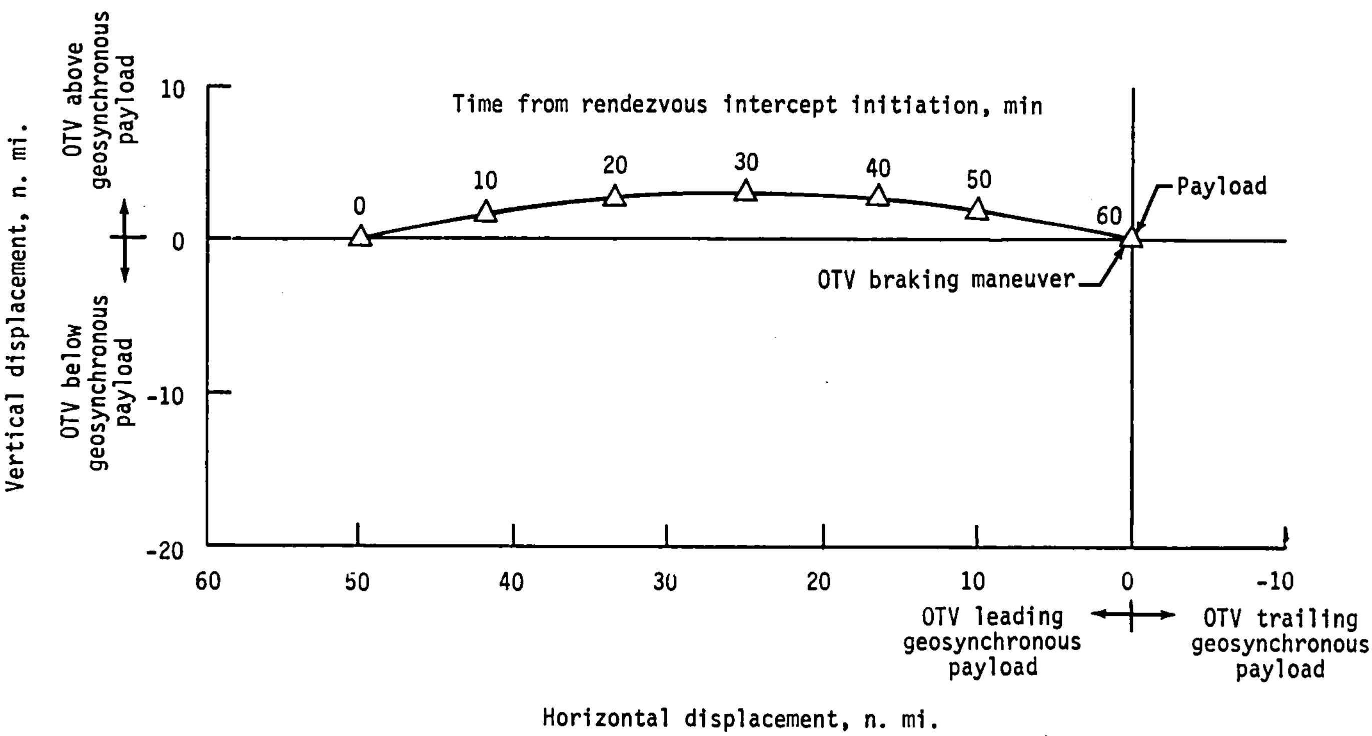


Option	Pre-intercept geo circularization	Terminal initiation position
A	No	50 n. mi. ahead
B	Yes	50 n. mi. behind
C	No	50 n. mi. behind
D	Yes	50 n. mi. ahead

Figure 2-11.- OTV energy requirements for geosynchronous rendezvous from low Earth orbit.



(a) Inertial coordinate system coincident with the local vertical at T = 0.



(b) Rotating local vertical curvilinear coordinate system.

Figure 2-12.- Typical relative motion during geosynchronous terminal rendezvous.

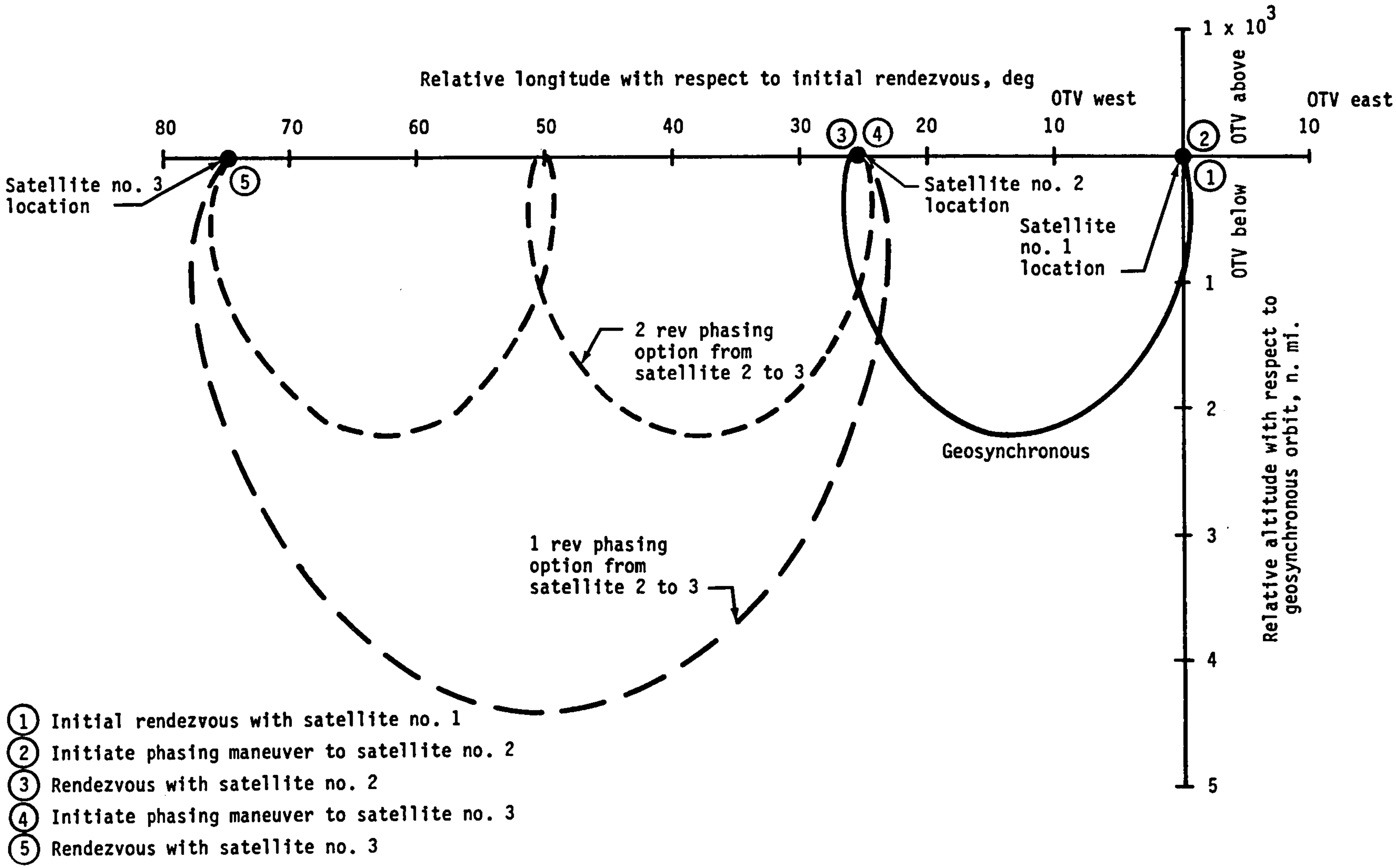


Figure 2-13.- Typical relative motion for servicing multiple geosynchronous satellites.

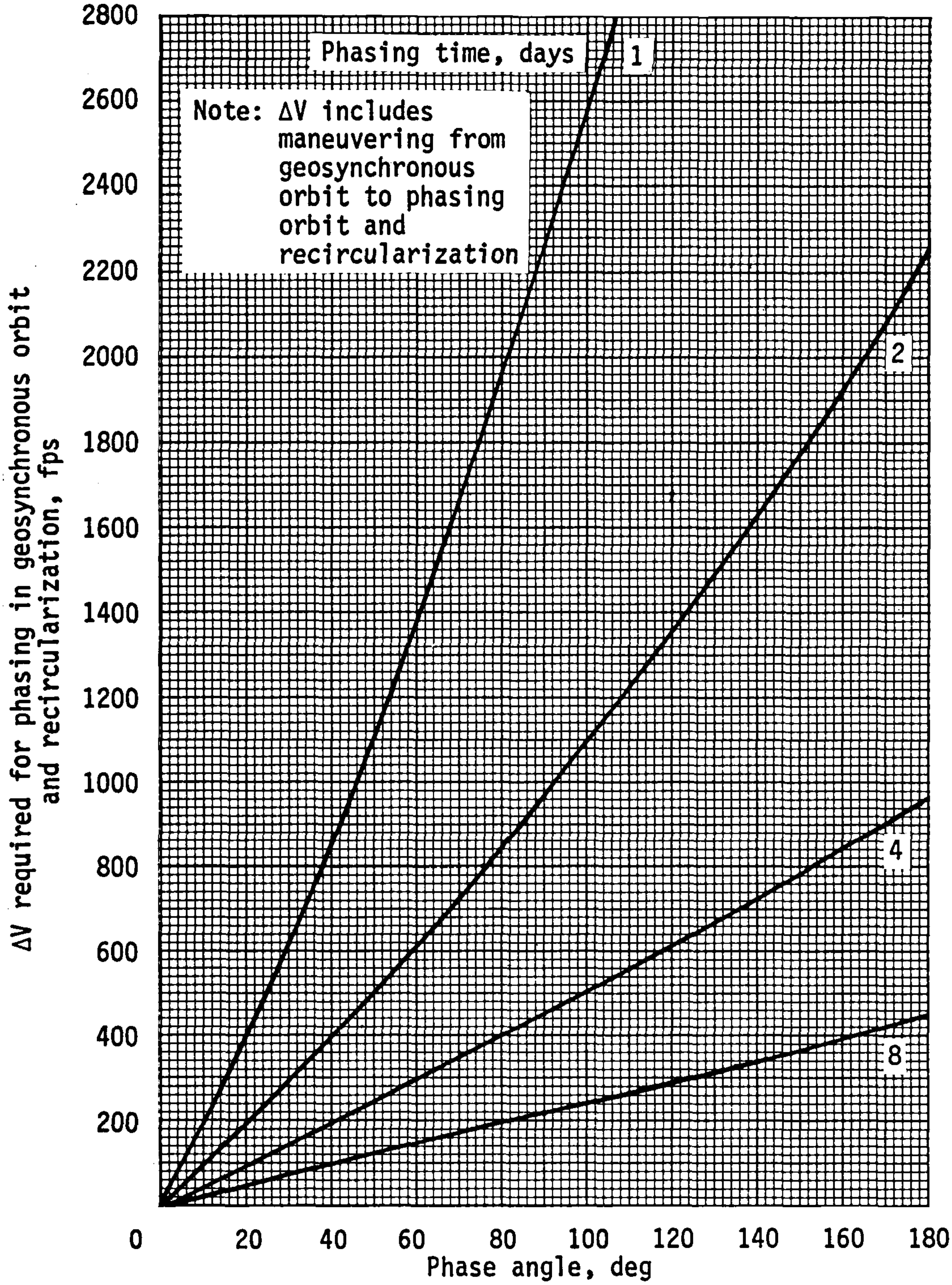
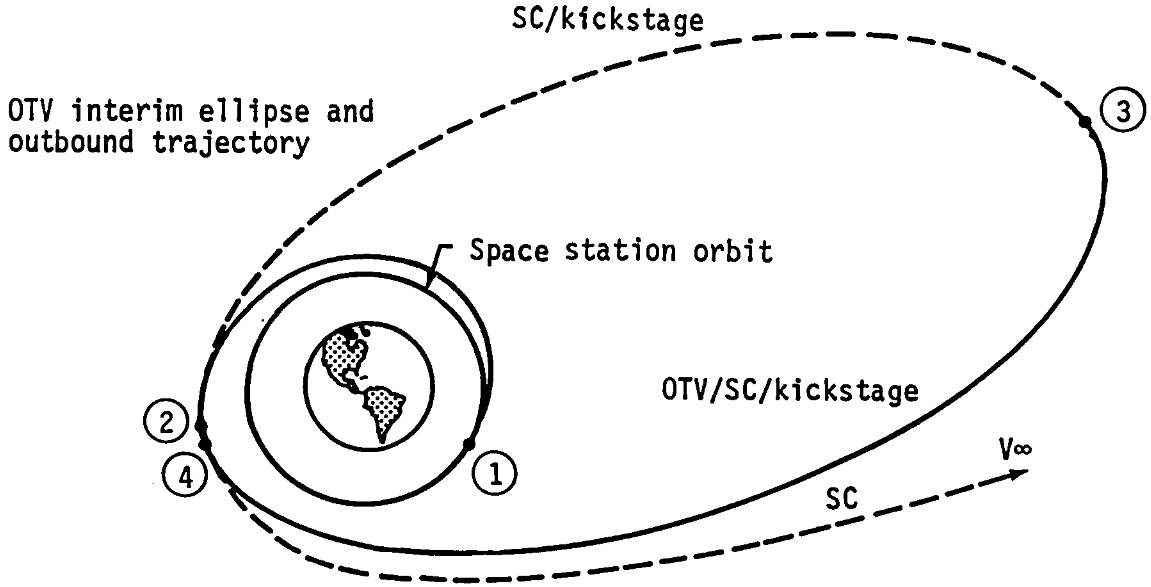


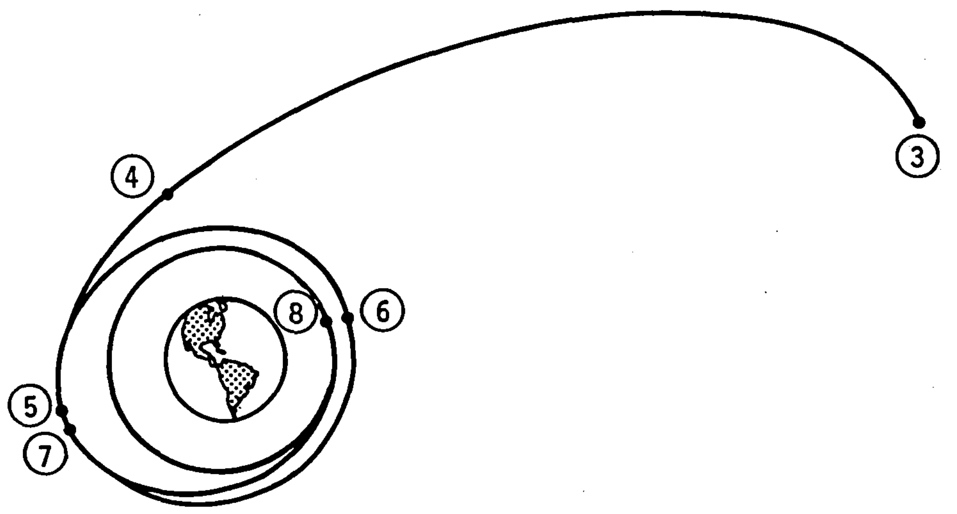
Figure 2-14.- ΔV required to phase in geosynchronous orbit.



	Elapsed time, hr:min	Event	ΔV, fps	H_A/H_P, n. mi.	θ, deg
1		OTV/SC separation from space station	20	210/200	0
2	0	OTV/SC transfer to high ellipse	8930	36450/210	
3	11:11	OTV separation from SC	TBD	36450/210	
4	22:22	SC kickstage injects on escape trajectory (typical C_3 requirement)	3000	∞/210	

(a) Planetary trajectory insertion.

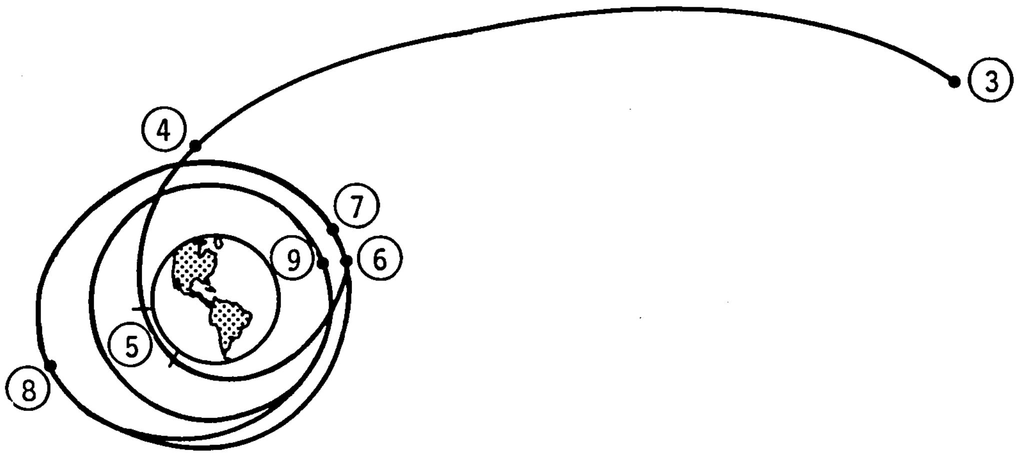
Figure 2-15.- Planetary support flight scenario.



	Elapsed time, hr:min	Event	ΔV, fps	H_A/H_P, n. mi.	θ, deg
3	11:11	OTV separation from SC/perigee height adjustment	77	36450/400	
4	21:55	Nodal alinement/plane change	982	36450/400	
5	22:26	OTV circularization maneuver	8602	400/400	-116
6	26:35	Phase correction maneuver	280	400/230	-50
7	28:59	Height maneuver to lower perigee	333	230/200	-25
8	46:44	Stable orbit maneuver	51	200/200	.2
	48:00	OTV final approach and docking	(small ΔV's)	200/200	0

(b) OTV return, all-propulsive option.

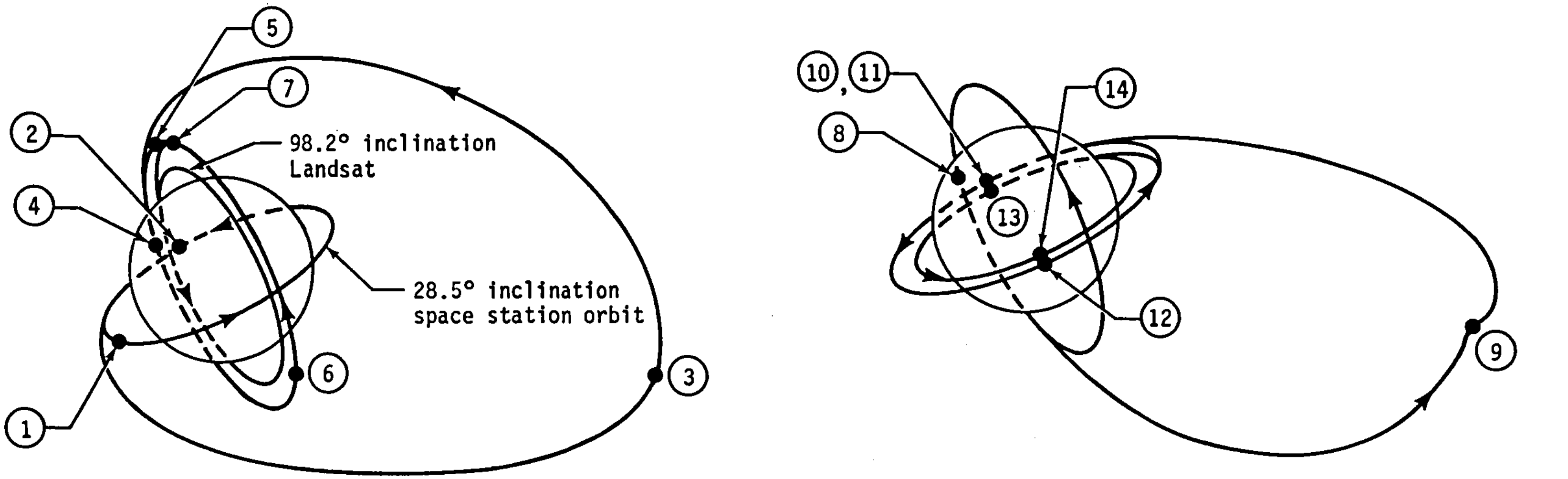
Figure 2-15.- Continued.



	Elapsed time, hr:min	Event	ΔV, fps	H_A/H_P, n. mi.	θ, deg
3	11:11	OTV separation from SC/perigee height adjustment	62	36450/45	
4		Nodal alinement/plane change	1025	36450/45	
5	22:17	Aerobraking phase		407/45	
6	23:04	OTV circularization	612	407/407	-148
7	26:18	Phase correction maneuver	240	407/260	-97
8	30:21	Height maneuver to lower perigee	344	260/200	-47
9	46:23	Stable orbit maneuver	104	200/200	.2
	48:00	OTV final approach and docking	(small ΔV's)	200/200	.2

(c) OTV return, aerobraking option.

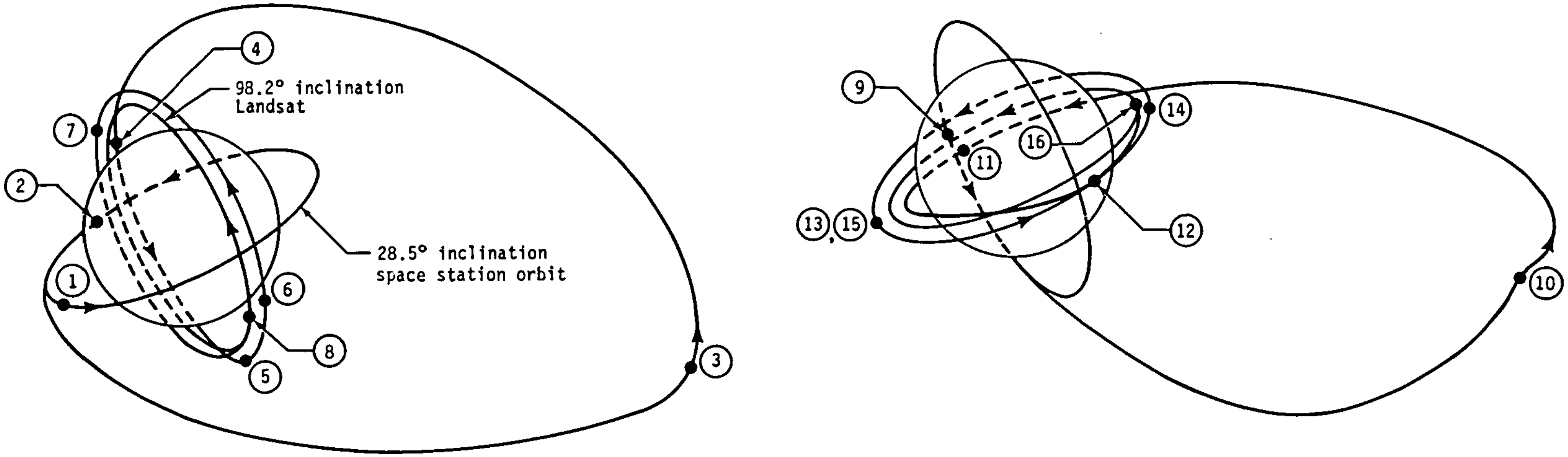
Figure 2-15.- Concluded.



