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PERFORMANCE CAPABILITY OF LASER-POWERED LAUNCH VEHICLES USING VERTICAL ASCENT TRAJECTORIES

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16. Abstract

The use of a ground-based high-power laser source to power a vertically launched rocket vehicle is investigated. By using a vertical ascent trajectory, only a single laser source is required. The vertical ascent mode is not applicable to Earth orbit destinations but is applicable to missions beyond Earth escape. Performance and trajectory characteristics are examined for vertical trajectories to Earth escape and solar escape (which may be of interest in the future for radioactive waste disposal). Specific impulse values from 2000 to 5000 seconds are considered. With these values, a single-stage vehicle can deliver payloads to Earth escape and beyond, but extremely high power sources (gigawatts) are required.

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SUMMARY

The use of a ground-based high-power laser source to power a vertically launched rocket vehicle is investigated. The principle advantage of a vertical ascent is that only a single laser source is required. Performance capability and trajectory characteristics are examined for vertical ascent trajectories to Earth escape and solar escape. The specific impulses used for the laser system range from 2000 to 5000 seconds. Launch sites include the Eastern Test Range (Cape Kennedy) and the Earth's two poles. The abort impact point traces for these trajectories are considered because of their special characteristics. The polar sites are included, in spite of their obvious disadvantages, because of their unique abort impact point traces (the launch site environs). This could be advantageous for possible future radioactive waste disposal missions.

Payload fractions for a single-stage laser-powered launch vehicle indicate that a vertical launch has potential for missions to Earth escape and beyond. However, extremely high power levels, on the order of gigawatts, are required.

INTRODUCTION

Laser power has been proposed for launch vehicle propulsion for the future (refs. 1 to 4). Due to the possible high specific impulse potential of such a system, using a remote laser power source for rocket propulsion could be very attractive. Several mission modes have been proposed for utilizing laser-powered propulsion. There are proposals for placing the laser source or sources in orbit and using them to raise the orbits of payloads, especially to geostationary orbit. A tug-like vehicle would be used to convert laser power into propulsive force. The laser-tug and the payload would first be placed in a low Earth orbit by a conventionally powered launch vehicle such as the Space Shuttle. Another proposal involves a ground-based laser power source, beaming the energy to an orbiting laser tug. Still another concept proposes using laser power to boost payloads into space from the ground (ref. 1). For a ground-based laser, however, a conventional ascent trajectory to orbit presents line-of-sight problems. A simple scheme to avoid the line-of-sight problem is to direct the thrust vertically such that the vehicle rises above the launch pad until the payload has achieved the required velocity. This ascent mode is not applicable without great modification for Earth orbit destinations, but if the objective is Earth escape or beyond, the method has possibilities.

In addition to space science missions, another potential use of launches beyond Earth escape could be for radioactive waste disposal (ref. 5). If such a program were undertaken, it could involve many launches. Also, for radioactive waste disposal missions, the payload impact point (in the event of an abort during ascent) is of particular interest, since it would be important to retrieve the radioactive payload.

The instantaneous impact point (IIP) trace (the track on the Earth's surface where the vehicle would impact in case of abort) for a vertical launch differs from the trace for a conventional trajectory. For vertical launches from the Eastern Test Range (ETR), the theoretical IIP trace remains near the launch site until some period of time prior to attaining escape velocity and then moves rapidly around the Earth at a near constant latitude nearly equal to the launch site latitude. A similar IIP trace would exist for other nonpolar launch sites. However, if a vertical launch were made from one of the Earth's poles (assuming such a site were feasible), a unique IIP trace would be obtained. For a polar launch, the IIP trace remains near the launch site until the vehicle has achieved Earth escape velocity.

This report presents the results of a study of the performance capability of a laser-powered single-stage rocket in a vertical launch mode. Earth escape and solar escape performance estimates are presented for nonpolar and polar launch sites. Earth escape is the lowest energy for which this launch mode is practical and solar escape represents a maximum energy of interest. The effect of launch thrust-to-weight ratio on performance and some trajectory characteristics are presented for specific impulses of 2000 and 5000 seconds. The data are based on numerically integrated trajectories.

SYMBOLS

C_3	vis-viva energy, km^2/sec^2
I_{sp}	specific impulse, sec
IIP	instantaneous impact point, latitude and longitude, deg
$\hat{P}$	unit vector in polar direction
$\bar{R}$	position vector, m
T/W_o	launch thrust-to-weight ratio

V	velocity magnitude, m/sec
$\bar{V}$	velocity vector, m/sec
$\bar{V}_p$	hyperbolic velocity in polar direction
$\bar{V}_1$	hyperbolic velocity for nonpolar launch site
μ	gravitational constant

Subscripts:

e	Earth
s	Sun

ASSUMPTIONS AND ANALYSIS

Vehicle Description

Since laser propulsion is an advanced concept, the characteristics of a laser propulsion system are described in available references in only the most general way. There has been virtually no effort to design an actual system. Such an effort is premature. The technology associated with intercepting laser energy beamed from the Earth's surface and converting that energy into propulsive force is not developed and is at this point in the conceptual stage. However, before effort is made to develop such technology, the advantages of such a system must be evaluated. But the absence of concrete design criteria forces those involved in a preliminary evaluation to make assumptions regarding vehicle characteristics which are unsupported by a preliminary design effort.

A launch vehicle performance evaluation requires more design information than does the evaluation of an upper stage which operates entirely above the atmosphere. Initial mass, hardware mass, engine thrust, and specific impulse information are requirements for any vehicle. In addition to these, vehicle parameters associated with the atmospheric effects are required to evaluate a launch vehicle. The atmosphere produces drag on the vehicle and degrades the thrust level and specific impulse.

