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STUDY OF EXTRATERRESTRIAL DISPOSAL OF RADIOACTIVE WASTES

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STUDY OF EXTRATERRESTRIAL DISPOSAL OF RADIOACTIVE WASTES

Part I

Space Transportation and Destination Considerations for Extraterrestrial Disposal of Radioactive Wastes

by

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SUMMARY

NASA has been requested by the AEC to conduct a feasibility study of extraterrestrial (space) disposal of radioactive waste. This report covers the initial work done on only one part of the NASA study, the evaluation and comparison of possible space destinations and space transportation systems. Only current or planned space transportation systems have been considered thus far. The currently planned Space Shuttle was found to be more cost-effective than current expendable launch vehicles by about a factor of 2. The Space Shuttle requires a third stage to perform the waste disposal missions. Depending on the particular mission, this third stage could be either a reusable space tug or an expendable stage such as a Centaur.

Of the destinations considered, high Earth orbits (between geostationary and lunar orbit altitudes), solar orbits (such as a 0.90 AU circular solar orbit) or a direct injection to solar system escape appear to be the best candidates. Both Earth orbits and solar orbits have uncertainties regarding orbit stability and waste package integrity for times of the order of a million years, which may be required for some nuclear waste isotopes to decay to a safe level. These problems can be avoided by injecting the waste package to solar system escape or impacting it into the Sun. The solar system escape mission requires a high Earth departure velocity, but the mission can be accomplished using two or three space tugs in tandem, each launched to Earth orbit by the Space Shuttle. However, the resulting space transportation system cost is about four times higher than for the high Earth orbit or solar orbit destinations. A direct solar impact mission requires a very high Earth departure velocity and cannot be accomplished with the current or planned launch systems considered in this study. As an alternate, the solar impact mission can be accomplished using a Jupiter swing-by trajectory which reduces the Earth departure velocity to values comparable to the solar system escape mission. However, the launch opportunity is limited to perhaps 40 days once every 13 months. The limited launch opportunity would make it difficult to achieve the high launch rates anticipated for disposing of significant amounts of radioactive waste.

Since the waste disposal problem will extend far into the future, new space technology and new development of advanced space transportation systems applicable to the waste disposal mission can be expected. This could provide a capability (performance and/or cost) superior to that considered in this report.

INTRODUCTION

The Atomic Energy Commission Division of Waste Management and Transportation has initiated a study to assess the feasibility of various long-term storage or disposal options for radioactive waste. Under this study several concepts are being investigated. NASA has been requested by the AEC to conduct a feasibility study of one of the concepts: extraterrestrial (space) disposal of radioactive wastes. The NASA study is to be used by Battelle Northwest Laboratories who have the responsibility of preparing a comprehensive report summarizing the feasibility, development requirements, and possible schedule and cost of development for each of the alternates.

This report is part of a series of NASA reports on the study of disposal of nuclear waste in space. It covers the initial work done on the evaluation and comparison of the various space destinations and launch vehicles considered. Reference 1 is a summary report of this study. Other parts of the NASA study (description of nuclear waste, design of containment vessels, shielding considerations, etc.) are reported in subsequent reports in this series (references 2 and 3).

The space destinations considered in this study include Earth orbits, solar orbits, solar system escape, and solar impact. The mission requirements for each destination are presented, and the relative advantages and disadvantages for each destination are discussed. In this report the destinations are referred to as disposal missions although, strictly speaking, some of the destinations could permit future retrieval of the nuclear waste, especially the Earth orbit destination.

The launch systems considered in this study include the current larger expendable launch vehicles as well as the reusable Space Shuttle (with a third stage such as the space tug) which is planned to be operational in 1980. They are shown in figure 1. Because the radioactive waste disposal problem will extend far into the future, new space transportation technology can be expected. Use of this new technology could result in more effective, lower cost transportation systems than those considered in this study. Similarly, the high launch rates anticipated for waste disposal (eventually one launch a week or more) could justify the development of a special launch vehicle dedicated to the disposal mission. However, this initial study is limited to current and planned capability where the basic development costs for the launch vehicle will have already been borne by other programs.

The two most important factors in assessing the feasibility of space disposal of radioactive wastes are safety and cost. In this report, safety has been considered only qualitatively in the comparisons of destinations, launch vehicles and their associated trajectories. The costs presented include the launch vehicles and their operations. These data can be used for comparative purposes for preliminary determination of the best launch vehicles and the most promising mission destinations. However, total cost of space disposal will have to include other elements such as the cost of separating and concentrating the waste material, transporting the nuclear waste and handling it at the launch site, and the cost of the flight containment system and its associated flight systems. These costs are not considered in this report.

POTENTIAL SPACE DESTINATIONS AND MISSION REQUIREMENTS

The space destinations considered in this study are discussed in the order of increasing mission energy requirement. They are listed in table 1. All launches are assumed to occur from the Eastern Test Range (ETR), Kennedy Space Center, in an easterly direction. For comparison purposes, it is assumed that the launch vehicle first enters a low circular Earth parking orbit, although this is not always necessary or advantageous. After parking in this orbit, the launch vehicle's upper stage or stages inject the waste package towards its final destination. A typical launch trajectory to a high Earth orbit destination is shown in figure 2. In general, for launch vehicles, best mission performance is achieved by using low parking orbit altitudes. For the mission and vehicle comparisons in this report, a parking orbit altitude of 185 kilometers (100 n.mi.), which is typical of current practice, is used. However, for the waste disposal mission a higher parking orbit altitude may be preferred from a safety standpoint, as is discussed later.

Mission energy is characterized by the mission ΔV requirement, which is defined as the sum of all the velocity increments that the launch vehicle has to provide after reaching low Earth orbit. In some cases the launch vehicle by itself can place, or inject, the waste package towards its final destination. In other cases the waste package, after separation from the launch vehicle, requires subsequent trajectory (midcourse) corrections or propulsion in order to reach its destination. In these cases the waste package becomes an active spacecraft requiring the addition of guidance, control, communications, and propulsion systems. These requirements are pointed out where needed.

High Earth Orbits

To achieve high Earth final orbits starting from a low circular parking orbit, two propulsion maneuvers are required (see figure 2). The first maneuver is made in the parking orbit and places the payload on an elliptical transfer orbit. After the payload coasts along the transfer orbit to the desired final altitude, the second maneuver is made to circularize the final orbit. It would be expected that both of these maneuvers would be performed by the launch vehicle's upper stage and that the waste package itself would require no additional guidance or propulsion capability. The orbital maneuvers from the initial parking orbit can be arranged so that in the event of a propulsion failure, the resulting orbit would have a lifetime of at least several months.

This would allow time for making a second launch which would rendezvous with the waste package so that corrective action could be taken.

Figure 3 depicts the total ΔV required to achieve high circular final orbits starting from several typical low circular parking orbits. The ΔV requirements shown in figure 3 result from having to provide two separate ΔV's, the first in the initial parking orbit and the second to circularize the final orbit. This velocity split is shown in figure 4 for a representative initial parking orbit altitude of 370 kilometers (200 n.mi.). The coast time to transfer from the initial parking orbit to the final orbit is shown in figure 5. More complete orbital characteristics are contained in reference 4.

It is not yet clear which orbit altitudes would be acceptable for the disposal of nuclear waste. Orbit lifetime is a primary factor. Orbit lifetimes of a million years or longer may be required if extremely long-half-lived wastes are to be disposed of into space. At reasonably high orbit altitudes, above several thousand kilometers, atmospheric drag is negligible; but other perturbations such as solar pressure and solar, lunar, and planetary gravitational perturbations must be considered. Orbits near the Moon should be avoided to minimize lunar perturbations. Furthermore, orbits beyond the Moon are subject to large solar perturbations. High-traffic regions or orbits important from a science or applications point of view (such as synchronous orbit altitude and some lower altitudes) should not be chosen. Therefore, probably the best choice for high Earth orbits would be those orbits lying between synchronous orbit altitude and the Moon. However, such orbits have the highest ΔV requirement of the high Earth orbits, as can be seen from figure 3.

If the payloads are launched due-East from ETR, their orbits will have an inclination of about 0.488 radians (28.5 deg.) to the Earth's equator. Gravitational perturbations will precess the orbits. Due to inherent limitations on placement accuracy, there will be slight differences in the orbits which will cause them to precess at slightly different rates. Eventually, the orbits of the waste packages will be randomly located in a belt around the Earth. Future lunar and planetary spacecraft would regularly penetrate this belt. However, because of the wide spacing between waste packages at such high placement altitudes, the probability of a collision would be extremely remote and could probably be ignored.

A principal advantage of the high Earth orbit destination is the relatively low ΔV required in comparison with some of the other destinations. In addition, launches can take place on any day. Another advantage is that the waste package could be retrieved at a later date either to recover the waste material or to remedy some unforeseen problem.

One of the more serious problems is that long time stability of the orbit elements (eccentricity, semi-major axis, etc.) and hence orbit lifetime, cannot be guaranteed. Intuitively, it might be expected that these high orbits will have satisfactory lifetimes. However, the complexity of the multi-perturbation problem precludes rigorously verifying the stability of these orbits over many thousands of years. In addition, there might be a problem with the package integrity. There is no assurance that the integrity of the relatively hot waste package can be maintained when it is exposed to the space environment over these long periods of time. Since neither orbit stability nor waste package integrity problems are well understood (for times of the order of a million years), high Earth orbits cannot be considered a permanent disposal site. Unless further studies can resolve these problems, Earth orbits should be considered only as a temporary (hundreds or a few thousands of years) storage site requiring further action at a later date.

Solar Orbits

If Earth orbit destinations for radioactive wastes are unacceptable, solar orbits are the next alternative from a ΔV standpoint. The solar orbits considered in this study are those achievable with relatively low ΔV's including (1) solar orbits achievable by injecting the payload to Earth-escape energy or slightly beyond, (2) circular solar orbits slightly inside or outside Earth's orbit around the Sun achievable by additional propulsion after Earth escape, and (3) solar orbits achievable by swinging by Mars or Venus.

Earth escape - The simplest method for achieving a solar orbit is to have the launch system inject the waste package to Earth-escape energy. This can be done with a single propulsive burn from Earth orbit providing a ΔV of approximately 3.23 kilometers per second (10,600 ft/sec). (This is actually somewhat less than the ΔV required for high Earth orbits as shown in figure 3.) The waste package would then be separated from the launch system and after escaping the Earth's gravitational field would be in an orbit around the Sun. The waste package would be in essentially the Earth's orbit around the Sun but in a different angular position.

The advantages of this approach, in addition to having a relatively low ΔV in comparison with some of the other destinations are that (1) the waste package (as in the Earth orbit case) could be passive, requiring no active spacecraft systems and (2) launches can be made every day during the year. A disadvantage is that there is a high probability of the waste package reencountering the Earth at some future time. Because of inherent limitations on injection accuracy and long-term gravitational perturbation effects-principally from the Earth, the waste package cannot be maintained at a fixed position from the Earth. As a result of these effects the waste package would tend to drift with respect to the Earth, and preliminary calculations indicate a high probability of reencountering the Earth within a few thousand years or less.

A better approach (to lessen this probability) would be to provide somewhat more ΔV than required for Earth escape, so that the waste package would be in a slightly elliptic solar orbit with a small inclination to the ecliptic plane (plane of the Earth's orbit around the Sun). As an example of such an orbit, a ΔV of 0.3 kilometers per second (1,000 ft/sec) is added to the Earth-escape ΔV.

This could result in the waste package being in a solar orbit having a 0.9 AU (astronomical unit) perihelion and a 1.0 AU aphelion with an inclination of about 0.07 radians (4 deg.) to the ecliptic plane. Initially, the orbit of the waste package would intersect the Earth's orbit at only one point (the Earth departure point). With time, gravitational effects could tend to precess the orbit of the waste package with respect to the Earth's orbit, making an Earth reencounter even less likely. Preliminary calculations indicate that such is the case for a few thousand years, and it is recommended that this approach should receive more study. However, there is no assurance that trajectories can be developed (and demonstrated analytically) which eliminate the possibility of reencounter with the Earth for times of the order of a million years. Because of this uncertainty Earth escape solar orbit cannot be established as a proven, acceptable destination at this time.

Circular solar orbits - In order to provide a positive separation between the orbit of the waste package and the orbit of the Earth, the waste packages could be placed in nearly circular solar orbits either inside or outside the Earth's orbit around the Sun. The Earth is in an elliptic orbit around the Sun at a distance ranging from 0.983 AU at perihelion to 1.017 AU at aphelion. These circular orbits should be at least outside this range to minimize the probability of a subsequent reencounter with the Earth. There is an incentive, however, for going no further than necessary outside this range since the required ΔV increases with increasing distance from the Earth's orbit.

Starting from Earth orbit, two propulsive burns are required to reach the desired circular solar orbit. The first burn which is performed by the launch vehicle's upper stage provides the ΔV to inject the waste package to slightly past Earth-escape energy. After escaping from the Earth the waste package is in an elliptical solar transfer orbit. This elliptical orbit intersects the Earth's orbit at the point of Earth departure. After coasting several months in this orbit, the payload reaches the desired final orbit radius. At this point the second burn provides the ΔV required to circularize the final orbit. Because of the long coast time it is impractical to accomplish this burn with the launch vehicle's upper stage. A propulsion system along with guidance, control and communications systems will have to be added to the waste package.

Figure 6 shows the total ΔV required to achieve circular solar orbit radii ranging from 0.90 AU to 1.10 AU starting from several typical low Earth circular parking orbits. These ΔV's are generated with the assumptions that (1) the Earth's orbit is circular at a distance of 1.0 AU from the Sun and (2) the final orbit is in the ecliptic plane. For a 1.0 AU solar orbit, the ΔV required is equal to the Earth-escape ΔV. The total ΔV is the sum of two discrete ΔV's. Figure 7 shows the ΔV split along with the transfer time for the range of final orbit radii considered. The ΔV required in Earth orbit to achieve the desired transfer elliptical orbit is only slightly above Earth-escape ΔV.

The long-term stability of solar orbits and the possible disintegration of the waste package in a solar orbit can influence the choice of an interior or exterior orbit. As was the case for orbits around the Earth, the problem of demonstrating the stability of solar orbits for times of the order of a million years is unresolved. Presumably, the final orbit could be placed sufficiently far from the Earth's orbit to preclude a subsequent reencounter with the Earth over the times required. The magnitude of the required separation is not known. Another problem is the possible disintegration of the waste package over long periods of time. If the waste package should disintegrate, the Poynting-Robertson effect will tend to draw the smaller fragments into the Sun. If part of the waste package should vaporize, the solar wind could tend to move some of the material out from the Sun. If the integrity of the waste package cannot be guaranteed these and other effects will have to be evaluated, not only in making the selection of the orbit location, but also to establish the ultimate destination of the waste material.

For comparison purposes, a final solar orbit radius of 0.90 AU is used in this study. The mission profile for a 0.90 AU solar orbit is shown in figure 8. The injection ΔV required to depart from a 185 kilometer (100 n.mi.) Earth parking orbit (point 1) is 3.26 kilometers per second (10,690 ft/sec). After escaping from the Earth and coasting about 170 days in an elliptical solar orbit, the second ΔV of approximately 0.80 kilometers per second (2,660 ft/sec) is applied at the perihelion (point 2) to the circularize the orbit at 0.90 AU. The total ΔV required is 4.07 kilometers per second (13,350 ft/sec) which is a relatively low ΔV in comparison with some of the other destinations.

The necessity of requiring an additional propulsion stage to perform the second burn introduces two disadvantages to this destination. The cost of the waste package payload will increase and the propulsion and associated systems added to the waste package must perform reliably over a 6-month time period. These disadvantages might be diminished by performing the circularization maneuver with a relatively simple spin-stabilized, solid rocket motor.

The consequences of several propulsion failure situations should be noted for this destination. If the first burn out of Earth parking orbit should fail prior to reaching Earth-escape velocity, the payload would be left in an elliptic Earth orbit. The departure trajectory can be designed so that if this should happen there would be sufficient orbit decay time (months) to permit a second launch for taking corrective action. If the first burn should fail after reaching Earth-escape velocity, or if the circularization burn should fail, the waste package would be left in an unplanned elliptic solar orbit, intersecting the Earth's orbit near aphelion and would be unrecoverable with present launch systems. For these cases there is a possibility that the waste package will eventually reencounter the Earth. This is a disadvantage shared by all destinations beyond Earth. The reencounter probability can be reduced by using departure trajectories similar to those suggested earlier for the Earth-escape case.

In summary, if the integrity of the waste package can be determined and if the stability of circular solar orbits (near Earth) can be established, circular solar orbits can be considered as a possible disposal destination. However, further study is required to evaluate the consequences of possible failure situations.

Solar orbit via Venus and Mars swing-bys[1]. Another method for achieving solar orbits that do not cross the Earth's orbit is to swing by another planet, using the gravitational attraction of that planet to change the initial swing-by trajectory. Both Venus and Mars are considered as candidate swing-by planets. Typical flight profiles for Venus and Mars swing-bys are shown in figures 9 and 10, respectively. For a Venus swing-by mission, the payload would be injected onto a Venus swing-by trajectory at point 1 of figure 9. After the payload coasts for several months along this trajectory (a solar orbit), it will swing by Venus at point 2. With a properly oriented swing-by, the preencounter solar orbit is altered so that it will no longer cross the Earth's orbit. This is the principal advantage of the swing-by missions. However, the post swing-by orbit will, periodically, cross the orbit of Venus. The waste package could collide with Venus on a subsequent orbit, or its orbit could be significantly perturbed on a subsequent close encounter, although these probabilities are small. To preclude such an encounter with Venus, the post swing-by orbit can be altered by a propulsive maneuver upon reaching the perihelion at point 3 of figure 9. The Mars swing-by mission profile shown in figure 10 is similar.

Seven Venus swing-by trajectories, having a minimum Earth departure ΔV, were examined in this study. The launch opportunities for these trajectories occur during the 1980 to 1989 time period at intervals of about 19.5 months. Some of the characteristics of the preencounter and postencounter trajectories are summarized in table 2.

The injection ΔV's for these trajectories range from 3.501 to 3.606 kilometers per second (11,490 to 11,830 ft/sec). The Earth-to-Venus and Venus-to-perihelion coast times are tabulated. With a properly oriented swing-by of Venus, the resulting post swing-by orbits have perihelia in the range of 0.50 to 0.55 AU and aphelia in the range of 0.72 AU to 0.75 AU. The range of aphelia is beyond Venus' orbit which has a perihelion of 0.718 AU. To preclude a subsequent encounter of the waste package with Venus, a propulsive burn can be performed at the perihelion of the post swing-by orbit (the optimum location to perform the ΔV) to lower the aphelion below Venus' orbit. After this burn the waste package would be in its final solar orbit, with an perihelion of about 0.5 AU and an aphelion of about 0.7 AU. The ΔV required to lower the aphelion to 0.70 AU for the cases shown in table 2 ranges from 300 to 600 meters per second (1,000 to 2,000 ft/sec). It should be noted that the ΔV required to obtain a 0.50 AU

(1) Data for these destinations were obtained from Victor Bond of NASA/JSC.

by 0.70 orbit without the Venus swing-by is about 7.80 kilometers per second (25,600 ft/sec). This ΔV is about double the ΔV required for the Venus swing-by cases. As another point of comparison, the Venus swing-by cases have a total ΔV requirement only slightly greater than the 0.90 AU circular solar orbit discussed earlier.

Six Mars swing-by trajectories, having a minimum Earth departure ΔV, were examined in this study. The launch opportunities for these trajectories occur during the 1979 to 1990 time period at intervals of 25 to 27 months. Some of the characteristics of the preencounter and post encounter trajectories are summarized in table 3.

The injection ΔV's for these trajectories range from 3.613 to 3.868 kilometers per second (11,850 to 12,690 ft/sec), which are slightly greater than those required for Venus swing-by trajectories. The coast times required for Earth-to-Mars are tabulated. (They are almost double the coast times required for Earth-to-Venus trajectories.) With a properly oriented swing-by of Mars, the resulting post swing-by orbits have perihelia of about 1.1 AU and aphelia in the range of 1.46 AU to 1.76 AU. The perihelia are outside the Earth's orbit which has an aphelion of 1.017 AU. However, the aphelia extend beyond Mars orbit which has a perihelion of 1.38 AU. To preclude a subsequent encounter with Mars, a propulsive burn can be performed at the perihelion to lower the aphelion inside Mars' orbit. The ΔV required to lower the aphelion to 1.35 AU for the cases shown in table 3 ranges from 60 to 180 meters per second (200 to 600 ft/sec). After the burn the waste package would be in its final orbit, with a perihelion of about 1.1 AU and an aphelion of about 1.35 AU.

The waste package will require a midcourse trajectory correction system (with currently achievable injection accuracies) to achieve a proper swing-by position at the swing-by planet. The midcourse ΔV's required and their point of application can only be determined from a detailed examination of the injection errors and how they effect the swing-by conditions and the post encounter trajectory. These kinds of detailed questions are beyond the scope of this study. The midcourse correction requirement will increase mission complexity and waste package cost. There will be further increases in mission complexity and waste package cost if a propulsion system is used to adjust the post swing-by orbit to preclude a subsequent encounter with the swing-by planet. In addition, it must perform reliably over long periods of time; e.g., it must perform a propulsive maneuver after coasting about 300 days in the Venus case and about 600 days in the Mars case.