Number	Event	Number	Event
1	OTV separation from space station	8	Height maneuver to raise apogee
2	Height maneuver to raise apogee	9	Major plane change; lower perigee
3	Major plane change; raise perigee	10	Circularize 200 n. mi. above space station
4	Circularize at 200 n. mi. above Landsat	11	Phasing maneuver
5	Phasing maneuver	12	Height maneuver
6	Height/plane change maneuver	13	Stable orbit maneuver
7	Stable orbit maneuver, dock with Landsat	14	OTV docks with space station

(a) All-propulsive.

Figure 2-16.- Sun-synchronous support flight scenario.



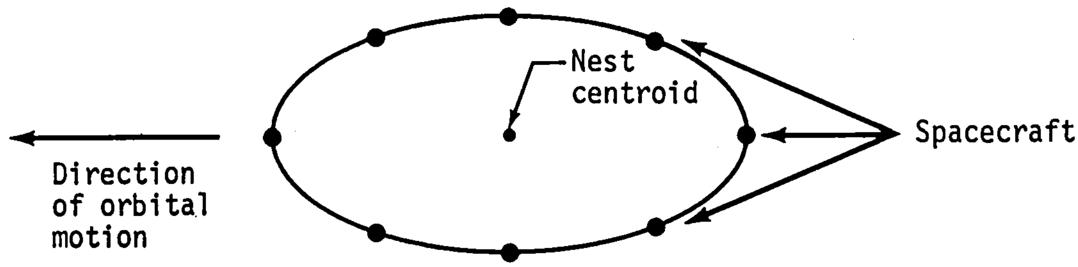
Number	Event
1	OTV separation from space station
2	Height maneuver to raise apogee
3	Major plane change at apogee, lower perigee to achieve aerobraking
4	Perigee at 45 n. mi.
5	Apogee resulting from aerobraking; circularize 200 n. mi. above Landsat
6	Phasing maneuver
7	Height/plane change maneuver
8	Stable orbit maneuver, dock with Landsat

Number	Event
9	Height maneuver to raise apogee
10	Major plane change at apogee, lower perigee to achieve aerobraking
11	Perigee at 45 n. mi.
12	Apogee resulting from aerobraking; circularize 200 n. mi. above space station
13	Phasing maneuver
14	Height maneuver
15	Stable orbit maneuver
16	OTV docks with space station

(b) Aerobraking.

Figure 2-16.- Concluded.

- A number of spacecraft can be placed in this orbit, each at a different relative motion phase angle or "time" position



- Several different spacecraft orbit planes can be established, each with a different range from the space station, allowing the "nesting" of co-orbiting spacecraft

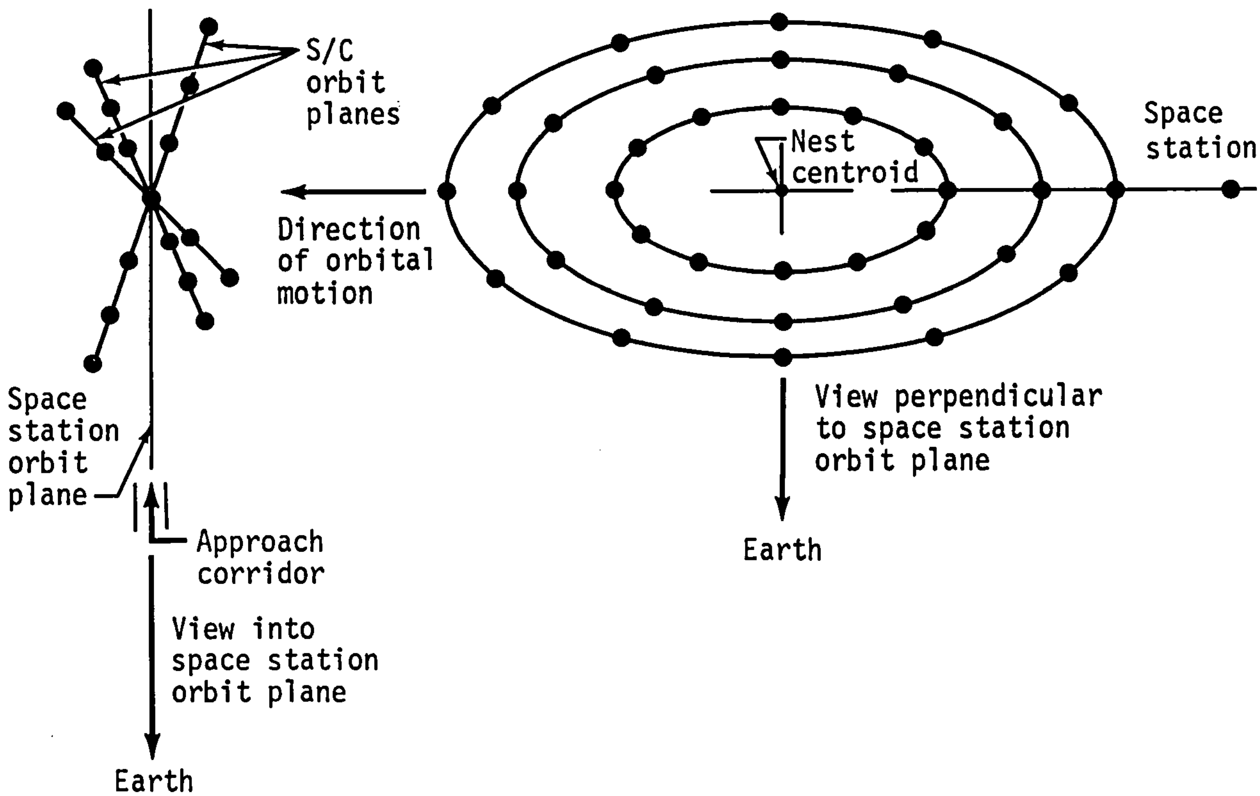
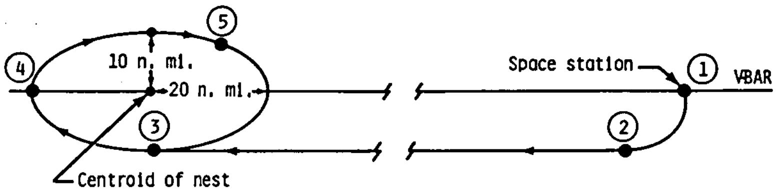
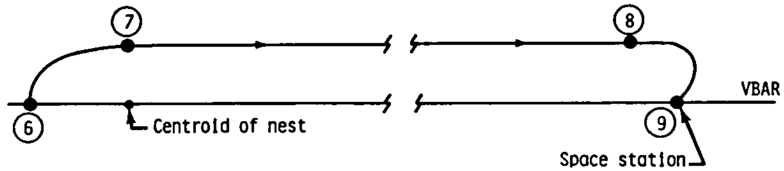


Figure 2-17.- Co-orbiting spacecraft.



a) Space station-to-nest.



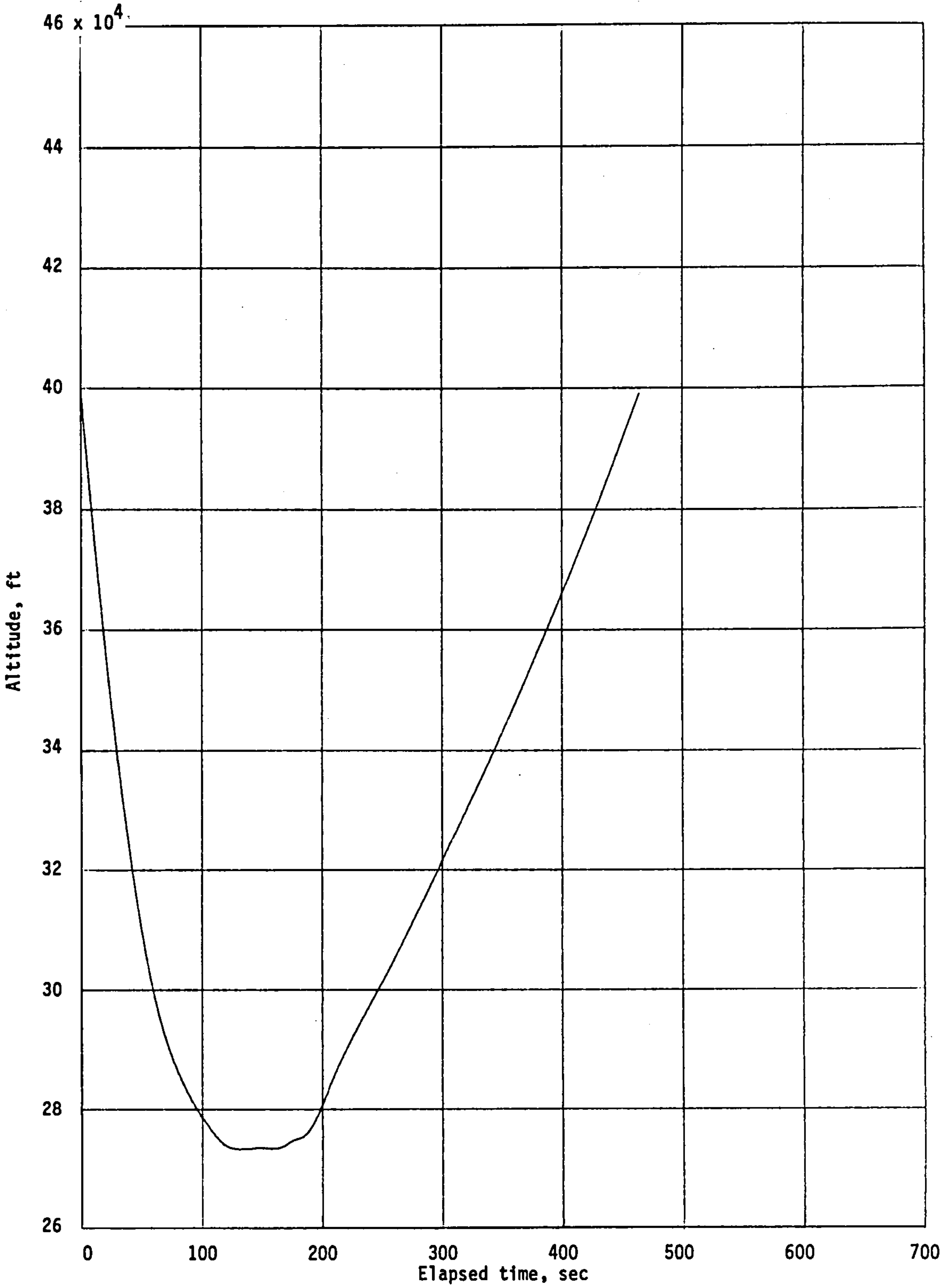
b) Return-to-space station.

Maneuver #	MET hr:min	Description	ΔV fps
1	00:00	Height maneuver to initiate transfer from space station to nest	22
2	00:45	Orbit maneuver to co-circular orbit	24
3	05:45	Maneuver to place FV in nest	36
4	06:08	Maneuver to change plane and initiate intercept	32
5	06:41	Brake at stable orbit offset	10

Maneuver #	MET hr:min	Description	ΔV fps
6	07:38	Maneuver to change plane	32
7	08:00	Maneuver to co-circular orbit	24
8	13:00	TPI to initiate space station intercept	22
9	13:32	Brake at stable orbit offset	24

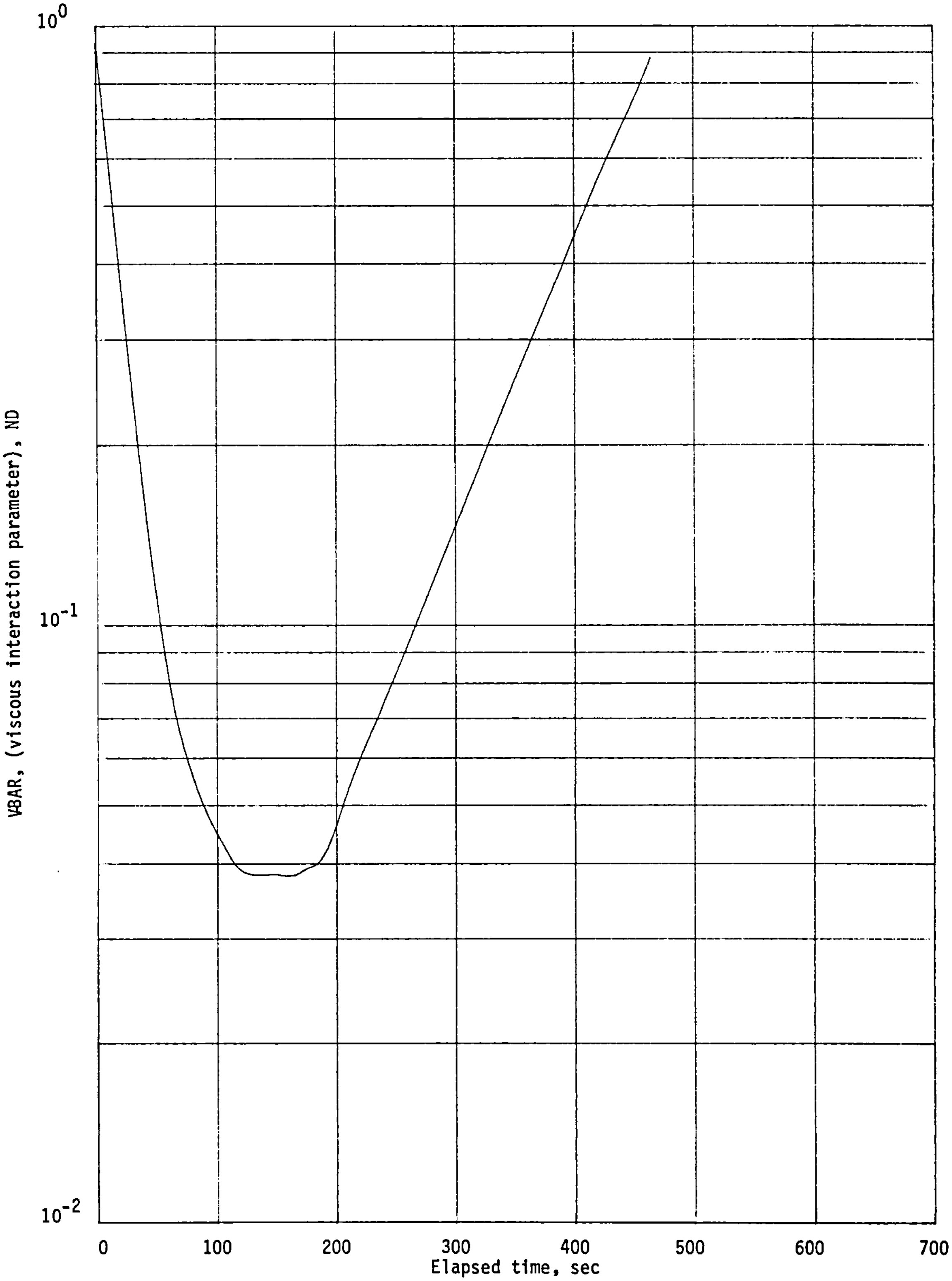
Total ΔV requirement: 226 fps

Figure 2-18.- Co-orbiting satellite mission scenario.



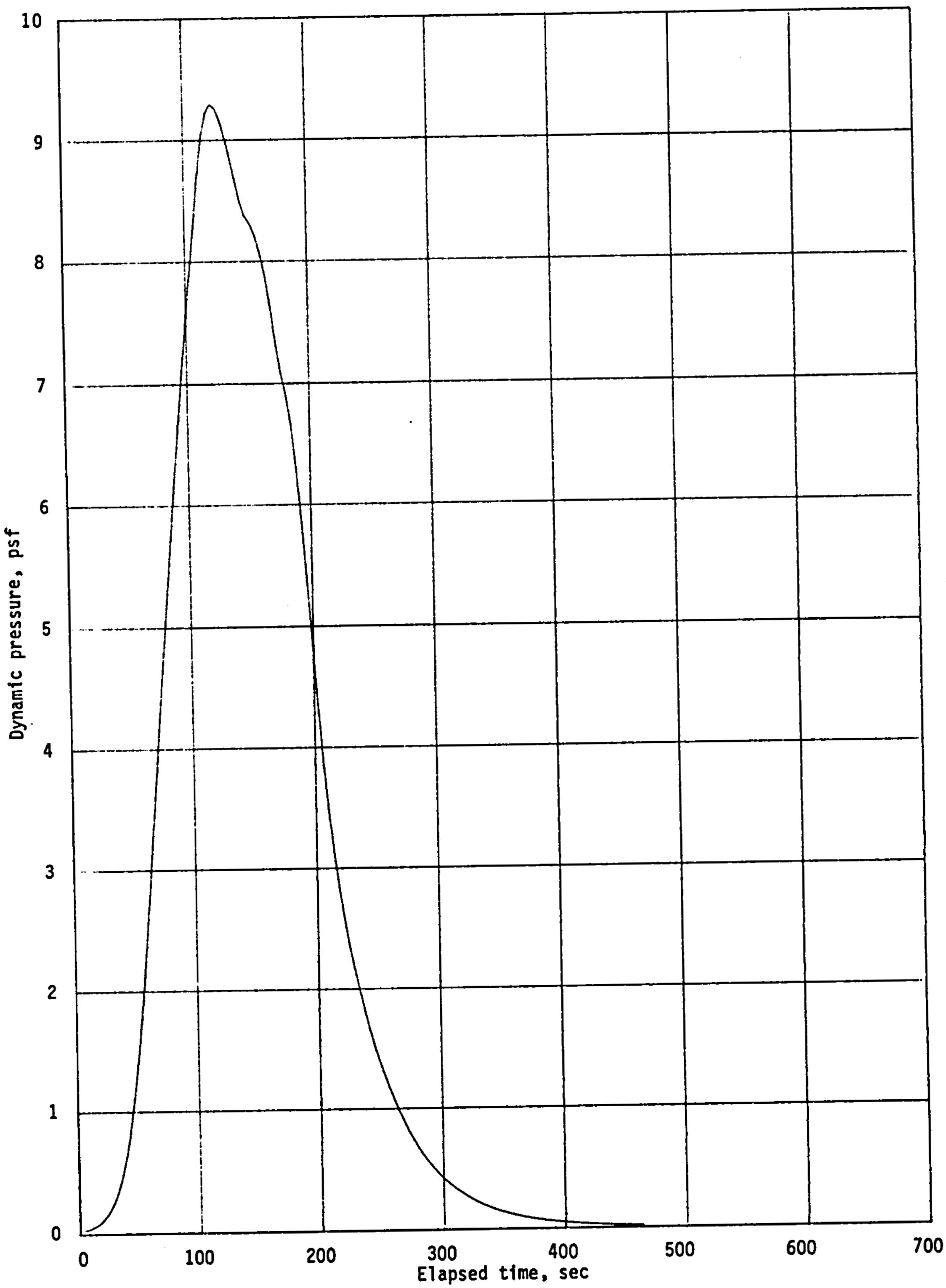
(a) OTV atmospheric flight altitude profile.

Figure 2-19.- Flight profiles.



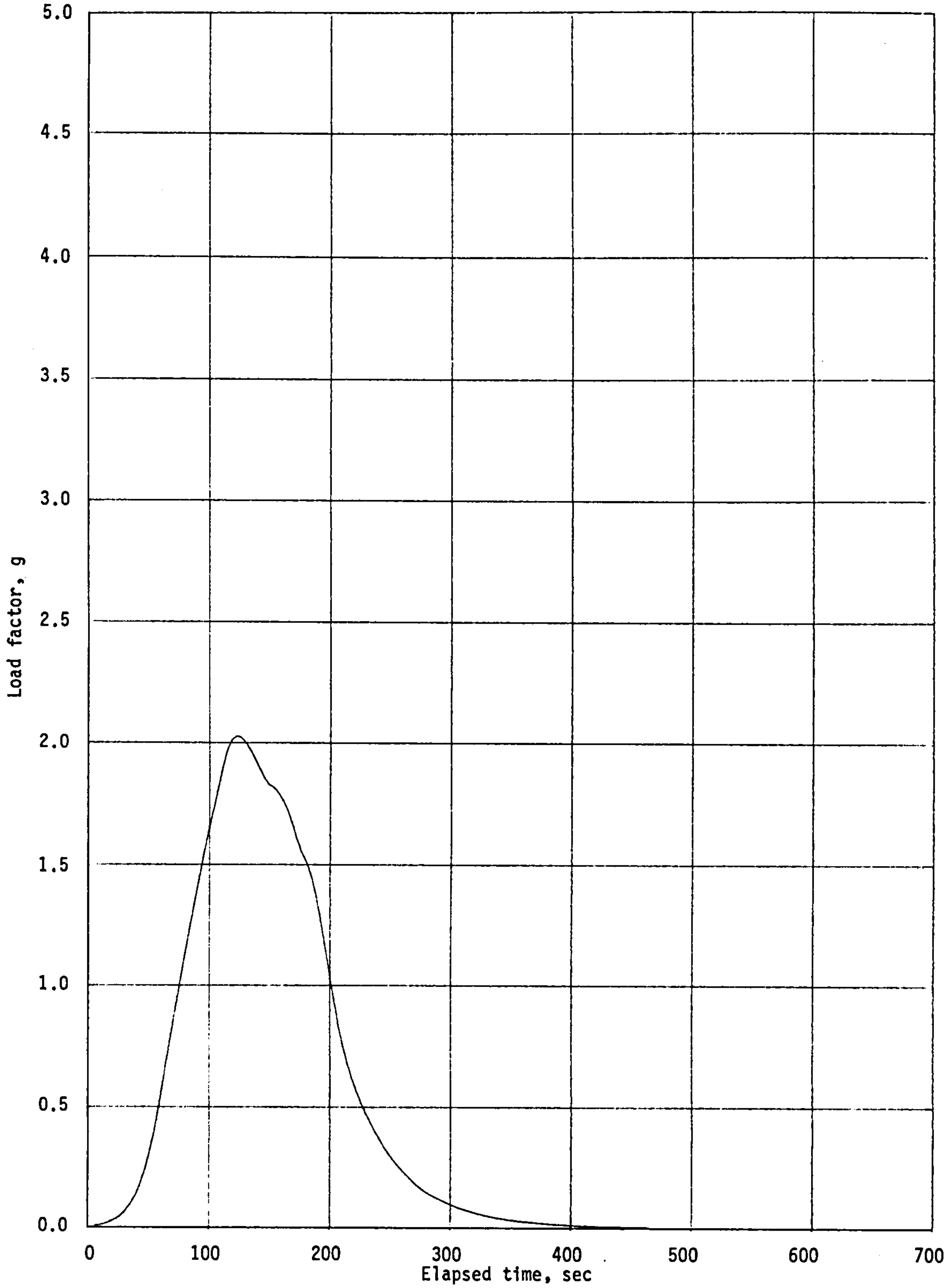
(b) Viscous interaction profile.