To evaluate the laser launch vehicle concept, the following vehicle characteristics were chosen:

(1) Launch weight of 45 360 kilograms (100 000 lb)
(2) Vehicle diameter of 4.6 meters (15 ft)
(3) Drag coefficient curve representative of current launch vehicles

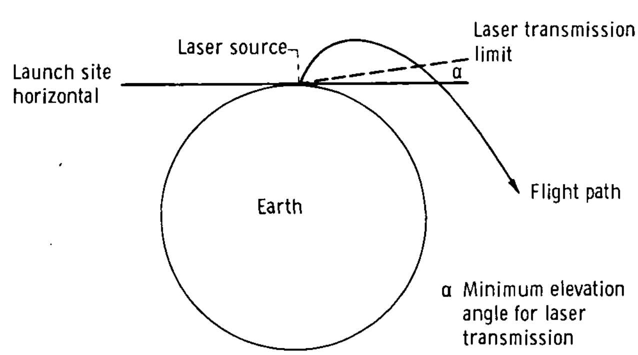
The vehicle diameter determines the drag area and, combined with the drag coefficient curve, determines the effect of drag on performance. Because of the lack of an adequately defined engine, the thrust level of the engine was assumed to be constant, thus ignoring the effect of the atmosphere on thrust level.

One of the greatest uncertainties is the mass of the propulsion system itself. Therefore the performance capability is shown in terms of the sum of payload and propulsion system mass. The mass of the vehicle exclusive of the propulsion system is assumed to be 20 percent of the propellant mass. A structure factor of 20 percent may be high, but the "fuel" heated and expended by the propulsion system will probably be hydrogen, which lacks the density usually associated with high mass fraction tankage. Adjusting the performance capability data to reflect a different structure factor is simple, since the amount of propellant associated with a given performance number is easy to determine.

The specific impulse and thrust-to-weight ratio are varied parametrically over the range of interest. Specific impulse is varied from 2000 to 5000 seconds. Five thousand seconds is an optimistic assessment of the maximum specific impulse of a laser propulsion system. Two thousand seconds is a relatively low value, below which the performance capability is marginal for the missions of interest.

Trajectory Description

A conventional launch vehicle trajectory is characterized by a short vertical rise of fixed duration after which the vehicle pitches over in the azimuth direction and thrusts in a near zero angle-of-attack mode until the atmosphere is left behind. From that point on, the thrust vector is directed optimally such that payload is maximized. The vehicle normally passes over the horizon before payload insertion. Furthermore, the elevation angle is quite low (the vehicle is near the horizon) for a significant portion of the flight as shown in the following sketch:



A conventional flight profile such as that just described would require more than one laser source to provide continuous power to the vehicle. As the elevation angle approaches zero and the vehicle is low on the horizon, the laser beam is required to pass through more and more of the atmosphere with the consequent scattering and absorption problem.

The problems just described for a conventional trajectory can be avoided by obtaining an optimum trajectory constrained such that the vehicle is always above some prescribed elevation angle from a single defined laser source. Such a trajectory could be established analytically with considerable effort, but more fundamental, determining the limiting elevation angle and transmitted power losses as a function of elevation angle and distance is a difficult problem. The results of a study would be dependent on those assumptions associated with elevation angle.

For a preliminary and somewhat specialized study, vertical ascent trajectories may be considered. A vertical trajectory is not practical for Earth orbit applications without great modification, but it does have possibilities for Earth escape energies and beyond. In the vertical ascent trajectories of this study, the thrust vector is constrained to be alined parallel to the radius vector passing through the launch site. This vector rotates with the Earth. Strictly speaking, if the vehicle does not remain over the launch site, the thrust is not vertical. Because of the centrifugal force due to the Earth's rotation and conservation of angular momentum, the vehicle subpoint for a nonpolar launch site does not remain over the launch site, but moves west and toward the equator. However, for the high thrust-to-weight ratios considered, the effect is small and does not affect the performance determination significantly.

There are other definitions of a vertical ascent mode than the one provided previously. For instance, a thrust vector direction could be defined such that it is alined parallel to the radius vector passing through the launch site, but does not rotate with the Earth. This would be an equally valid definition. The trajectory profile for such a definition would be only slightly different from the one chosen and performance characteristics would be virtually identical. There are other conceivable definitions that might be chosen, but the performance characteristics would be almost indistinguishable.

A conventional launch vehicle ascent trajectory has a characteristic instantaneous impact point (IIP) trace. The path of the trace is roughly a great circle if the ascent is planar. For a planar Eastern Test Range (ETR) launch, the trace propagates eastward at first very slowly, then progressively faster until the perigee radius of the instantaneous orbit exceeds the radius of the Earth, at which point it is said to lift off. The IIP trace for a vertical ascent trajectory from a nonpolar launch site propagates west at a virtually constant latitude. This is in small part the result of the westward movement of the subpoint. The major contribution to the propagation of the IIP trace is the rotation of the Earth during the time period between cessation of thrust (due to an abort) and the subsequent surface impact. As this period becomes a significant portion of a day, the IIP trace will propagate around the Earth at an almost constant latitude. When the time to impact becomes near infinity (as the vehicle approaches Earth escape energy), the Earth may complete several rotations before impact. Oblateness, solar, lunar, and other perturbations as well as dispersions will expand the IIP trace into a zone girding the Earth.

Theoretically, for a polar launch, the IIP trace would be confined to the launch site. Perturbing forces and dispersions will expand the IIP footprint beyond a point, but nevertheless, it should be much smaller than a conventional IIP trace.

