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Mission Capabilities of Ion Engines
Contract NAS5-935
Phase II - Final Report

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Report R-1297-9

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TABLE OF CONTENTS

Report R-1297-9

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SUMMARY

Payloads and mission times were calculated for space vehicles propelled by ion rockets using nuclear power supplies having specific weights from 10 to 50 lb/kw. Included in the study were five missions: low-altitude lunar satellite, low-altitude Venus satellite, solar probe, Saturn probe, and a Jupiter satellite with a circular orbit at the altitude of Jupiter's fourth moon. The variation of payload with the ratio of power supply weight to gross weight was studied and the optimum power levels thereby determined. The ion rocket payload capabilities were compared with those of high-thrust vehicles using hydrogen-oxygen rockets and tungsten-core nuclear rockets; in addition the performance of high- and low-thrust systems staged in combination has been investigated. Launch vehicles considered in this study were the Atlas-Centaur, the Saturn C-1, and the Saturn C-5.

CONCLUSIONS

1. A power supply weight between 20 and 30% of vehicle gross weight will result in near-minimum flight time for a given payload, except for payloads under about 5 to 10% of vehicle gross weight. This conclusion is relatively independent of the mission, flight time, power supply specific weight, and accelerator efficiencies.

2. The best power levels for spacecraft orbited by Atlas-Centaur, Saturn C-1, and Saturn C-5 vehicles would thus be about 170 to 250 kw, 380 to 570 kw, and 4.4 to 6.6 mw, respectively, for a power supply specific weight of 10 lb/kw, or half these values if the specific weight were 20 lb/kw.

3. Assuming compatibility of physical dimensions, a power supply of near-optimum weight for Saturn C-1 space vehicles could also be used with an

Atlas-Centaur vehicle rather than one optimized for the Atlas-Centaur, although with significant reduction in payload capability. On the other hand, two power supply units of near-optimum weight for the Atlas-Centaur would also provide near-optimum performance for a Saturn C-1 space vehicle assuming that parallel operation of two reactors and power supplies is feasible.

4. Because of the very large characteristic velocities required for a solar probe, the ion rocket is the only system among those considered which is more than marginally capable of performing the mission.

5. Ion-rocket vehicles can deliver greater payloads than high-thrust systems for all flight times on the Saturn probe and Jupiter satellite missions if the power supply specific weight is about 10 lb/kw or less. If the specific weight is as great as 20 lb/kw, this conclusion still holds except for marginally small payloads.

6. The flight times for Jupiter satellite and Saturn probe missions calculated on the basis of the efficiencies and specific weights assumed in the study are greater than the planned operating life of the SNAP-8 power supply. The provision of longer lifetimes for advanced power supplies would allow the exploration of these planets and those beyond.

7. Ion-rocket vehicles can deliver greater payloads than the high-thrust systems on lunar and Venus satellite missions, although flight times are longer.

8. A spacecraft which uses a high-thrust stage to accelerate to slightly beyond escape velocity and then uses an ion rocket to complete the mission will generally be superior to both all-low-thrust and all-high-thrust vehicles over an intermediate range of payloads and trip times. Such a combination appears to be most attractive for two categories of missions: (a) those for which the low-thrust Earth escape spiral would be a large fraction of the total trip time and the characteristic velocity requirement is large, such as solar probes, Mercury probes, and out-of-the-ecliptic probes; (b) those for which both the all-high-thrust and all-low-thrust systems provide only marginal performance, such as manned interplanetary missions.

9. The optimum specific impulse is a function of mission, trip time, and power supply specific weight. It can range from the specific impulse for maximum thrust (2540 sec in this study) for short trip times and small payloads on up to 12,000 sec or greater for large payloads and long trip times.

RECOMMENDATIONS

A number of avenues of further and continuing study are suggested by the results contained in this report:

1. The feasibility of developing a single basic nuclear power supply unit for use with both the Atlas-Centaur category launch vehicle and the Saturn C-1 launch vehicle should be investigated. This study would include the feasibility of operating two reactors in parallel so that if for example, a 2100 to 2500 lb power supply unit were developed for use with an Atlas-Centaur, two of these could be used with the C-1. Alternatively, it should be determined whether a power supply of around 3800 lb designed for use with the C-1 would be compatible with the Centaur in terms of physical dimensions.

2. The capability to operate the powerplant continuously for periods in excess of a year and ranging up to 2 to 3 years should be investigated, since it appears that these operating times will be necessary to carry out advanced missions such as the Saturn probe mission and the Jupiter satellite mission, at least until power supplies which weigh less than 10 lb/kw can be developed. In addition, such capability will be required for manned interplanetary voyages.