Figure 2-19.- Continued.



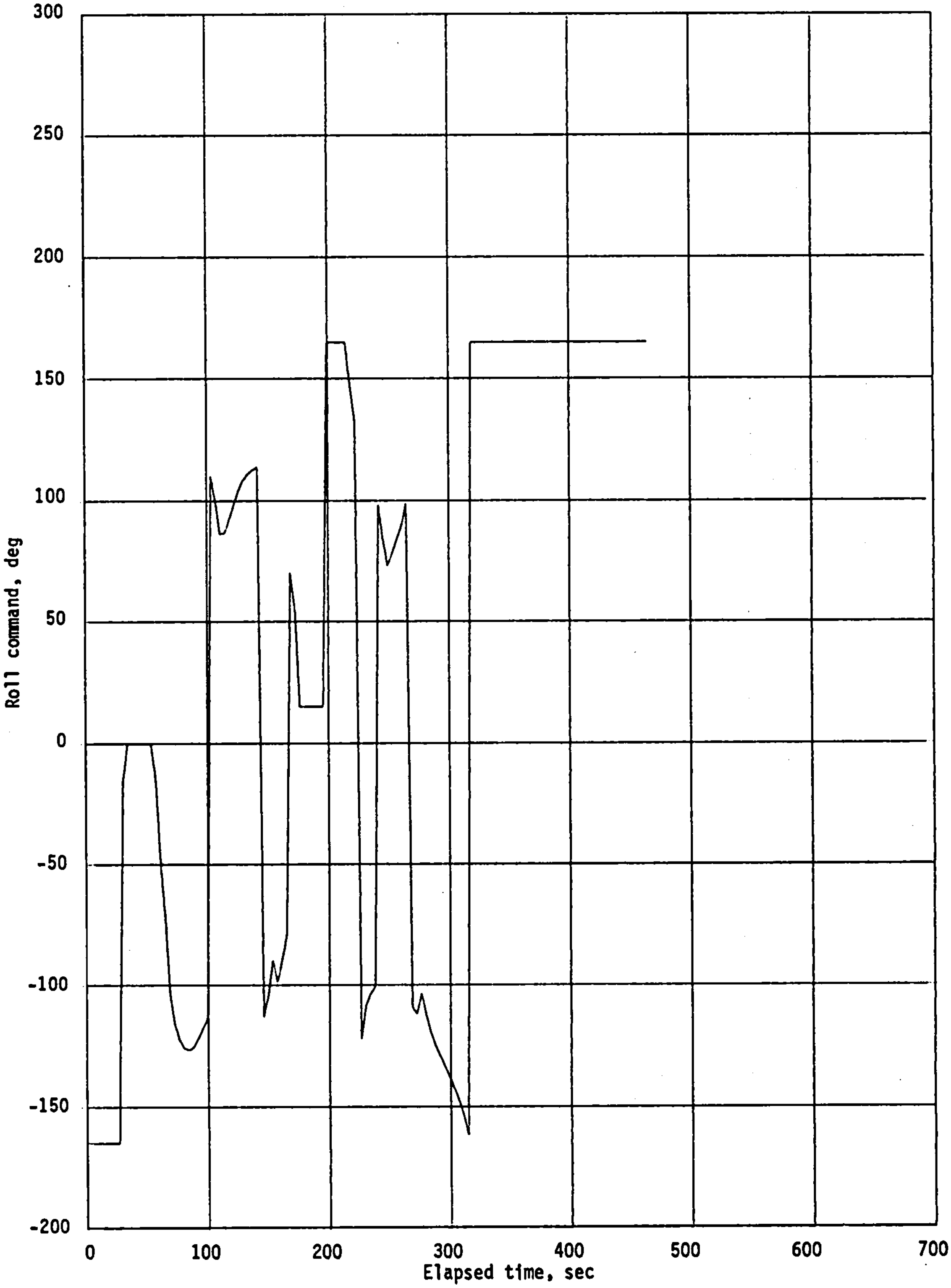
(c) Dynamic pressure profile.

Figure 2-19.- Continued.



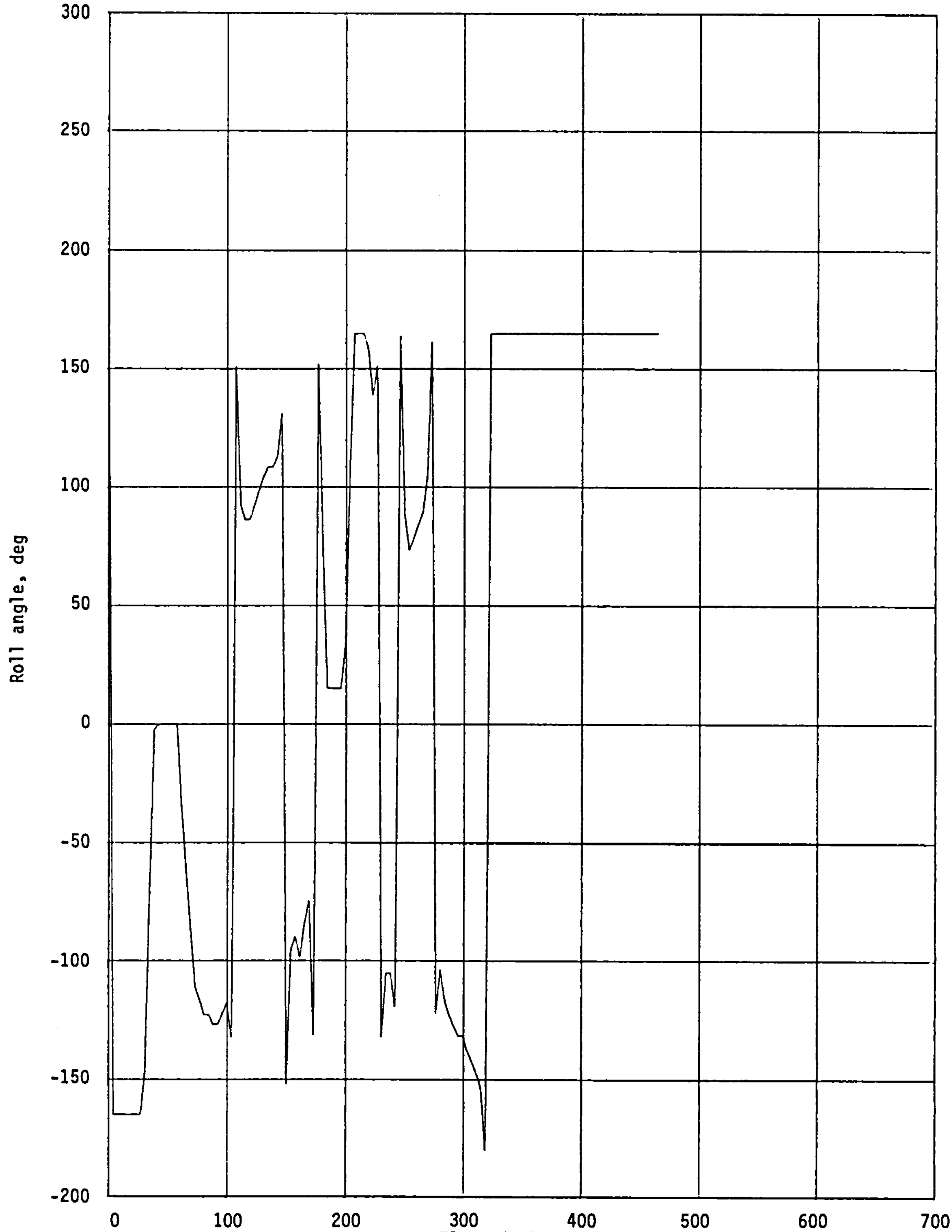
(d) Load factor profile.

Figure 2-19.- Continued.



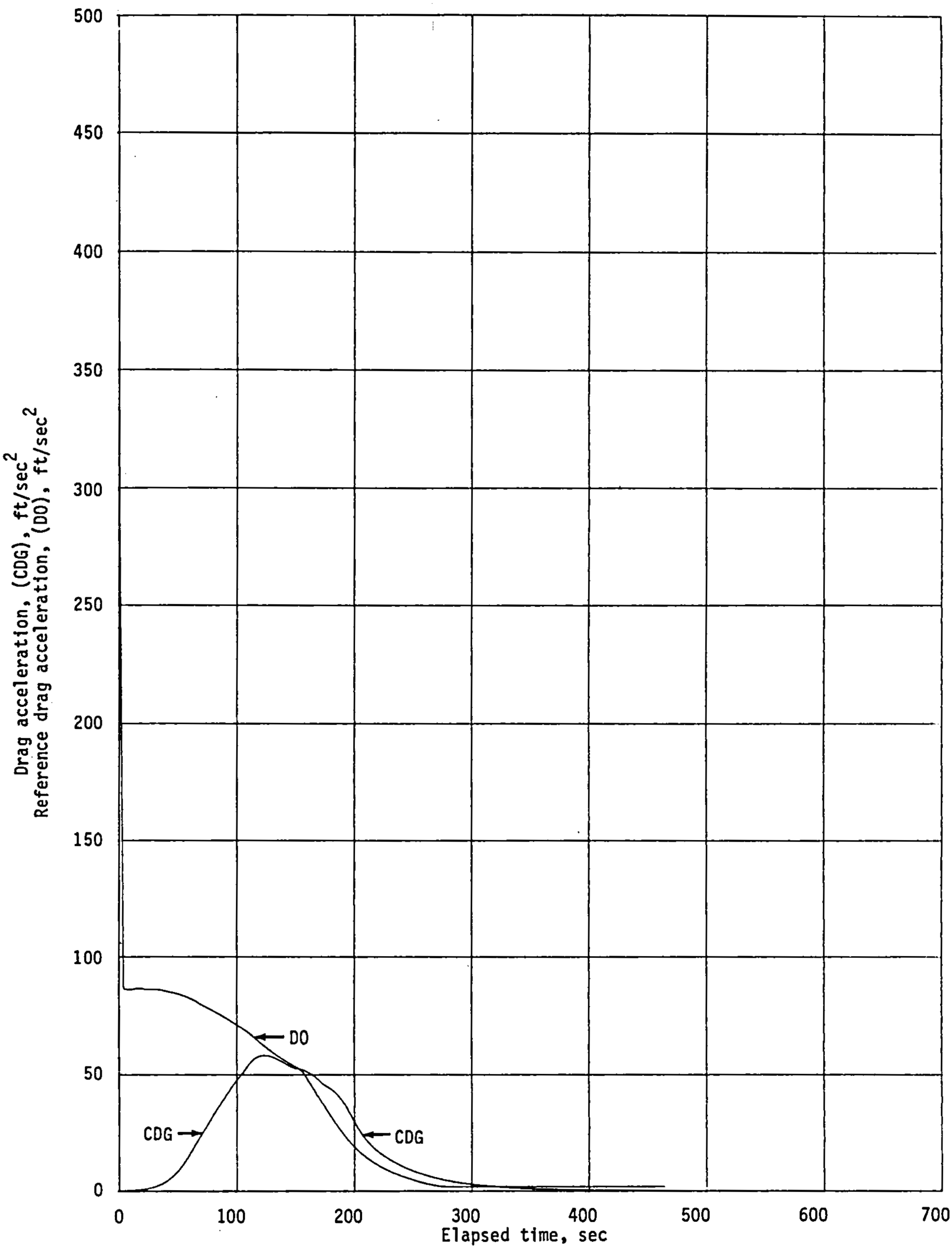
(e) Roll command profile.

Figure 2-19.- Continued.



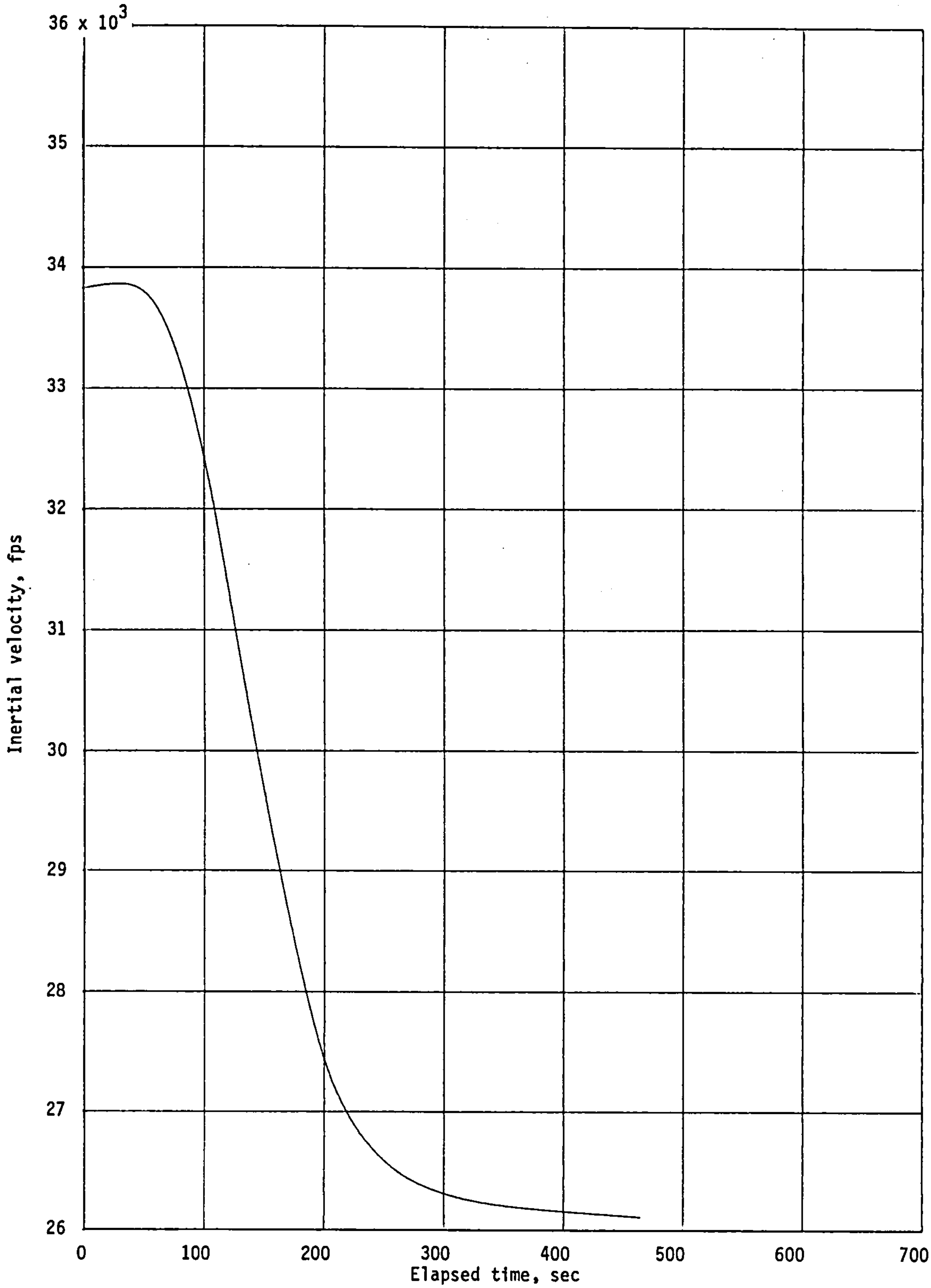
(f) Roll angle profile.

Figure 2-19.- Continued.



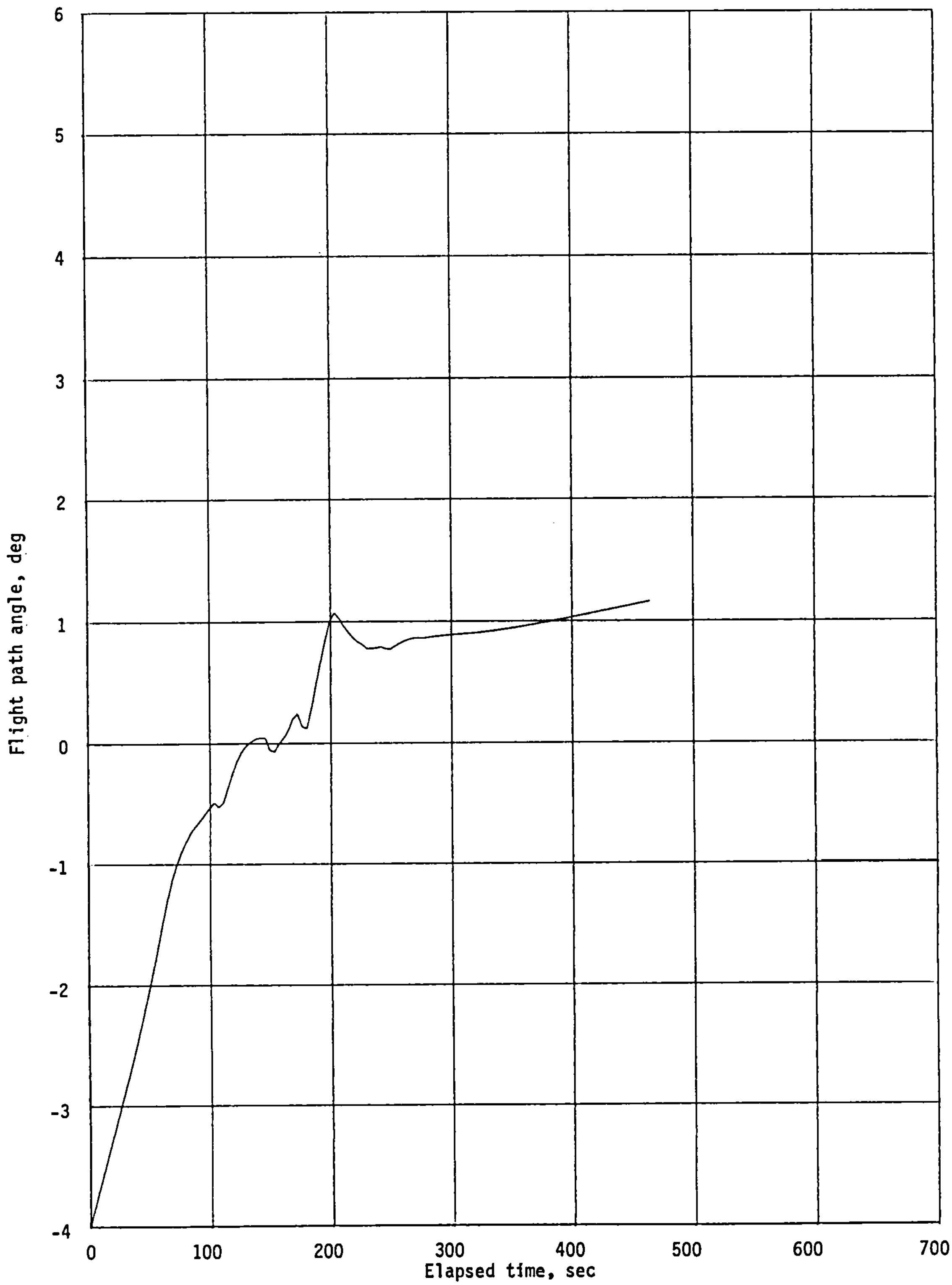
(g) Reference drag acceleration and drag acceleration profile.

Figure 2-19.- Continued.