Solar Escape Energy Requirements

As stated earlier, a vertical ascent launch mode is most applicable for Earth escape missions and beyond. The energy required to reach Earth escape (or some specific energy beyond Earth escape) is almost constant regardless of launch site. One of the greatest energies beyond Earth escape that might be sought is solar escape. This is not a constant energy if referenced to the Earth, but is a function of launch site location, launch time, and launch date. Performance determinations were made for solar escape missions for vertical launches from ETR and from the poles. The energy requirements to solar escape as a function of launch date for the designated launch sites were determined.

The polar sites are considered first since the analysis is simpler. The position and velocity vectors $\overline{R}_e$ and $\overline{V}_e$ of the Earth with respect to the Sun are obtained from an ephemeris as a function of launch date. The polar direction $\hat{P}$ with respect to the Sun is obtained in the same manner. For solar escape from the poles,

$$\frac{(\overline{V}_p + \overline{V}_e)\cdot(\overline{V}_p + \overline{V}_e)}{2} - \frac{\mu_s}{|R_e|} = 0$$

The magnitude of $\overline{V}_p$ (velocity in the appropriate polar direction) can be easily obtained from this equation. The magnitude of $\overline{V}_p$ is designated as V_p. In an Earth-centered coordinate system, V_p is the hyperbolic velocity with respect to Earth and the vis-viva energy C_3 required is equal to the square of the hyperbolic velocity. In the polar launch site cases, the rotation of the Earth and, hence, launch time of day are not pertinent.

For solar escape missions from nonpolar launch sites, the rotation of the Earth is pertinent. Although solar escape is possible at any time with sufficient expenditure of energy, there is a daily launch opportunity which minimizes the energy required. As before, $\overline{R}_e$ and $\overline{V}_e$ are functions of date. It is obvious that the minimum energy for solar escape on any given day occurs when the angle between the Earth's velocity vector around the Sun and the vehicle velocity vector is minimal. This problem is most easily solved in Earth-centered nonrotating coordinates. In this coordinate system, the Earth's velocity around the Sun on any given day is virtually fixed in magnitude and direction. Thus, there is a fixed declination and right ascension for that vector. The launch site vector, on the other hand, has a fixed declination but the right ascension rotates 360° every day. The minimum angle between $\overline{V}_e$ and $\overline{V}_l$ (vehicle velocity vector) at nonpolar launch sites occurs when the right ascensions are equal; therefore, the minimum angle is the difference in declination. The minimum energy is then calculated as in the polar launch case.

RESULTS AND DISCUSSION

A vertical ascent trajectory is not an efficient method for achieving a desired energy. By definition, the thrust vector in a vertical ascent trajectory is alined almost antiparallel to the gravity vector such that so-called gravity loss (g-loss) is maximum for such a launch mode. In an optimum trajectory to a prescribed energy with the usual constraints (drag, etc.), the thrust vector is directed such as to minimize alining any component of the vector against the force of gravity. Vertical ascent trajectories are not feasible for Earth escape missions with conventional multistage launch vehicles ($I_{sp} \le 450$ sec) since the increased gravity losses result in very low payload fractions. Solar escape missions are certainly not feasible because of the much higher energies required. With high thrust-to-weight ratios and high specific impulses, a single-stage laser-powered vertically launched vehicle may be feasible from performance capability considerations. High thrust-to-weight ratios reduce the g-loss; an impulsive velocity would theoretically eliminate it. But high thrust-to-weight ratios have disadvantages. High accelerations increase structure weight. High velocities in the low atmosphere result in high drag, high atmospheric heating, and large maximum dynamic pressures. Figures 1 to 4 show the effects of launch thrust-to-weight ratio on performance, maximum dynamic pressure, altitude, and propulsion time to Earth escape for a specific impulse of 2000 seconds. Payload plus propulsion system mass increases as a function of launch thrust-to-weight ratio over the range considered. The payload plus propulsion mass as a function of thrust-to-weight ratio is also shown for a conventional trajectory. There is an optimum thrust-to-weight ratio for a conventional trajectory since the vehicle is required to pass through a 185-kilometer (100-nm) circular parking orbit, which is an arbitrary constraint. For the higher thrust-to-weight ratios, a lower circular orbit altitude would improve performance. This is not explored since dynamic pressure limitations are violated in any case. For both cases, payload plus propulsion system mass is reported because of the difficulty in predicting the mass of a laser propulsion system. A hardware mass corresponding to 20 percent of the propellant mass has been subtracted from the mass at burnout. The impact of thrust level on propulsion system mass and the hardware mass is not considered.

Even though the structure factor and propulsion system mass are not penalized for higher thrust-to-weight ratios, as launch thrust-to-weight ratio exceeds 3.5, the performance advantage of higher thrust is small. In figure 2, maximum dynamic pressure as a function of launch thrust-to-weight ratio shows that the maximum pressure exceeds the current limits (40 000 to 50 000 N/m^2) for launch thrust-to-weight ratios in excess of about 3. Before choosing a thrust-to-weight ratio for the remainder of the study, another factor should be considered. Figure 1 includes on the abscissa the power required for a 45 360-kilogram (100 000-lb) vehicle. The power required is proportional to thrust level. The power levels indicated for even a modest vehicle are extremely high. Thus, there is motivation to keep the thrust level as low as possible. From performance considerations, dynamic pressure criteria, and power level, a launch thrust-to-weight ratio of 2 appears to be acceptable. The performance capability of 16 700 kilograms (36 800 lb) is less than the 19 700 kilograms (43 400 lb) available with a conventional trajectory for a similar vehicle model, but it is high enough to encourage further consideration of the vertical launch mode. The optimum thrust-to-weight ratio and maximum payload would depend on the specific mass of the propulsion system, which is unknown.