As was the case for the near-Earth solar orbits, the problem of demonstrating the long-term stability of solar orbits achieved using a planetary swing-by (of the order of a million years) is unresolved. Also, the integrity of the waste package is not known over a long period of time. However, these problems may be of somewhat lesser importance than for near-Earth solar orbits since the final solar orbits resulting from a planetary swing-by are further away from Earth's orbit.

A basic disadvantage of all swing-by missions is that they cannot be launched every day as could the previous destinations discussed. A launch opportunity to Venus occurs only once every 19 months and to Mars about every 26 months. The duration or "width" of each of these launch opportunities, however, can be about three to four months long without major increases in injection ΔV. To illustrate this, injection ΔV's required for several Earth departure periods for a 50-day launch opportunity to Venus and Mars are shown in tables 4 and 5, respectively. These ΔV's can be compared with the minimum injection ΔV's previously presented in tables 2 and 3. Even though the "width" of these launch opportunities can be about 3 to 4 months long, they may be too limited to support effectively the anticipated number of launches required. Even if only the long-half-lived material were placed in space, it is anticipated that eventually one launch a week or more on a continuing basis would be required. For a Venus swing-by mission, this means that there would have to be almost daily launches over perhaps a 90-day period, once every 19 months. For a Mars swing-by mission, there would have to be at least one launch per day over perhaps a 90-day period, once every 26 months. In either case, such an operation would be expensive in terms of the required Space Shuttle fleet size, the required number of launch facilities, and use of ground crews. (For example, the reusable Space Shuttle is expected to have a two week turn-around-time between launches.) This problem could be alleviated somewhat by using both the Venus and Mars swing-by launch opportunities. However, since the swing-by missions offer no outstanding advantages over the 0.90 AU solar orbit case (which can be launched any day), the latter case seems the better choice.

Solar System Escape

Since both the Earth orbit and solar orbit destinations involve uncertainties regarding long-time orbit stability and containment system integrity, solar system escape and solar impact should also be considered as possible waste package destinations. For solar system escape the waste package is removed from the solar system; for solar impact the waste package is destroyed. Both of these destinations can be achieved either directly or via Jupiter swing-by. Of the two, it takes less ΔV to escape the solar system, and this case is discussed first.

Direct solar system escape - Solar system escape can be achieved with a single propulsive burn from Earth parking orbit with all the propulsion and guidance provided by the launch vehicle. The waste package can be passive and requires no additional propulsion and thus no astrionics systems. A sketch of a direct solar system escape trajectory is shown in figure 11. The ΔV required from an initial Earth parking orbit is applied in the proper direction such that the resultant heliocentric velocity change is added in the direction of the Earth's orbital velocity. This results in a hyperbolic solar escape trajectory where the waste package leaves the solar system. The most efficient trajectories (those which result in maximizing payload) are those in or near the ecliptic plane, and consequently the waste package would traverse the asteroid belt. The probability of the waste package encountering an asteroid requires further study. Based on a circular Earth orbit around the Sun at a distance of 1.0 AU and a solar escape trajectory in the ecliptic plane, the ΔV required from a 185 kilometer (100 n.mi.) Earth parking orbit is 8.75 kilometers per second (28,700 ft/sec). The launch can be made on any day although there is a small variation in injection ΔV required throughout the year since the Earth's orbit is slightly elliptical. It would take about 20 years for the waste package to reach the mean orbital distance of Pluto, but there is not difficulty in selecting a trajectory that would miss the outer planets. And it would take over a million years for it to reach the distances of the nearest stars. Thus, except for its high ΔV requirement, direct solar escape is the most attractive destination discussed thus far. It shares one problem with all destinations beyond Earth. That is, in the event of a propulsion system failure after reaching Earth-escape velocity, the waste package would be left in an unplanned orbit around the Sun.

<u>Solar system escape via Jupiter swing-by</u> - As will be discussed in a later section on launch vehicles, it is difficult to provide the high ΔV required for the direct solar escape mission with current launch vehicles. One means for reducing this ΔV requirement is to use a Jupiter swing-by trajectory. A sketch of a Jupiter swing-by trajectory is shown in figure 12. A Jupiter swing-by trajectory can be achieved with a single propulsive burn (performed by the launch vehicle) from Earth parking orbit. After the waste package coasts for over a year along this trajectory, it will swing by Jupiter. As the waste package swings by Jupiter, the energy of the preencounter trajectory is changed. The amount of change depends on the approach velocity vector of the waste package with respect to Jupiter's orbital velocity vector and the proximity of the waste package's closest approach to Jupiter. In general, if the waste package passes behind and in proximity to Jupiter, its velocity vector with respect to Jupiter will be turned as shown in figure 12, such that its velocity with respect to the Sun is increased and it escapes the solar system. The waste package will require a midcourse correction system (with currently achievable injection accuracies) to achieve a proper swing-by position at Jupiter. A good example of a Jupiter swing-by mission is the Pioneer F spacecraft which flew by Jupiter in December of 1973 and was swung into a solar escape path.

A launch opportunity to Jupiter occurs about once every 13 months. The ΔV required in Earth orbit to escape the solar system via Jupiter swing-by is largely dependent upon the year of launch, the duration or "width" of each launch opportunity, and the trip time. To determine the velocity requirements to achieve solar escape via Jupiter swing-by, five typical launch opportunities were examined using the data of reference 5. These data indicate that for an assumed swing-by radius of five Jupiter radii and a 30-day launch duration, the injection ΔV would range from about 6.71 to 7.32 kilometers per second (22,000 to 24,000 ft/sec) with corresponding trip times to Jupiter ranging from about 600 to 1,000 days. For higher swing-by radii than that assumed or for faster trip times than those above, the ΔV requirement will increase.

The principal advantage of using a Jupiter swing-by gravity assist to achieve solar escape energy rather than the direct mode is that it reduces the ΔV by about 1.5 to 2.0 kilometers per second (5,000 to 7,000 ft/sec). However, it suffers from many of the same disadvantages that were discussed previously for the Venus and Mars swing-by missions. The waste package can no longer be passive since it will require a trajectory midcourse correction capability.

The capability of launching every day is lost since the Jupiter launch opportunity only occurs once every 13 months. As was the case for the Venus and Mars swing-by missions, this launch opportunity is considered too limited to support effectively the anticipated number of launches required. In general, the Jupiter swing-by could be more restrictive than the Venus and Mars swing-bys. It would appear to be much simpler to use the direct solar escape mission, even though the ΔV is somewhat higher than for the Jupiter swing-by mission to solar escape. The effect of the higher ΔV on launch vehicle payload capability will be shown later.

Solar Impact

A solar impact is possible either directly or indirectly via Jupiter swing-by to turn the trajectory into the Sun. Again the purpose of using a Jupiter swing-by is to reduce the ΔV requirement.

Direct solar impact - Direct solar impact can be achieved with a single propulsive maneuver from Earth parking orbit. A sketch of a direct solar impact trajectory is shown in figure 13. The injection ΔV is provided by the launch system in such a way that the resultant heliocentric velocity change is nearly equal to and opposite to the Earth's mean orbital velocity about the Sun. (The Earth's mean orbital velocity is 29.79 kilometers per second (97,700 ft/sec) at a distance of 1.0 AU from the Sun.) The heliocentric velocity change tends to cancel the Earth's orbital velocity, so that the waste package "falls into" the Sun. The range of ΔV's shown in figure 13 reflect the difference between reaching the Sun's surface (or photosphere) or approaching the Sun along a straight path. In the first case, the heliocentric velocity change does not cancel out all of the Earth's orbital velocity and the resulting solar orbit is a highly eccentric ellipse having a perihelion at the Sun's surface (0.005 AU). This is assumed to be sufficient for solar impact. In the second case, all of the Earth's orbital velocity is canceled and the waste package essentially "falls into" the Sun along a straight path. For ΔV's within this range the waste package would presumably enter the Sun's atmosphere and be consumed. There is a slight variation in the range of ΔV's depending on the launch day. A representative trip time for a direct solar impact is about 65 days.

The direct solar impact shares many of the advantages and disadvantages of the direct solar system escape case. The waste package can be passive and can be launched any day. Except for failures, there are no problems of orbit stability or encounters with the Earth or other planets. The problem of a launch vehicle failure occurring near Earth-escape velocity is similar to the solar system escape mission.

The main disadvantage of the direct solar impact mission is that it requires an extremely high ΔV. For this mission the ΔV's required are far beyond the capability of current or planned launch systems, and therefore it is considered impractical.

Solar impact via Jupiter swing-by - A solar impact can be achieved with a single propulsive maneuver from Earth parking orbit using a Jupiter swing-by to turn the trajectory back into the Sun. A sketch of this trajectory is shown in figure 12. The outbound trajectory is similar to that of the solar escape via Jupiter swing-by except at the Jupiter approach. In this case the swing-by is properly designed such that the waste package passes in front of and in proximity to Jupiter. As a result, the waste package's velocity vector with respect to Jupiter is turned such that it is nearly equal and opposite to Jupiter's orbital velocity. These velocities then tend to cancel one another resulting in the waste package having zero or near zero velocity with respect to the Sun. Under these conditions the waste package will impact the Sun as shown in figure 12.

Solar impact trajectories using a Jupiter swing-by were briefly analyzed assuming circular coplanar Earth-Jupiter orbits. The purpose in doing this preliminary analysis was to determine the approximate injection ΔV's and the swing-by distances required. The required Jupiter swing-by distance to ensure a solar impact (perihelion $\leq$ 0.005 AU) is shown in figure 14 as a function of the ΔV above circular Earth orbit. There is a minimum ΔV required below which a solar impact cannot be achieved. Above that ΔV the required swing-by distance is a band representing grazing impacts on either side of the Sun. The lower portion of the band represents post Jupiter orbits previously discussed (i.e., orbits which "fall into" the Sun after Jupiter encounter). The upper dotted portion of the band represents post Jupiter orbits which first move out from Jupiter after the encounter before falling back towards the Sun. It appears that a ΔV of about 7.6 km/sec (25,000 ft/sec) is a reasonable choice for this mission. This would allow the swing-by radius to range from about 5 to 15 Jupiter radii. For this case the trip time to Jupiter is about 500 days, and the total trip time to solar impact ranges from about 1,100 to 1,500 days, depending on the swing-by radius.

The principal advantage of using a Jupiter swing-by to achieve a solar impact rather than a direct solar impact is that the ΔV requirement can be reduced appreciably, from about 24.08 kilometers per second (79,000 ft/sec) down to about 7.62 kilometers per second (25,000 ft/sec). However, a solar impact achieved via Jupiter swing-by suffers the same disadvantages as was previously discussed for the Jupiter swing-by to achieve solar system escape. The Jupiter launch opportunity occurs only every 13 months. Even if the ΔV were increased to the same value as for the direct solar escape mission, 8.75 kilometers per second (28,700 ft/sec), the width of each launch opportunity would be only of the order of 40 days. Again, the Jupiter launch opportunity is too limited to support effectively the high launch rates expected for waste disposal missions, and it would appear simpler to use the direct solar escape mission which could be launched on any day.

Other Destinations

Many other space destinations in addition to the ones discussed have been suggested. Examples include depositing the waste packages on the Moon, on planets, in planetary orbits, on asteroids, and at Lagrangian equilibrium points. These destinations were not considered in this study although in some cases they could warrant further investigation. The general arguments against these destinations include the following:

(1) A landing failure could result in widespread contamination.
(2) The regions are unexplored and/or are of scientific interest.
(3) Some of the regions could be of future value from an applications standpoint.
(4) Launch opportunities are limited.
(5) Deep-space propulsion is required and in many cases the retropropulsion ΔV's are high.

COMPARISON OF DESTINATIONS

As a summary of the destinations discussed, table 6 lists the typical ΔV requirements for the various missions and their principal advantages and disadvantages. The ΔV's shown are representative for each destination, although there will be some variation depending on the particular launch opportunity and the details of the mission profile. These ΔV's are used to determine the performance of the various launch systems considered in the LAUNCH VEHICLE PERFORMANCE AND COST section which follows. The ΔV for high Earth orbits is an upper value for orbits between synchronous and lunar orbit altitudes. The Earth-escape ΔV includes some provision for additional ΔV (beyond Earth-escape ΔV) in an effort to minimize the probability of a subsequent Earth reencounter as was previously discussed.

The ΔV's for the other solar orbits include the ΔV's required after departing from Earth to prevent the waste package from encountering any planet. The term "passive waste package" implies that the package will require no special space propulsion, midcourse correction or associated astrionics systems. The abort possibility past Earth-escape velocity (referred to as the abort gap) is a disadvantage associated with all destinations beyond the Earth. As was previously discussed, if the launch vehicle should fail after reaching Earth-escape velocity, the waste package would be left in an unplanned solar orbit with a possibility of subsequently encountering the Earth or another planet. With current state-of-the-art launch systems it would be impractical to recover the waste package from these orbits.

The conclusions reached thus far indicate the most promising candidate mission destinations are high Earth orbits, solar orbits (near Earth), and direct solar system escape. The payload capabilities of possible launch systems for these mission destinations are discussed in the next section.

LAUNCH VEHICLE PERFORMANCE AND COST

As was discussed in the INTRODUCTION, only the larger current and planned launch vehicles have been considered in this study. They are shown in figure 1. The Titan IIIE launch vehicle consists of two solid rocket motors strapped to two tandem earth-storable propellant stages. The Centaur upper stage is fueled with liquid hydrogen-liquid oxygen propellant. The Titan IIIE/Centaur is the expendable booster that will launch the 1975 Viking mission to Mars. The Saturn V is the three-stage expendable Apollo booster. The first stage is fueled with liquid kerosene-liquid oxygen propellant and the upper stages are fueled with liquid hydrogen-liquid oxygen propellant. Its two stage version has been used to launch Skylab. The Space Shuttle is primarily reusable and is to be operational in 1980. It is planned as a replacement for virtually all the nation's space boosters in operation today. The Space Shuttle consists of two strap-on solid rocket motors, the manned orbiter, and an external liquid oxygen liquid hydrogen propellant drop tank which fuels the orbiter. As is discussed later, the Space Shuttle will require an additional stage for the disposal mission.

EXPENDABLE LAUNCH VEHICLES

Performance

Performance data for the Titan IIIE/Centaur and the Saturn V are shown in figure 15. The data are based on a due-East launch from ETR into a 185 kilometer (100 n.mi.) parking orbit. The upper stage of the launch vehicle provides the ΔV needed to accelerate the payload to higher velocities from the parking orbit. Typical ΔV requirements for the various destinations previously discussed are shown in the figure. A ΔV of 4.11 kilometers per second (13,500 ft/sec) is used to characterize the high Earth orbit and solar orbit destinations. Actual ΔV's will vary somewhat depending on the details of the specific mission design. As is evident from figure 15, the direct solar impact mission (24.08 kilometers per second or 79,000 ft/sec) is well beyond the capability of current vehicles. The Titan IIIE/Centaur can deliver 3,860 kilograms (8,500 lbs) to high Earth orbit and solar orbit destinations. It has no payload capability for the solar escape mission. The Saturn V can deliver 32,700 kilograms (72,000 lbs) to high Earth orbit and solar orbit destinations, but it also has no payload capability for a direct solar escape mission. The use of the Centaur as an upper stage on the Saturn V provides a direct solar escape mission payload capability of about 7,260 kilograms (16,000 lbs).

Cost

The costs of the expendable launch vehicles depend greatly on the use rate. The Titan IIIE/Centaur cost is about $27 million at a production rate of four per year. At the higher launch rates expected for space disposal of radioactive waste, the costs would be expected to be considerably lower. For this study, it is assumed that the cost of the Titan IIIE/Centaur at high launch rates can be reduced about 30 percent, and thus its cost would be $19 million. Similarly, the costs of the Saturn V and Saturn V/Centaur are taken at $150 and $155 million, respectively. As mentioned in the INTRODUCTION, the launch vehicle costs used in this study cover the normal recurring production and operations costs. They do not include operational costs associated with handling the nuclear waste at the launch or the integration of the waste package with the launch vehicle.

SPACE SHUTTLE

The Space Shuttle by itself can deliver payloads only to low Earth orbit. (Shuttle ascent trajectory characteristics and considerations, particularly for a Shuttle carrying a nuclear waste package are presented in Appendix A.) Missions beyond low Earth orbit will be accomplished by having the Space Shuttle carry both a propulsion stage and the mission payload to Earth orbit in its cargo bay. The propulsion stage is generally referred to as a Space Shuttle third stage. After the third stage and payload are deployed in Earth orbit from the shuttle orbiter, the third stage will inject the payload to its destination. Existing expendable upper stages are currently being evaluated for early use as Space Shuttle third stages. These stages would be expended on each flight. However, it is planned to eventually develop a new space tug explicitly for use as a Space Shuttle third stage which would have the capability of being recovered and reused. The Space Shuttle would launch the tug and payload into low Earth orbit. After the tug and payload are deployed from the shuttle orbiter, the tug will inject the payload to its mission destination. Following the injection burn, the payload is separated from the tug and the tug does a series of burns to return to the waiting shuttle orbiter for recovery and reuse. (There is a brief discussion of the consequences of an upper stage abort during the orbital launch phase in Appendix A.)

Several Space Shuttle/third stage options were considered in this study:

(1) One of the reusable space tug concepts under study by NASA.
(2) A similar reusable tug but optimally sized for the waste disposal mission.
(3) The current expendable Centaur stage.
(4) A similar expendable Centaur stage resized for the waste disposal mission.

The high launch rates envisioned for the waste disposal mission could justify resizing the tug or Centaur stage if the performance gains are worthwhile.

Performance

The performance of the various Space Shuttle/third stage combinations is shown in figure 16. The performance data are based on a Space Shuttle delivery capability of 29,500 kilograms (65,000 lbs) into a due-East 185 kilometer (100 n.mi.) orbit. The data are based on ideal ΔV calculations. A two percent ΔV contingency is included in all performance computations to account for flight performance reserve (for make-up of off nominal performance) and for various performance losses which were not specifically accounted for. Payload mounting and adapter weights (attaching payload to third stage) were not accounted for and would have to be subtracted from the payloads quoted. Typical ΔV's for the various mission destinations are indicated in the figure.

The reusable space tug performance is based on one of the higher performing configurations studied to date. The tug is designed to perform a round trip mission to geostationary (synchronous) orbit with a 1,360 kilogram (3,000 lb) payload. All the groundrules for defining the reusable space tug concept are summarized in table 7. It is a hydrogen-oxygen fueled stage with an engine specific impulse of 470 seconds and it has a propellant capacity of approximately 23,880 kilograms (53,000 lbs). (Note an allowance of 1,360 kilograms (3,000 lbs) has been set aside for tug support weight in the orbiter cargo bay.) The space tug gross weight is 26,760 kilograms (59,000 lbs). The tug propellant capacity is sized to fully utilize the Space Shuttle capability with a 1,360 kilogram (3,000 lb) payload. Therefore, for payloads greater than this, tug propellant must correspondingly be off-loaded. Optimally resizing the space tug (decreasing the propellant capacity) therefore will improve its performance as shown by the dashed curve in figure 16. As can be seen from this figure, the only destinations which result in useful payloads are high Earth orbits and solar orbits (which for convenience are all characterized by a ΔV of 4.11 kilometers per second (13,500 ft/sec)). At its current size, the reusable tug can deliver a payload of 4,180 kilograms (9,200 lbs) to these destinations. The optimally sized tug (about 20,900 kilograms (46,000 lbs) propellant) can deliver a payload of 4,680 kilograms (10,300 lbs).

The current Centaur stage also uses hydrogen-oxygen propellants. It has an engine specific impulse of 444 seconds and a propellant capacity of about 13,630 kilograms (30,000 lbs). For the waste disposal missions, it is too small to use the full 29,500 kilogram (65,000 lb) low-Earth orbital capability of the Space Shuttle. Consequently, the performance of the Centaur can be improved by increasing its propellant capacity as shown by the dashed curve on figure 16. For the high Earth orbit and solar orbit destinations, the current Centaur stage can deliver a payload of 6,500 kilograms (14,300 lbs). An optimally sized Centaur (about 17,250 kilograms (38,000 lbs) propellant) can deliver a payload of 8,490 kilograms (18,700 lbs).

It should be recognized that the higher payload capability shown for the Centaur in figure 16 is a consequence of its being an expendable stage. For the reusable tug, a portion of its propellant is required for its return to the shuttle orbiter waiting in low Earth orbit. If the tug were expended, its performance would be comparable to that for the optimally sized Centaur stage.

Cost

The recurring cost per Space Shuttle flight is estimated at approximately \$10.5 million. In addition, the cost per reusable tug flight is estimated to be \$1.75 million. This cost includes propellants, operations, refurbishment and amortization of a unit production cost of \$20 million. Totaling the two, the cost per flight of a Space Shuttle/reusable tug is about \$12.25 million. The cost of the expendable Centaur stage at the high launch rates expected for waste disposal would be about \$5.5 million. In total, the cost of a Space Shuttle/expendable Centaur launch is about \$16 million.