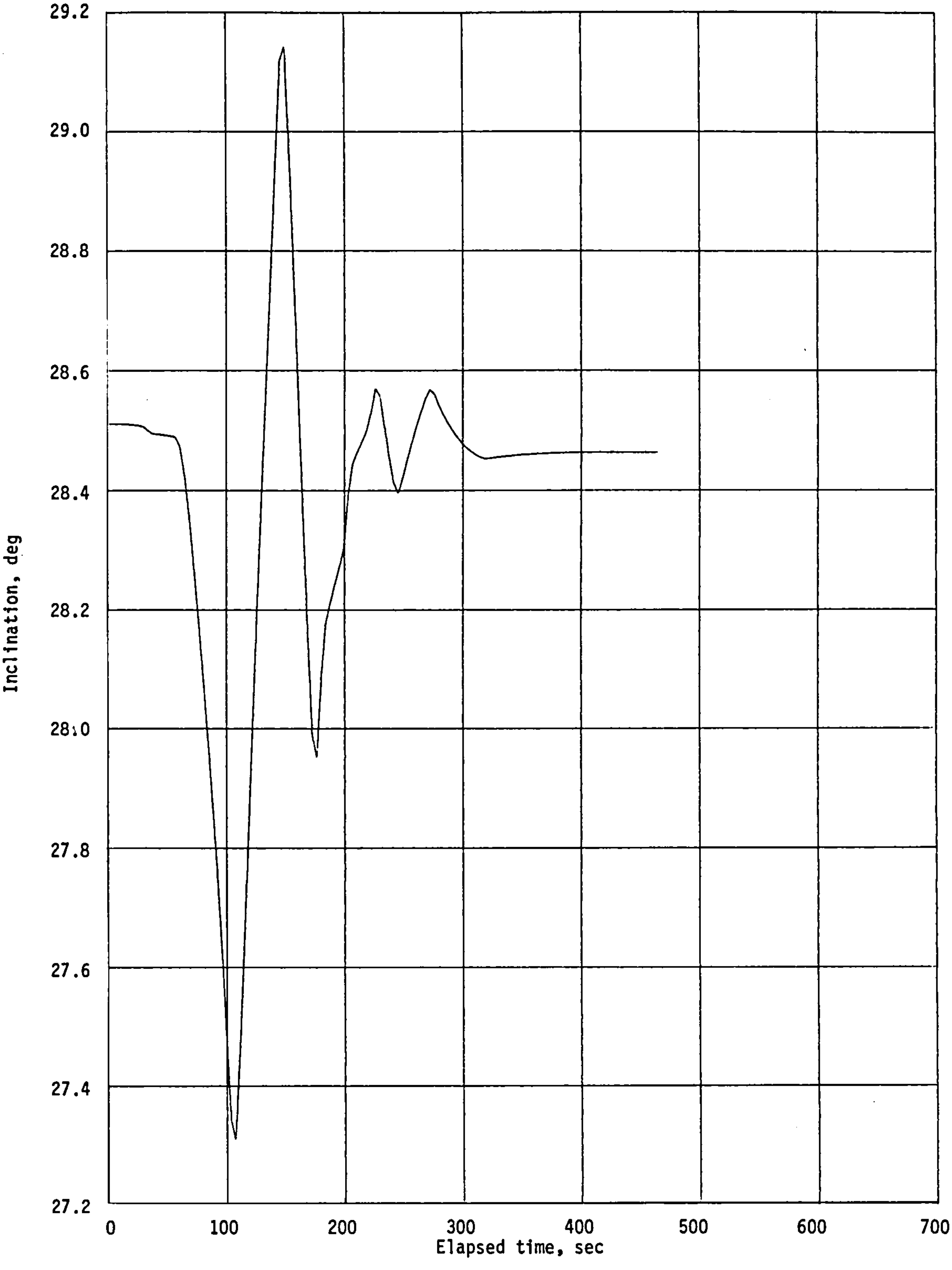
(h) Inertial velocity profile.

Figure 2-19.- Continued.



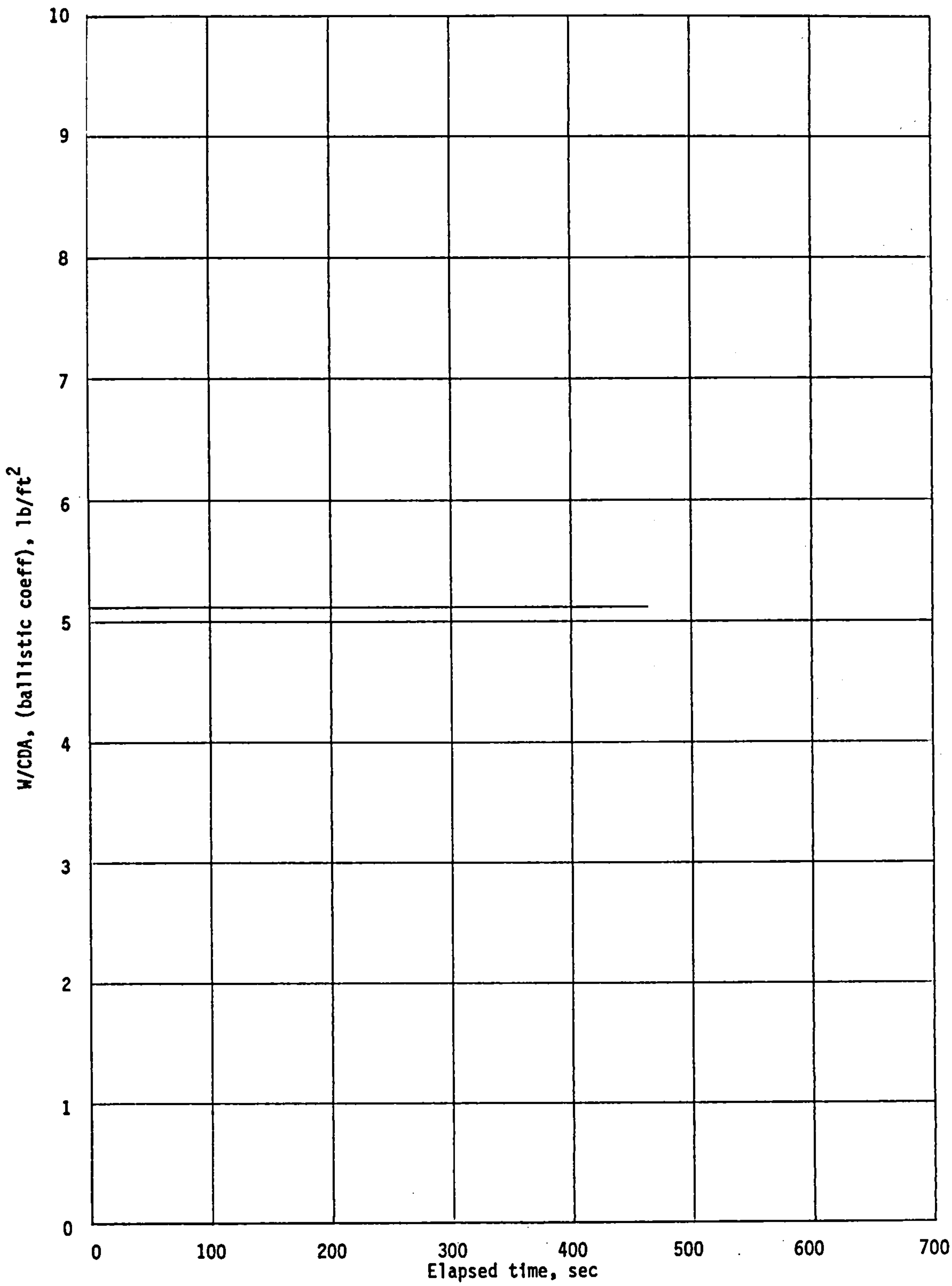
(i) Inertial flightpath angle profile.

Figure 2-19.- Continued.



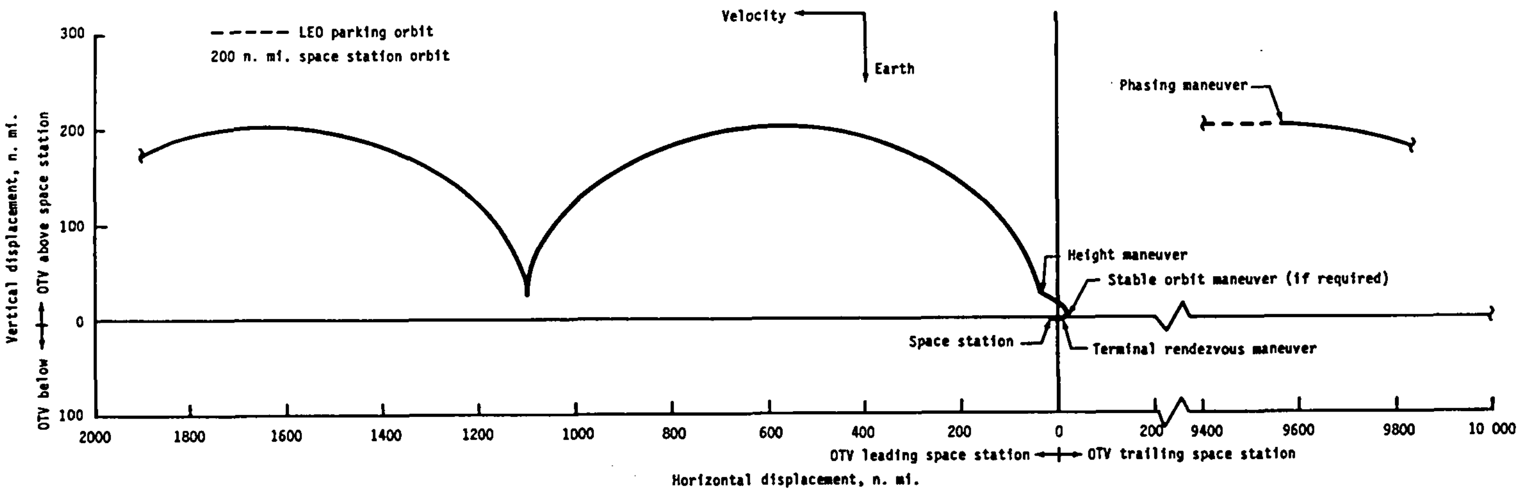
(j) Inclination profile.

Figure 2-19.- Continued.

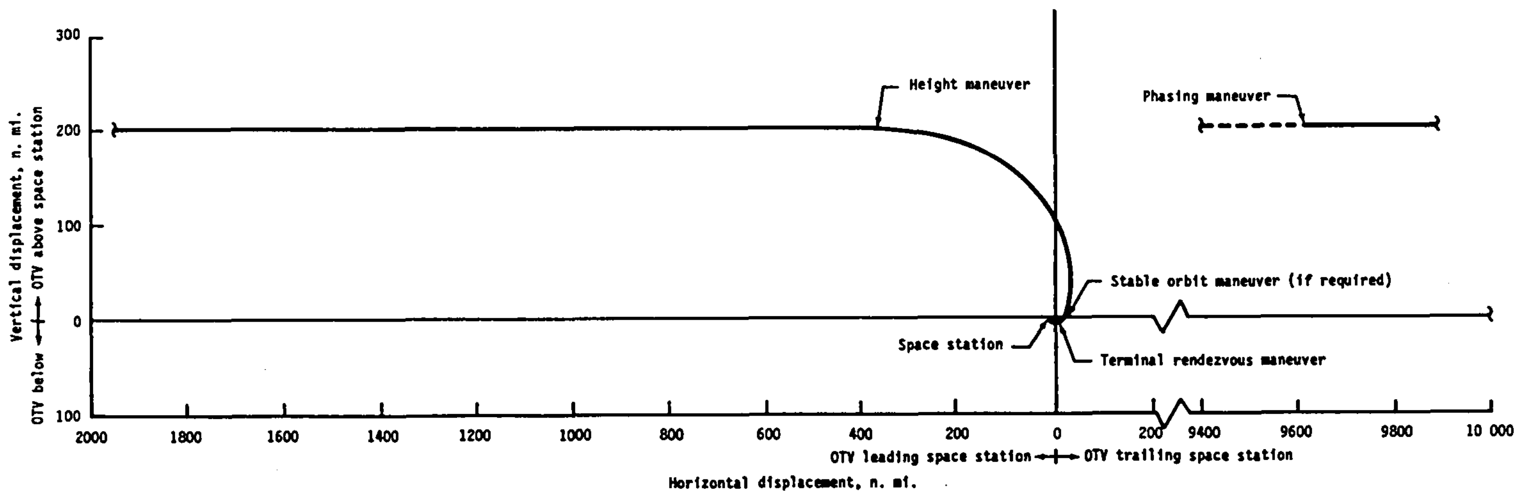


(k) Ballistic coefficient profile.

Figure 2-19.- Concluded.



(a) 400 n. mi./225 n. mi. phasing orbit (minimum).



(b) 400 n. mi. circular phasing orbit (maximum).

Figure 2-20.- Typical relative motion during OTV/space station rendezvous phase (rotating curvilinear coordinate system).

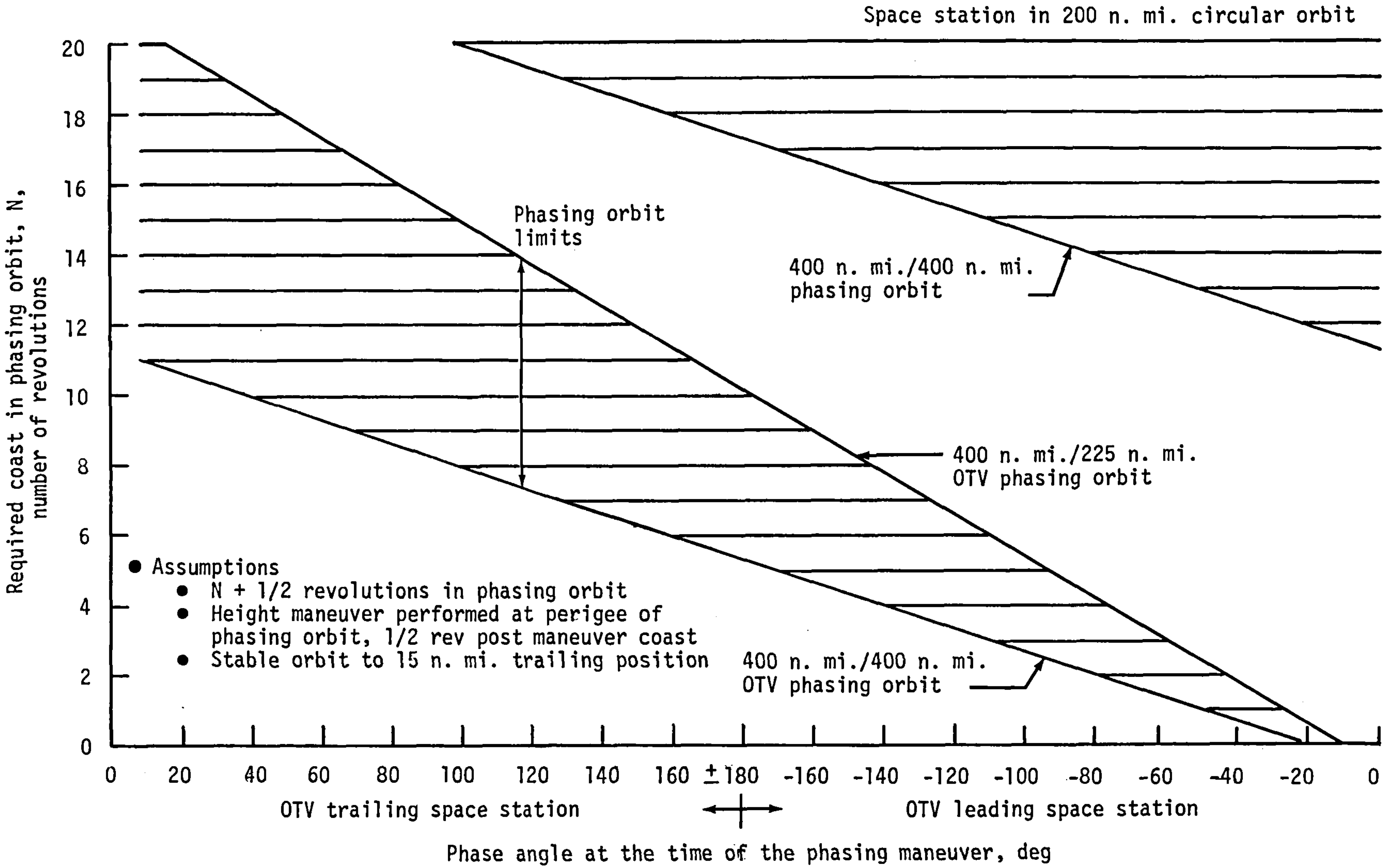


Figure 2-21.- Phasing orbit coast requirements as a function of phase angle.

3.0 OTV SIZING

3.1 INTRODUCTION

The concept of space based operations pivots around the development of an efficient, high energy, orbital transfer vehicle stage. The OTV's are fueled and mated to their payload(s) at the space station and then deliver the payloads to their final destination.

The direct transportation costs for the payloads then reduce to the cost of delivering the various materials to LEO plus replacement cost of any expended stages. When the OTV's are recovered, their replacement costs can be amortized over 10 to 15 flights, making them a very small percentage of the operation's cost. Thus the direct operations dollar cost of delivering a pound of payload to GEO is simply the Shuttle launch cost per pound of cargo to the space base multiplied by the number of pounds of cargo required at the space base per pound of payload delivered to GEO.

This latter number is given for each case in this study and is used in comparison of costs or cost effectiveness.

Sizing of the OTV is crucial to this operational cost effectiveness. Previous studies have tended to tie OTV sizing to some particular payload requirement coupled with some specific Shuttle lift capability. This can easily result in an OTV stage that is inefficient when considering overall operations.

This present study views the OTV as part of a generalized transportation system with the following characteristics:

A. The Shuttle will be the primary launch vehicle to low Earth orbit.

B. The OTV will be a high energy stage that is restartable, has long duration (weeks) mission time capability, has extensive on-board intelligence, and is designed to fit inside the Shuttle for delivery to LEO.

C. The propellant is LOX/H_2.

D. Space based missions are not constrained to the weight of propellant and payload that can be delivered in an integral number of full Shuttle flights.

3.2 GROUNDRULES AND CONSTRAINTS

A. OTV ISP = 460 sec.

B. OTV burnout wt. = 4000 lb. + .08 X (propellant wt.)
Propellant wt. = usable propellant in full stage.

C. Aerobraker capability is a kit add-on to regular stage.

D. Two-stage operation uses two identical fully fueled OTV stages in series.

E. Propellant capacity quoted is usable propellant. Residuals and vehicle reserves are included in burnout weight.

F. ΔV Budgets are

1. Geosynch Impulsive

   Outbound ΔV = 14 000 fps
   Return ΔV = 14 000 fps

2. Geosynch Aerobraking

   Outbound ΔV = 14 000 fps
   Return ΔV = 7 000 fps

3. Sun-Synchronous Impulsive

   Outbound ΔV = 20 800 fps (best departure time)
   Return ΔV = 21 700 fps (worst return time)

4. Sun-Synchronous Aerobraking

   Outbound ΔV = 12 600 fps (best departure time)
   Return ΔV = 12 600 fps (worst return time)

3.3 OTV OPERATIONS

3.3.1 Shuttle Based Operations

Shuttle based operations use the OTV as a direct extension of the Shuttle vehicle. For a GEO delivery mission the Shuttle is launched with a fueled OTV in the bay attached to the GEO bound payload. The total weight (OTV, propellant, and payload) may not exceed the Shuttle deployed cargo capability.

When LEO is reached the Shuttle deploys the OTV which then transfers its payload to GEO. If the OTV is to be recovered it deploys the payload and returns to LEO where it is recovered by the Shuttle either immediately or on a later launch.

3.3.2 Space Based Operations

Space basing utilizes a space station as a storage and operations depot. The Shuttle, over a number of flights, delivers the various mission elements (OTV's, propellant, and payloads) to the depot. The payloads and OTV's are checked out and readied for flight. They are mated to each other and the OTV's are fueled. The OTV then departs from the space station and transports the payload to its final destination. After the payloads are separated, the OTV's return to the space station for use on subsequent missions.

Rather extensive storage, checkout, repair, and handling facilities at the space station are implied in this concept.

3.4 FLIGHT PROFILES

3.4.1 Space Station to GEO - Impulsive Transfer

Impulsive transfer to GEO from a low earth orbit (see section 2.4.1) at 28.5° inclination, begins with a large (8000) fps posigrade burn as the OTV crosses the equator. This raises apogee to the GEO altitude of 19 323 n. mi. and makes $\approx 2^{\circ}$ of the plane change. At apogee a second burn of $\approx$6000 fps circularizes and provides the remainder of the plane change (26.5°). The payload is deployed after which the OTV rendezvous with any payload that is to be returned and retrieves it. The OTV then coasts around the GEO orbit until it crosses the plane of the space station (one of the space station nodal crossing points) where a mirror image of the outbound transfer is performed to return to the space station orbit for rendezvous.

3.4.2 Space Station to GEO - Aerobraking Transfer

An aerobraking OTV uses the standard all-impulsive transfer outbound to GEO. Aerobraking is only of use during the return trip.

The first burn of the return leg is a retrograde burn that makes the entire 28.5° plane change and drops perigee to $\approx$45 n. mi. This dips the lower end of the orbit into the atmosphere where a controlled aerobraking maneuver dissipates energy to bring apogee down to near space station altitude. The OTV exits the atmosphere, coasts up to apogee and circularizes in preparation for rendezvous. The net result is a reduction in the ΔV required from the rocket engine of $\approx$7000 fps.

3.4.3 Space Station to Sun-Synchronous - Impulsive Transfer

The sun synchronous orbits considered here are at an inclination of 98° and an altitude of 380 n. mi. The space station is at an inclination of 28.5° and an altitude of 200 n. mi. The plane change necessary to transfer between the two orbits (i.e., the wedge angle between the two orbit planes) varies from a minimum of 69.5° when the nodes are alined to 126.5° when they are 180° out of phase. The relative precession rates of the two orbits are such that they rotate through the entire cycle of plane changes requirements, from minimum to maximum and back to minimum again, in approximately 40 days.

It was assumed that the transfer to Sun-synch was started at the best possible time (minimum plane change required). Because the round trip time is fairly lengthy, it was further assumed that the return was at the <u>worst</u> possible time (maximum plane change).

For large plane changes such as these, a "three-impulse" transfer is utilized to minimize propellant (ΔV) requirements. At the nodal crossing of the two orbit planes the OTV makes a large posigrade burn (≈10 000 fps) raising apogee to as high a value as practical (here 100 000 n. mi. was used) and leaving perigee unchanged. The vehicle then coasts up to apogee where a properly directed burn makes the plane change (while the OTV is moving very slowly) and also raises perigee to near the Sun-synchronous altitude. The OTV then coasts down to the new perigee where a large retrograde burn (≈10 000 fps) recircularizes.

Returning is a mirror image of the process.

3.4.4 Space Station to Sun-Synchronous - Aerobraking Transfer

Transferring to Sun-synch with an aerobraker again uses a "three-impulse" type transfer. An initial large posigrade burn transfers from a 200 n. mi. circular to a 100 000 x 200 n. mi. ellipse. After coasting to apogee a second burn makes the plane change and also lowers perigee to about 45 n. mi. so that the OTV will enter the atmosphere. The OTV coasts down into the atmosphere around perigee and uses guided aerobraking to reduce apogee altitude to near Sun-synchronous orbit altitude. The OTV then coasts up to this altitude and performs a circularization burn.

The returning sequence is essentially the same. Both outbound and return are transfers from one low circular orbit to another in a different plane using an intermediate high ellipse to reduce the cost of the plane change. Aerobraking is used in each case to dissipate the energy of the intermediate ellipse.