As stated earlier, the IIP trace for a vertical launch mode is very different from that of a conventional vehicle. The IIP trace for a vertical ascent vehicle propagates west from the launch site at a virtually constant latitude as shown in figure 5. Figure 6 shows the angular separation in longitude of the launch site and IIP as a function of time for both a vertical ascent and a tilted ascent. Because the vehicle is not pitched over early in the flight as in a conventional trajectory, the IIP trace tends to linger about the launch site longer than the IIP trace of a conventional trajectory. For some payloads under certain circumstances (such as a radioactive payload), this might be desirable. The figure shows that the IIP stays within 10° of the launch site for approximately the first 400 seconds and then moves away very rapidly. The dwell time of the IIP trace near ETR could be prolonged by tilting the thrust vector slightly to the east. To accomplish this, the thrust vector was constrained to be parallel to a vector emanating from the center of the Earth having the same declination as the launch site vector but tipped 3.7° to the east. The tilted ascent increases the dwell time near the launch site about 25 seconds, which is probably not significant. For such a small tilt, the performance variation is negligible. The dwell time might be increased by tilting the thrust vector by greater angles, but the impact point would only move further away from the launch site to the east before moving back over the site and proceeding on westward.

Figure 7 shows performance beyond Earth escape for an initial thrust-to-weight ratio of 2 as a function of C_3 (the square of the hyperbolic velocity) for the specific impulses of 2000 and 5000 seconds. Figures 8 and 9 show the corresponding injection altitudes and propulsion times. The information from these figures covers the probable range of interest for the proposed launch vehicle and launch mode for Earth escape and solar escape missions.

Figures 10 and 11 show data for a solar escape mission from ETR. The C_3's required range between approximately 150 and 250 km^2/sec^2. C_3 is a function of launch site, as discussed earlier. Because of Earth-Sun geometry, C_3 is periodic and roughly sinusoidal as a function of date. The payload plus propulsion system mass is approximately 10 000 kilograms (22 000 lb) over a large opportunity for a 2000-second specific impulse and an initial launch thrust-to-weight ratio of 2. A reasonable payload fraction is a possibility for a launch opportunity of over a hundred days a year. The IIP trace for the ETR launch site is shown in figure 5 and would be identical for every launch, regardless of date.

Solar escape mission results are presented in figures 12 and 13 for both North and South Pole launches. The practical difficulties (inaccessibility, severe climate, etc.) are recognized, but results are presented because of the unique IIP trace for these sites. Theoretically a vertical launch from the poles would impact the launch site should a vehicle failure occur short of reaching Earth escape energy.

Figure 12 shows the required C_3 and the performance capability for a vertical ascent solar escape mission from the poles. The initial thrust-to-weight ratio is 2 and the specific impulse is 5000 seconds. The C_3 required for solar escape is a function of launch date and is sinusoidal and periodic in nature, as a result of the relation between the polar direction and the position of the Earth in its orbit about the Sun. The C_3's required are enormous, ranging between 400 and 1900 km^2/sec^2. Figure 13 shows the time from launch and injection altitude as functions of launch date.

Figure 14 shows maximum and minimum performance capability for a polar launch as a function of specific impulse and launch date. For a specific impulse of 2000 seconds, the maximum capability is marginal; there is no capability for the highest C_3 of the year (minimum capability). For the higher specific impulses, performance capability appears to be adequate to make the concept interesting. There are, however, disadvantages to the high specific impulses. The high temperatures associated with high specific impulses make realization of such a system difficult. Also, the power levels for a 2000-second specific impulse are already quite large and doubling the specific impulse doubles the power requirement of the ground-based laser source since power level is directly proportional to specific impulse. These disadvantages to a polar launch are in addition to the obvious geographic difficulties already mentioned.

CONCLUSIONS

Several of the characteristics and constraints of the laser-powered vehicle in a vertical ascent mode suggest the use of a vertical ascent mode. The high specific impulse makes a vertical launch a possibility, even though more capability is available with a conventional launch profile. In turn, the need to be within line-of-sight of the laser source makes a vertical launch attractive from an operational standpoint. The instantaneous impact point (IIP) trace for a vertical launch is quite different from that of a conventional trajectory. Its characteristics may make it attractive for some missions, such as nuclear waste disposal mission. Vertical polar launches theoretically eliminate the IIP problem, but performance for solar escape energies is severely degraded. In addition, there are the obvious climatic and operational difficulties of a polar launch.

If the propulsion system mass is not excessive, the results indicate that a solar escape launch opportunity yielding a reasonable payload fraction is a possibility for an Eastern Test Range (ETR) launch with a 2000-second specific impulse. A 2000-second specific impulse is too low for a solar escape mission from the poles with a single-stage vehicle. A specific impulse closer to 5000 seconds might be required to make a polar launch a possibility.

The power required to launch the vehicle described in this study is extremely high. For the vehicle of this study (2000-sec specific impulse, thrust-to-weight ratio equal to 2, initial mass equal to 45 360 kg), the 8.7 gigawatts of power required is equal to more than twice the power generated by the Colorado River dam system (1972 World Almanac and Book of Facts), even if no atmospheric attenuation or conversion losses are assumed. However, since the use of a laser-powered system is far in the future, large amounts of power for short durations may be available from some advanced system.

Lewis Research Center,
National Aeronautics and Space Administration,
Cleveland, Ohio, April 18, 1974,
502-04.