LAUNCH VEHICLE PERFORMANCE AND COST COMPARISON

Except for the Saturn V/Centaur, the launch vehicles considered thus far can deliver useful payloads only to high Earth orbit or solar orbit destinations. In order to provide an overall vehicle comparison for these destinations, the payload, cost, and cost per pound of payload delivered to a ΔV of 4.11 kilometers per second (13,500 ft/sec) are summarized in table 8. These data should be used only for making preliminary comparisons since other factors will have to be considered in making a vehicle selection. For example, there are limits on the desired waste package size. The desired waste package size is influenced by the mix of the waste material (fission products) being disposed of, the allowable radiation dose rates specified, and the ground impact and reentry design requirements. Also, the nuclear waste is only a small fraction of the total waste package weight, and this fraction will vary with waste package size. These and other factors will influence the choice of a launch vehicle for a particular destination. Nonetheless, table 8 shows that the Space Shuttle vehicles are more cost-effective than the current expendable launch vehicles. The cost per kilogram of total payload delivered by the Space Shuttle vehicles is of the order of one-half of that delivered by expendable launch vehicles.

For the shuttle-launched missions, it appears worthwhile to resize the upper stages for the waste disposal mission. The improved performance and cost effectiveness should readily justify the non-recurring costs associated with resizing the stages. The cost per kilogram of payload delivered with the resized Centaur stage is about 25 percent lower than for the resized reusable tug. This indicates that an expendable third stage would be more cost effective than a reusable stage. However, it is recommended that both reusable and expendable shuttle third stages continue to be considered in further evaluations, since in addition to cost effectiveness, safety considerations and specific mission details can influence the final choice. For example, the reusable tug performance is very sensitive to mission ΔV. If the selected mission requires a ΔV somewhat lower than the 4.11 kilometers per second (13,500 ft/sec), the reusable tug performance will improve significantly. If the required mission ΔV were below about 3.3 kilometers per second (10,800 ft/sec), a reusable third stage (tug) could be more cost-effective than an expendable stage (Centaur).

If an expendable stage such as a Centaur is used for the waste disposal mission, it will still be necessary to provide a reusable tug to recover from possible mission failures. If the Centaur stage should fail before reaching Earth-escape velocity, the waste package would be left in an unplanned Earth orbit. In this case, a shuttle/reusable tug launch could be made to either retrieve or properly inject the waste package. Such a retrieval mission will involve rendezvous and docking with the payload, and only the reusable tug will have this capability. To perform this orbital operation an adequate payload orbit lifetime is essential. In this regard, the 185 kilometer (100 n.mi.) parking orbit which has been used throughout the study for comparison purposes is not a good choice. To illustrate this, figure 17 shows how circular orbit lifetime varies with parking orbit altitude for a range of ballistic coefficients (W/C_dA). Although the payload configurations weight, size and shape have not been firmly established, a range of ballistic coefficients was assumed which will most likely bracket the ballistic coefficient of the final payload configuration. Configurations having low ballistic coefficients will have the lowest orbital lifetimes. For example, for a ballistic coefficient of 244 kilograms per square meter (50 lbs/ft^2), the orbital decay time of a package left in a 185 kilometer (100 n.mi.) parking orbit is of the order of a few days. In this case, if the injection stage should fail at or shortly after ignition, there would be insufficient time for taking corrective action. To ensure adequate orbit lifetime, a parking orbit for deployment of the payload from the shuttle orbiter of the order of 370 kilometers (200 n.mi.) would be a better choice. At this altitude and for the same ballistic coefficient (244 kg/m^2), the orbital decay time of a package would be several hundred days, as can be seen from figure 19. This would allow adequate time for making a second shuttle/reusable tug launch which could either retrieve or properly inject the waste package to its destination.

The performance data presented for the missions using the Space Shuttle have been based on a 29,500 kilogram (65,000 lb) payload capability to a 185 kilometer (100 n.mi.) parking orbit. But the Space Shuttle can also deliver 29,500 kilograms (65,000 lbs) to a 370 kilometer (200 n.mi) parking orbit. Figure 18 shows the Space Shuttle payload capability as a function of circular orbit altitude. The payload curve shows that payloads higher than 29,500 kilograms (65,000 lbs) could be carried to a 370 kilometer (200 n.mi.) parking orbit, but the current shuttle structure design limits the payload carrying capability to 29,500 kilograms (65,000 lbs). Thus, the performance data presented for the missions using the Space Shuttle are essentially unaffected by the choice of a 370 kilometer (200 n.mi.) parking orbit.

MULTIPLE SPACE TUG CONFIGURATIONS

The only launch vehicle considered thus far that has a useful payload capability for the direct solar escape mission is the Saturn V/Centaur. As shown in figure 15, it can deliver a payload of about 7,260 kilograms (16,000 lbs) to this destination. At a launch cost of $155 million, this results in a specific cost of $21,300 per kilogram ($9,700 per lb). This is roughly an order of magnitude higher than for the shuttle launches to high Earth or solar orbits. One possibility for providing a more cost-effective solar escape capability is to use several shuttle/tug launches to assemble a larger vehicle (consisting of several space tugs) in Earth orbit. This approach could also be used to provide higher payloads (if desired) for the Earth orbit and solar orbit destinations.

A preliminary study of the use of multiple shuttle/tug launches has been previously done for NASA missions (reference 6). The ground rules and assumptions of that study are not all specifically applicable to the waste disposal mission. However, the results are discussed since they serve to show the potential for achieving the more difficult (high ΔV) waste disposal missions.

The procedure would be to use several shuttle launches to place several tugs along with the payload into low Earth orbit. The tugs, which have a rendezvous and docking capability, would be assembled in orbit to form a tandem vehicle. In performing the mission, the tugs burn sequentially. And each, if it is to be recovered, returns to its waiting shuttle orbiter.

In this preliminary evaluation of a tandem vehicle, only the fixed-sized tug (defined in table 7) is considered. It is assumed to be available in both reusable and expendable configurations. The tug and shuttle performance parameters and costs are the same as discussed previously. Each shuttle flight is assumed to cost $10.5 million and each reusable tug flight, $1.75 million. The one exception is the cost of an expended tug. The expected unit cost of the reusable tug is of the order of $20 million. If the waste disposal mission required expending a tug, the cost of the expendable tug could be considerably lower. The production rate for an expended tug would be much higher than for a reusable tug since each disposal mission would require a new tug. The high use rate would probably justify development of an expendable tug that incorporated only the features necessary for accomplishment of the waste disposal mission. As an alternate approach, a modified version of the existing Centaur stage could be used as an expendable tug. An accurate cost for the expendable tug cannot be established at this time, but for the purposes of this study it is taken as $6 million per flight.

An overall performance map of the various multiple shuttle/tug combinations is presented in figure 19. The performance shown is optimistic since gravity losses during space tug burns were not included. Gravity losses can be quite high for the tandem tug configurations as is discussed later. The most cost-effective vehicle for each region on the map is indicated by the coding shown. The first digit indicates the number of shuttle flights required to launch the tugs and payload. The second digit indicates the number of expendable tugs in the assembled vehicle and the third digit the number of reusable tugs. While the combinations of expendable and reusable tugs could be extended to reach even higher payload-ΔV regions, only configurations costing less than $50 million are shown. When a mix of recoverable and expendable tugs is used, the recoverable tugs are the lower stages (burned first) since this is the optimum arrangement. When the number of shuttle launches exceeds the number of tugs; for example (2, 0, 1), it implies that the waste package is brought up in a separate shuttle launch.

The number following the three digit coding is the total transportation cost per launch. In all cases, it is assumed that the recoverable tugs are brought back to Earth with the shuttle orbiters used to initially launch the tugs and waste package. That is, no additional shuttle cost is charged for returning a tug.

As can be seen from figure 19, the direct solar impact with a ΔV of 24.08 kilometers per second (79,000 ft/sec) cannot be achieved. However, several of the configurations can accomplish the direct solar escape mission with a ΔV of 8.75 kilometers per second (28,700 ft/sec). The payload for the (1, 1, 0) configuration is too low to be useful. The (2, 2, 0) configuration cannot be used since it requires a rendezvous of two expendable tugs in orbit which do not have a rendezvous or docking capability. (A similar argument precludes the use of the (3, 1, 1) configuration.) This leaves the (2, 1, 1) and (3, 1, 2) configurations which have a direct solar escape payload capability of 3,910 and 6,090 kilograms (8,600 and 13,400 lbs), respectively. As mentioned previously, gravity losses will significantly reduce the actual performance of these multi-stage tug configurations. The gravity losses have been determined for the (2, 1, 1) and (3, 1, 2) configurations assuming the tug has a thrust of 88,940 newtons (20,000 lbs). The actual capability of the (2, 1, 1) and (3, 1, 2) configurations for a direct solar escape mission is 2,270 and 3,040 kilograms (5,000 and 6,700 lbs), respectively.

A higher tug thrust level could be used to reduce the gravity losses. However, since the tug is being designed for single stage mission applications, it is not expected that the new tug engine will have a thrust level higher than 88,940 newtons (20,000 lbs). If, however, a dedicated tug were developed for a multi-stage tug configuration waste disposal mission to direct solar escape, it would likely have a higher thrust level. Another approach for reducing the gravity losses is to use a technique referred to as perigee propulsion. (It is explained in some detail in reference 6.) This is operationally more complicated and necessitates carrying the waste package once around the Earth in an elliptical orbit between tug burns. However, using perigee propulsion increases the payload capability of the (2, 1, 1) and (3, 1, 2) configurations for the direct solar escape mission to 3,270 and 4,400 kilograms (7,200 and 9,700 lbs), respectively.

An overall comparison of launch vehicle performance and cost for the direct solar escape mission is shown in table 9. The expendable Saturn V/Centaur provides the highest payload weight, but at a cost of about $21,300 per kilogram ($9,700 per lb). The multiple shuttle/tug configurations using perigee propulsion have lower payloads, at a cost of about $9,000 per kilogram ($4,100 per lb). This lower cost, however, is of the order of four times higher than the cost for the high Earth orbit and solar orbit destinations using the shuttle/third stages considered (table 8).

The multiple shuttle/tug vehicles can also provide more capability, if needed, for the higher Earth orbit and solar orbit destinations. The (1, 0, 1) configuration of figure 19 is identical to the current size reusable tug of table 8, and the (1, 1, 0) configuration, as would be expected, has essentially the same performance as the optimum size Centaur. From figure 19, only those multiple tug configurations which can be assembled in Earth orbit are considered. Their performance and cost are summarized in table 10. These configurations have not been studied in detail; e.g., gravity losses during the tug burns were not included. However, they can provide higher payload capability and in some cases, at slightly lower specific costs than the single shuttle launch configurations. Of the configurations shown, the (3, 0, 2) configuration provides the highest payload of 20,000 kilograms (44,000 lbs) at the lowest specific cost of $1,750 per kilogram ($800 per lb). Comparing this configuration with the optimum size Centaur (single shuttle launch), its payload is more than double and its specific cost is about 10 percent lower.

CONCLUDING REMARKS

Of the space destinations considered, high Earth orbits, solar orbits and direct solar system escape remain as candidate destinations and all three should continue to be studied. The final selection of the destination will depend on cost and safety considerations beyond those considered in this report.

For high Earth orbits, circular orbits between geostationary (synchronous) and lunar orbit altitudes appear to be the best choice. However, since neither orbit stability nor waste package integrity can be guaranteed for times of the order of a million years, high Earth orbits cannot now be considered a permanent disposal site. Unless further studies can resolve these uncertainties, Earth orbits should only be considered a temporary storage site, since package retrieval or placement to more remote destinations may eventually be required.

Among the possible solar orbit destinations considered, circular solar orbits suitably displaced from the Earth's orbit around the Sun (e.g., at 0.90 AU) to prevent Earth reencounter possibilities currently appear to be the best choice. This destination requires a propulsive maneuver about six months after departing from Earth. The guidance and control requirements associated with this maneuver can be minimized by performing it with a relatively simple spin-stabilized, solid rocket motor. A disadvantage with this destination and all other destinations beyond the Earth is that they have an "abort gap" during the Earth departure burn. If the departure propulsion or guidance system should fail at or beyond Earth-escape velocity, the waste package will be left in an unplanned solar orbit. In this case, there could be a high probability of eventually reencountering the Earth. This abort problem, as well as the long-term stability of solar orbits, needs more investigation. The possibility of achieving acceptable solar orbits using only a single departure burn from Earth orbit should also be studied further since this would eliminate the need for a waste package propulsion system.

The problems of long-term orbit stability and waste package integrity can be avoided by injecting the waste package to solar system escape or impacting it into the Sun. The direct solar escape mission can be accomplished by using two or three stage space tugs in tandem. The resulting space transportation system cost is about four times higher than for the high Earth orbit and solar orbit destinations. If this is acceptable, solar system escape can eliminate potential orbit instability, long-term package integrity, and future planet encounter problems. A direct solar impact mission cannot be achieved with the current or planned launch systems considered in this study. It can be accomplished using a Jupiter swing-by for roughly the same ΔV as for the direct solar escape mission. It should be noted that a solar system escape can also be accomplished using a Jupiter swing-by for a ΔV somewhat lower than the direct mode. However, in both cases, solar impact or solar system escape using a Jupiter swing-by, the launch opportunity will be quite limited.

Regarding the space transportation system, the currently planned Space Shuttle is more cost-effective than current expendable launch vehicles by about a factor of 2. The Space Shuttle will require a third stage to perform the disposal missions. Depending on the particular mission, this third stage could be either a reusable stage such as the space tug or an expendable stage such as a Centaur. In either case, the third stage should be resized for the selected disposal mission. In fact, the launch rates required for waste disposal are expected to be sufficiently high that it will probably be worthwhile to develop a version of the entire launch vehicle dedicated to providing maximum performance, lowest cost, and higher reliability for the disposal mission.

In this study, only current or planned space transportation systems were considered. However, the waste disposal will extend far into the future, and new space technology and systems development can be expected. Consequently, the performance and cost data presented in this study may be conservative as far as future capability is concerned.

APPENDIX A[1]

SPACE SHUTTLE TRAJECTORY CONSIDERATIONS

A general discussion of the Space Shuttle launch phase is presented, and those areas which could be of particular concern for a shuttle carrying a nuclear waste package to orbit are discussed. The Space Shuttle performance data and other shuttle information presented in this appendix are based on the baseline Space Shuttle configuration as it is defined in reference 7. As the shuttle development proceeds, updated design modifications could change this configuration somewhat. A brief description of a typical shuttle launch operational sequence to orbit is given. Nominal shuttle abort techniques for the entire ascent phase are described wherein the orbiter returns intact to the landing site. Impact points for several shuttle ascent trajectories are examined in the event of a shuttle failure causing the shuttle and payload to impact the Earth in an uncontrolled manner. The effect on shuttle performance of these ascent trajectories is discussed. In addition, an upper stage malfunction occurring during the orbital launch phase which results in an aborted mission is considered.

NOMINAL SHUTTLE ASCENT TRAJECTORY

The Space Shuttle flight system is an integrated vehicle composed of the recoverable manned orbiter vehicle containing the main engines, an external drop tank containing the orbiter liquid hydrogen/liquid oxygen ascent propellant, and two solid rocket motors (SRM). The shuttle vehicle is shown in figure 1A.

The shuttle vehicle carrying the nuclear waste package and injection stage in the orbiter cargo bay will be launched from ETR into a 92.6 by 185.2 kilometer (50 by 100 n.mi.) initial orbit using a standard shuttle launch trajectory. A typical Space Shuttle trajectory is shown in figure 2A.

At liftoff, the SRM's and orbiter main engines fire in parallel and propel the Space Shuttle to the desired solid motor staging velocity and altitude. At staging, the SRM's burnout and the SRM cases separate from the orbiter, drop into the ocean down-range and are recovered. The orbiter main engines continue burning to orbit insertion. After orbit insertion the external tank is separated from the orbiter, and the expendable tank is de-orbited by a small retro-rocket. The orbiter vehicle circularizes the orbit at the desired altitude

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(1) Part of the data presented in this appendix was obtained from Victor Bond of NASA/JSC.

using the orbital maneuvering system and performs its on-orbit mission operations, in this case deploying the injection stage and nuclear waste package. After performing its on-orbit mission operations, the orbiter and its crew are returned to Earth.

INTACT SHUTTLE ABORT

In the event of a premature mission termination prior to payload deployment, the payload will be returned to Earth inside the orbiter cargo bay. The capability for intact orbiter recovery is provided throughout the entire mission phase. The launch into the standard 92.6 by 185.2 kilometer (50 by 100 n.mi.) initial orbit will require around 8 minutes depending on the Shuttle configuration and performance. In order to describe the intact abort modes used during the ascent phase, typical time intervals during which they could occur were chosen. Aborts during the first 4.7 minutes of the shuttle ascent phase are suborbital trajectories and result in a flyback to the launch site. For aborts occurring during the first 1.5 minutes after liftoff, the orbiter is separated, and a glide return to the launch site is performed. After 1.5 minutes and up to 4.7 minutes, the orbiter returns to the launch site by means of a powered maneuver using the orbiter main engines. After 4.7 minutes of flight, the shuttle is targeted to once-around abort conditions. That is, the orbiter lands at the launch site after making one revolution around the Earth. Figure 3A shows ground tracks for once-around aborts for launch azimuths from 90 degrees to 130 degrees in 10 degree increments. Entry points are shown for a landing target at ETR. An approximate maximum capability landing envelope is shown for the 90 degree azimuth trajectory. The orbiter cross-range capability is about 2,590 kilometers (1400 n.mi.) at the chosen target down-range distance of 9950 kilometers (5,370 n.mi.). Figure 3A indicates that a landing at either ETR or Vandenberg is possible from any of the trajectories shown.

For aborts after 6.3 minutes of flight along the ascent trajectory, the shuttle has the capability of going into the 92.6 by 185.2 kilometer (50 by 100 n.mi.) initial orbit, circularizing at 185.2 kilometers (100 n.mi.) and later de-orbiting and landing in its normal manner.

CATASTROPHIC SHUTTLE FAILURE

As previously described, the capability for intact orbiter recovery is provided throughout the entire mission phase. However, there is a possibility, although small, of a catastrophic failure occurring during the shuttle ascent phase. This would result in the shuttle (and payload) falling back and impacting the Earth in an uncontrolled manner. As a means of determining the approximate impact point, the shuttle and payload are assumed to fall back along a vacuum ballistic path. (In actuality, launch vehicle dispersions and atmospheric effects will affect the fall back path.) The point of impact is called the instantaneous impact point (IIP). During the shuttle ascent phase the IIP's will trace a line across the surface of the Earth. IIP's are examined for several shuttle launch trajectories.

Shown in figure 4A are several conventional shuttle IIP's representing easterly launches from ETR at launch azimuths ranging from 90 degrees to 108 degrees. They all pass over Africa, but at a time in the ascent profile which is only about three seconds prior to achieving the 92.6 by 185.2 kilometer (50 by 100 n.mi.) initial orbit. The launch azimuth at 108 degrees represents the most southerly launch azimuth normally allowed for current expendable launch vehicles. It provides a miss distance of the IIP's of about 340 kilometers (180 n.mi.) from the down-range Carribean Islands. The IIP dwell time over Africa (the time that the IIP remains over a land mass) for the 90 degree launch azimuth is only of the order of one second.

If the launch azimuth is increased from 90 degrees to reduce dwell time over Africa, the shuttle performance capability decreases. In general, payload as a function of launch azimuth tends to describe a sine curve, with the due-East launch giving the maximum performance. Shuttle performance as a function of launch azimuth can be obtained from reference 7.

It is impossible to launch into orbit using a conventional launch trajectory without the IIP crossing some land mass. However, using yaw steering it is possible to design a launch trajectory such that the IIP misses Africa, further decreasing the probability of a land impact in the event of a catastrophic failure. This type of trajectory, commonly called a dog-leg, requires that the launch vehicle initially fly an easterly heading that will avoid IIP overpasses of land in the Carribean area and then must yaw right to swing the IIP around the southern tip of Africa. However, even with a dog-leg trajectory there is still a possibility of land impact if a catastrophic failure should occur prior to reaching an acceptable lifetime orbit. For example, if a catastrophic failure occurred while the shuttle and payload were in the initial 92.6 by 185.2 kilometer (50 by 100 n.mi.) orbit, the payload would reenter before a retrieval mission could be launched.

The payload for a dog-leg trajectory can be optimized by trading off launch azimuth, time of yaw initiation and yaw steering profile. The trajectory is targeted to an orbit inclination of 40 degrees with respect to the equatorial plane so that the IIP trace will miss the southern tip of Africa.

Preliminary studies have shown that the launch azimuth which maximizes payload is about 97.5 degrees. The resulting maximum payload is about 17,700 kilograms (39,000 lbs), which is significantly lower than the due-East shuttle payload capability of 29,500 kilograms (65,000 lbs). Figure 5A shows the IIP trace for this launch trajectory. The IIP trace misses the down-range Carribean Islands, South America and Africa by a distance of 340 kilometers (180 n.mi.).

The shuttle payload capability for the dog-leg trajectory is considered too small for the injection stage plus waste package combination. To achieve a higher payload capability, multiple shuttle launches are required. In this case the injection stage and the waste package would be carried to orbit in separate shuttle launches, each using a different launch trajectory. As an example of this launch mode, consider a dual shuttle launch. The first shuttle launch would carry the injection stage to orbit using a conventional trajectory. The second shuttle launch carrying the waste package would use a launch trajectory using yaw steering similar to the one described earlier for maximizing payload. After orbit insertion of the waste package the injection stage would then rendezvous and dock with the waste package. (Only the space tug is planned to have this capability.) A launch azimuth of 121 degrees is required for the first shuttle launch so that the injection stage orbit inclination is compatible with the waste package orbit inclination which was selected as 40 degrees. As a result of this launch azimuth requirement, there is a small loss in shuttle payload delivery capability. It is reduced from the full capability (29,500 kilograms or 65,000 lbs) to about 28,000 kilograms (61,700 lbs).