3. Since electric propulsion can yield much greater payload fractions for difficult missions than any currently planned high-thrust system, the use of electric propulsion for manned interplanetary missions should be studied, and the gross weight required in orbit to carry out typical missions should be compared with the gross weights which would be necessary if nuclear rockets or combination staging of nuclear rockets and electrical rockets were used.

4. Further studies should be undertaken of vehicles which use high thrust to escape the Earth and then utilize ion propulsion to carry out the remainder of the mission. The possibility of usefully employing the SNAP-8 system on interplanetary missions by means of this technique bears specific investigation. In addition,the use of this technique with higher performance electric systems should be studied with the purpose of defining missions and trip times for which it is the most desirable mode of operation. It is believed that this technique will be particularly attractive for missions such as solar probes, Mercury probes, and out-of-the-ecliptic probes.

5. Data for ion rocket missions contained in this report and in other studies should be updated as more precise estimates of accelerator efficiencies and accelerator weights become available. Similarly, data on nuclear rocket capabilities should be updated as more precise estimates of over-all propulsion system weight and specific impulse are obtained. In addition, more precise estimates of insulation requirements for planetary satellite missions should be

incorporated into the performance data for both nuclear rockets and cryogenic chemical rocket systems. Small changes in any of the above will make important differences for small payload fractions and shorter trip times but will be less significant at higher payload fractions and longer trip times.

6. A more detailed preliminary design study of an ion propulsion system for a particular launch vehicle should include a realistic assessment of scaling effects. It is expected that in general there will be some decrease in specific weight with increasing power level. This is not expected to alter the general conclusions of this study but would tend to favor somewhat higher power levels.

INTRODUCTION

During Phase I of this contract the performance of electrically propelled spacecraft using the SNAP-8 nuclear power supply was investigated. The results as given in Ref. 1 indicate that SNAP-8-powered systems will have application mainly to 24-hr satellite and possibly lunar missions. Because of the relatively high weight of the SNAP-8 system (3000 lb for the 60-kw version and 2000 lb for the 30 kw system) it does not appear promising for interplanetary missions. It is generally recognized, however, that ion-rocket vehicles using power supplies with lower specific weights should be capable of performing highly useful "deep space" missions. Therefore, the purposes of this phase of the study were to (1) determine the weights and power levels of the power supplies which should be developed for use with these vehicles, (2) establish payload capabilities of advanced electric propulsion systems using currently programmed NASA launch vehicles, and (3) provide a preliminary comparison with high-thrust vehicles using hydrogen-oxygen rockets or advanced nuclear rockets.

SCOPE AND METHOD OF ANALYSIS

Missions and Boosters

Five missions are considered in this study, each of which originates from a 300 n mi Earth orbit. These missions include three transfers to a satellite orbit: a 100 n mi lunar orbit, a 500 n mi Venus orbit, and a 1,015,000 n mi Jupiter orbit; and two probes: passing close to Saturn, and within 20 radii of the Sun. For each mission the performance of ion-propelled spacecraft is compared with that of chemically-propelled and nuclear-propelled spacecraft launched by the same booster and originating from the same orbit. In addition, the use of chemical and electric propulsion in combination is investigated for the Venus satellite mission. The boosters considered are the Atlas-Centaur with a payload capability of 8500 lb in a 300 n mi orbit, and the Saturn C-1 with a 19,000 lb payload capability. For the lunar satellite mission the Saturn C-5 with a 220,000 lb payload capability is also considered.

Ion Engine Efficiency

Selection of an operating condition for ion-propelled spacecraft is characteristically a trade-off between trip time and payload. This circumstance arises from the inverse proportionality between thrust, F, and specific impulse, I, in the thrust equation which shows the dependence of thrust on engine efficiency, η, input power, P, and specific impulse.

$$F = \frac{2\eta P}{Ig} \tag{1}$$

The over-all engine efficiency, η, which consists of two parts, a power efficiency η_p and a propellant utilization efficiency η_u, is given by Eq. (2).

$$\eta = \frac{\eta_u}{1 + \frac{\eta_u^2 L^2}{I^2}} \tag{2}$$

where L is a constant for a given type of ion engine. This equation demonstrates the dependence of engine efficiency on specific impulse and, in combination with Eq. (1), indicates the existence of a specific impulse for maximum thrust.