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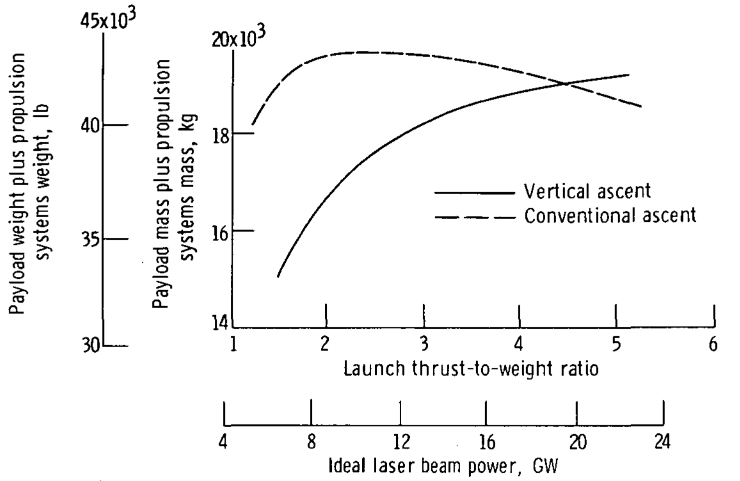

Figure 1. - Performance to Earth escape. Launch weight, 45 360 kilograms (100 000 lb); specific impulse, 2000 seconds; vertical ascent.

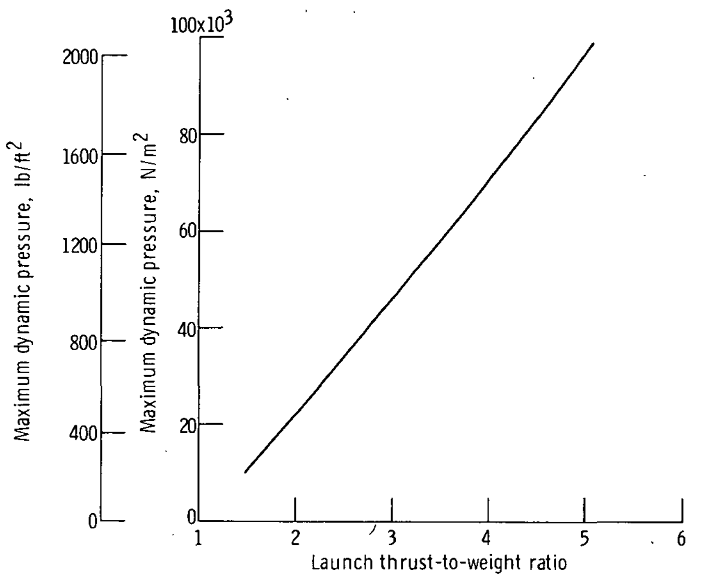

Figure 2. - Maximum dynamic pressure. Launch weight, 45 360 kilograms (100 000 lb); specific impulse, 2000 seconds; vertical ascent.

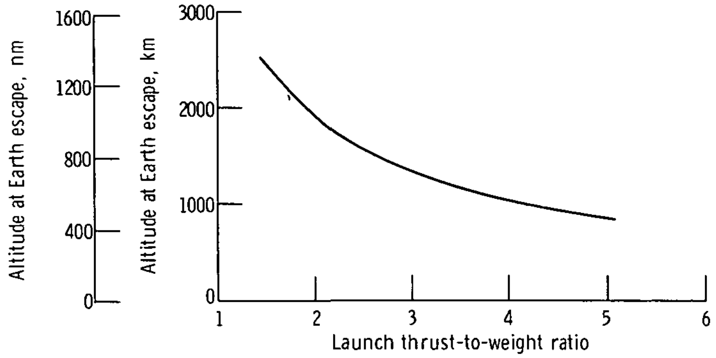

Figure 3. - Altitude at Earth escape. Launch weight, 45 360 kilograms (100 000 lb); specific impulse, 2000 seconds; vertical ascent.

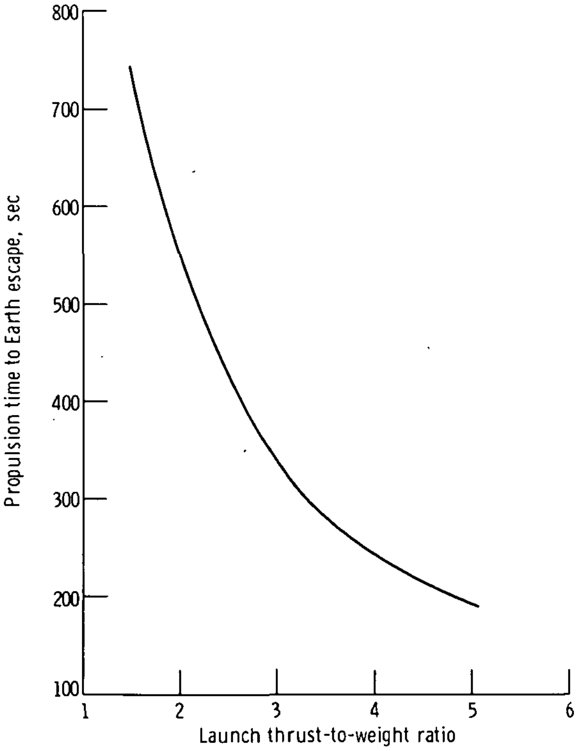

Figure 4. - Propulsion time to Earth escape. Launch weight, 45 360 kilograms (100 000 lb); specific impulse, 2000 seconds; vertical ascent.