A catastrophic failure resulting in land impact could also occur while returning to Earth during an intact abort. In this case, a non-catastrophic failure occurred during ascent and the orbiter was targeted to return intact to the launch site with the payload still in the cargo bay. A second failure occurring during orbiter reentry could cause a subsequent land impact of the payload. As an example, the vacuum IIP's for a once-around abort from an easterly launch trajectory after the reentry phase has been initiated are shown in table 1A and figure 6A. Seven representative points along the entry trajectory were selected as initial points of the second failure (catastrophic) for the impact trajectories. The cases selected, ranging from 75 kilometers altitude (entry plus 8 minutes) to 30 kilometers altitude (entry plus 29.2 minutes), yield impact points from the Baja Peninsula to the east coast of Florida.

UPPER STAGE ABORT DURING ORBITAL LAUNCH PHASE

During the injection stage burn from the initial parking orbit, a stage malfunction such as a guidance or control system failure could occur. Such a failure would alter the trajectory in an uncontrolled manner as long as the engine burn continues. It is possible, although very unlikely, that an inadvertent reentry trajectory could be achieved. However, precautions can be taken to ensure that a reentry trajectory is not attained. This requires identifying that a failure has occurred and terminating the thrust quickly. Identifying that a failure has occurred will be accomplished by continually tracking the injection stage burn. As soon as a failure has been identified, it is estimated that the thrust can be terminated in sufficient time to prevent immediate reentry and the waste package is left in Earth orbit. A tug could be launched to rendezvous and dock with the waste package to either retrieve the waste package or send it to its mission destination.

REFERENCES

1. Zimmerman, A. V.; Thompson, R. L.; and Lubick, R. J.: Summary Report of Space Transportation and Destination Considerations for Extraterrestrial Disposal of Radioactive Waste. NASA TM X-68211, 1973.

2. Hyland, R. E.; Wohl, M. L.; Thompson, R. L.; and Finnegan, P. M.: Study of Extraterrestrial Disposal of Radioactive Wastes. Part II: Preliminary Feasibility Screening Study of Extraterrestrial Disposal of Radioactive Wastes in Concentrations, Matrix Materials, and Containers Designed for Storage on Earth. NASA TM X-68147, 1972.

3. Hyland, R. E.; Wohl, M. L.; and Finnegan, P. M.: Study of Extraterrestrial Disposal of Radioactive Wastes. Part III: Preliminary Feasibility Screening Study of Space Disposal of the Actinide Radioactive Wastes with 1 Percent and 0.1 Percent Fission Product Contamination. NASA TM X-68216, 1973.

4. Space Flight Handbooks. Orbital Flight Handbook, Volume 1, Part 1 - Basic Techniques and Data. NASA SP-33, 1963.

5. Clarke, V. C. Jr.; Bollman, W. E.; Feitis, P. H.; Roth, R. Y.: Design Parameters for Ballistic Interplanetary Trajectories. Part II: One-Way Transfers to Mercury and Jupiter. JPL-TR-32-77, Jet Propulsion Lab. (NASA CR-70154), 1966.

6. Zimmerman, Arthur V.: Performance of Recoverable Single and Multiple Space Tugs for Missions Beyond Earth Escape. NASA TM X-68136, 1972.

7. Space Shuttle Baseline Accommodations for Payloads. NASA TM X-68642, 1972.

TABLE 1.— POTENTIAL DESTINATIONS FOR NUCLEAR WASTE DISPOSAL IN SPACE

High Earth Orbit

Solar Orbits via:

- Single burn beyond Earth escape
- Hohman transfer to circular solar orbit (near Earth)
- Venus or Mars swing-by

Solar System Escape:

- Direct
- Via Jupiter swing-by

Solar Impact:

- Direct
- Via Jupiter swing-by

TABLE 2.—CHARACTERISTICS OF MINIMUM ENERGY VENUS SWING-BY TO SOLAR ORBIT TRAJECTORIES

Calendar date of Earth departure	Earth-to-Venus flight time, days	Flight time, Venus encounter to perihelion, days	ΔV to depart Earth, km/sec (ft/sec)	Post encounter perihelion and aphelion, A.U.	Flyby altitude at Venus, Venus radii
4/3/80	172	113	3.603 (11,830)	0.517-0.744	1.0
11/7/81	156	112	3.562 (11,690)	0.528-0.744	1.0
5/21/83	160	107	3.501 (11,490)	0.573-0.725	1.0
1/8/85	127	71	3.565 (11,700)	0.543-0.749	1.0
8/15/86	114	67	3.595 (11,800)	0.516-0.748	1.0
4/2/88	171	112	3.606 (11,830)	0.539-0.737	1.0
11/4/89	157	106	3.566 (11,700)	0.548-0.732	1.0

TABLE 3.—CHARACTERISTICS OF MINIMUM ENERGY MARS SWING-BY TO SOLAR ORBIT TRAJECTORIES

Calendar date of Earth departure	Earth-to-Mars flight time, days	Flight time, Mars encounter to perihelion, days	ΔV to depart Earth, km/sec (ft/sec)	Post encounter perihelion and aphelion, A.U.	Flyby altitude at Mars, Mars radii
11/13/79	260	281	3.676 (12,060)	1.108-1.556	2.871
12/20/81	215	337	3.636 (11,930)	1.098-1.546	4.302
2/8/84	185	356	3.628 (11,900)	1.097-1.506	3.418
4/11/86	185	356	3.613 (11,850	1.091-1.467	4.008
7/8/88	195	388	3.743 (12,280)	1.098-1.624	7.839
9/4/90	380	198	3.868 (12,690)	1.103-1.755	7.839

TABLE 4. — ΔV REQUIRED FROM LOW EARTH ORBIT FOR 50-DAY LAUNCH OPPORTUNITY TO VENUS

Year	Velocity, km/sec (ft/sec)
1980	3.694 (12,120)
1981	3.691 (12,110)
1983	3.691 (12,110)
1984-85	3.716 (12,190)
1986	3.746 (12,290)

TABLE 5. — ΔV REQUIRED FROM LOW EARTH ORBIT FOR 50-DAY LAUNCH OPPORTUNITY TO MARS

Year	Velocity, km/sec (ft/sec)
1979	4.191 (13,750)
1981-82	3.898 (12,790)
1983-84	3.837 (12,590)
1986	3.652 (11,980)
1988	3.898 (12,790)
1990	4.322 (14,180)

TABLE 6.— SUMMARY OF DESTINATIONS

Destination	ΔV-km/sec (ft/sec)	Advantages	Disadvantages
High Earth orbit	4.11 (13,500)	Low ΔV Launch any day Passive waste package Can be retrieved	Long-term container integrity required Orbit lifetime not proven
Solar orbits via: Single burn beyond Earth escape	3.65 (12,000)	Low ΔV Launch any day Passive waste package	Earth reencounter possible (may not be able to prove otherwise) Abort gap past Earth-escape velocity
Hohman transfer to circular solar orbit (0.90 AU)	4.11 (13,500)	Low ΔV Launch any day	Orbit stability not proven Requires space propulsion system Abort gap past Earth-escape velocity
Venus or Mars swing-by	4.11 (13,500)	Low ΔV	Limited launch opportunity Requires midcourse systems Need space propulsion or have possibility of planet encounter Abort gap past Earth-escape velocity
Solar system escape: Direct	8.75 (28,700)	Launch any day Passive waste package Removed from solar system	High ΔV Abort gap past earth-escape velocity
Via Jupiter swing-by	7.01 (23,000)	Removed from solar system	High ΔV Limited launch opportunity Requires midcourse systems Abort gap past Earth-escape velocity
Solar impact: Direct	24.08 (79,000)	Package destroyed Launch any day Passive waste package	Extremely high ΔV Abort gap past Earth-escape velocity
Via Jupiter swing-by	7.62 (25,000)	Package destroyed	High ΔV Limited launch opportunity Requires midcourse systems Abort gap past Earth-escape velocity

TABLE 7.— SPACE TUG PERFORMANCE GROUND RULES

Shuttle 185 km (100 n.mi.) due-East payload capability	29,484 kg (65,000 lb)
Shuttle payload support weight	1,361 kg (3,000 lb)
Shuttle deployed payload capability	28,123 kg (62,000 lb)
Tug hydrogen/oxygen propellant capacity	23,879 kg (52,643 lb)
Tug empty weight	2,883 kg (6,357 lb)
Round-trip geostationary payload	1,361 kg (3,000 lb)
Tug specific impulse	470 sec
Tug mass fraction	.892
Flight performance reserve	2% ΔV

TABLE 8.—LAUNCH VEHICLE PERFORMANCE AND COST SUMMARY FOR HIGH EARTH ORBIT AND SOLAR ORBIT DESTINATIONS

[ΔV = 4.11 km/sec (13,500 ft/sec)]

Launch Vehicle	Payload kg (lb)	Cost 10^6 Dollars	Cost per Kilogram Dollars/kg ($/lb)
Saturn V	32,700 (72,000)	150	4,580 (2,080)
Titan IIIE/Centaur	3,860 (8,500)	19	4,920 (2,240)
Space Shuttle:			
Reusable tug (current size)	4,170 (9,200)	12.25	2,940 (1,330)
Reusable tug (optimum size)	4,670 (10,300)	12.25	2,620 (1,180)
Centaur (current size)	6,490 (14,300)	16.0	2,460 (1,120)
Centaur (optimum size)	8,480 (18,700)	16.3	1,920 (870)

TABLE 9.—LAUNCH VEHICLE PERFORMANCE AND COST SUMMARY FOR DIRECT SOLAR ESCAPE MISSION

[ΔV = 8.75 km/sec (28,700 ft/sec)]

Launch Vehicle	Payload kg (lb)	Cost 10^6 Dollars	Cost per kilogram Dollars/kg ($/lb)
Saturn V/Centaur	7,260 (16,000)	155	21,300 (9,700)
(2,1,1) Shuttle tug config.:[a]			
Without perigee propulsion	2,270 (5,000)	28.75	12,660 (5,750)
With perigee propulsion	3,270 (7,200)	28.75	8,790 (4,000)
(3,1,2) Shuttle tug config.:[b]			
Without perigee propulsion	3,040 (6,700)	41.0	13,490 (6,120)
With perigee propulsion	4,400 (9,700)	41.0	9,320 (4,230)

[a] Two shuttle launches, one expendable tug, one reusable tug.

[b] Three shuttle launches, one expendable tug, two reusable tugs.

TABLE 10.– MULTIPLE SHUTTLE/TUG PERFORMANCE AND COST SUMMARY
FOR HIGH EARTH ORBIT AND SOLAR ORBIT DESTINATIONS
(No gravity losses)
[ΔV = 4.11 km/sec (13,500 ft/sec)]

Launch Vehicle	Payload kg (lb)	Cost 10^6 Dollars	Cost per Kilogram Dollars/kg ($/lb)
Shuttle/tug configurations:			
(2,0,2)	11,700 (25,700)	24.5	2,090 (950)
(2,1,1)	16,000 (35,200)	28.75	1,800 (820)
(3,0,2)	20,000 (44,000)	35.0	1,750 (800)

TABLE 1A.—VECTORS AND CORRESPONDING VACUUM IMPACT CONDITIONS DURING ENTRY AFTER A ONCE-AROUND ABORT

	Case Number						
	1	2	3	4	5	6	7
Initial condition vector:							
Time from entry (min:sec)	8:00	12:00	16:00	20:00	24:00	28:00	29:54
Latitude, deg.	22.5	26.4	28.4	29.4	29.3	29.1	29.0
West longitude, deg.	137.2	123.0	109.3	97.5	88.0	83.5	82.2
Altitude, km (n.mi.)	74.6	71.6	67.1	59.4	51.8	39.0	30.5
	(40.3)	(38.7)	(36.2)	(32.1)	(28.0)	(21.1)	(16.5)
Velocity, km/sec (ft/sec)	7.32	6.86	6.10	5.03	3.35	1.52	0.88
	(24,000)	(22,500)	(20,000)	(16,500)	(11,000)	(5,000)	(2,900)
Flight-path angle, deg.	0.1	-0.1	-0.4	-0.4	-0.7	-3.0	-6.0
Azimuth, deg.	71.0	77.5	82.5	87.5	93.5	95.5	96.4
Vacuum impact conditions:							
Time to impact (min:sec)	5:53.2	4:05.2	2:56.2	2:18.3	1:49.3	1:23.1	1:10.6
Range to impact, deg.	23.1	14.7	9.3	5.9	3.0	1.1	0.6
Range to impact, km (n.mi.)	2,572	1,639	1,033	652	333	122	61
	(1,389)	(885)	(558)	(352)	(180)	(66)	(33)

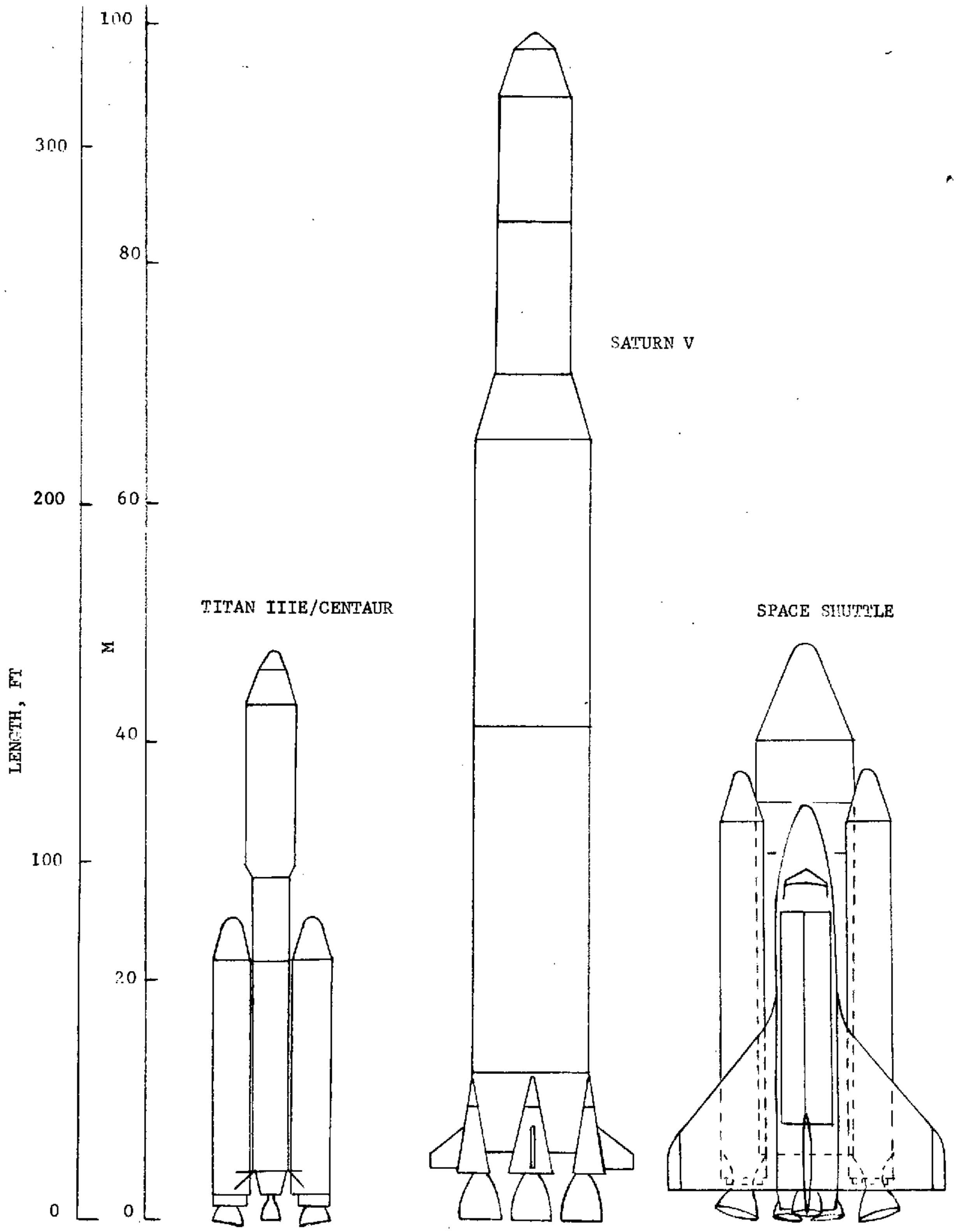


Figure 1.— Space transportation systems.

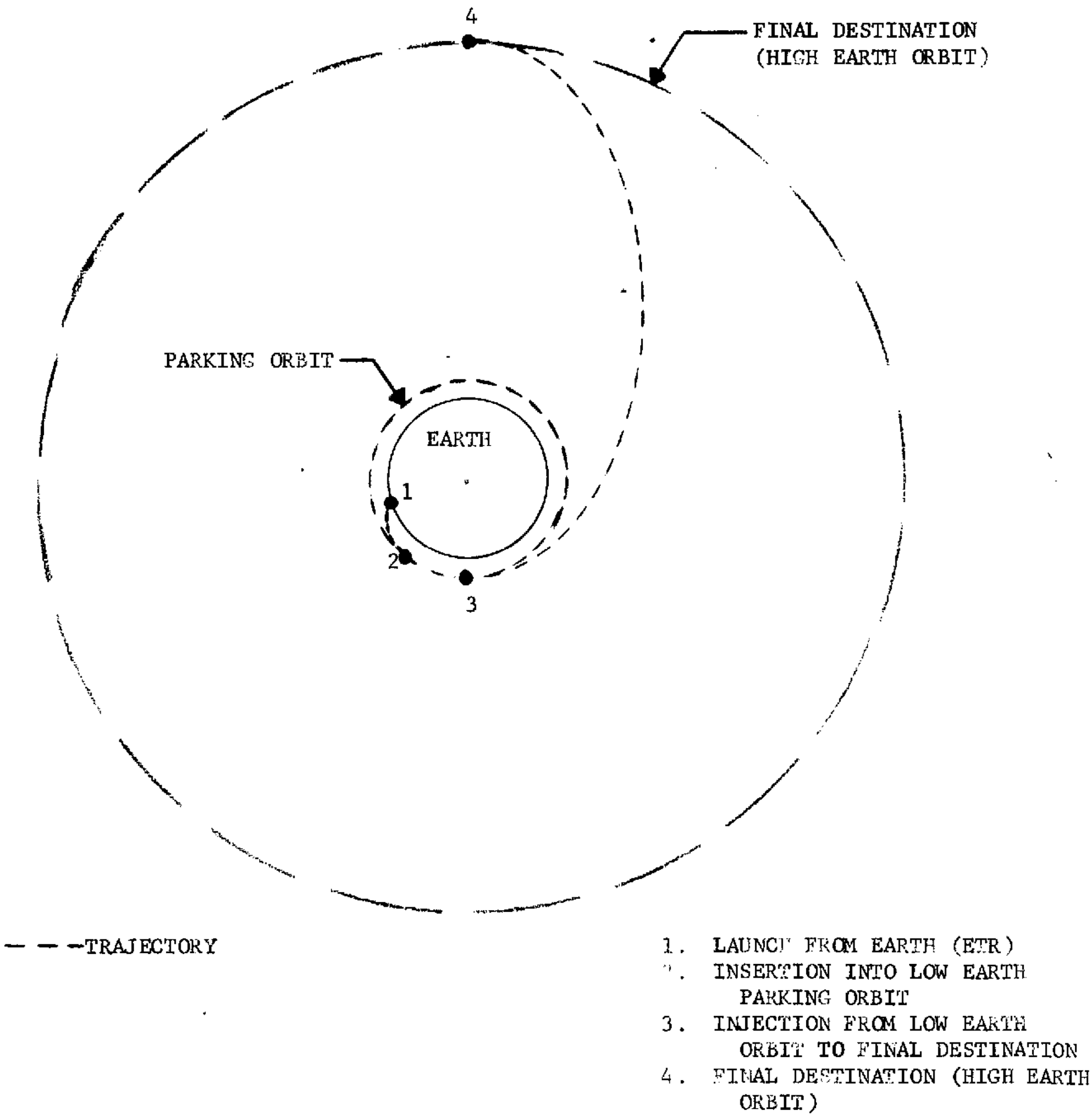


Figure 2.— Launch vehicle trajectory from liftoff at ETR to final destination, high Earth orbit.