3.5 OTV DESCRIPTION

The OTV stages were assumed to have the following characteristics:

The propellant is LOX/Hydrogen with a mixture ratio of 6:1 and ISP = 460 sec.

The stage is designed to be delivered in the Shuttle bay. A single stage would be basically a cylinder 15 ft in diameter and of varying length depending on propellant capacity but no greater than 60 ft long. The stage would have one or more rocket motors at one end, a set of propellant tanks in the middle, and an attitude control/avionics/payload-interface module at the other end. Stages with propellant capacities of at least 100 000 lb could be fitted within the 60 ft limit.

The nominal inert weight scaling law used was burnout weight equals 4000 lb + .08 x usable propellant wt.

It was assumed that the aerobraking capability could be added to the stage as an add-on kit. Kit weights of 4000 and 8000 lb were examined.

Stages could be stacked one behind the other and provide sequential propulsion for two-stage operations. All stages were identical and interchangable between single stage and dual stage operations.

The OTV's are reusable with a long term (several weeks) onorbit fueled operation.

3.6 TWO-STAGE OPERATIONS

Two-stage operations consist of taking two identical OTV's and mounting them one behind the other with the payload in front of the forward (top) stage. When used in this manner the first (bottom) stage is burned nearly to depletion yielding a ΔV of $\approx$6000 fps (depending on the payload). It then shuts down and separates and the top stage completes the required initial ΔV. The bottom stage coasts 1 rev ($\approx$4 hr) and uses its remaining propellant to recircularize in its original orbit and rerendezvous with the space station. The top stage continues on to complete the normal mission.

For an aerobraking OTV, after separation the bottom (empty) stage dips into the atmosphere to slow itself back down before final circularization.

For this study it was assumed that the bottom stage was always recovered even when the top stage is expended.

3.7 RESULTS

The results of the performance study are shown in figures 3-1 through 3-10.

Each figure is in two parts, both plotted against OTV propellant capacity (OTV size). Part (a) of each figure, delivery capability, shows cargo delivered per OTV flight. Part (b), delivery effectiveness, converts this to pounds of weight delivered to LEO (Shuttle cargo) for each pound delivered to the final destination. Weight to LEO includes propellant for the OTV, weight of the payload to be delivered, and replacement weight for any OTV's expended.

Figure 3-1 shows performance for a Shuttle based operation (i.e., no space station) to a final cargo destination of GEO. This would cover standard operations for the period after OTV availability and before space station operations commence. It was assumed that aerobraking would not initially be available so no such cases were included here.

Performance is shown for OTV expended (one-way case) and for the case with OTV returned empty. These are given for Shuttle deployed cargo per flight of 50K lb, 60K lb, and 70K lb. These are the only cases where Shuttle lift capability is a parameter. Note that if the OTV is optimally sized for the expendable case it is too small for any return capability at all and while under-sizing has a sharp impact on capability, oversizing the OTV is not terribly costly in performance.

Figure 3-2 also gives capability to GEO for a one-way (stage expended) flight, but in this case for a space based system. Both single stage and dual stage operations are shown. In the two-stage case only the top stage is expended (left at GEO). The bottom stage is recovered. No aerobraking is shown as there is no aerobraking opportunity outbound to geosynch. Two-stage operations show a slight advantage in Shuttle delivery requirements (figure 3-2b) and in addition, only expend half as many stages. Expending the top stage of a two-stage stack provides a method of delivering very large payloads to GEO.

Figures 3-3, 3-4, and 3-5 show space based performance to GEO with the OTV's returned. The cases considered were: no payload returned, payload returned equal to one-half the payload delivered, and the payload returned equal to the payload delivered (round trip case). Two-stage and single stage operations with and without aerobraking are considered.

The results shown in these figures follow some very definite trends.

A. Two-stage operations show substantially better performance, both in payload per flight and in efficiency, than single stage operations and allow the use of smaller stages.

B. Aerobraking shows considerable advantage only where there is returned payload.

C. The delivery cost curves (figure 3-3b, 3-4b, 3-5b) exhibit a typical shape throughout the study. Direct delivery costs decrease <u>sharply</u> with increasing OTV size until the knee of the curve is reached. Thereafter the performance efficiency increases but only slowly.

The OTV size necessary for efficient operation (i.e., to clear the knee of the curve) varies with the manner of operation. For the most part, however, it seems that to an OTV stage size of at least 50K lb, with probably 60 to 70K lb being better, would be required for efficient two-stage operations, and roughly twice that size is indicated for the equivalent single stage. At least a 70K lb to 80K lb OTV size (propellant capacity) is required for any reasonable efficiency for a single stage vehicle flying round-trip to GEO.

All of the sizes indicated are considerably larger than the optimum Shuttle based OTV size, which is in the 30 000-40 000 lb range of OTV propellant (depending upon Shuttle lift capability).

Figure 3-6 shows the sensitivity of the results to changes in the basic OTV inert weight scaling laws. These were impulsive delivery missions to GEO with the stages returned empty. The general trends remain the same but the actual optimum size will be very much a function of the actual scaling law of the final design. Note also that a stage weight reduction of 2000 lb reduces the weight required in orbit by 15 to 20 percent. This means that for a spaced based OTV, if the OTV inert weight can be reduced by designing the structure for OTV engine thrust level only and the OTV is launched unfueled, the added efficiency may more than make up for the lost Shuttle payload on the OTV launch flight. In this context one must consider that for an OTV with a ten mission lifespan, approximately one Shuttle launch in 15 will carry a replacement OTV. Thus, the total (fuel) loss of payload to LEO for launching an empty OTV would be a maximum of around 6 percent. The efficiency gain of a light OTV could well be greater than that value.

Figure 3-7 gives OTV one-way delivery capabilities to a Sun-synchronous orbit (inclination = 98°, 500 x 500 n. mi. altitude) from a space station in a 28.5° orbit. The final (top) stage is expended in each case. These transfers are "three-impulse" utilizing an intermediate elliptic orbit with a very high (100 000 n. mi.) apogee. The differential nodal regression of space station and destination orbits results in a minimum plane change occurrence about every 40 days. It was assumed the transfer took place at that time.

As in previous results where high ΔV's were involved the two-stage operations provide a definite advantage. Aerobraking, which for this transfer reduces impulsive ΔV by some 40 percent, more than doubles the payload capability. Moreover, two-stage aerobrakers indicate the possibility of delivering 60K lb class payload to the Sun-synchronous orbit. This is beyond the currently planned Shuttle capability for a single direct launch to these high inclinations.

Figures 3-8, 3-9, and 3-10 give OTV round-trip capability to these same Sun-synchronous orbits, again starting from a 28.5° inclination space station.

The performance shown is for OTV aerobraking both outbound and returning. There is insufficient performance, with the inert weights considered here, to return the OTV if either leg of the trip is all-impulsive.

Again, performance is shown for three cases: the OTV returned empty, one-half the outbound payload returned, and as much payload returned as delivered (round-trip).

The resulting trends are similar to those for the geosynchronous destination. Generally OTV sizes of 60 to 70K lb and two-stage operations provide reasonably efficient operations. In effect we find that a stage sized and designed for efficient transport to GEO with the flexibility of multistage operations, provides the capability for operations to sun-synchronous or any other very distant Earth orbit.

3.8 CONCLUSIONS

A. Optimum OTV sizing for space based operations is considerably different (larger) than for Shuttle based operations.

B. Undersizing the OTV sharply reduces efficiency; oversizing does not.

C. Aerobraking provides significant advantage over all-impulsive operations and the required technology should be vigorously pursued. It is worth noting, however, that most of this advantage is lost unless the OTV can aerobrake while carrying the various OTV payloads.

D. Dual staging (identical stages) is more efficient than single stage operations, and allows use of smaller stages. The ability to stack stages offers considerable operational flexibility.

E. Manifesting of multiple payloads per OTV flight will be required, since most current GEO satellites are smaller than the capability shown.

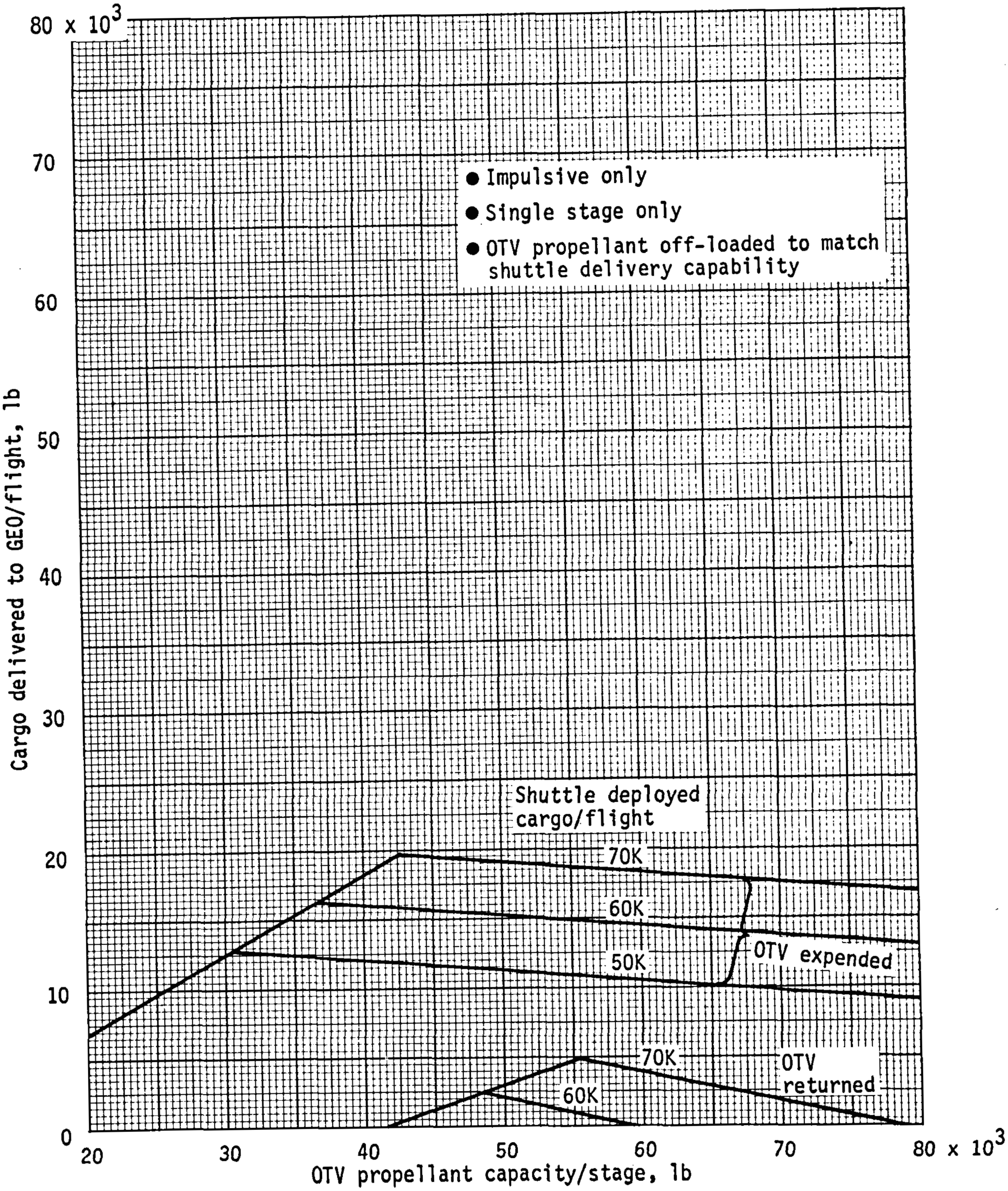
F. Space basing decouples OTV performance from Shuttle lift capability per flight. Shuttle performance can then be optimized in terms of lb/dollar rather than lb/flight.

G. Space basing would allow effective use of dual staging, aerobraking, and OTV scaling. It would make possible development of extensive payload retrieval and/or refurbishment capabilities with the resultant substantial potential cost savings.

<u>However</u>

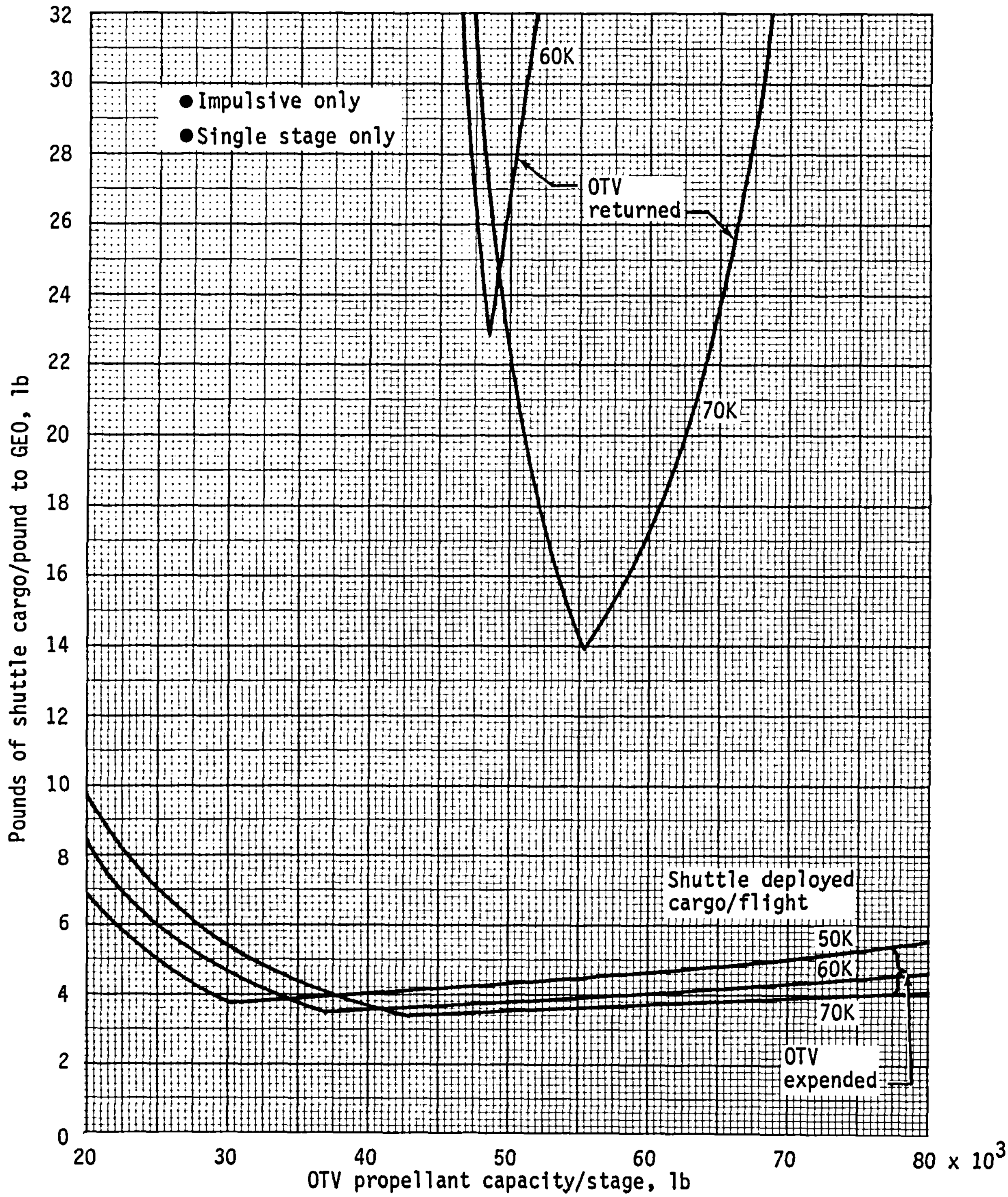
To achieve these benefits, the OTV must be sized and designed for space based operations. The operations and operational techniques must not be constrained to those of a Shuttle based operation.

H. Optimum OTV sizing will be very dependent on the inert weight scaling law of the OTV design. The sizing study should be redone during the design phase with exact inert weight relationsips. For the inert weights used in this study an OTV propellant capacity of at least 60 000 to 70 000 lb seems to be indicated.



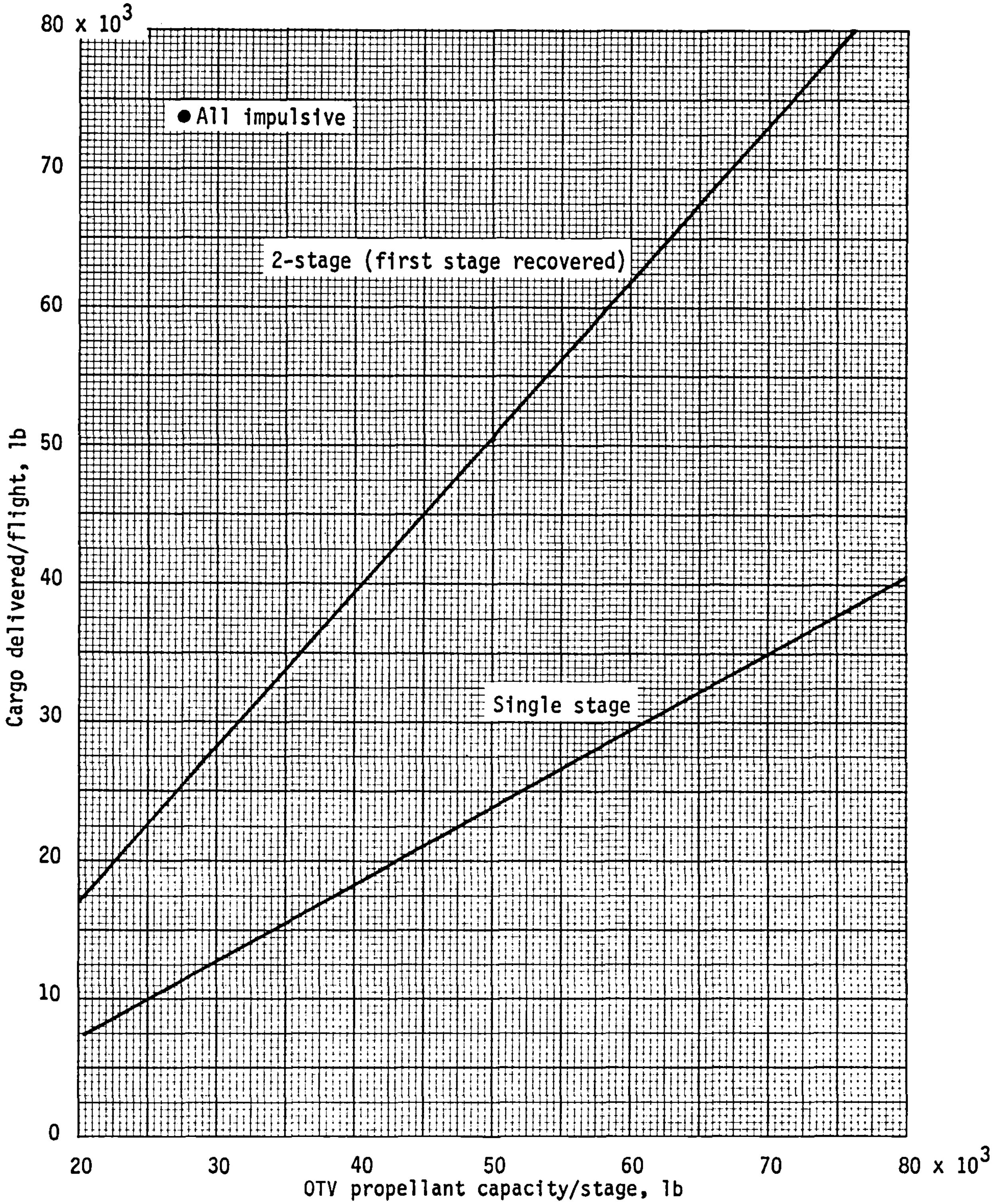
(a) Delivery capability.

Figure 3-1.- Shuttle based OTV performance to GEO.



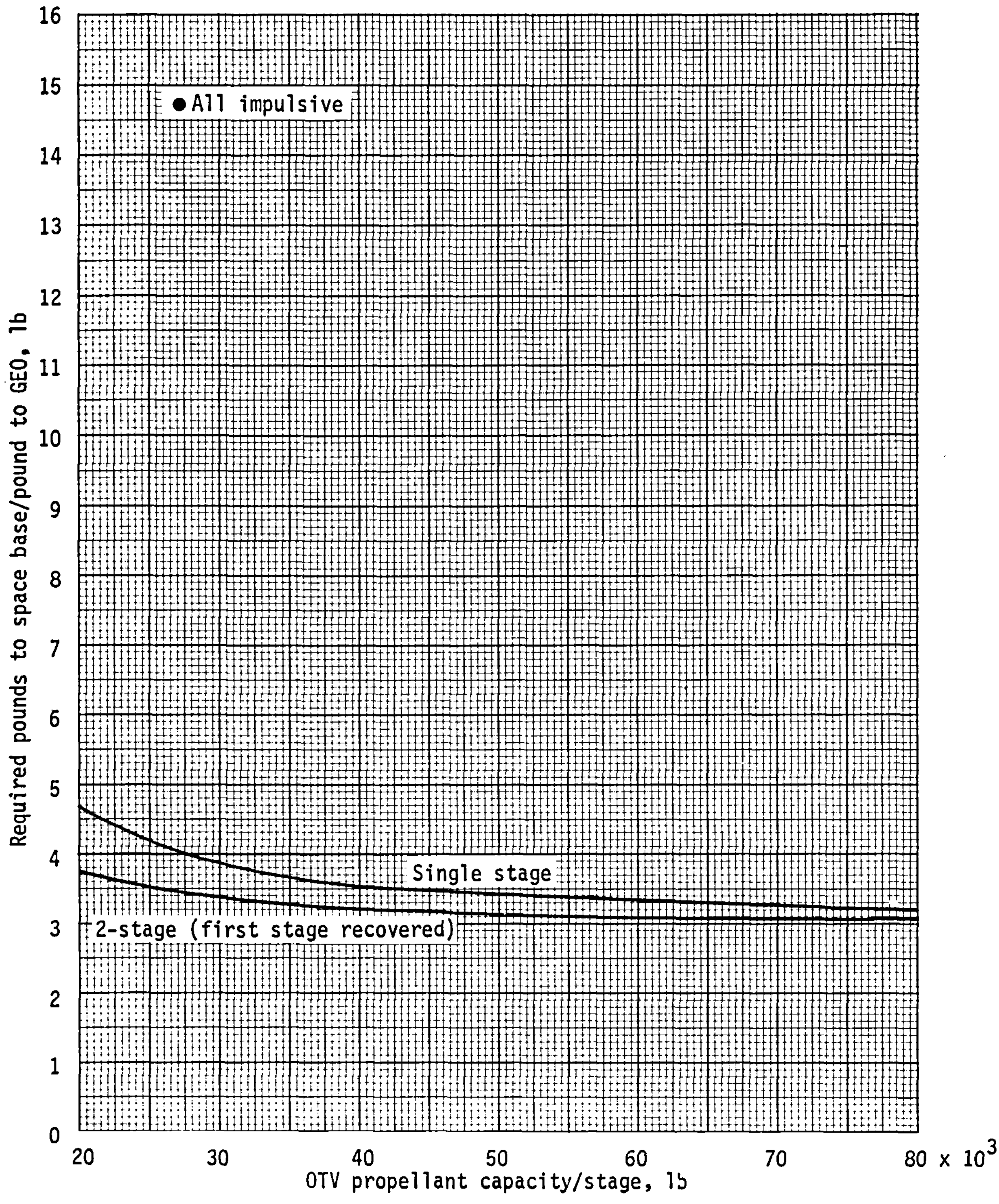
(b) Delivery effectiveness.