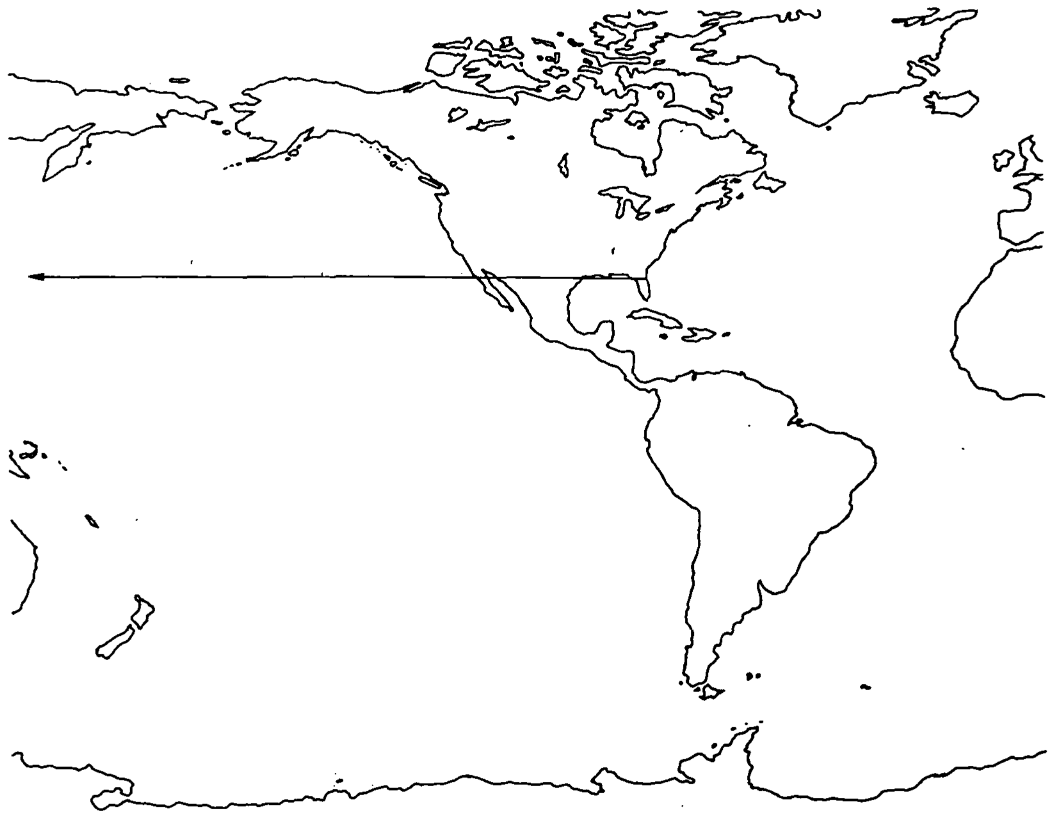

Figure 5. - Instantaneous impact point trace. Launch weight, 45 360 kilograms (100 000 lb); specific impulse, 2000 seconds; launch thrust-to-weight ratio, 2; vertical ascent.

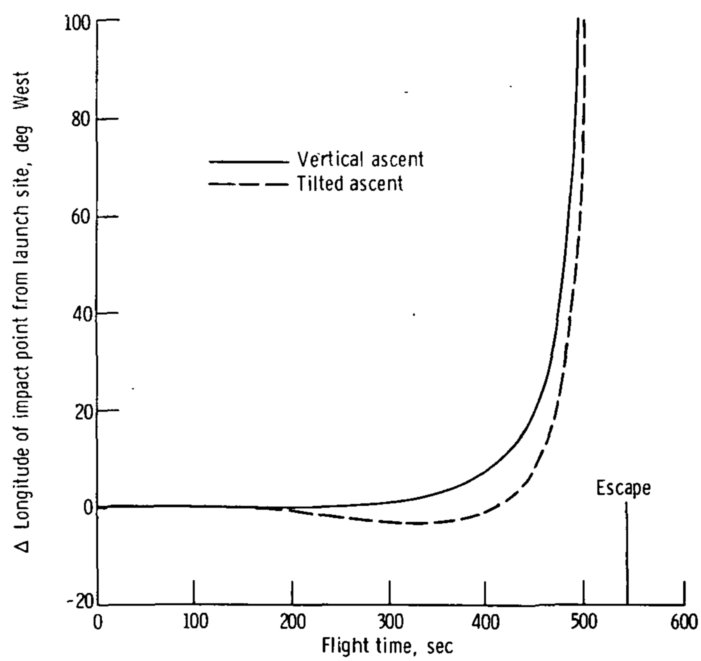

Figure 6. - Longitudinal distance of impact point from launch site. Launch weight, 45 360 kilograms (100 000 lb); specific impulse, 2000 seconds; launch thrust-to-weight ratio, 2.

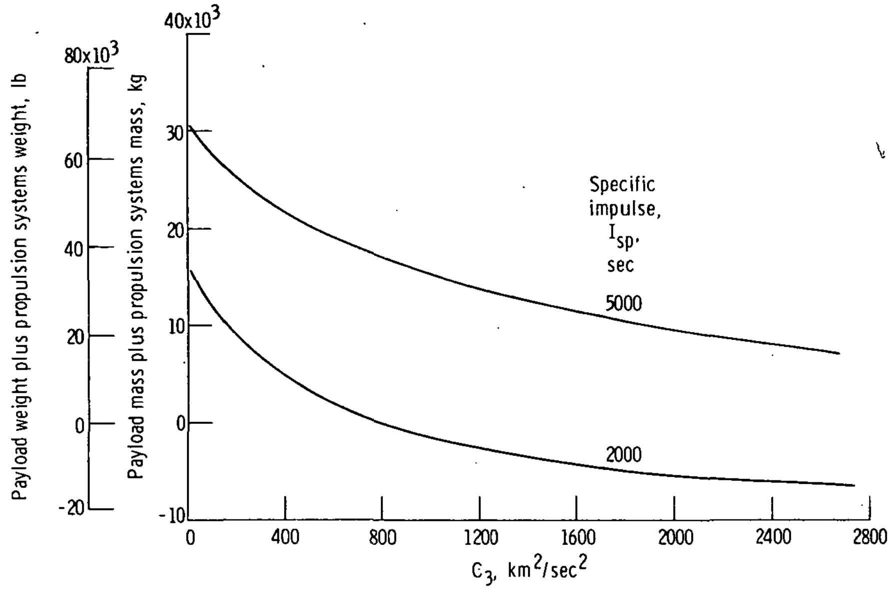

Figure 7. - Performance capability beyond Earth escape. Launch weight, 45 360 kilograms (100 000 lb); launch thrust-to-weight ratio, 2; vertical ascent.