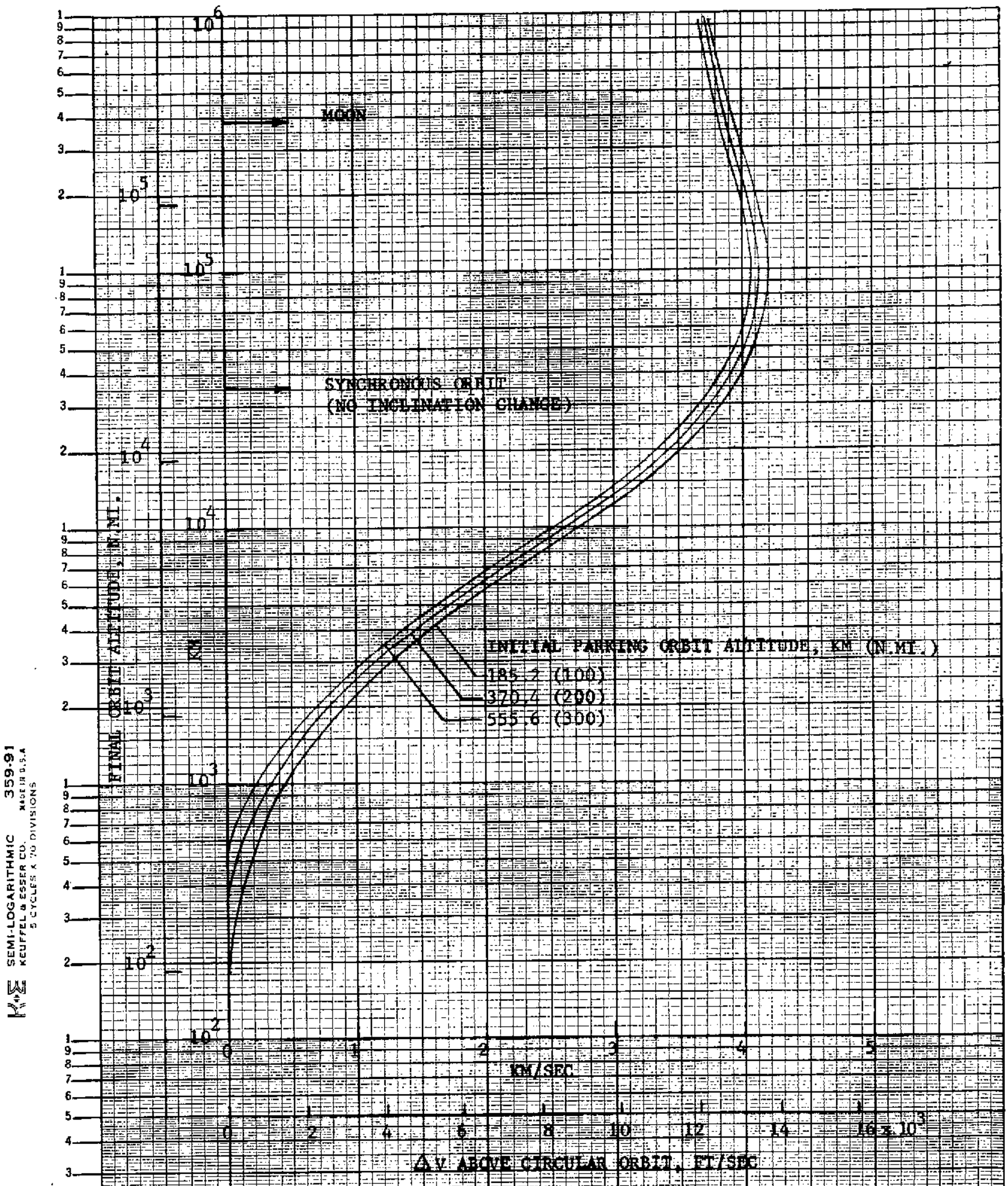


Figure 3.- Total ΔV required for circular Earth orbits starting from several initial Earth parking orbits.

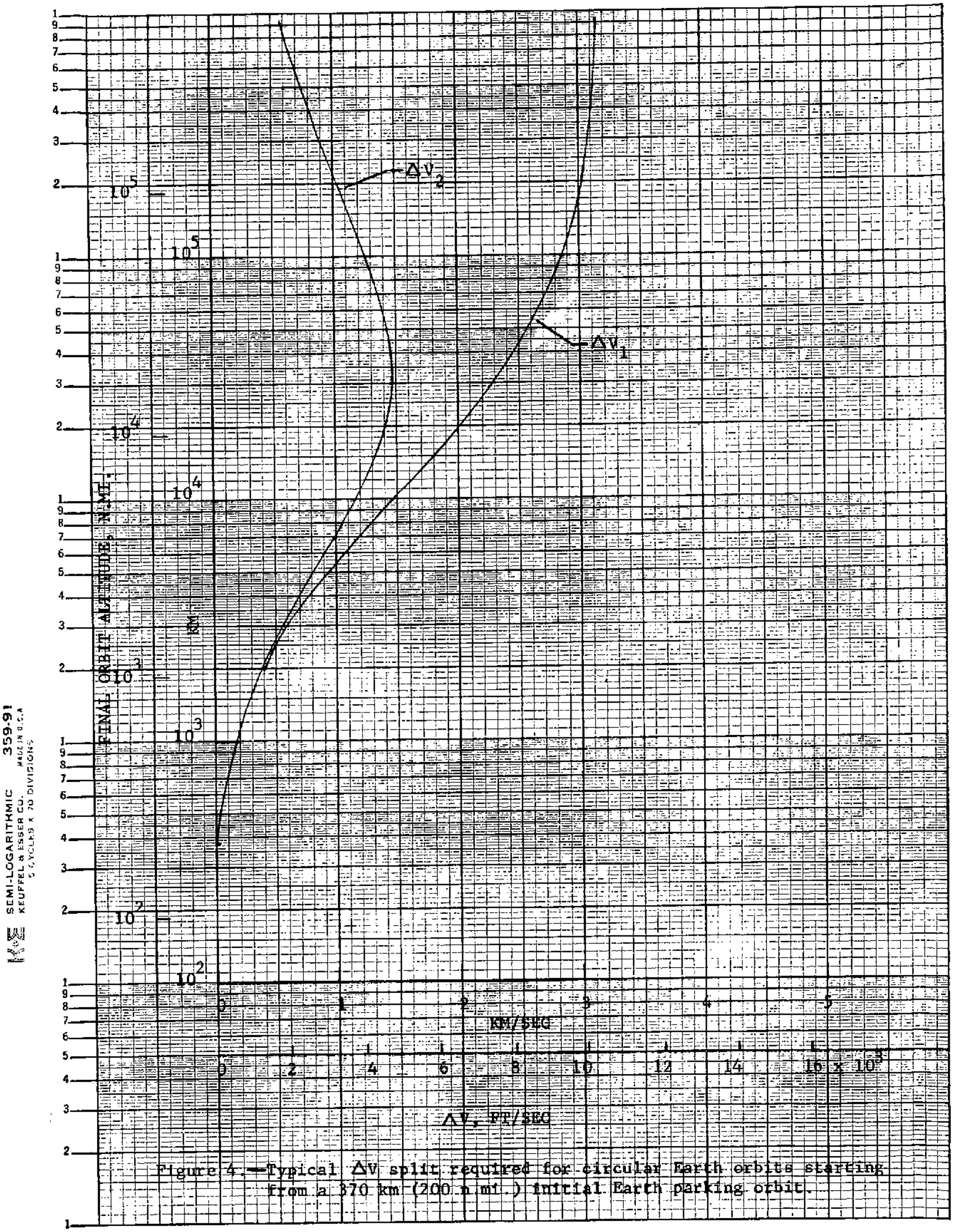


Figure 4.—Typical ΔV split required for circular Earth orbits starting from a 370 km (200 n.mi.) initial Earth parking orbit.

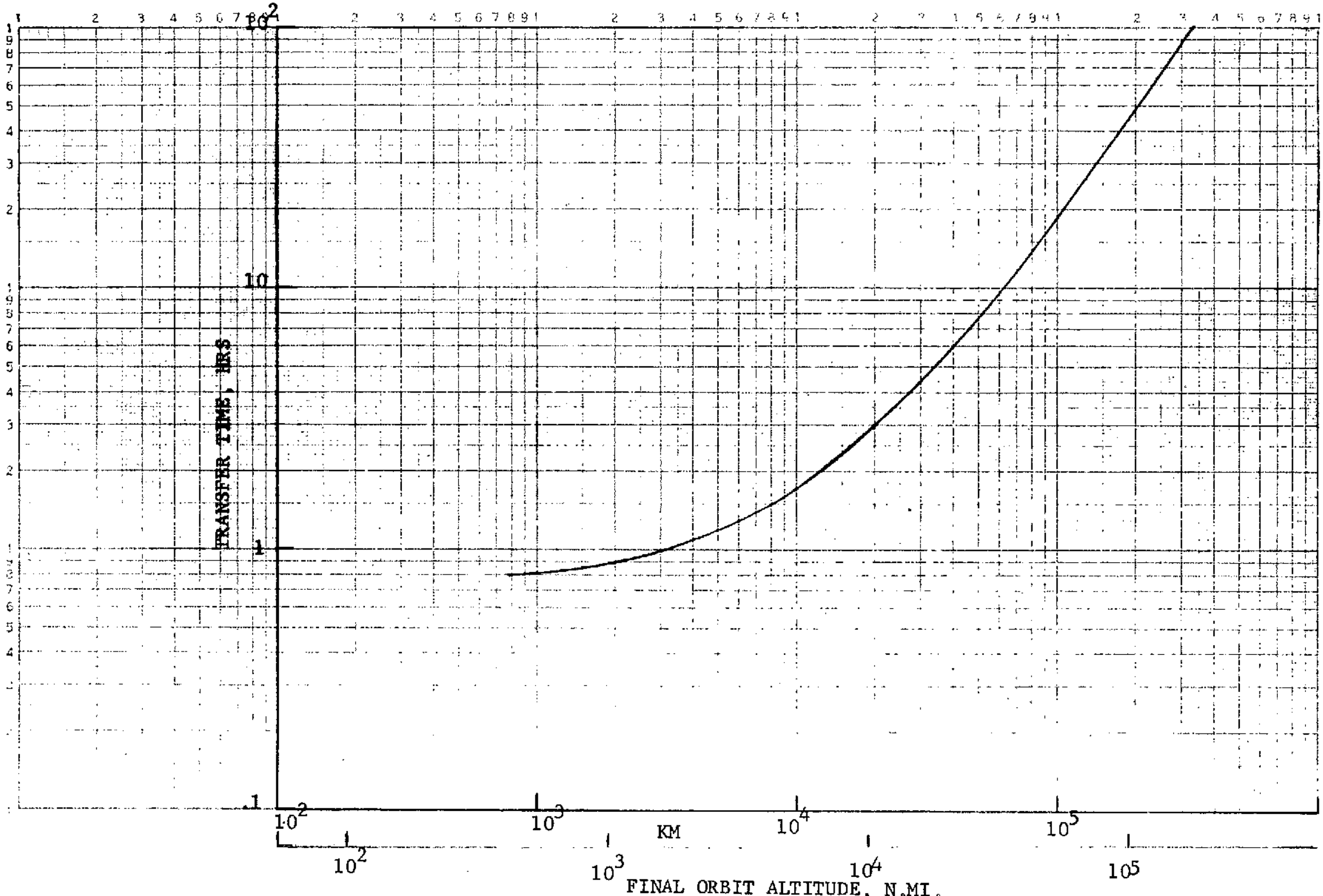


Figure 5.— Typical transfer time required for circular Earth orbits starting from a 370 km (200 n.mi.) initial Earth parking orbit.

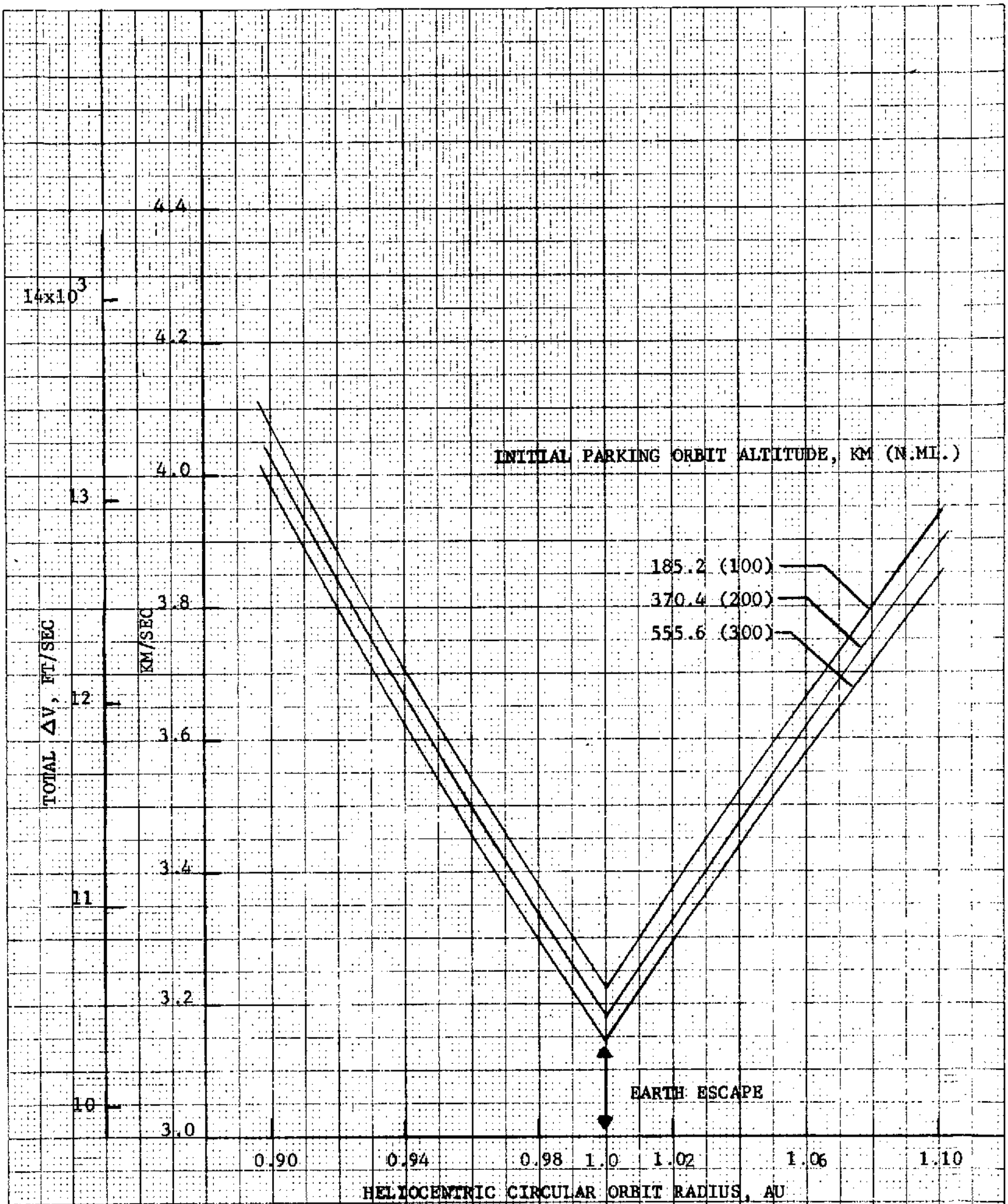


Figure 6.—Total ΔV required for circular heliocentric orbits starting from several initial Earth parking orbits.

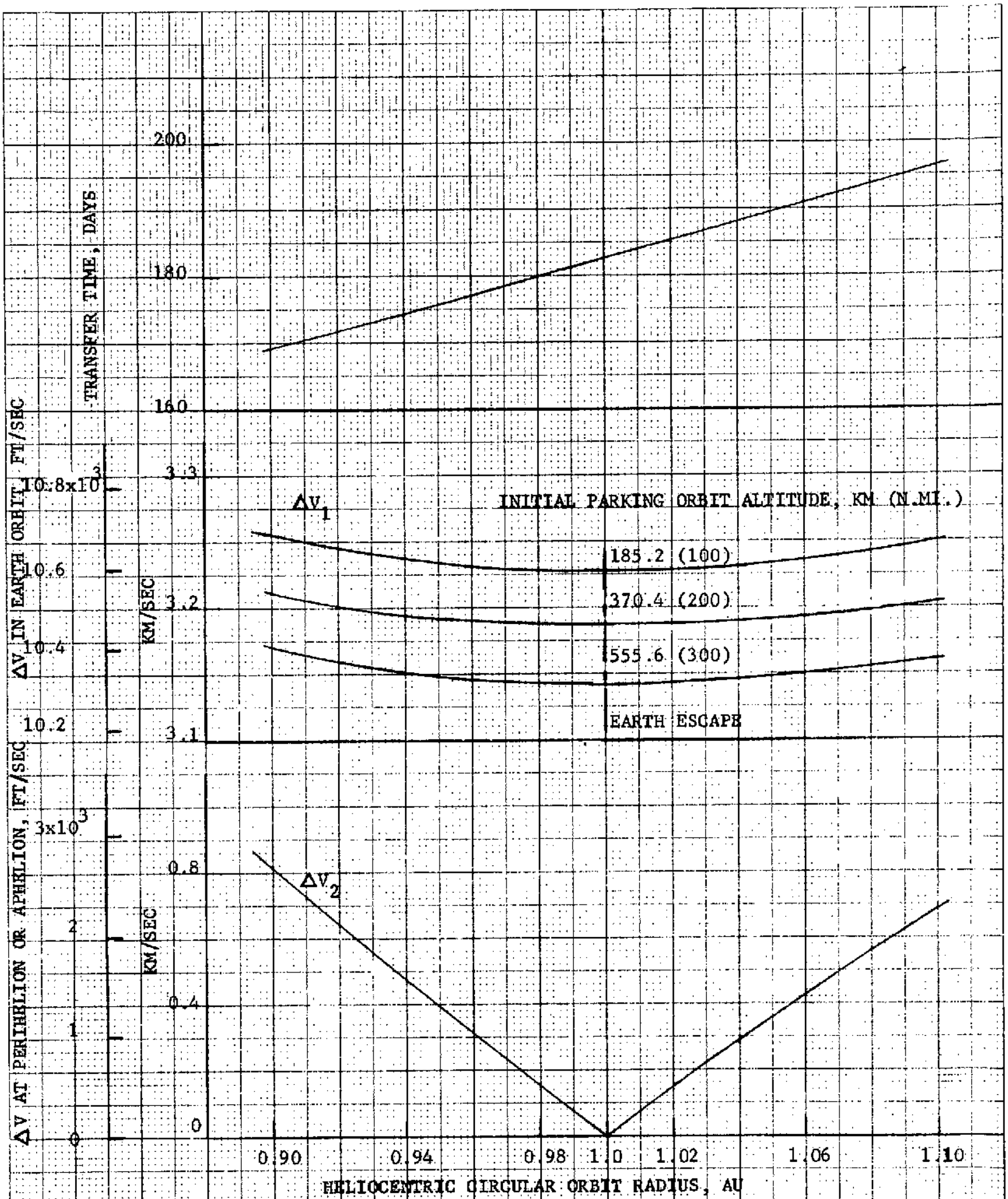
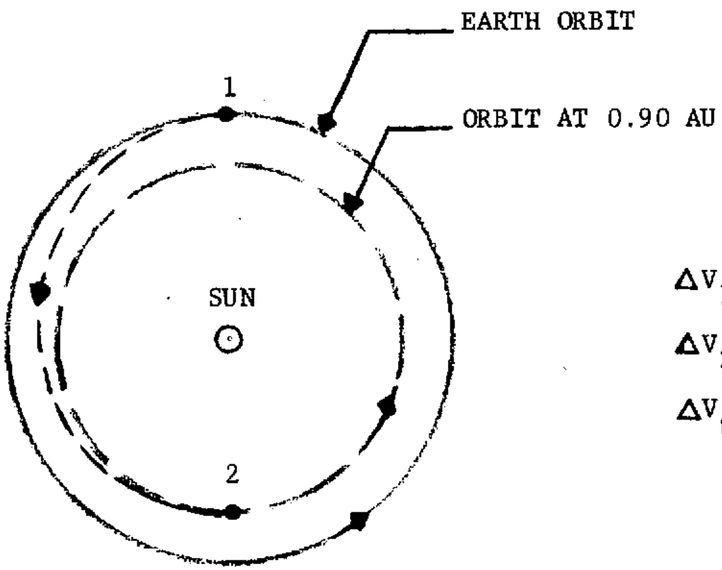


Figure 7. — ΔV split and transfer time required for heliocentric circular orbits starting from several initial Earth parking orbits.



ΔV_1 = 3.26 KM/SEC (10,690 FT/SEC)

ΔV_2 = 0.81 KM/SEC (2,660 FT/SEC)

ΔV_{total} = 4.07 KM/SEC (13,350 FT/SEC)

1 - DEPART EARTH

2 - ARRIVE AT 0.90 AU

Figure 8.—Mission profile for 0.90 AU circular solar orbit.