In Fig. 1 the efficiencies of the ion engines considered in this study are shown as functions of specific impulse. The curve for the bombardment ion engine, corresponding to the configuration being developed at the NASA Lewis Research Center, is based on a loss of 1000 electron volts per mercury ion and 80% propellant utilization. The curve for the cesium engine applies to a surface-contact engine of the type being developed by Hughes Aircraft Company and is based on their estimates (Ref. 2) of 20 ma/cm^2 ion source current, 40% heater efficiency for the ionizer, and a neutralizer power requirement which is 25% of that for the ionizer. A 95% propellant utilization has been assumed for the Hughes engine. The lowest specific impulse shown for each engine is the one at which maximum thrust is obtained for a given power input. Since operation at lower specific impulses involves a reduction rather than an increase in thrust, there is no advantage to be gained by operation below this point.

Figure 1 shows that the bombardment engine has the higher efficiency for specific impulses less than about 7000 sec, while the cesium engine is superior at higher specific impulses. In this study the higher efficiency of the two is always used so that operation below 7000 sec assumes the use of the bombardment engine and operation above 7000 sec assumes use of the cesium engine. The maximum thrust of the bombardment engine occurs at a specific impulse of 2540 sec which is the minimum specific impulse considered in this study.

High-Thrust Systems and Trajectories

In general the high-thrust systems are analyzed on the basis of state-of-the-art assumptions as regards weights (see Appendix I) and engine performance. Although optimum operating conditions vary for the different vehicles and missions considered here, the H_2/O_2 chemical rockets are assumed to operate at a mixture ratio of 6 to yield a specific impulse of 420 sec. These figures represent a typical operating condition for H_2/O_2 stages currently being considered for future use.

With regard to the nuclear high-thrust stages, the use of graphite-moderated reactors was found to result in prohibitively large fixed weights and correspondingly poor payload carrying capacity for stages of the size under consideration here. Consequently, a fast unmoderated system was selected with a core consisting of uranium oxide dispersed in tungsten. According to Ref. 3 such a system could have a minimum weight of 1000 lb as compared with 4500 lb for a graphite reactor engine. The fast system is assumed to operate at a specific impulse of 800 sec at which 1 mw corresponds to 45 lb of thrust.

In the comparison of low- and high-thrust upper stages, several high-thrust combinations are considered: two chemical stages, a nuclear stage plus a chemical stage, two nuclear stages, and a $1\frac{1}{2}$-stage nuclear system. The last assumes disposability of a propellant tank which carries enough propellant to fulfill one-half of the characteristic velocity requirement for the probe missions or the first velocity impulse of the satellite missions. This tank is jettisoned as soon as it becomes empty.

Trajectory calculations for the high-thrust Jupiter satellite, Venus satellite, and Saturn probe missions were carried out with the use of an existing program which employs the usual assumptions of impulsive application of thrust and motion according to Kepler's laws. These calculations, as well as all others in this study, are based on the assumptions of circular, coplanar Earth and destination planet orbits. This assumption provides acceptable accuracy for the planets considered in this study. In each case the minimum ΔV is found for each trip time by considering a wide range of launch positions, arrival positions, and firing angles.

In the case of the Jupiter satellite mission, the approach to the planet consists of a three-impulse capture maneuver explained in Ref. 4. The three-impulse capture was selected because for high-altitude satellite orbits it requires the lowest ΔV of any known capture maneuver and consequently results in significantly greater payloads for a given trip time for the Jupiter satellite mission.

Results for the lunar mission are based on previous work done at the Research Laboratories (Ref. 5) involving three-body calculations in the Earth-Moon system. The only high-thrust system which yields a positive payload for the solar probe mission is the $1\frac{1}{2}$-stage nuclear rocket using a Saturn C-1 launch vehicle.

Ion Engine Systems and Trajectories

The performance criterion which is used in this study is minimum total trip time for a given payload. With the assumptions of constant thrust plus coast trajectories, it is necessary to determine the optimum operating specific impulse for each trip time. To accomplish this optimization for a given mission, power supply specific weight, α , and power supply weight, W_{PP}, calculations are made for several values of specific impulse. Power supply weight and specific weight as referred to herein include the reactor, heat exchanger, energy conversion unit, and space radiator. The optimum specific impulse for any trip time is the one which yields the greatest payload. In this way payload vs trip time curves are generated for each mission.

The low-thrust trajectories begin with a spiral escape phase in which the spacecraft moves from the initial 300 n mi orbit to the Earth's gravisphere where transition to the heliocentric phase is assumed to begin. In general the excess velocity of the spacecraft at the gravisphere is a significant fraction of the ΔV requirement of the mission. This is especially important for the Jupiter satellite mission wherein the hyperbolic excess velocity upon approach to Jupiter can be as great as Jupiter's heliocentric velocity.