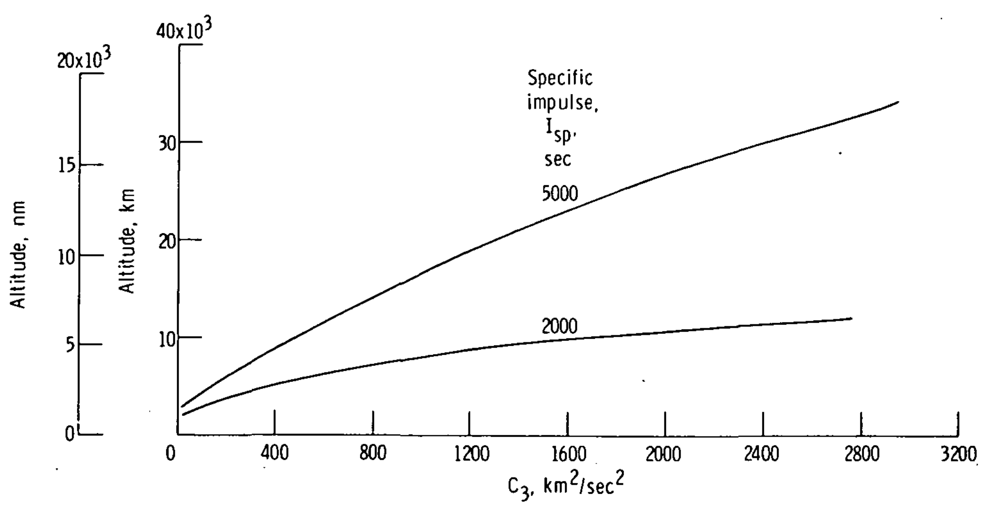

Figure 8. - Altitude at injection. Launch weight, 45 360 kilograms (100 000 lb); launch thrust-to-weight ratio, 2.

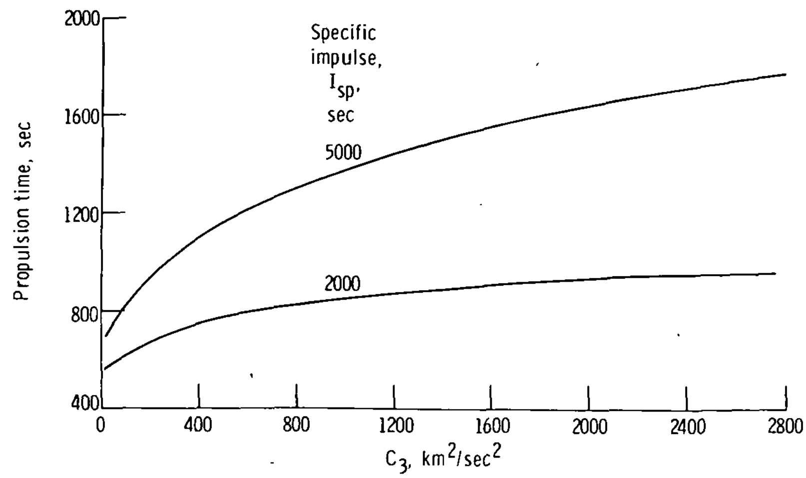

Figure 9. - Propulsion time to injection. Launch weight, 45 360 kilograms (100 000 lb); launch thrust-to-weight ratio, 2.

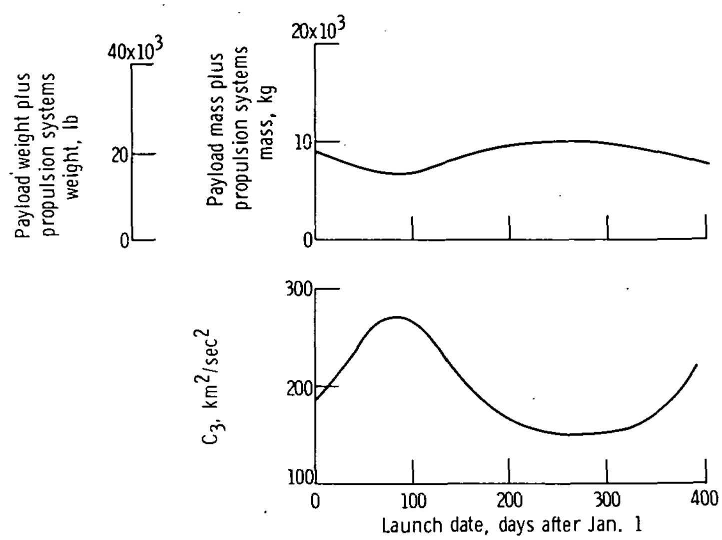

Figure 10. - Performance requirements and capability for solar escape; launch from Cape Kennedy. Launch weight, 45 360 kilograms (100 000 lb); specific impulse, 2000 seconds; launch thrust-to-weight ratio, 2; vertical ascent.