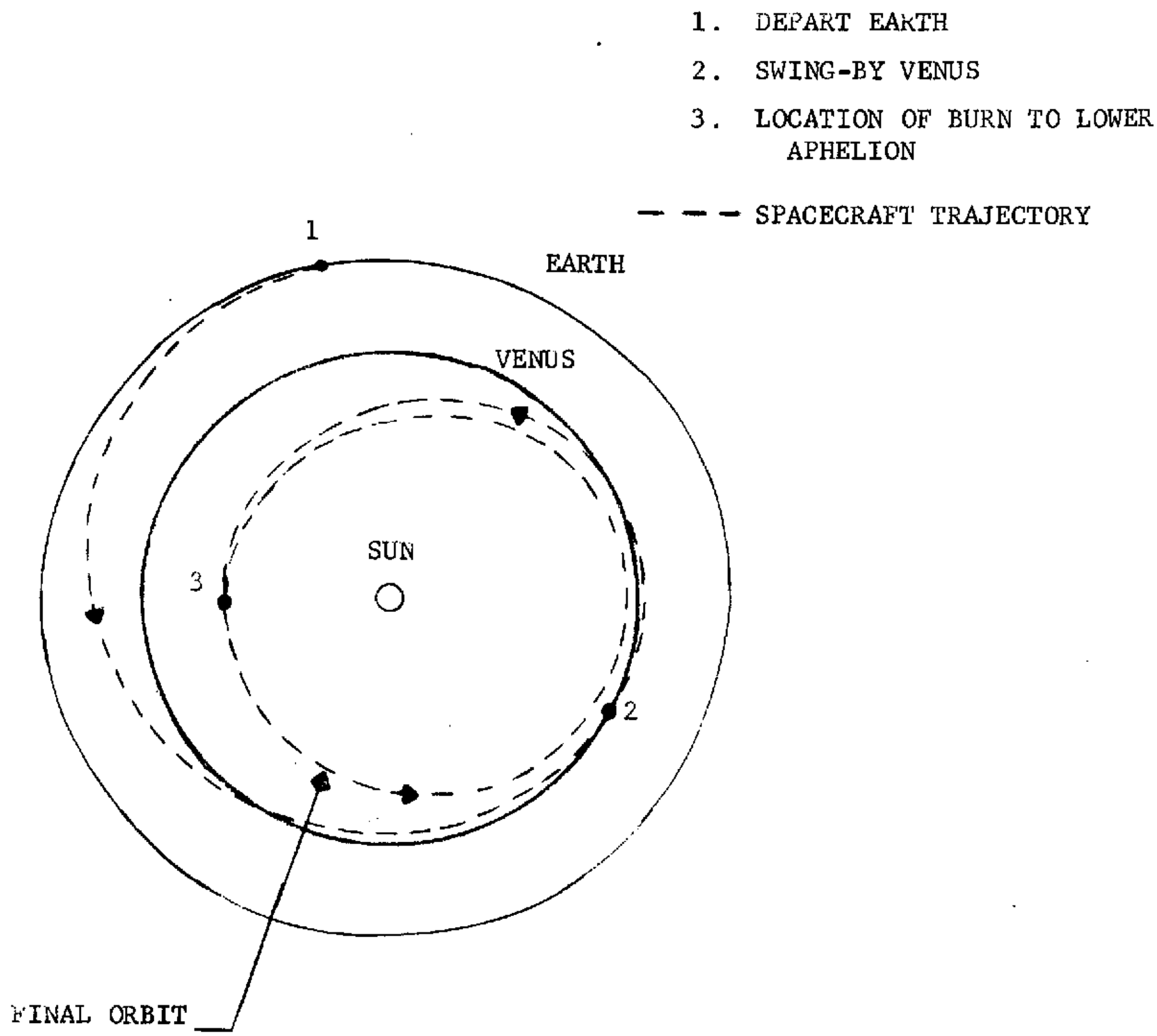


Figure 9.— Mission profile for Earth-Venus swing-by into a solar orbit inside Earth's orbit.

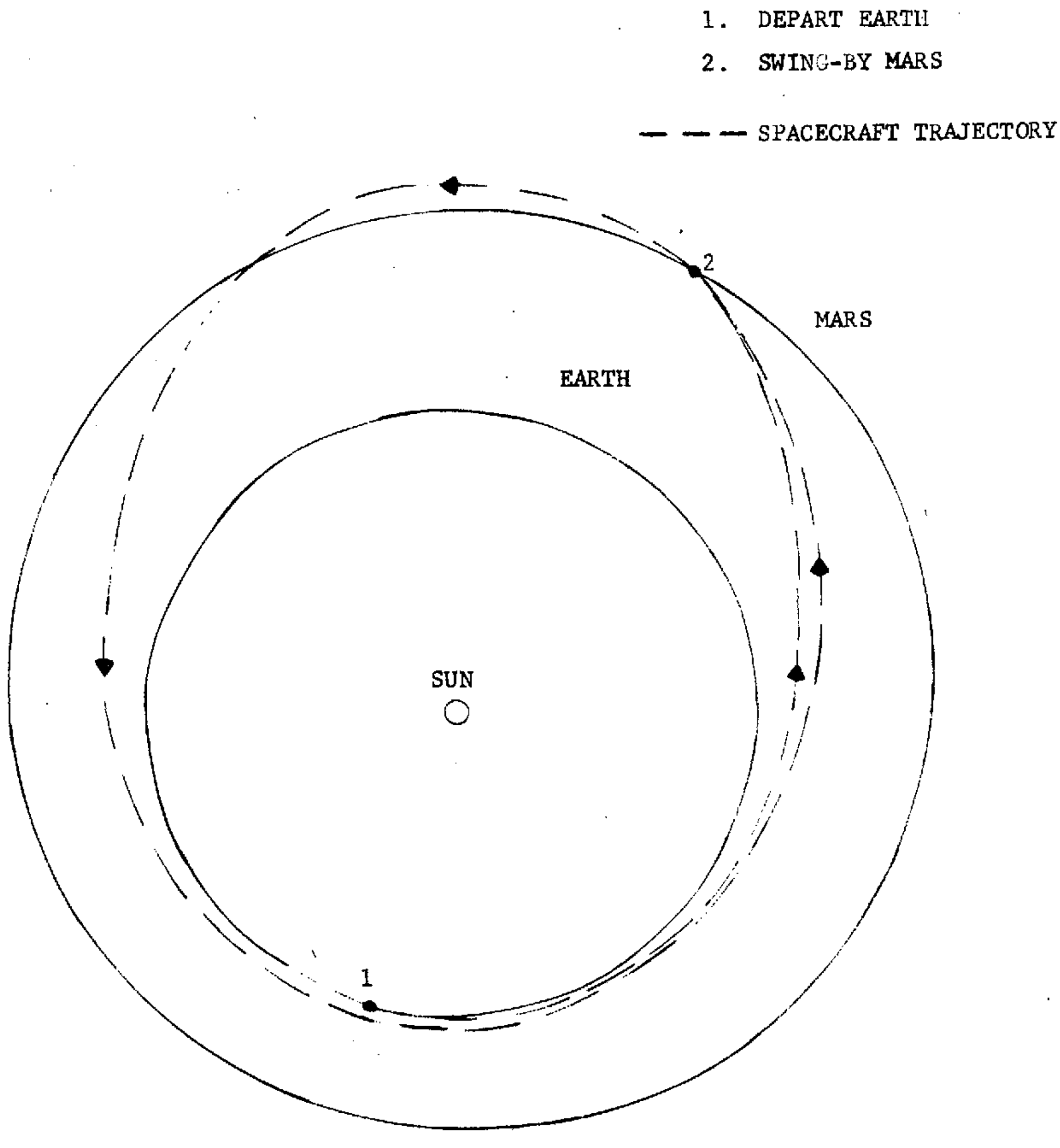


Figure 10.—Mission profile for Earth-Mars swing-by into a solar orbit outside Earth's orbit.

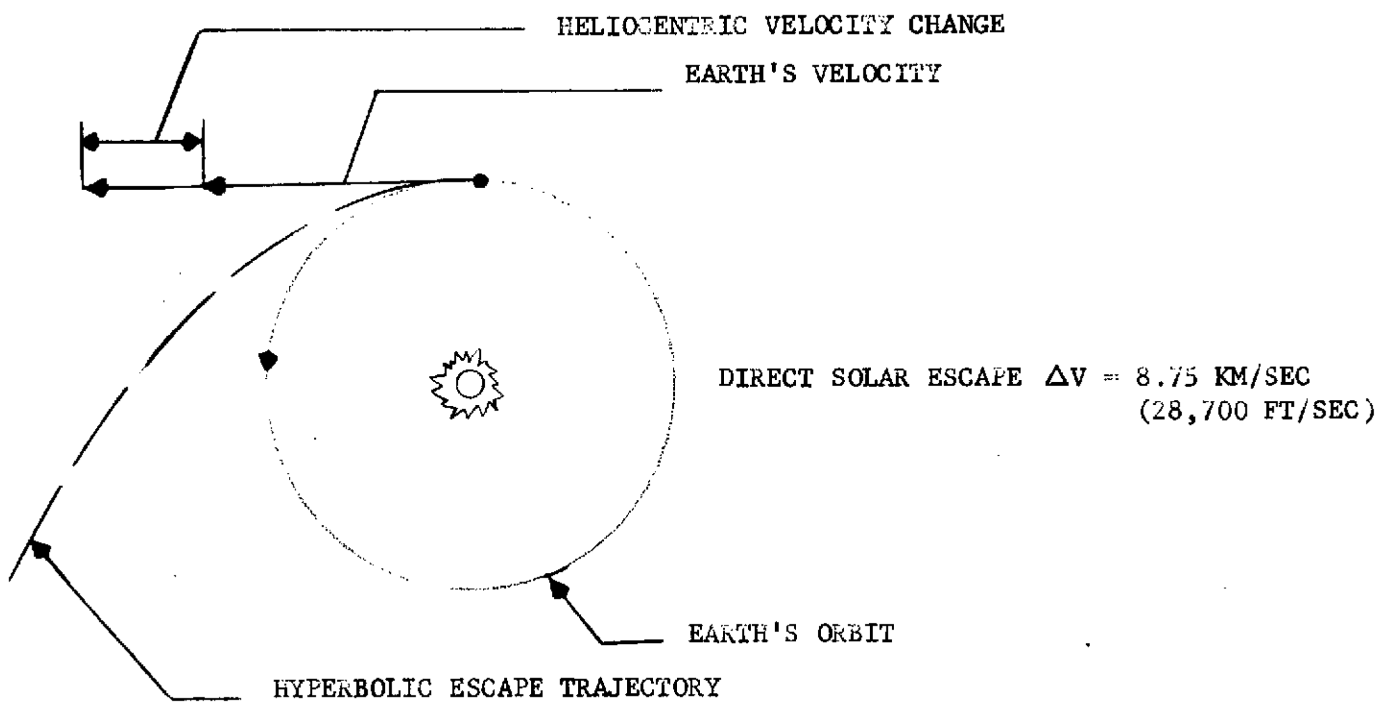
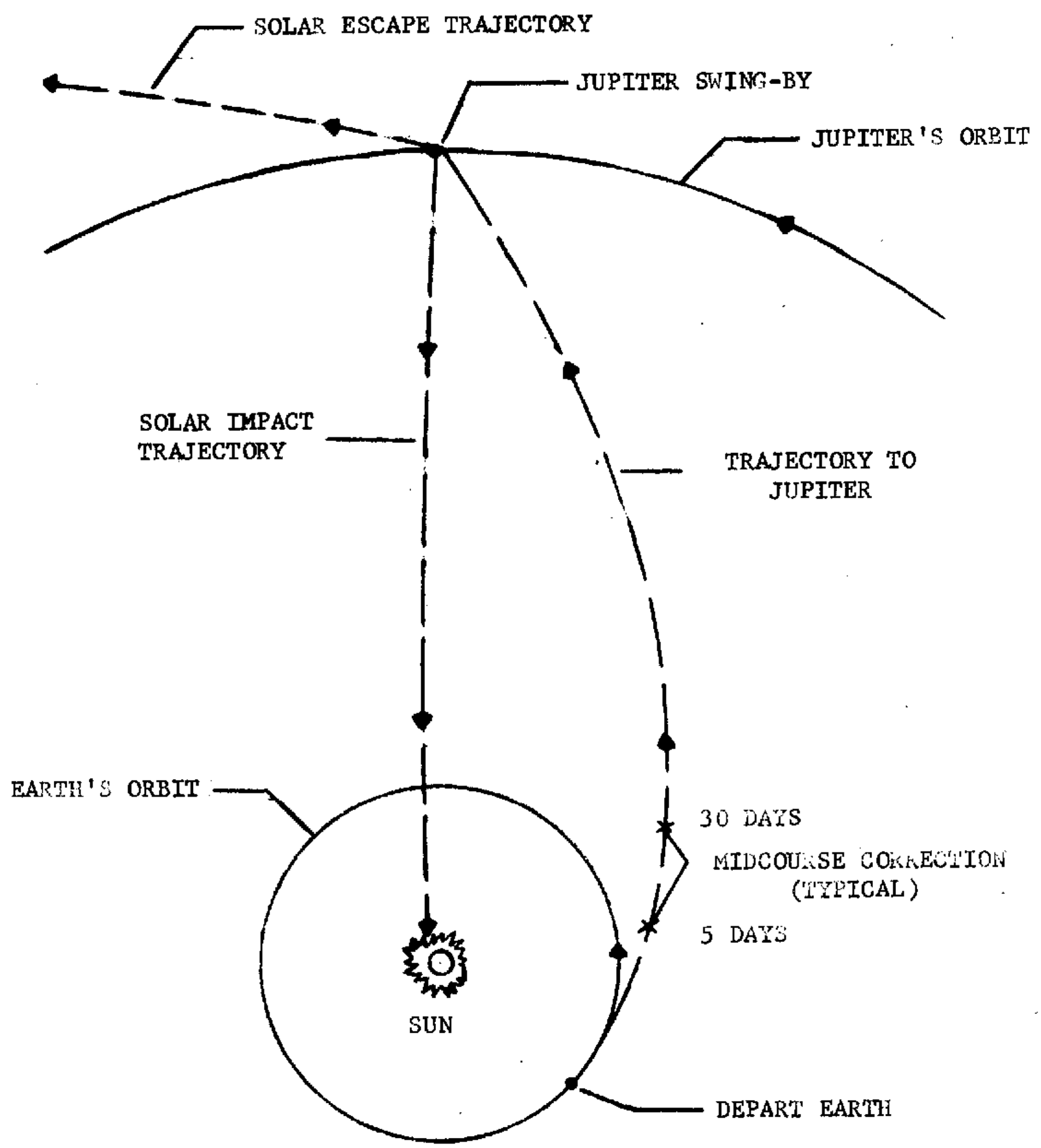


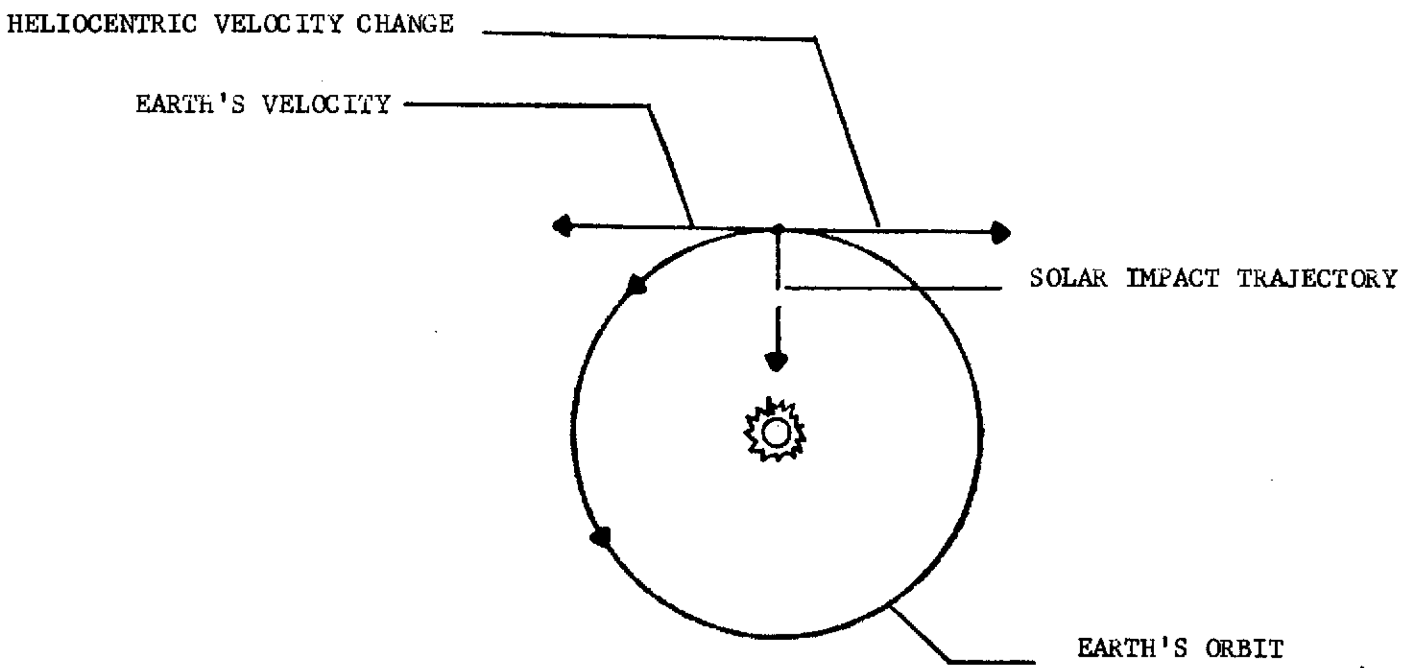
Figure 11.—Mission profile for a direct solar escape.



ΔV COMPARISON - KM/SEC (FT/SEC)

SOLAR ESCAPE:	
DIRECT	8.75 (28,700)
VIA JUPITER	6.71 - 7.32 (22,000 - 24,000)
SOLAR IMPACT:	
DIRECT	21.34 - 24.08 (70,000-79,000)
VIA JUPITER	7.62 (25,000)

Figure 12. Mission profile for a Jupiter swing-by mode resulting in solar escape or solar impact.



DIRECT SOLAR IMPACT ΔV = 21.34 - 24.08 KM/SEC (70,000 - 79,000 FT/SEC)

Figure 13.— Mission profile for a direct solar impact.

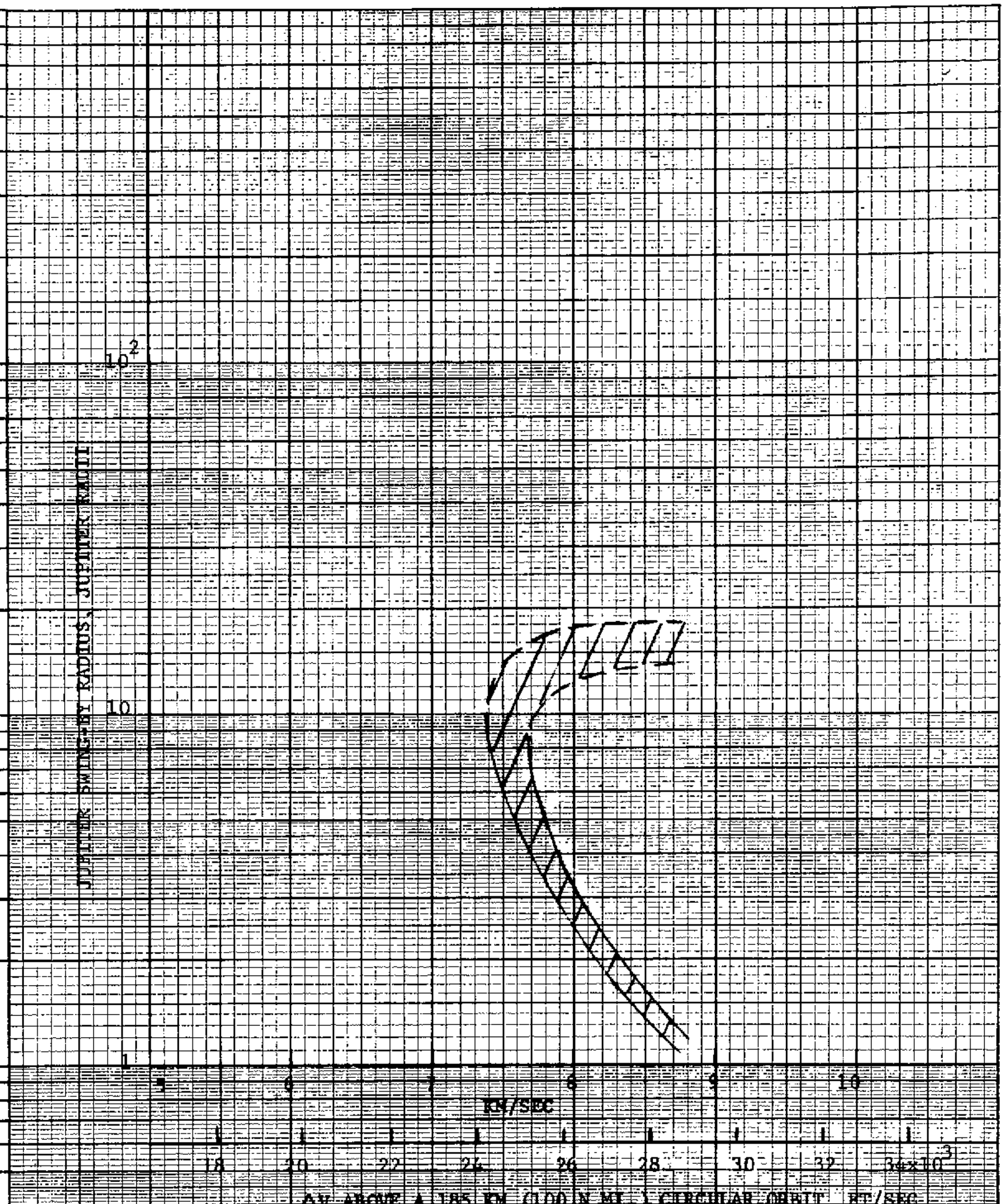


Figure 14.—Proximity to Jupiter required for solar impact as a function of ΔV above a 185 KM (100 n.mi.) circular Earth parking orbit.