Figure 3-1.- Concluded.



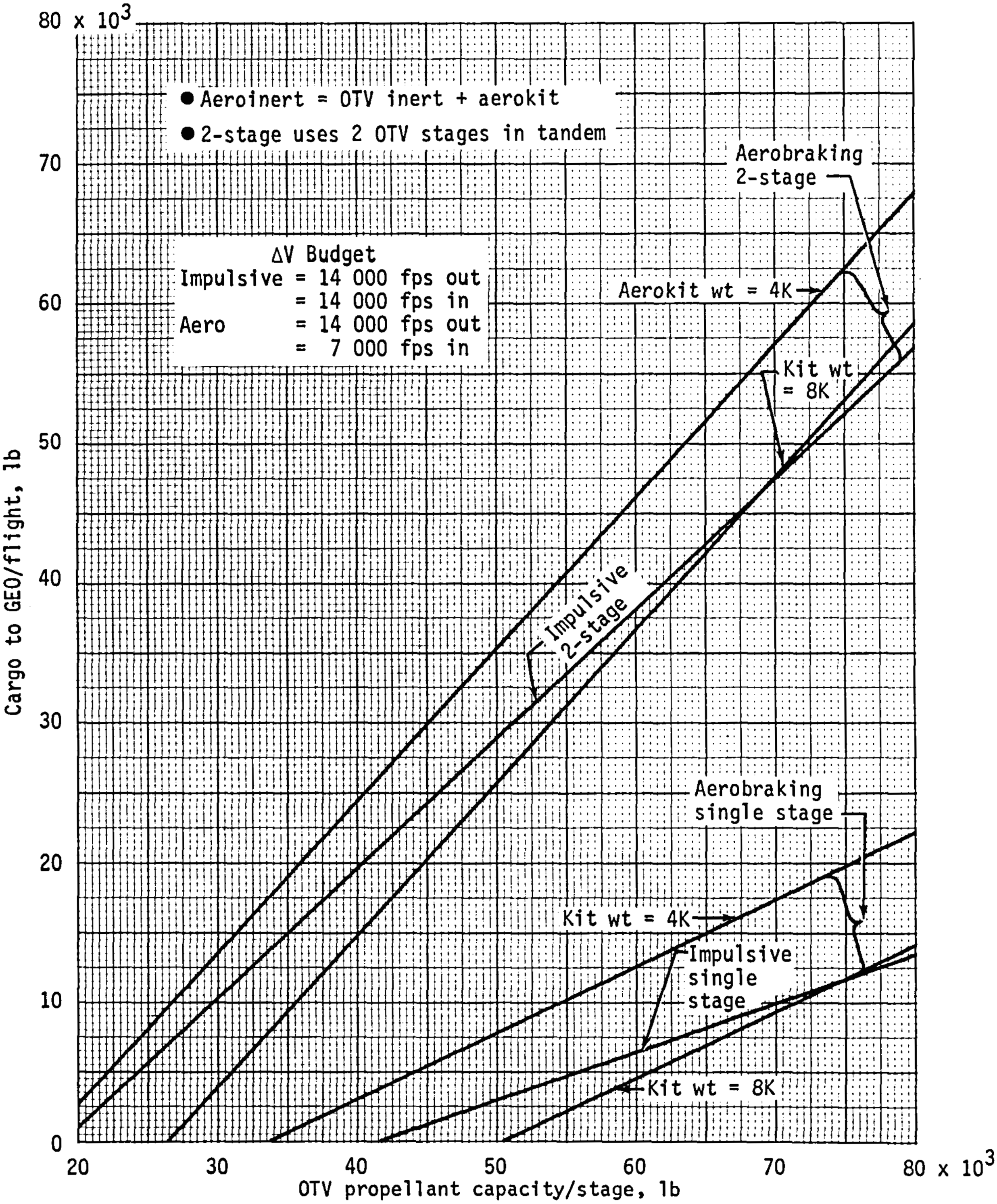
(a) Delivery capability.

Figure 3-2.- Space based OTV one-way cargo to GEO (top stage expended).



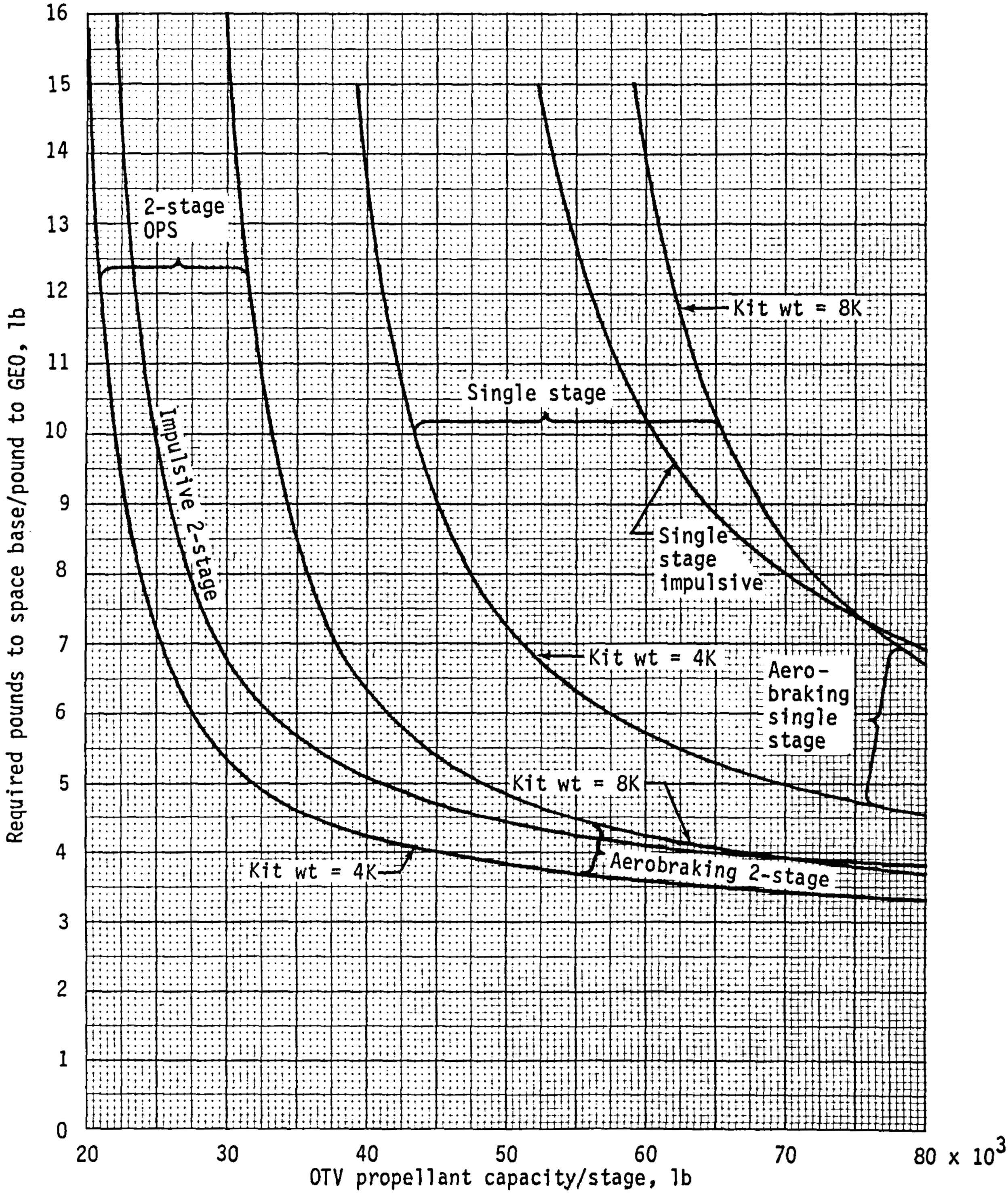
(b) Delivery effectiveness.

Figure 3-2.- Concluded.



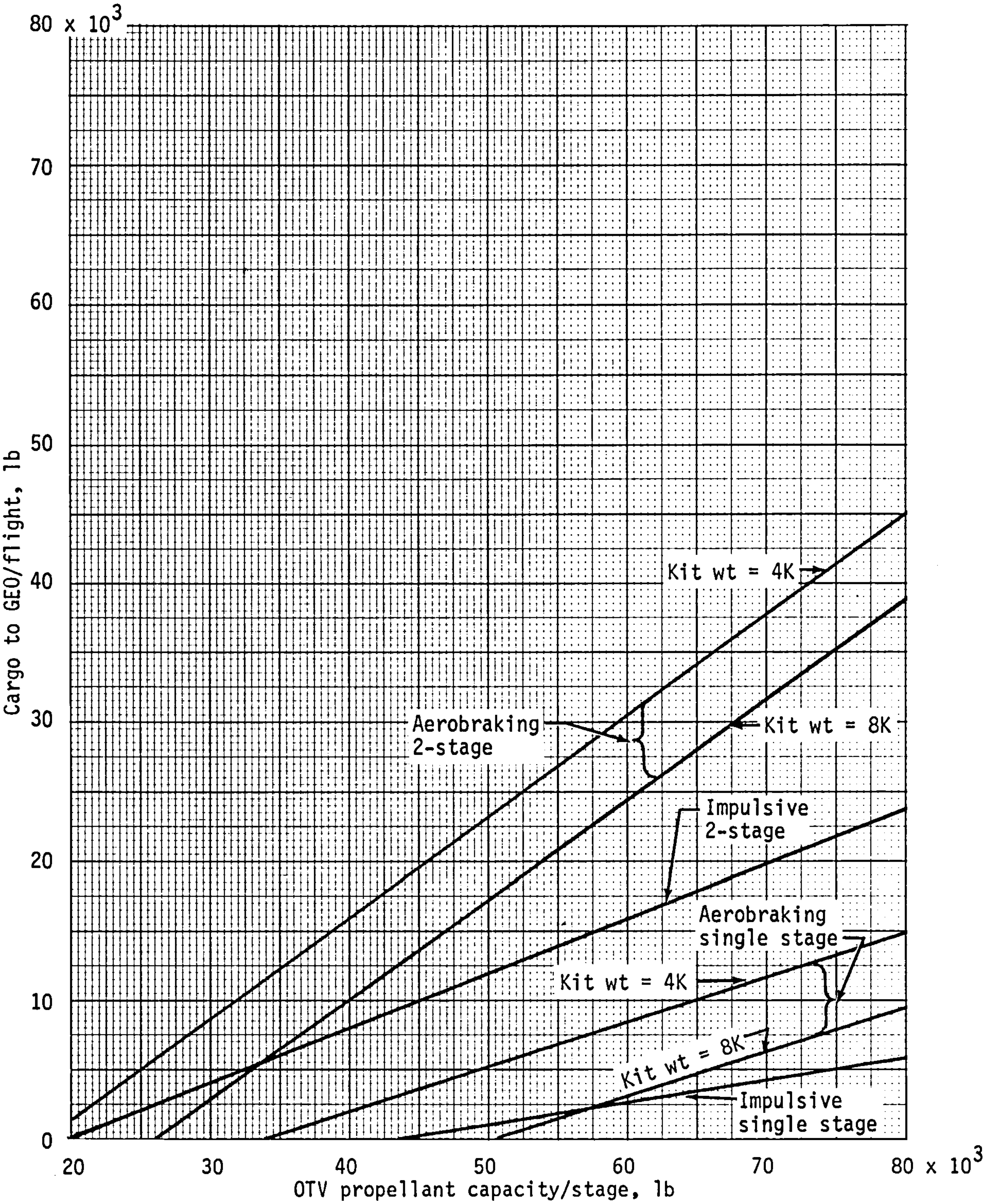
(a) Delivery capability.

Figure 3-3.- Space based OTV performance to GEO (stage(s) returned empty).



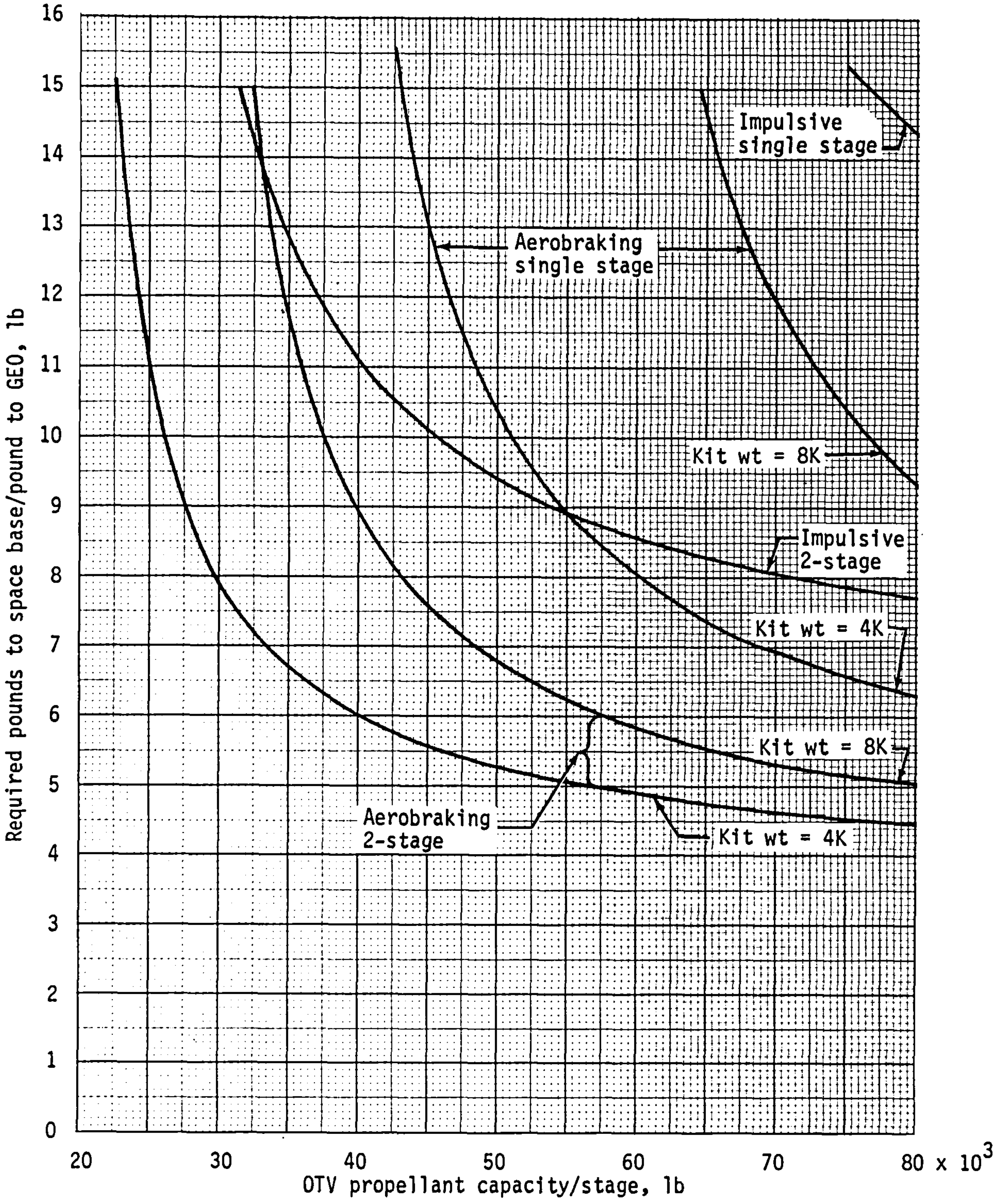
(b) Delivery effectiveness.

Figure 3-3.- Concluded.



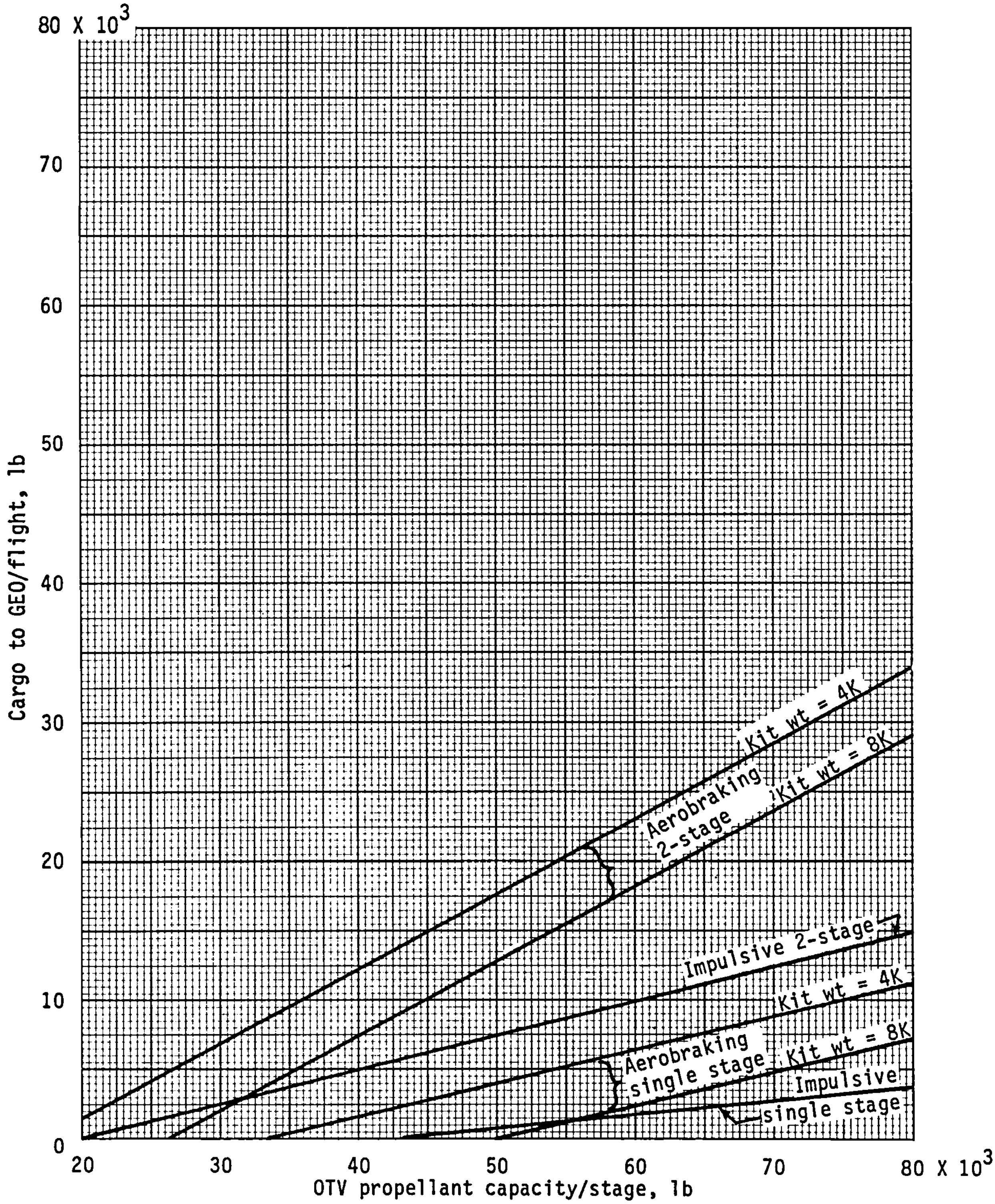
(a) Delivery capability.

Figure 3-4.- Space based OTV performance to GEO (returned cargo = 1/2 outbound cargo).