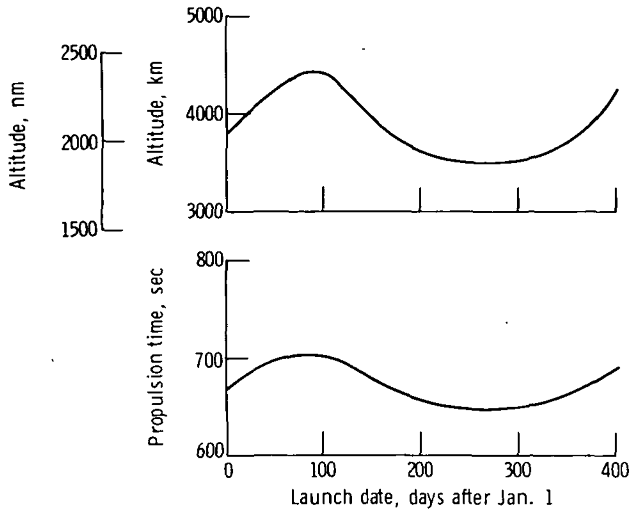

Figure 11. - Propulsion time and altitude at solar escape; launch from Cape Kennedy. Launch weight, 45 360 kilograms (100 000 lb); specific impulse, 2000 seconds; launch thrust-to-weight ratio, 2; vertical ascent.

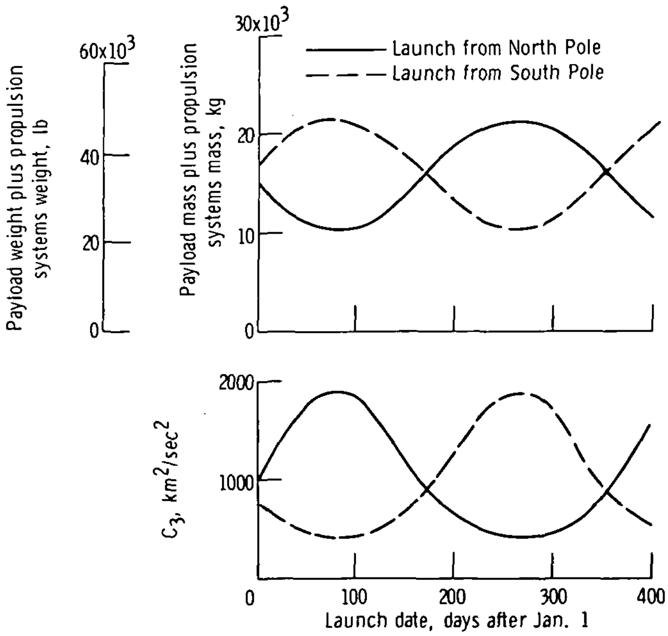

Figure 12. - Performance requirements and capability for solar escape from poles. Launch weight, 45 360 kilograms (100 000 lb); specific impulse, 5000 seconds; launch thrust-to-weight ratio, 2; vertical ascent.

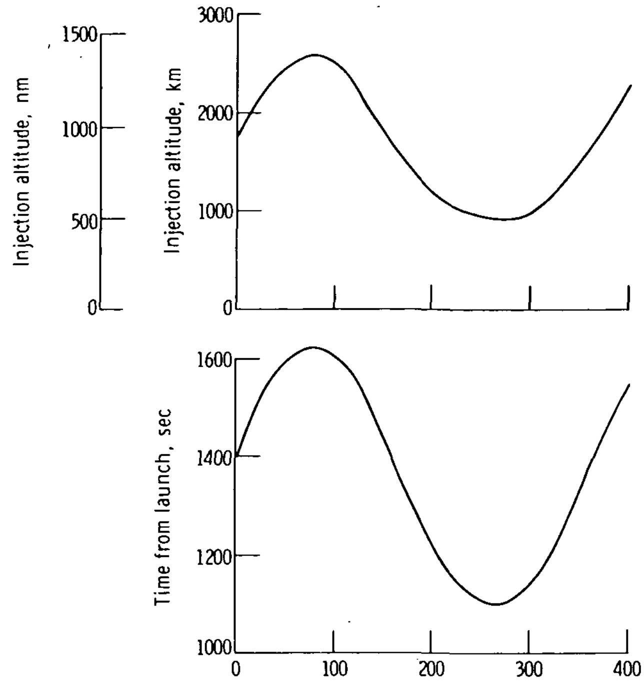

Figure 13. - Propulsion time and altitude for solar escape from poles. Launch weight, 45 360 kilograms (100 000 lb); launch thrust-to-weight ratio, 2; vertical ascent.

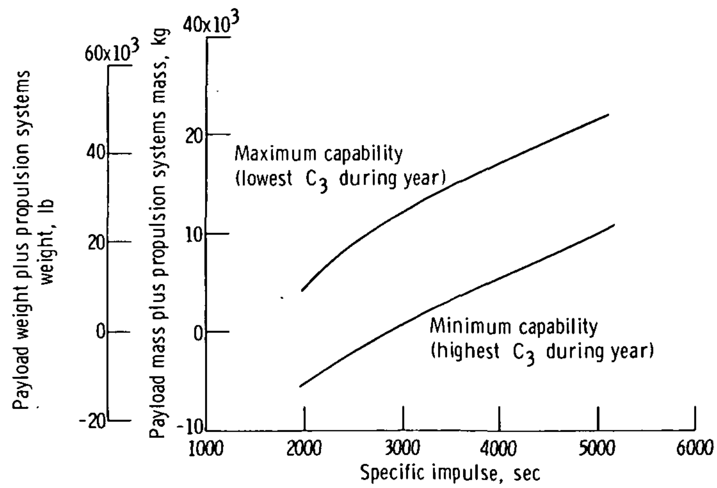

Figure 14. - Performance capability for solar escape from poles. Launch weight, 45 360 kilograms (100 000 lb); launch thrust-to-weight ratio, 2; vertical ascent.

NASA-Langley, 1974 E-7929

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—NATIONAL AERONAUTICS AND SPACE ACT OF 1958

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