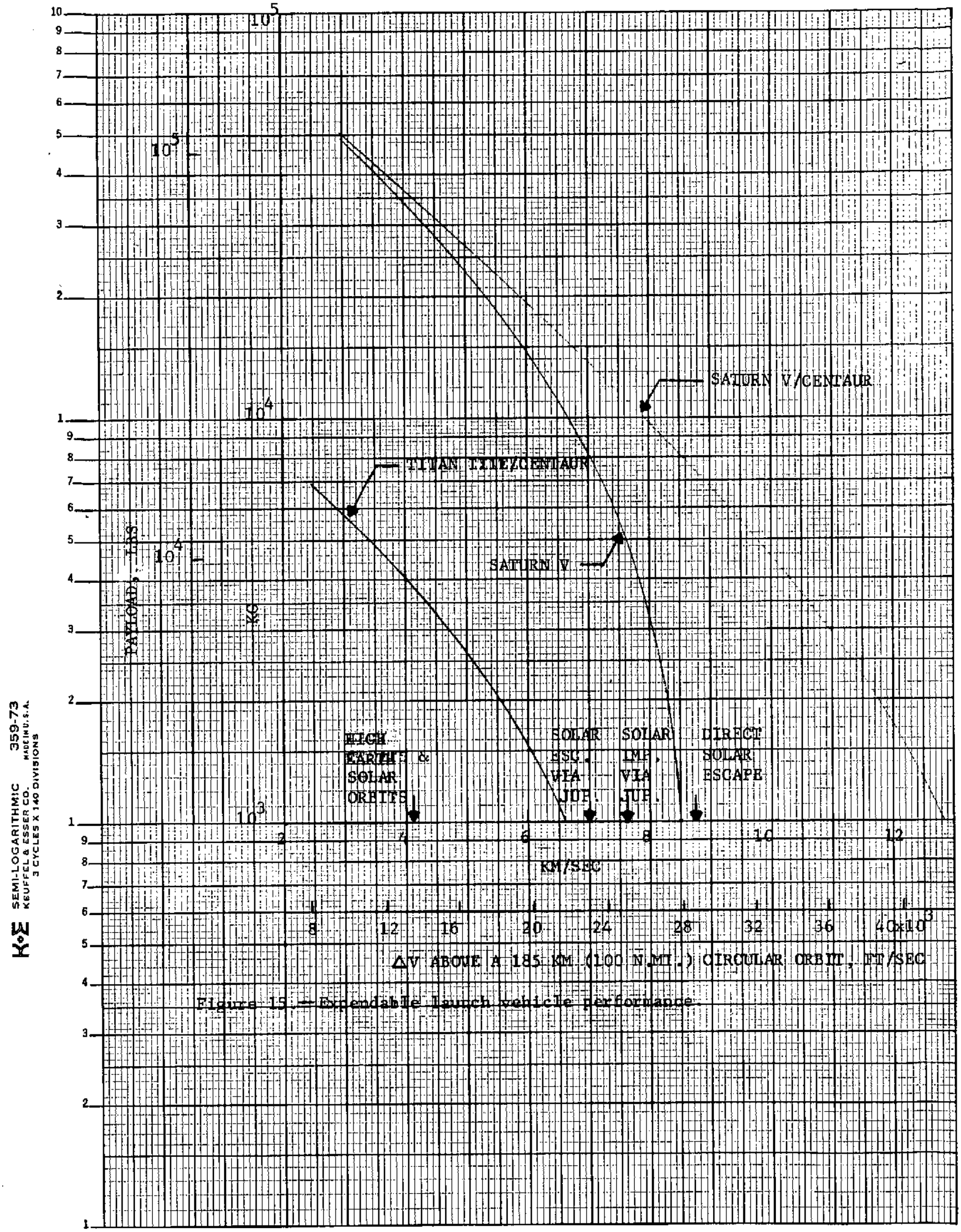


Figure 15.— Expendable launch vehicle performance

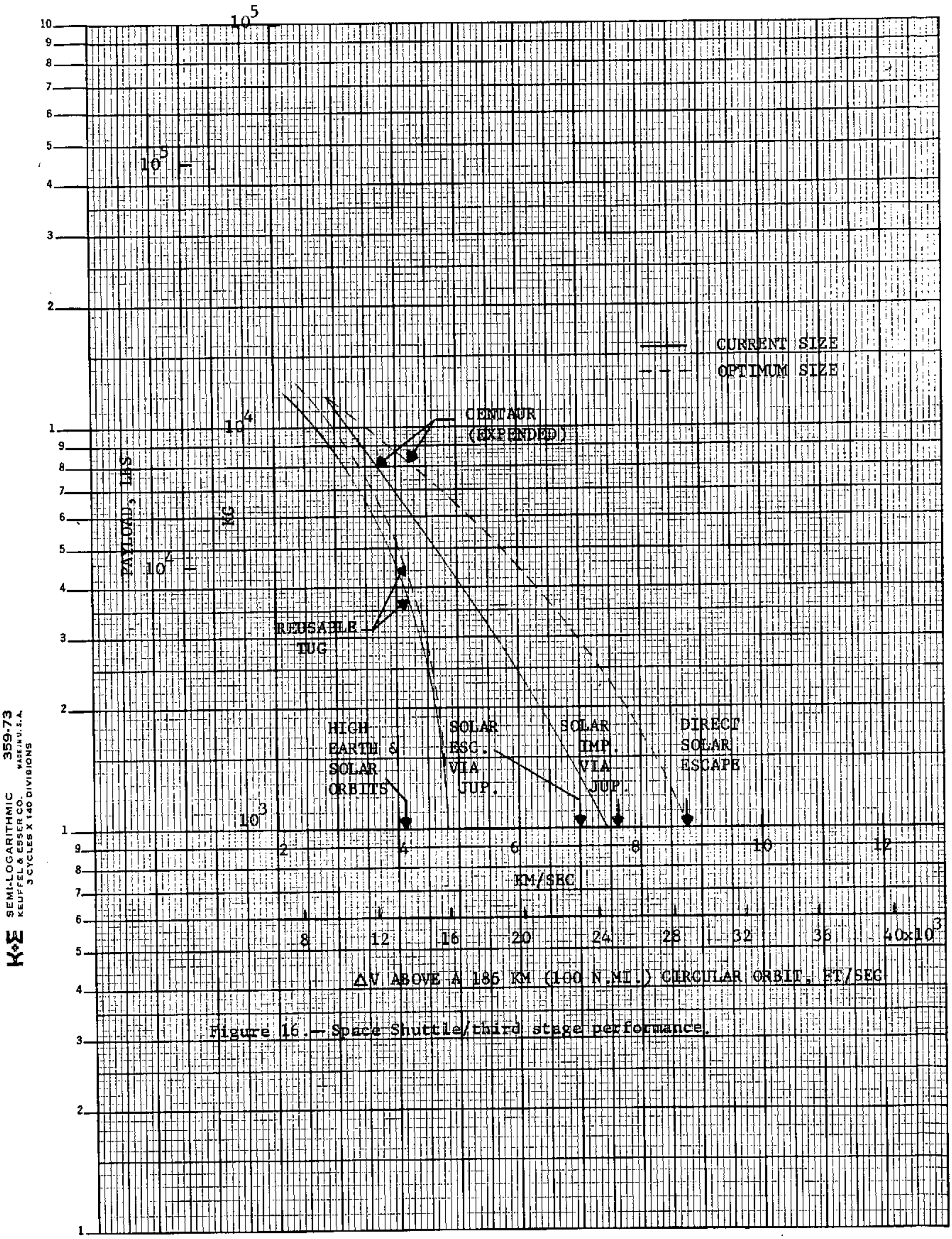


Figure 16. — Space Shuttle/third stage performance.

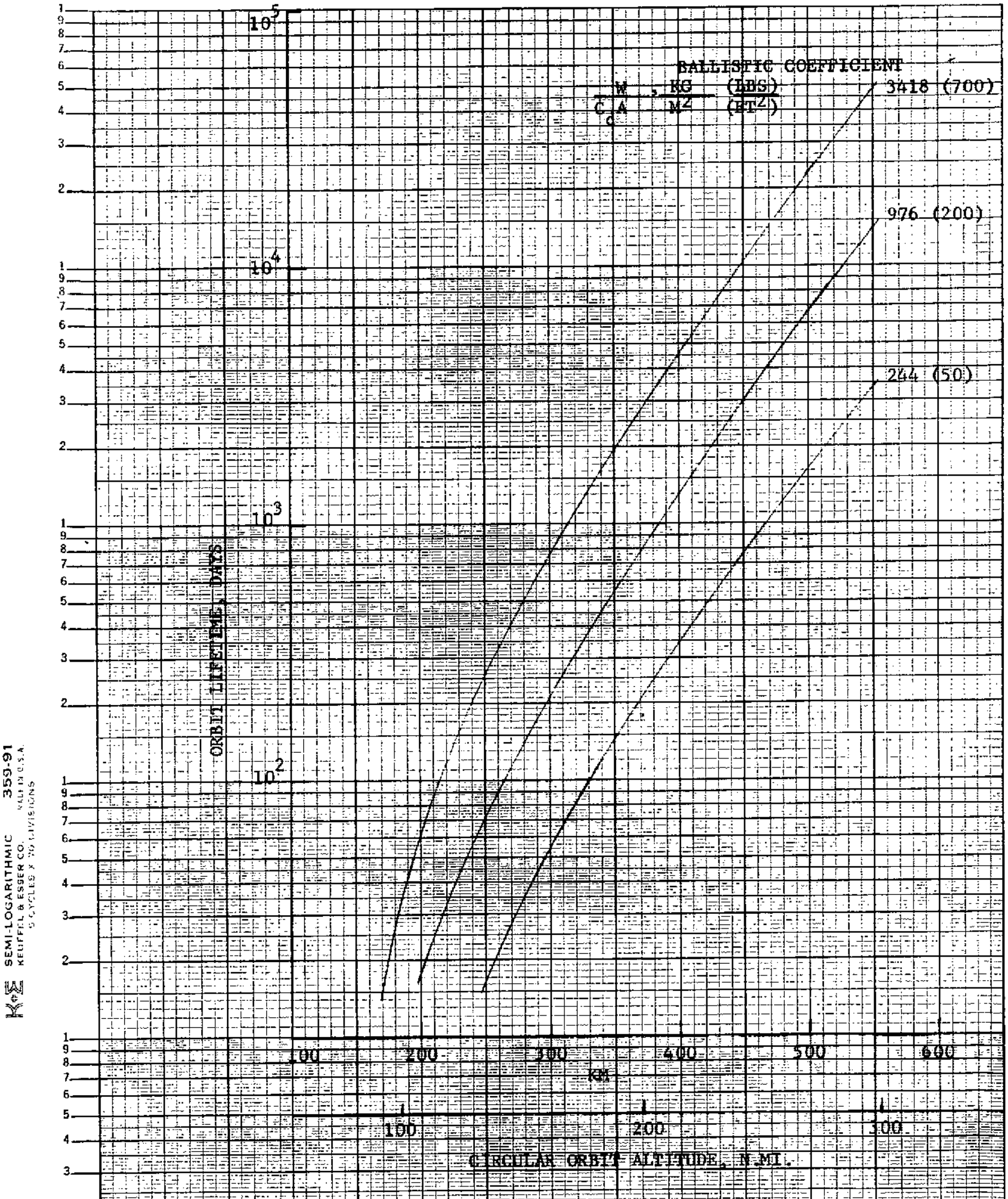


Figure 17.— Circular orbit lifetimes of payloads with several ballistic coefficients; W - weight; C_d - drag coefficient; A - frontal area. Assumes free molecular flow.

SQUARE 10 X 10 TO THE HALF INCH AS-1113 .OT

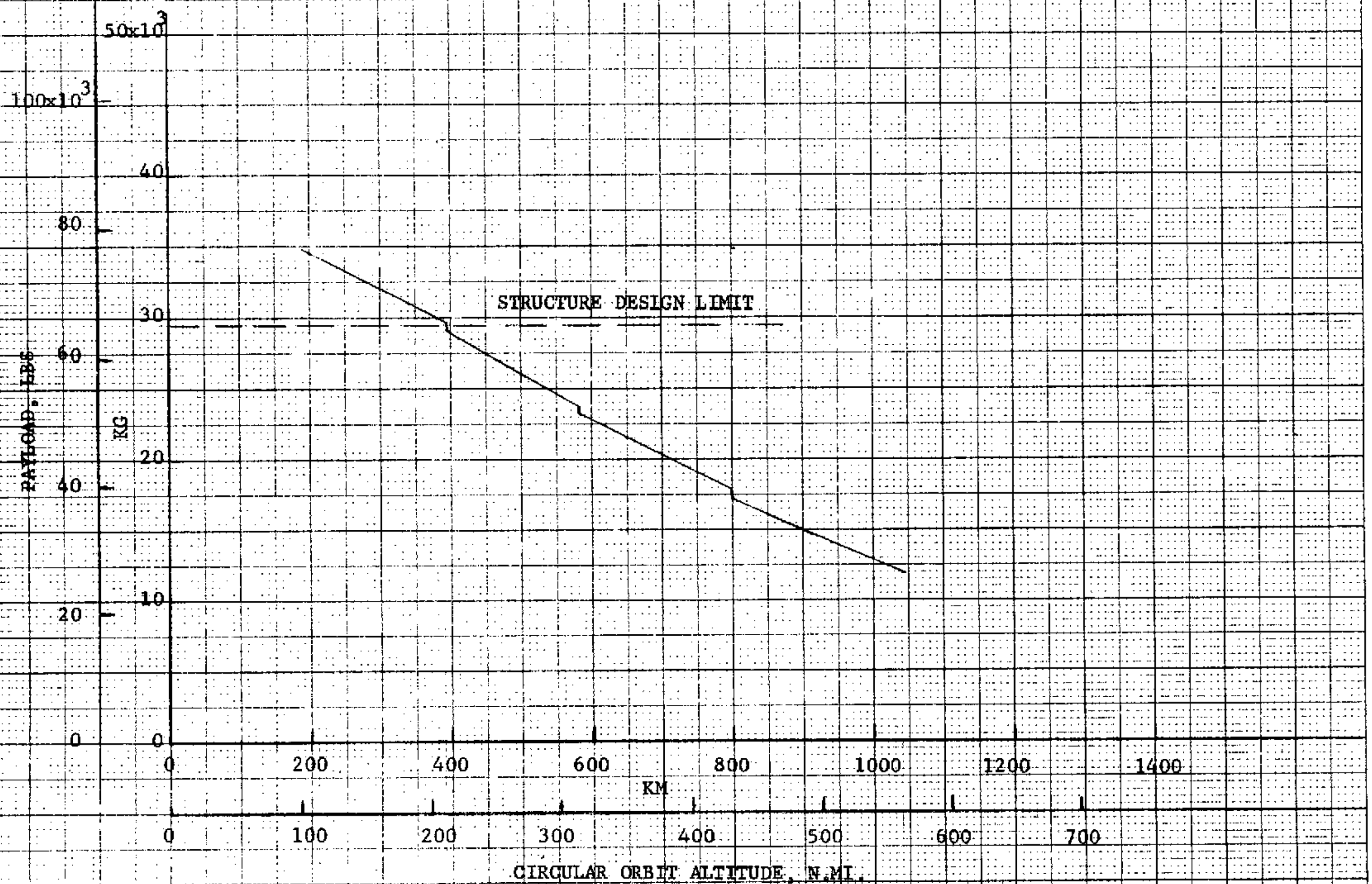


Figure 18.—Space Shuttle payload to circular orbits with rendezvous. Inclination = 28.5 deg.

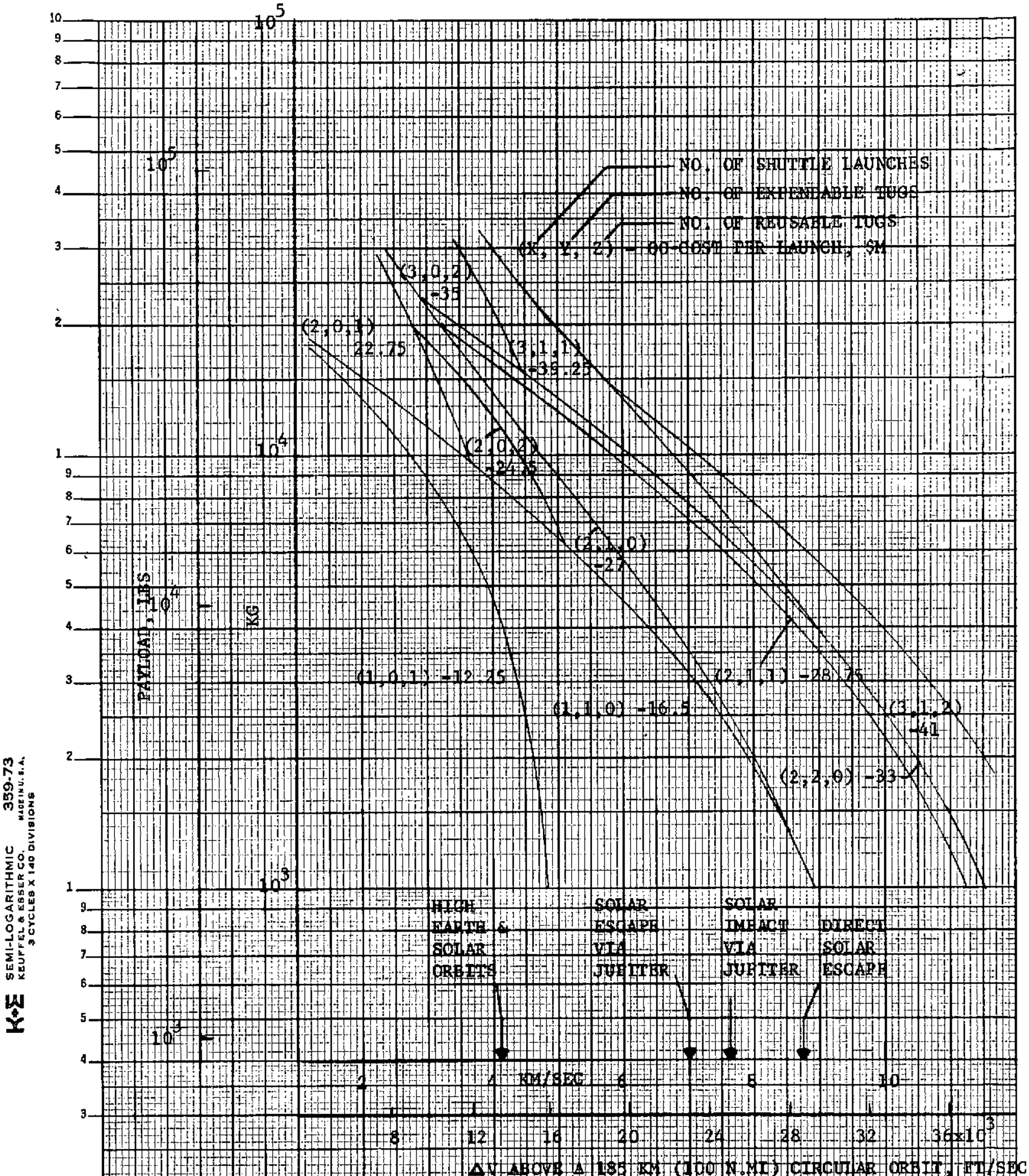


Figure 19. - Ideal performance of multiple space tug vehicles. Cost per shuttle launch, $10.5M; cost per reusable tug, $1.75M; cost per expendable tug, $6.0M.