For all but the Jupiter mission the method of Ref. 6 is employed in the calculation of escape and capture spiral trajectories. This analysis is based on constant tangential acceleration but can be modified by the method of Ref. 7 so as to be applicable to the constant-thrust case. This method is not useful for the capture maneuver of Jupiter satellite missions, however, because the ratios of thrust to local weight involved are always greater than the maximum (0.01) for which the method is applicable. Trajectory data for the Jupiter capture were obtained from Ref. 8.

Previous studies at JPL and NASA Lewis Research Center have shown that for the heliocentric phase of a low-thrust mission, the use of two constant-thrust-angle burning periods provides an excellent approximation to the optimum trajectory. In the Jupiter satellite mission the thrust is assumed to have a component in the direction of increasing radius for part of the trip, followed by a coast period and a second powered phase in which the radial component is the same magnitude but in the direction of decreasing radius. The Venus satellite mission is performed in the same way but in this case the radial component of thrust is first pointed inward and then outward.

In order to perform the calculations for these missions it is necessary to work forward from the Earth and backward from the destination planet, matching energy and angular momentum of the probe at some point in between. The coast period is then determined by the corresponding burnout radii and the parameters of the resultant transfer ellipse. By repeating the calculations for a range of thrust angles a curve of payload vs trip time is determined.

If impulsive thrust transfer is considered as a limiting case, the best thrust direction would be expected to be in the same direction as the excess velocity at the gravisphere radius. This is borne out by exact solutions employing the calculus of variations. In the variational analysis, transversality conditions applied at the end points require that the thrust angles be directed parallel to the excess velocity at each terminal.

A computer program based on a gradient optimization technique was developed for analyzing the heliocentric phase of the Saturn probe mission. The optimum steering program for minimizing heliocentric flight time is selected for each input powered time. In order to generate data for a wide range of power supply weights and power levels without prohibitively long computing times, information for a single case (α = 10 lb/kw, W_{pp}/W_0 = 0.25) was obtained on the machine and transformed so as to be applicable to other cases. This generalization of the data was based on the assumption of an average thrust-weight ratio which can be expressed as a function of trip time and powered time for the entire mission. From these generalized data, curves of payload vs trip time can be generated for any desired power level or power supply weight.

Preliminary results showed that if the constant-thrust assumption were made, successful performance of the solar probe mission would require operation of the ion engine at high thrust levels and consequently at reduced fuel efficiency. In order to obtain results over a wide range of trip times, it was decided to use variable thrust for this mission.

In Ref. 9 results were obtained for a low-thrust solar probe mission in which the probe approaches tangentially to 20 solar radii. The trajectories were optimized with respect to magnitude and duration of thrust and the results correspond to a particular ratio of power to gross weight. This information is directly applicable to the mission desired in this study and is only limited by the restricted range of allowable power levels, power supply weights, and trip times.

In order to extend these results use is made of the J factor (Ref. 10) which represents the integral of the square of thrust acceleration over the total burning time. This factor is analogous to a characteristic velocity in the sense that it represents a performance requirement for a low-thrust mission. Payload can be expressed as a function of J, α, and W_{PP}/W_0, and the J factors obtained from the curve of Ref. 9 can be used to produce new curves within the given range of trip times.

Since information on shorter trip times is desirable, an extrapolation of these data must be made. To accomplish this extrapolation it is noted that for very short trip times the thrust acceleration becomes so large that the gravitational acceleration at the Earth's orbit can be neglected. The solution for field-free space which is derived in Ref. 10 shows that J is inversely proportional to the cube of total mission time. With this approximation for short trip times and the data of Ref. 9 at the longer times, a curve can easily be drawn in the region between.

In addition to the high- and low-thrust modes of propulsion considered in this report, the dual-thrust mode (i.e., when both high- and low-thrust engines are used in combination) was analyzed for the Venus satellite mission. A typical trajectory begins with a high-thrust impulse applied in the initial orbit and of sufficient magnitude to effect escape from the Earth with a residual velocity at the gravisphere. A low-thrust phase then begins with the thrust applied in the same direction as the excess velocity. At some point thrust is terminated and a coast period ensues, followed by a second low-thrust powered phase which includes a capture spiral about Venus. The calculation procedure is identical to the low-thrust Venus satellite case with the exception of the high-thrust escape from Earth.