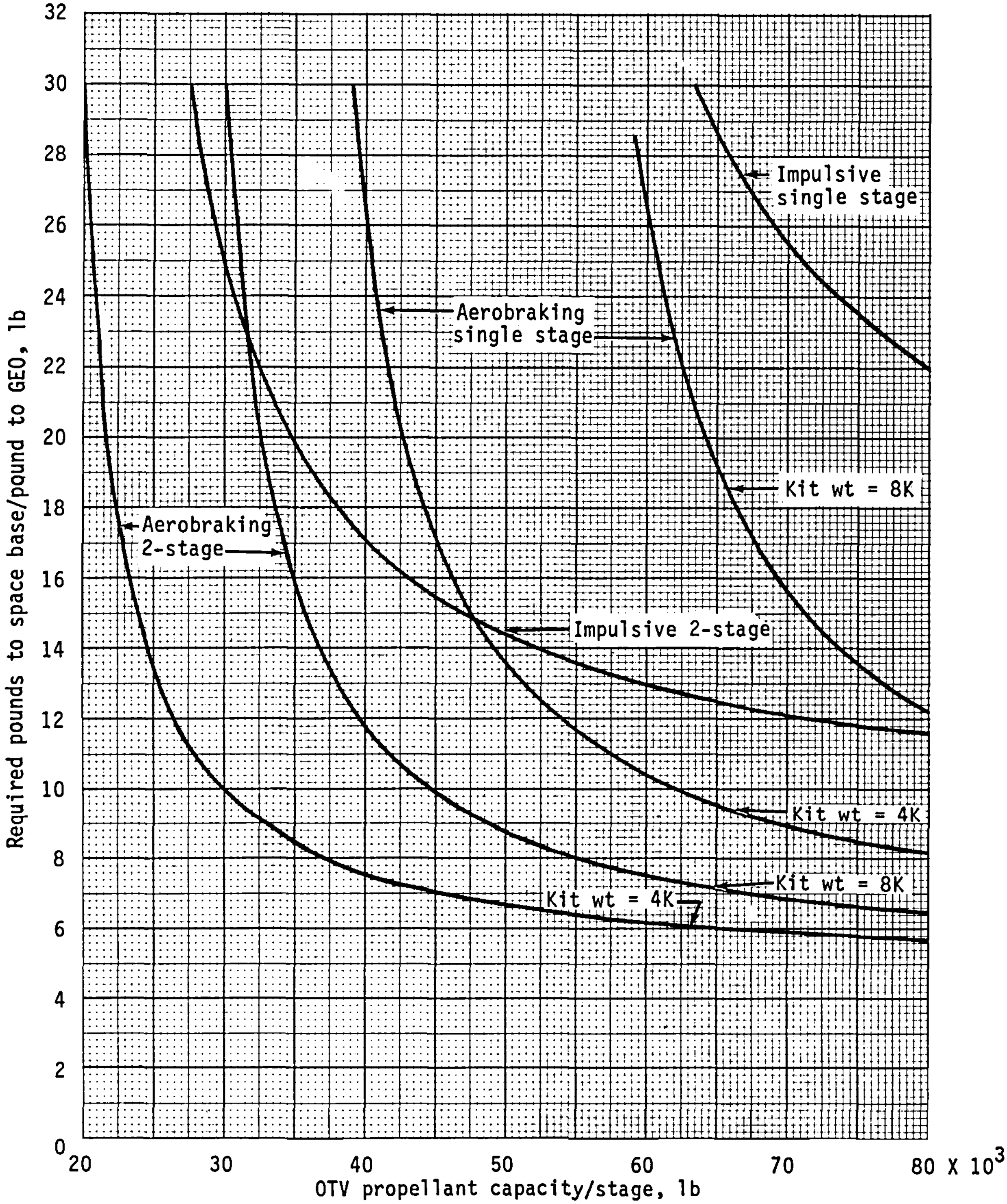
(b) Delivery effectiveness.

Figure 3-4.- Concluded.



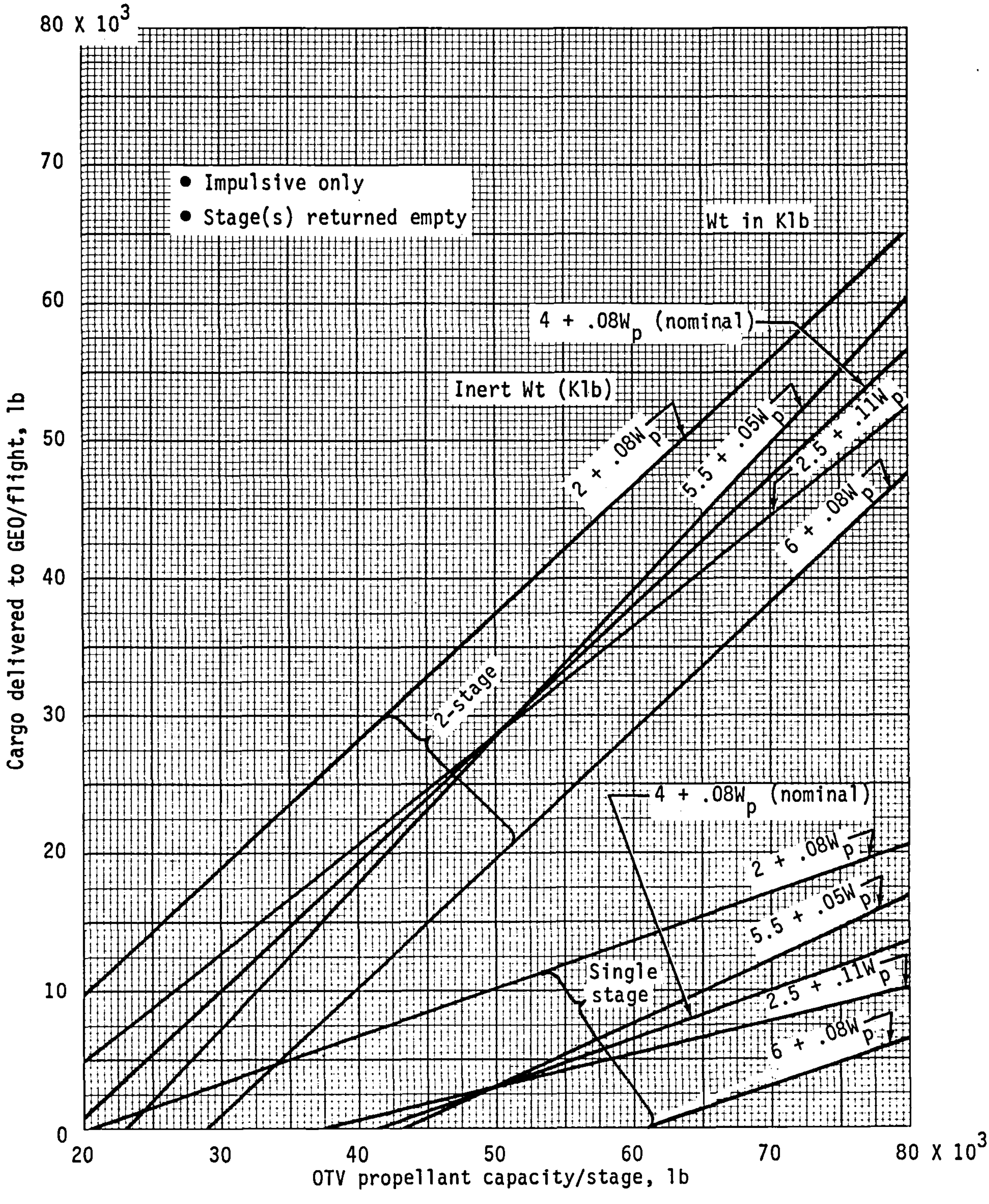
(a) Delivery capability.

Figure 3-5.- Space based OTV performance to GEO (cargo returned = cargo delivered).



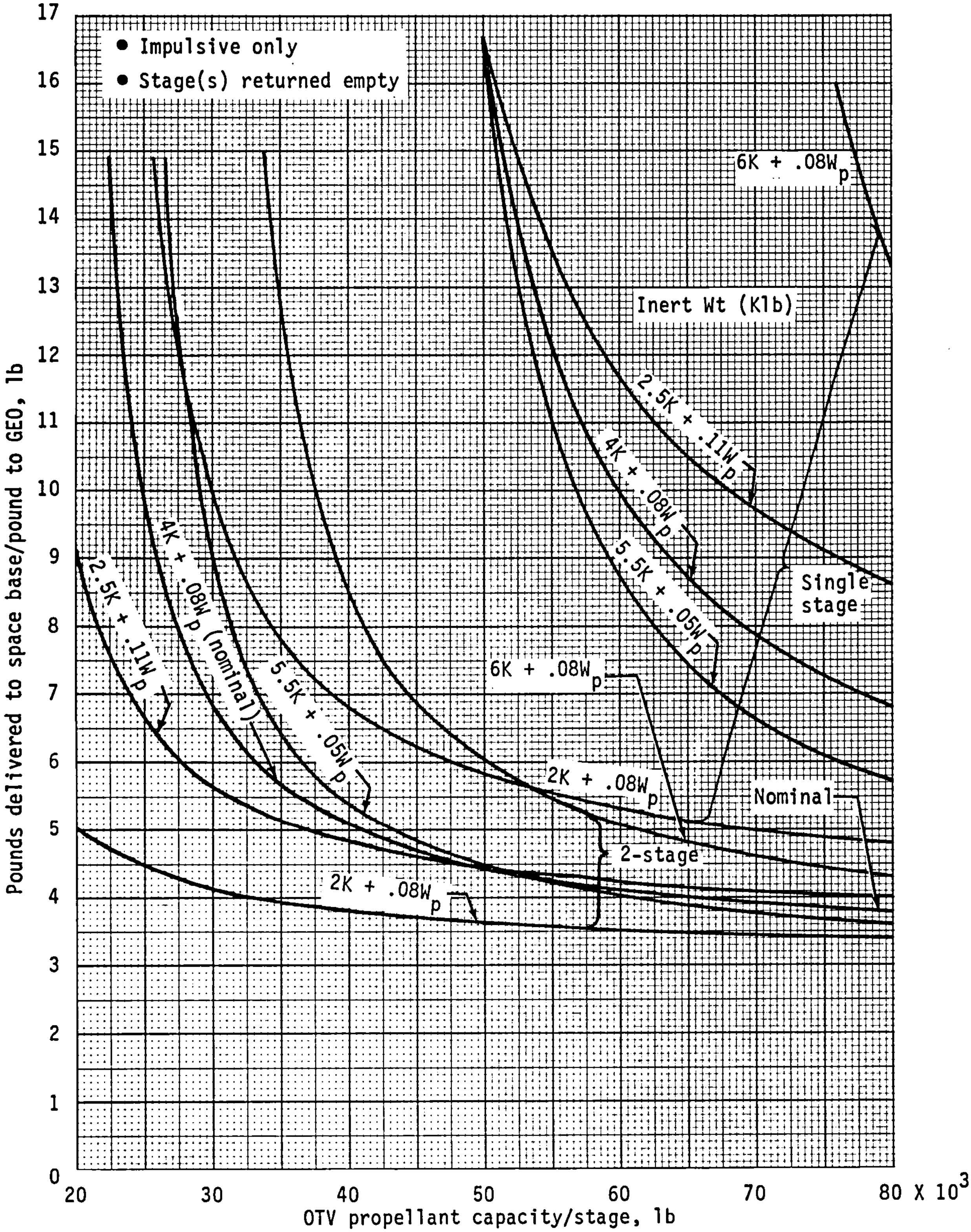
(b) Delivery effectiveness.

Figure 3-5.- Concluded.



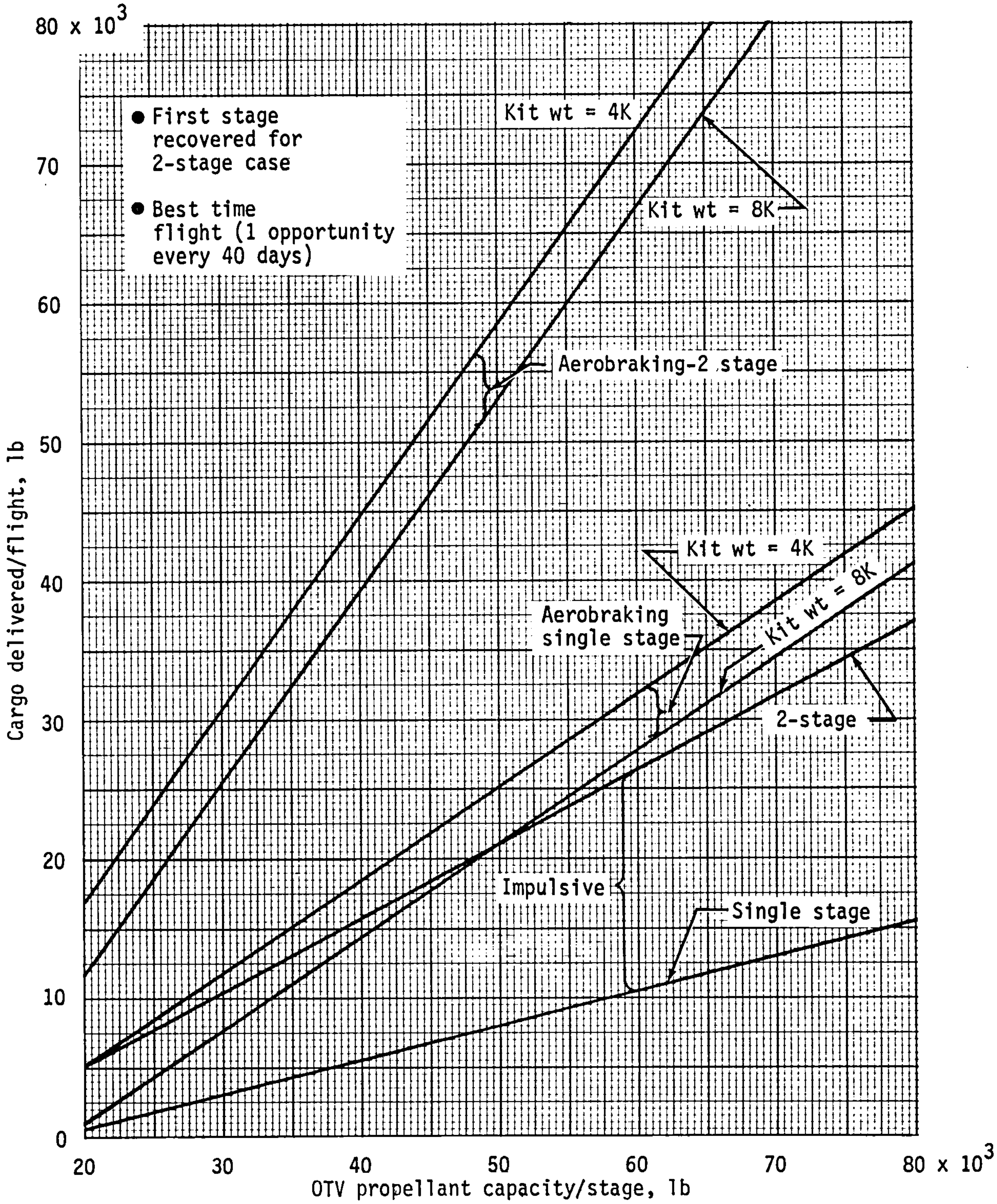
(a) Delivery capability.

Figure 3-6.- Space based OTV performance to GEO for various OTV inert weight scaling laws.



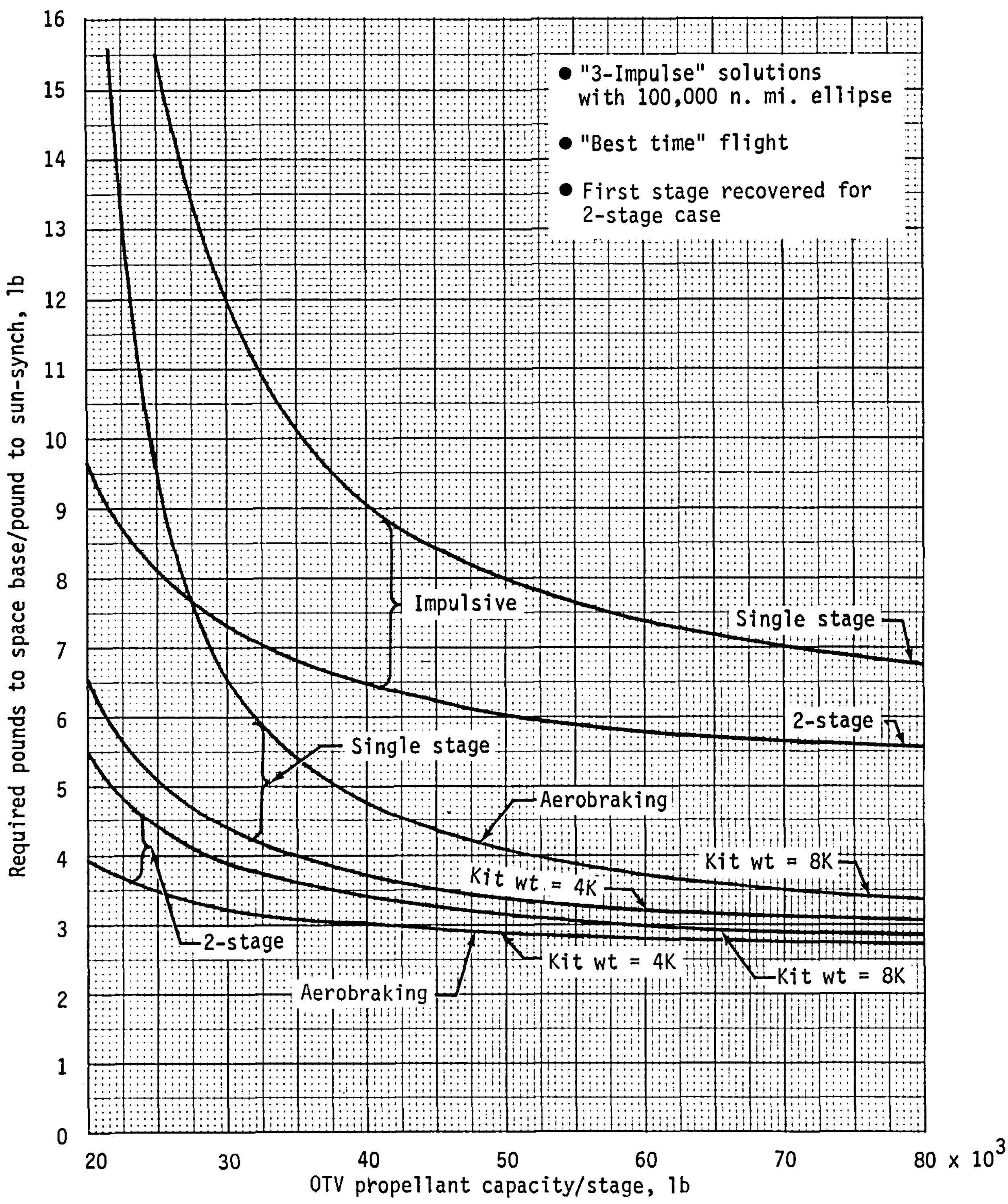
(b) Delivery effectiveness.

Figure 3-6.- Concluded.



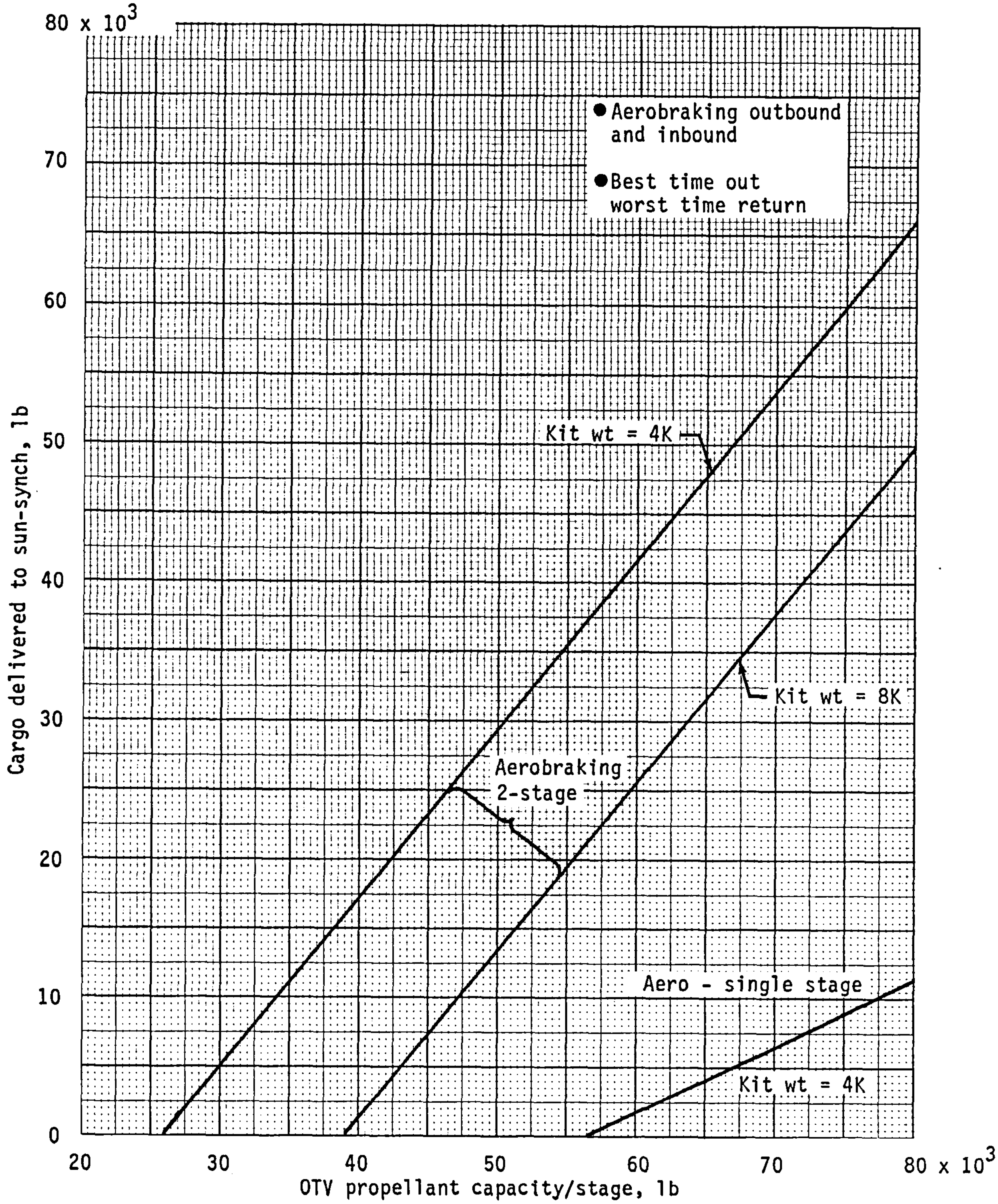
(a) Delivery capability.

Figure 3-7.- Space based OTV one-way cargo to Sun-synchronous orbit (top stage expended).