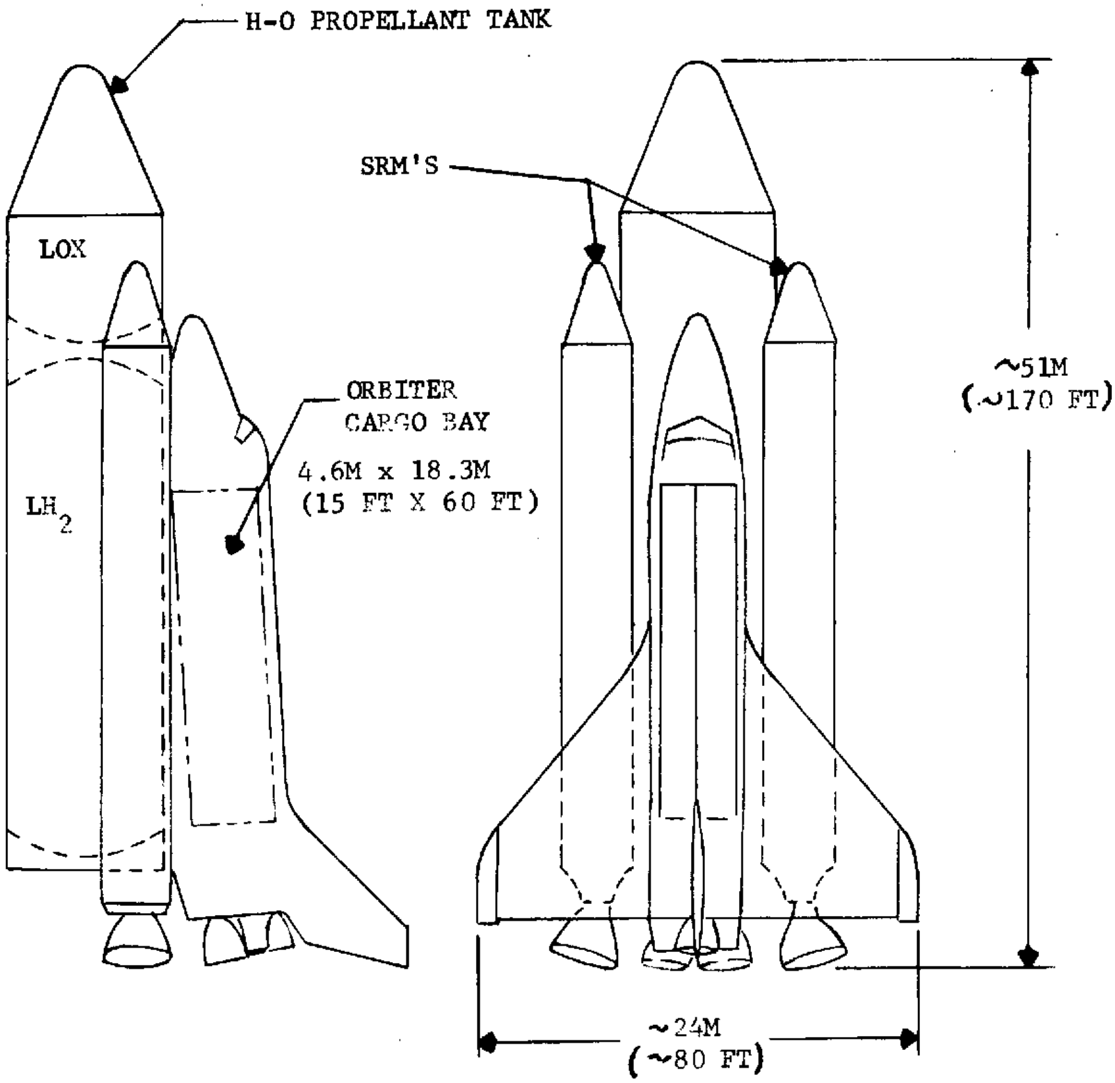


Figure 1A.—Space Shuttle launch vehicle.

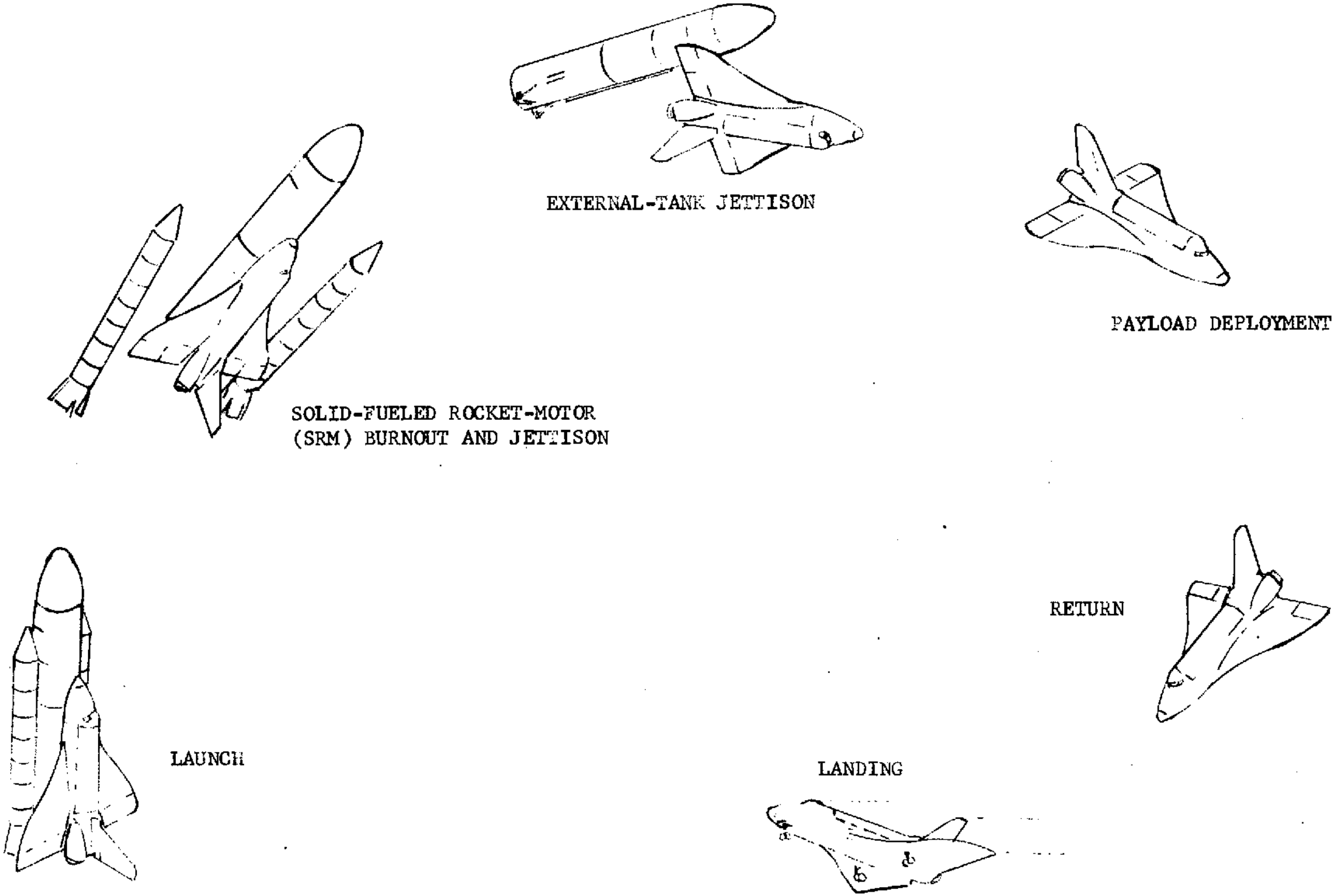


Figure 2A.—Space Shuttle launch-to-landing sequence.

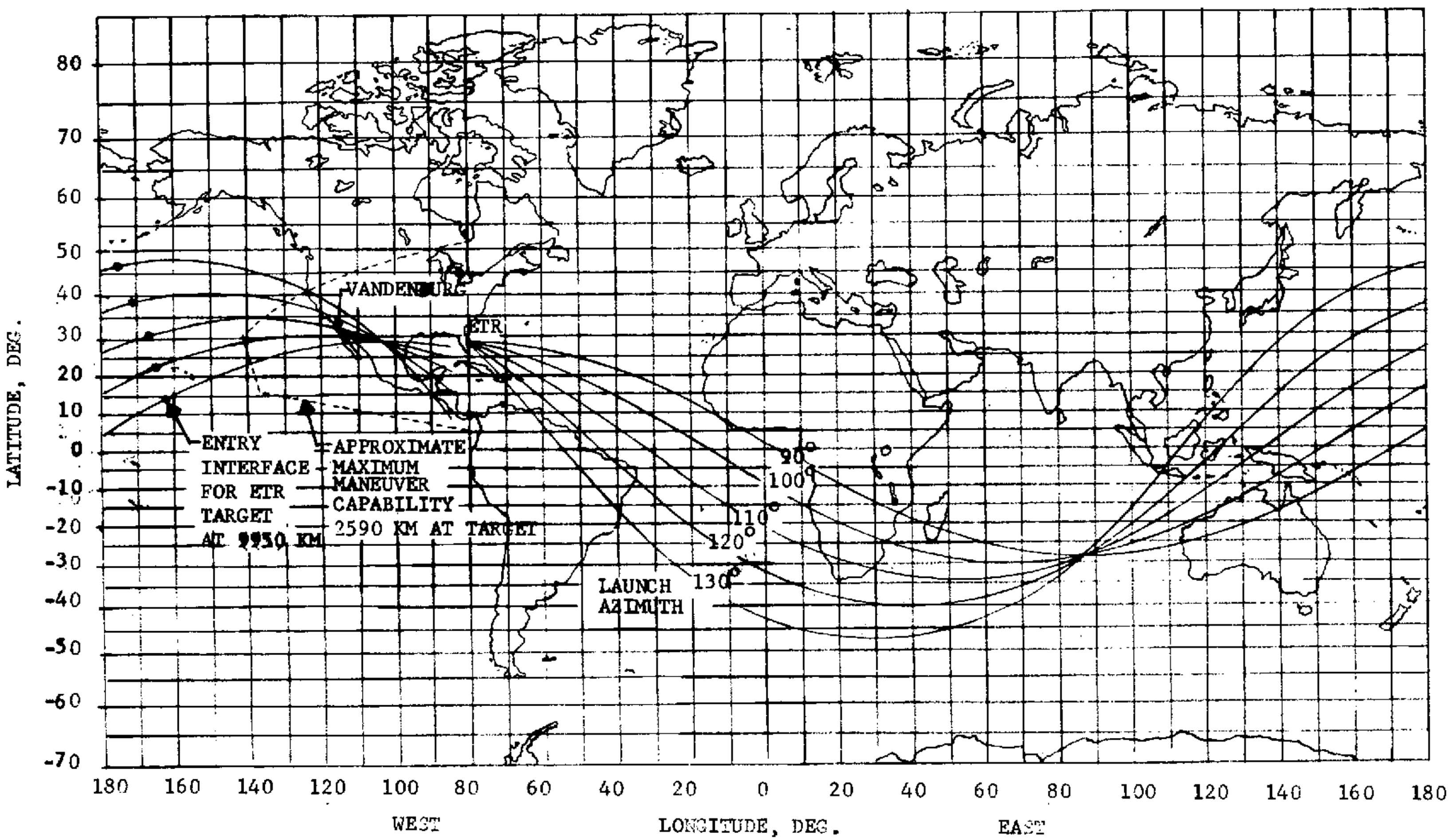


Figure 3A.—Once-around abort ground tracks for five launch azimuths from 90 deg. to 130 deg.

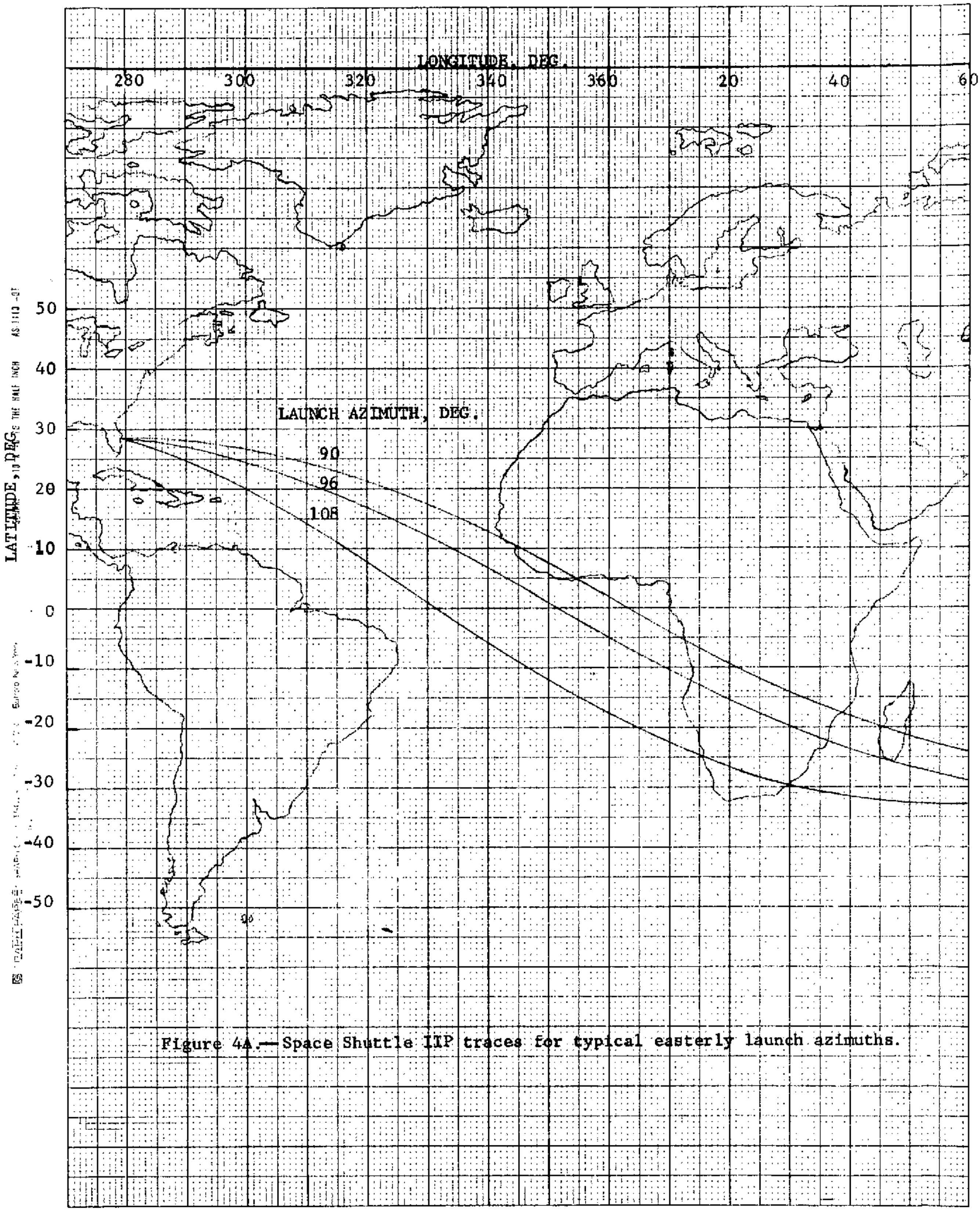


Figure 4A.—Space Shuttle IIP traces for typical easterly launch azimuths.

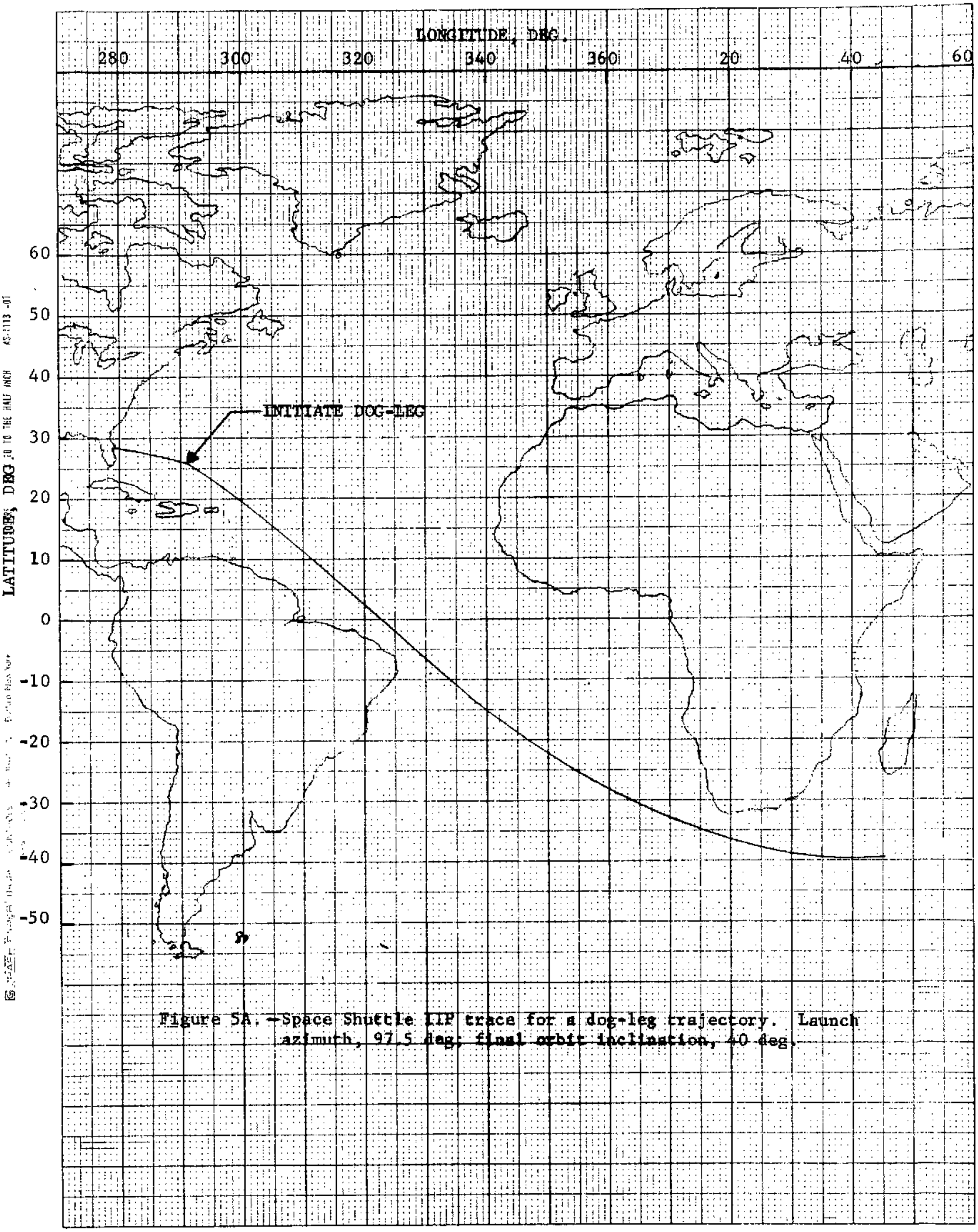


Figure 5A.—Space Shuttle IIP trace for a dog-leg trajectory. Launch azimuth, 97.5 deg; final orbit inclination, 40 deg.

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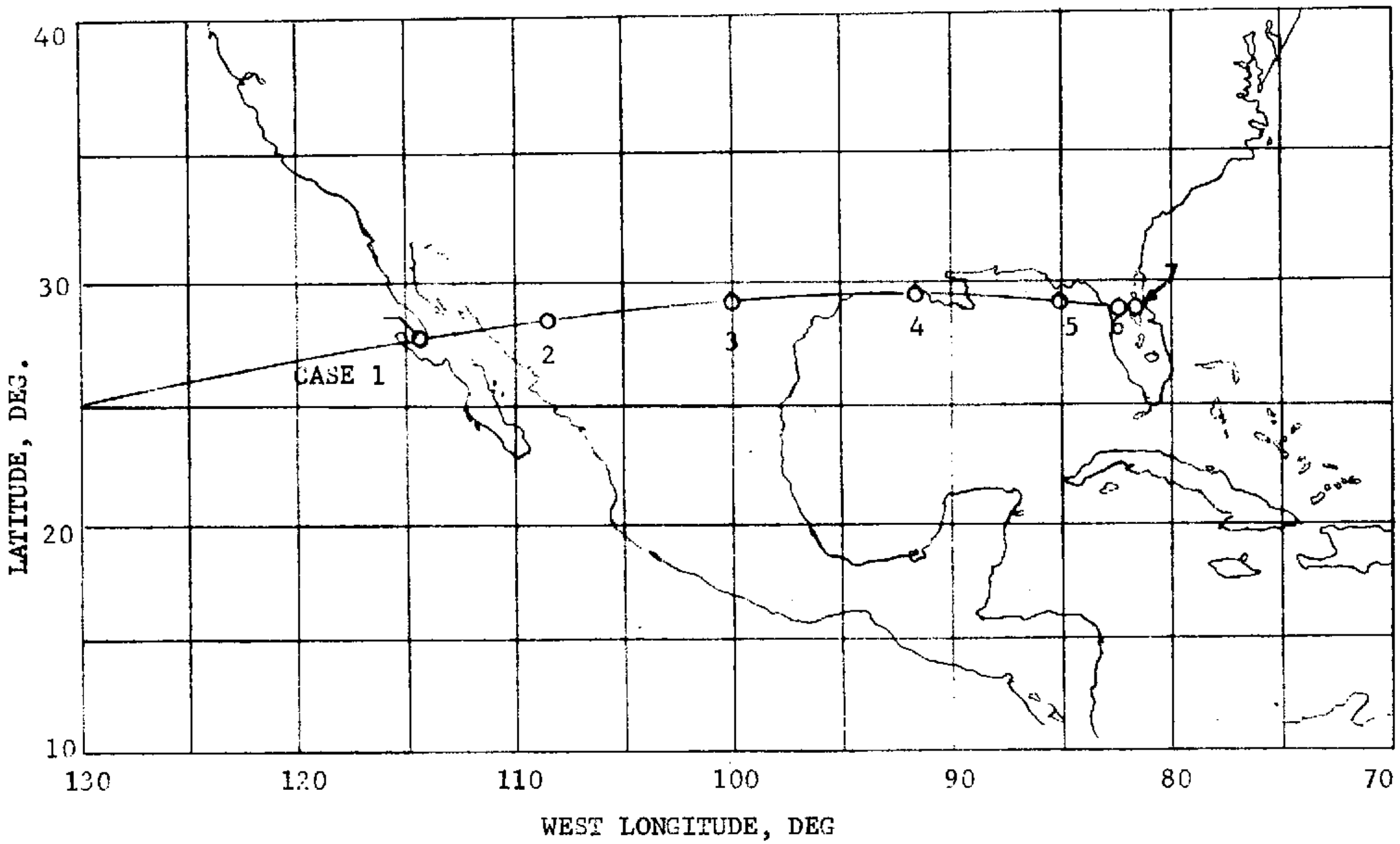

Figure 6A.—Space Shuttle vacuum IIP's for a once-around abort from an easterly launch trajectory. Case numbers correspond to conditions listed in table 1A.