DISCUSSION OF RESULTS

Optimization of Power Supply Weight Fraction

Optimum Power Supply Fraction for Missions in Field-Free Space

The dependence of spacecraft performance on the ratio of power supply weight to vehicle gross weight can be understood in principle by consideration of a much simplified mission; namely, the acceleration of a vehicle to a velocity ΔV in a time t by an electrically propelled vehicle in field-free space. For each value of the ratio of power supply weight to gross weight, thrust and specific impulse can be traded off in accordance with the constant power relationship, Eq. (1). A specific impulse and the corresponding value of thrust are selected and the time required to reach the given ΔV is determined. The combination of thrust and specific impulse which results in the required value of t is determined and the corresponding payload fraction calculated. This analysis is repeated for different values of the ratio of power supply weight to gross weight, and a variation of W_L/W_0 with W_{PP}/W_0 for constant values of ΔV and t is thereby obtained. Such an analysis has been carried out for various values of the parameter $2t\eta/\alpha\Delta V^2$, under the further simplifying assumptions that structural weight is zero and that the accelerator efficiency η is a fixed value independent of specific impulse. The smaller the value of this parameter, the shorter the time t for a given ΔV or the greater the resulting ΔV for a fixed time t, for fixed values of α and η.

The results are shown in Fig. 2. It is seen that if W_{PP}/W_0 is varied from 0 to 1.0 for a fixed value $2t\eta/\alpha\Delta V^2$, an optimum, i.e., a maximum value of payload weight fraction, W_L/W_0, occurs. For large payloads, i.e., small ΔV or long acceleration time t, the optimum power supply weight fraction approaches zero; as the payload is reduced the optimum value increases to a maximum of around 25%. It is furthermore seen that over a large range of payloads a power supply weight of between 20% and 30% of gross weight provides near-optimum performance. Figure 3 compares the payload fractions using a 25% power supply fraction with those obtained by optimizing the power supply fraction; it is seen that there is very little difference in this theoretical case except at large payloads. It is important to note that these results are independent of the power supply specific weight, α. This, then provides a preliminary basis for taking a power supply weight of 25% of gross weight as a rule of thumb for near-optimum performance. The ensuing results based on more realistic mission studies, in fact, largely confirm this approximation; the only significant difference in the results is that because the accelerator efficiency actually decreases with decreasing specific impulse rather than remaining constant, the optimum power supply fraction shifts to higher values as the payload approaches zero rather than tending to level off or decrease.

Optimum Power Supply for Actual Missions

The optimization of power supply weight has been studied in detail for the Saturn probe, Venus satellite, and lunar satellite missions. In Fig. 4 the results are shown for the Saturn probe mission as curves of payload fraction vs power supply fraction for constant trip times, with $\alpha = 10$ lb/kw. The optimum value of W_{PP}/W_0 varies from 25% at a trip time of 1000 days and a payload fraction of 56% to about 36% for a trip time of 450 days and a payload fraction of about 2%. Even at a trip time of 500 days the payload for a 25% power supply fraction would be only about 25% less than the optimum value. Similar results are shown in Fig. 5 for the Venus satellite mission. The optimum power supply fraction in this case varies from about 12% for a 450-day mission with a 62% payload fraction to 35 or 40% for a 135-day trip and 11% payload fraction. It is seen that a 25% power supply fraction again yields near-optimum results except for small payloads and short trip times. Although the results shown in Figs. 4 and 5 are for a particular value of

10 lb/kw, other values of α merely shift the time scale. This can be seen from the results for the lunar satellite mission, which is somewhat easier to analyze in generality than the other missions. As shown in Fig. 6, the optimum power supply fraction for this mission can be determined in terms of the single parameter t/α , similar to the ideal case of Figs. 2 and 3.