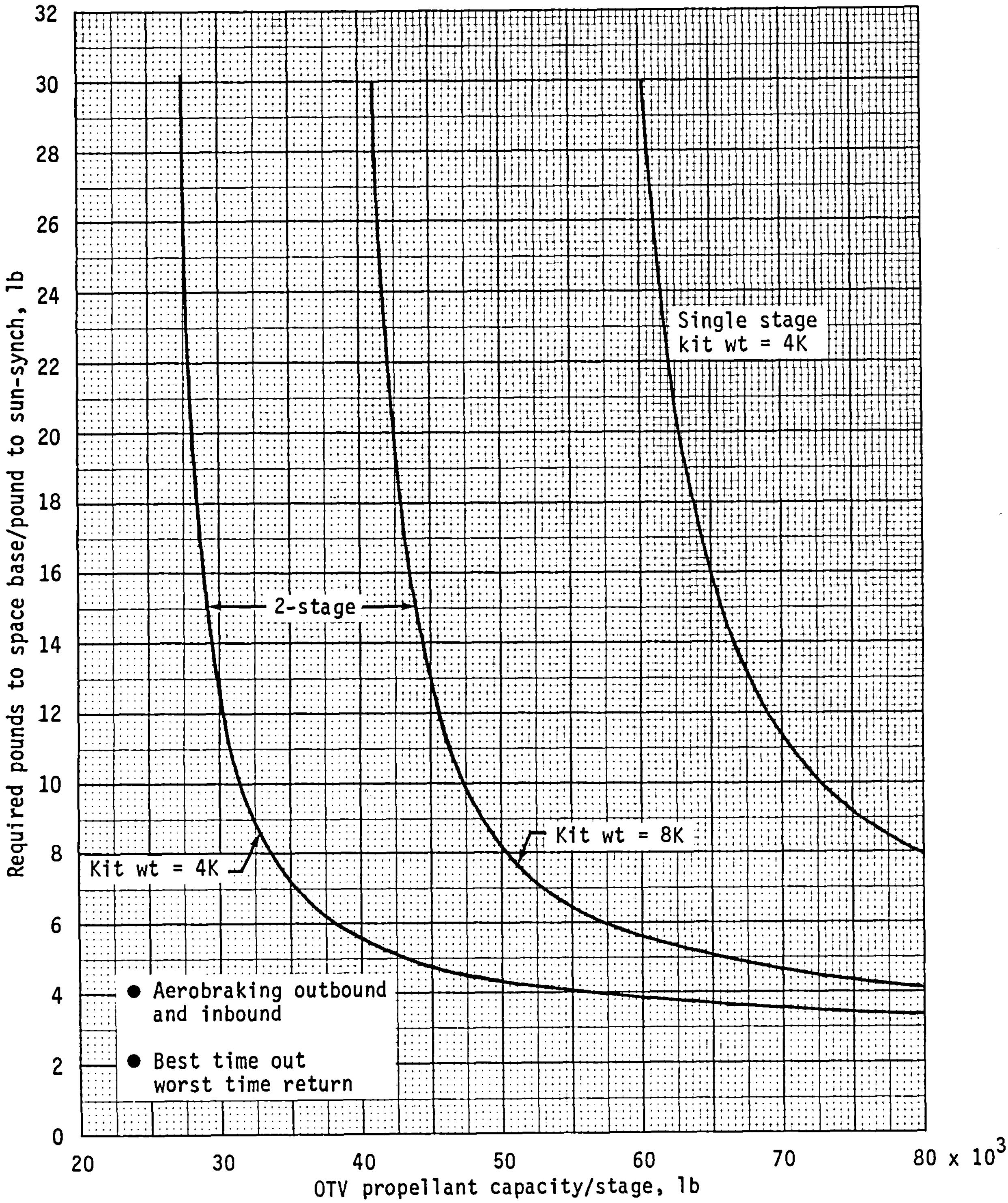
(b) Delivery effectiveness.

Figure 3-7.- Concluded.



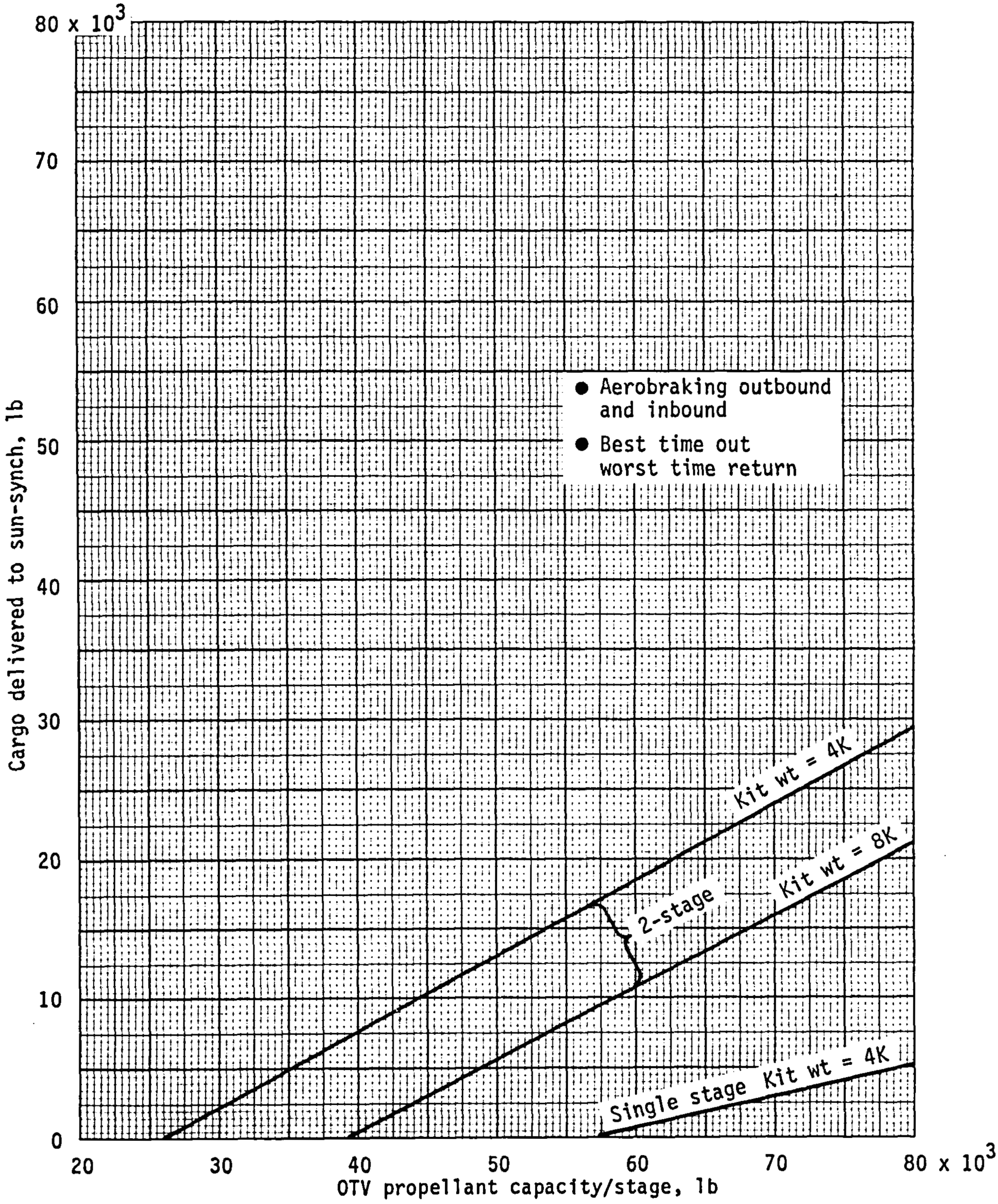
(a) Delivery capability.

Figure 3-8.- Space based OTV performance to Sun-synchronous orbit (stage(s) returned empty).



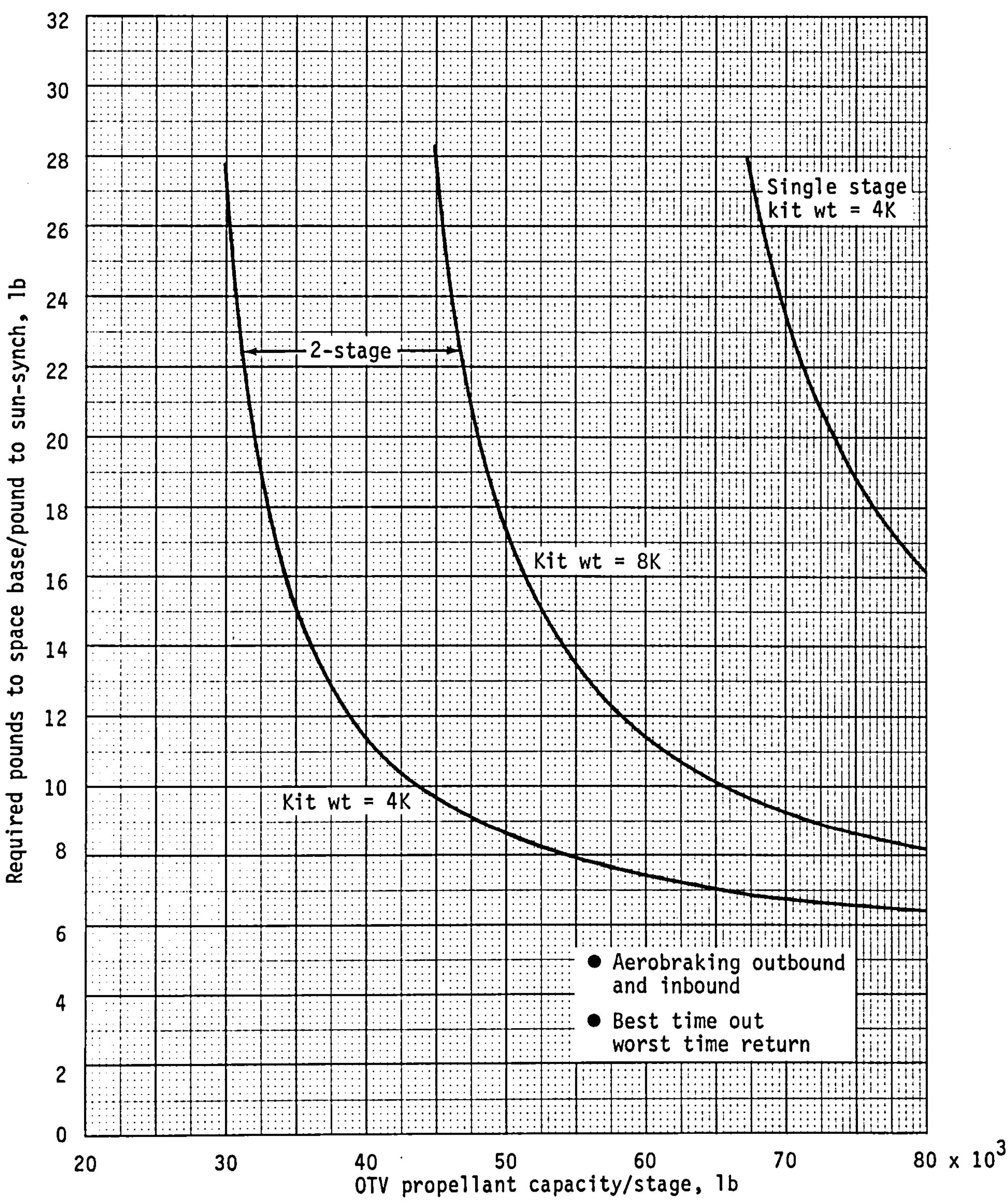
(b) Delivery effectiveness.

Figure 3-8.- Concluded.



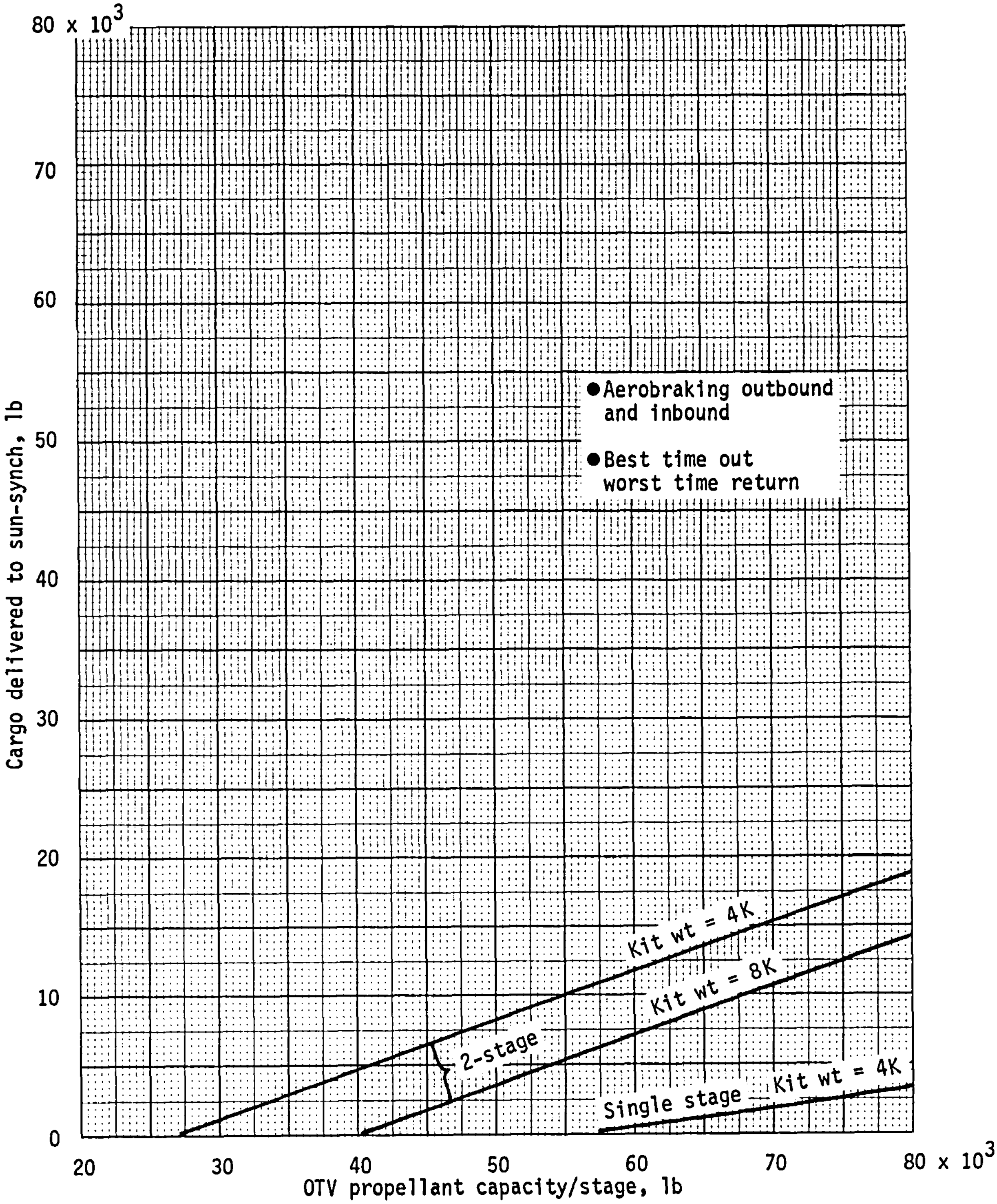
(a) Delivery capability.

Figure 3-9.- Space based OTV performance to Sun-synchronous orbit (returned cargo = 1/2 delivered cargo).



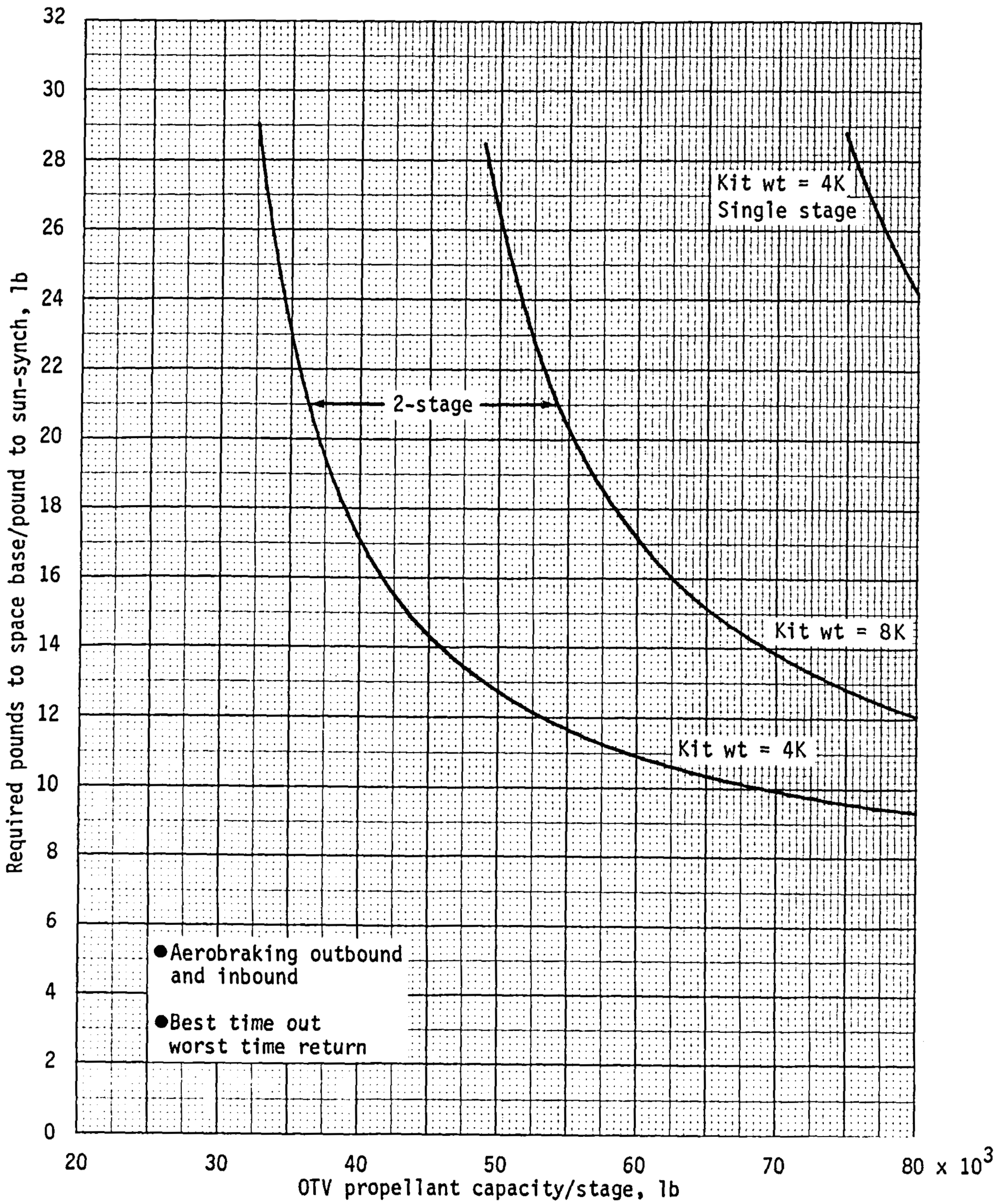
(b) Delivery effectiveness.

Figure 3-9.- Concluded.



(a) Delivery capability.

Figure 3-10.- Space based OTV performance to Sun-synch (returned cargo = delivered cargo).



(b) Delivery effectiveness.

Figure 3-10.- Concluded.

4.0 CONCLUDING REMARKS

The delivery and retrieval of payloads utilizing a Space Station have different constraints and opportunities than those utilizing ground-based transportation. These space-based operations would need to employ a high energy orbital transfer vehicle (OTV) to perform all necessary propulsive maneuvers in the sequence of orbit transfers from and return to the Space Station. In addition, detached operations in the vicinity of the Space Station would also require a space-based vehicle to transport this class of payload to and from the space base.

A Space Station would permit assembly-on-orbit of payloads and upper stages, thus relieving many configuration size and performance constraints. The Space Station as a depot would permit Shuttle transportation of some propellant to orbit at marginal costs. These considerations will favor some space-based options. The suitability of a Space Station as a transportation node for a given mission application will depend upon mission-window requirements, performance requirements, and vehicle capability and constraints. This performance requirements analysis is intended to provide the data base from which potential Space Station users and upper stage developers are able to assess the suitability and impact of a Space Station on their missions.

1. Report No. NASA RP-1110	2. Government Accession No.	3. Recipient's Catalog No.
4. Title and Subtitle PERFORMANCE REQUIREMENTS ANALYSIS FOR PAYLOAD DELIVERY FROM A SPACE STATION		5. Report Date July 1983 6. Performing Organization Code
7. Author(s) A. L. Friedlander, J. K. Soldner, J. Bell, G. W. Ricks, R. E. Kincade, D. DeAtkine, R. Reynolds, B. A. Nader, O. Hill, G. R. Babb, and E. C. Lineberry		8. Performing Organization Report No. S-523
9. Performing Organization Name and Address NASA Lyndon B. Johnson Space Center Houston, Texas 77058		10. Work Unit No. 925-85-00-00-72 11. Contract or Grant No.
12. Sponsoring Agency Name and Address National Aeronautics and Space Administration Washington, D. C. 20546		13. Type of Report and Period Covered Reference Publication 14. Sponsoring Agency Code

15. Supplementary Notes

A. L. Friedlander and J. K. Soldner, Science Applications, Inc., authors of Section 1. Other Sections by JSC authors

Publication edited by J. Bell, JSC

16. Abstract

Operations conducted from a Space Station in low Earth orbit have different constraints and opportunities than those conducted from direct Earth launch. While a Space Station relieves many size and performance constraints on the Space Shuttle, the Space Station's inertial orbit has different launch window constraints from those associated with customary Earth launches which reflect upon upper stage capability. A performance requirements analysis has been developed to provide a reference source of parametric data, as well as specific case solutions and upper stage sizing trade to assist potential Space Station users and Space Station and upper stage developers assess the impacts of a Space Station on missions of interest.

17. Key Words (Suggested by Author(s))	18. Distribution Statement
Performance, requirements, independent variables, orbital assembly, space bases, orbital space stations, payload delivery, orbit transfer vehicles, satellites	Unclassified - Unlimited Subject Category 15

19. Security Classif. (of this report)	20. Security Classif. (of this page)	21. No. of Pages	22. Price*
Unclassified	Unclassified	229	A11

*For sale by the National Technical Information Service, Springfield, Virginia 22161

JSC Form 1424 (Rev Nov 75) ☆U.S. GOVERNMENT PRINTING OFFICE:1983 639-008/24 NASA — JSC

National Aeronautics and
Space Administration

Washington, D.C.
20546

Official Business
Penalty for Private Use, $300

SPECIAL FOURTH CLASS MAIL
BOOK

Postage and Fees Paid
National Aeronautics and
Space Administration
NASA-451

U.S.MAIL

NASA

POSTMASTER: If Undeliverable (Section 158 Postal Manual) Do Not Return