The results for the lunar satellite mission are somewhat different due to the fact that practically the entire trip time consists of the tangential-thrust Earth departure and lunar approach circular spirals. Consequently the trip time depends essentially only on the thrust-weight ratio. This differs from missions in which the coast period is a substantial portion of the total time since on such missions a given trip time can be obtained either with a relatively high thrust-weight ratio and short powered time or with a smaller thrust-weight ratio and a longer powered time. If the power supply fraction is decreased below the optimum it is only necessary to increase the powered time. In the lunar mission, however, if the power supply fraction is decreased below the optimum then in order to maintain a given trip time the specific impulse must also be decreased so as to operate at a higher value of the ratio of thrust to powerplant weight. However, this results in a lower efficiency (as in Fig. 1) with a consequent loss in payload which is greater than that resulting from off-optimum operation on other missions. Furthermore, because the variation of efficiency with specific impulse results in a value of specific impulse below which the thrust actually decreases, there is a maximum value of the ratio of thrust to powerplant weight, and it follows that for a given trip time there is a minimum power supply fraction below which the mission cannot be carried out even though the payload at this minimum point is not zero. This is in contrast to the other missions for which the limiting payload for a given time as the power supply fraction is decreased is always zero. It is therefore necessary, as seen in Fig. 6, to resort to higher power supply fractions for short trip times than in the other missions. These results are illustrated in Fig. 7 where the payload fractions corresponding to a constant power supply fraction of 0.25 are compared with those corresponding to optimum power supply fractions. It is seen that for α = 10 lb/kw, for example, the payload penalty for using 25% power supply is about 15% at a trip time of 60 days (the minimum possible for this value of power supply fraction), decreases to zero at a trip time of 75 days, and then gradually increases again to about 15% at a trip time of 200 days. Thus even for the lunar mission, for which the penalties for off-optimum design are more severe than for other missions, the 25% rule of thumb gives near-optimum performance over a significant range of flight times.

From the above results it is clear that a reasonable design value of power supply weight is between about 20% and 30% of the space vehicle gross weight. It should be emphasized that this result is independent of the power supply specific weight and that the performance can therefore be optimized simply on the basis of power supply weight rather than on the power level.

Thus, the optimum power supply weight for an 8500 lb vehicle orbited by an Atlas-Centaur would be between about 1700 and 2500 lb, so that the power level would be between 170 and 250 kw for α = 10 lb/kw and between 85 and 125 kw for α = 20 lb/kw. Similarly, for a 19,000 lb vehicle orbited by a Saturn C-1 the optimum power supply weight would be between about 3800 and 5700 lb and the corresponding power levels would be 380 to 570 kw for α = 10 lb/kw or 190 to 285 kw for α = 20 lb/kw.

Use of Same Power Supply in Two Vehicles

An important question to be explored in the development of electrically propelled spacecraft for future NASA missions is the possibility of using the same basic power supply unit for more than one vehicle; in particular for spacecraft designed for launch by both Atlas-Centaur and Saturn C-1. One possibility is to develop a power supply sized for a Centaur-launched vehicle and to use two of these for Saturn C-1 vehicles. If a 2125 lb power supply (25% of the 8500 lb nominal orbital capability of the Atlas-Centaur) were developed for Centaur spacecraft then two of these would be 22.4% of the 19,000 lb orbital capability of the Saturn C-1. A somewhat larger power supply of 2500 lb could just as well be designed for the Centaur; this would equal 30% of the spacecraft weight and two of these would be 26.4% of the weight of a Saturn C-1 spacecraft. Therefore, if it can be ascertained that parallel operation of two power supplies is feasible, it would be very attractive from the point of view of vehicle performance to develop a power supply of between 2100 and 2500 lb weight for use with the Centaur and to use two of these with the Saturn C-1 since this would result in power supply weight fractions which are near-optimum for both vehicles.

The other possibility is to design a single larger power supply which could be used with both vehicles. This is not quite as attractive a possibility from the point of view of vehicle performance since a power supply weight of 4750 lb, equal to 25% of the Saturn C-1's 19,000 lb orbital capability, would be 56% of the Centaur spacecraft's weight. This ratio can be improved by using a 3800 lb power supply which would then be 20% of the C-1 spacecraft's weight and 45% of the Centaur spacecraft's weight. This would result in near-optimum performance for the C-1 vehicle, and compromised but still substantial payloads for the Centaur as shown in Figs. 8 to 10. For the Saturn probe mission, as shown in Fig. 8, a 3800 lb power supply used with a Saturn C-1 vehicle results in payloads within about 7% of those which could be obtained with a 25% power supply fraction, whereas the same power supply results in a maximum payload of about 2800 lb for a Centaur-launched vehicle as compared with about 4200 lb maximum using a 25% power supply fraction.

The results for the Venus satellite mission are shown in Fig. 9. For the C-1 vehicle the 3800 lb power supply actually gives somewhat larger payloads than does the 4750 lb power supply for trip times greater than 270 days, and below this cross-over point requires only about 10 to 15 days more trip time for any given payload than does the larger power supply. However, using the 3800 lb power supply rather than an optimum one with the Centaur results in a reduction of the maximum payload from 5000 to 3500 lb, but at trip times below about 165 days the 3800 lb unit gives somewhat larger payloads.

Similar results are shown in Fig. 10 for the lunar satellite mission. For the C-1 vehicle the 3800 lb power supply gives greater payloads than the larger power supply for trip times greater than about 90 days, whereas the 4750 lb power supply gives greater payloads for trip times between 60 and 90 days. If the same 3800 lb power supply is used with a Centaur vehicle instead of a near-optimum 2125 lb unit, the maximum payload is reduced from about 5700 lb to 4100 lb, but the 3800 lb power supply now gives larger payloads for trip times between about 40 and 60 days.

Payload Capabilities and Comparison with High-Thrust Systems

Since it has been ascertained that a power supply weight of between 20 and 30% of vehicle gross weight generally produces near-optimum performance, 25% has been selected as a nominal design point as a basis for calculating actual payloads with the different launch vehicles. These payloads are compared with the payloads that could be obtained with high-thrust chemical or nuclear rocket propelled spacecraft orbited by the same launch vehicle. The missions are considered below in detail.

Solar Probe

The solar probe mission is one of the most difficult to perform in that although the trip times involved are relatively short (80 days for a high-thrust vehicle to get within 20 solar radii of the center of the Sun on a ballistic trajectory), the characteristic velocities required are extremely high since it is necessary to almost cancel out the Earth's orbital velocity around the Sun. It was found, in fact, that of the high-thrust systems considered none are capable of performing this mission with a Centaur launch vehicle and only the $1\frac{1}{2}$ stage nuclear rocket is even marginally capable of doing it with a Saturn C-1 launch vehicle. Ion-rocket vehicles on the other hand, are capable of delivering large payloads, as shown in Figs. 11 and 12.

Figure 11 shows that for $\alpha = 10$ lb/kw a Centaur-launched vehicle could deliver 1000 lb in about 135 days, 2000 lb in about 165 days, or as much as 3600 lb in 300 days. The trip times are of course increased for greater values of α , but are still not prohibitive. Thus for example, 1000 lb payload can be carried in about 190 days if $\alpha = 20$ lb/kw or 250 days if $\alpha = 30$ lb/kw.

Figure 12 gives similar results for the Saturn C-1 launch vehicle; the ion rocket payloads for a given value of α and trip time are scaled up according to the ratio of the orbital payloads of the Saturn C-1 and Atlas-Centaur. The $1\frac{1}{2}$-stage nuclear rocket is capable in this case of delivering a 500 lb payload in 80 days. It should be remembered, of course, that at this small a payload fraction small changes in the assumed performance parameters (specific impulse, powerplant weight, structure weight, etc.) will make a large percentage difference in the payload.

Saturn Probe Mission

In Fig. 13 the payloads of Centaur-launched space vehicles are compared for three types of space vehicles: ion rocket, $1\frac{1}{2}$-stage nuclear rocket, and a nuclear rocket first stage plus an H_2/O_2 second stage. With Centaur-size space vehicles, the nuclear-plus-chemical vehicle is apparently superior to the all-nuclear vehicle for payloads under about 500 lb. A two-stage H_2/O_2 vehicle would in this case give payloads which at all trip times are about 300 lb less than the better of the two high-thrust curves shown. It is seen that the ion rockets give greater payloads for all trip times if α = 10 lb/kw; if α is 20 lb/kw the same is true for payloads greater than about 250 lb and if α = 30 lb/kw for payloads greater than about 500 lb.

The results are similar for the Saturn C-1 launch vehicle as shown in Fig. 14. The ion rocket is again superior to the nuclear rocket for all trip times if α = 10 lb/kw; it is still superior for payloads greater than about 1300 lb if α = 20 lb/kw. However, if α is as great as 30 lb/kw the nuclear rocket becomes superior for all trip times and payloads. In this case the $1\frac{1}{2}$-stage nuclear rocket is always superior to the nuclear-plus-chemical vehicle, and the latter is therefore omitted from the comparisons. This is true for the C-1 vehicle for the other missions as well. Comparison with two-stage H_2/O_2 vehicles is also shown in Fig. 14. In this comparison the ion rockets are superior for all trip times at α = 20 lb/kw except for vanishingly small payloads, and even for 30 lb/kw the ion rockets are superior for payloads greater than about 800 lb.

It should be noted that the powered times required for ion rockets on this mission are nearly the same as or longer than presently projected powerplant lifetimes. The powered times are about 350 days or greater for α = 10 lb/kw, and about 450 days or greater for α = 20 lb/kw, as compared with an estimated operating lifetime of 420 days for the SNAP-8 system and, unofficially (Ref. 11) one year for early SPUR units. Therefore the feasibility of this mission will depend on either extending the powerplant lifetimes or else reducing the power supply specific weights and/or increasing accelerator efficiencies.

Venus Satellite Mission

The Venus satellite mission, which is similar in propulsion requirements to a Mars satellite mission, is less difficult to perform and consequently the high-thrust systems deliver greater payloads for trip times up to the ballistic Hohmann transfer time of 145 days. It is seen in Fig. 15 that with the Centaur launch vehicle, a $1\frac{1}{2}$-stage nuclear rocket would have about a 1250 lb payload capability on a Hohmann transfer and a two-stage H_2/O_2 rocket about 1000 lb. The ion rocket, with α = 10 lb/kw, can match the 1000 lb rocket at 140 days, but by taking a somewhat longer trip time the payload can be greatly increased. If α = 20 lb/kw, 240 days are required for a 2000 lb payload and 290 days for a 3000 lb payload. Also shown for comparison is a curve for α = 50 lb/kw, corresponding to SNAP-8 technology; in this case the flight times are always in excess of the SNAP-8 design lifetime of 420 days.

The reason for the cross-over of the nuclear rocket and chemical rocket curves at a payload of about 800 lb is that the fixed weight of the nuclear rocket powerplant (1100 lb) detracts substantially from the potential payload at the shorter trip times for vehicles of this small size.

A vehicle with a nuclear rocket first stage and an H_2/O_2 second stage was also considered. The estimated payload was slightly worse than for the $1\frac{1}{2}$-stage nuclear rocket on a Hohmann transfer but was slightly greater at shorter trip times. However, since the differences are inconsequential for present purposes the latter results have been omitted from Fig. 15.

The corresponding results for vehicles placed into orbit by the Saturn C-1 are given in Fig. 16. In this case the breakeven points for ion propulsion in terms of payloads are about 4500 lb in comparison with the $1\frac{1}{2}$-stage nuclear rocket and about 2300 lb in comparison with the two-stage H_2/O_2 rocket. For α = 10 lb/kw these payloads correspond to trip times for the ion rocket of about 160 and 140 days respectively; an increase in trip time to 200 days results in a payload of about 7500 lb. If α = 20 lb/kw the trip times for the breakeven-point payloads are increased to about 245 days and 220 days respectively, and a trip of 300 days yields a 7000 lb payload.

Lunar Satellite Mission

The payload capabilities on the lunar satellite mission are given in Figs. 17, 18, and 19 for vehicles placed in a low-altitude Earth orbit by the Centaur, Saturn C-1, and Saturn C-5 launch vehicles respectively. It is unlikely that special power supplies would be developed for the Centaur and C-1 vehicles especially for the purpose of orbiting scientific payloads around the Moon; therefore, the payloads of Figs. 17 and 18 are based on the same power supply weights as for the other missions; namely, 2125 lb and 4750 lb respectively. In this case the minimum trip times are about 60 days for α = 10 lb/kw and 120 days for α = 20 lb/kw. It is seen that for either launch vehicle large payload advantages are to be had in using ion propulsion if lunar trip times of these lengths or greater are acceptable.

In assessing the payload capabilities of ion rocket vehicles which would be placed in orbit by a C-5 vehicle (W_0 = 220,000 lb), the power supply weight might well be chosen specifically on the basis of the lunar cargo carrier mission since at present there do not appear to be other unmanned missions which could make use of such a large launch vehicle. Therefore, in Fig. 19 the payloads are shown on the basis of the optimum power supply weight for each trip time; the payload vs time curves for W_{pp}/W_0 = 0.25 are also shown for comparison. It is seen that by optimizing the power supply weight, trip times as low as about 20 days can be obtained if α = 10 lb/kw or 40 days if α = 20 lb/kw. However, payloads greater than those for the chemical rocket are obtained only for trip times greater than 30 to 40 days if α = 10 lb/kw, and for trip times greater than about 65 to 75 days if α = 20 lb/kw. Comparing ion rockets with the assumed $1\frac{1}{2}$-stage nuclear rocket, the breakeven trip times are about 60 to 70 days if α = 10 lb/kw, and 110 to 130 days for α = 20 lb/kw.

Jupiter Satellite Mission

Jupiter satellite missions are in general exceedingly difficult to perform. Previous discussions of such missions have been limited to the so-called "capture" mission, in which only a minimal ΔV is used for capture and the vehicle is consequently placed on an elliptical satellite orbit of near unit eccentricity around Jupiter. More ambitious missions are difficult because of the long interplanetary travel time involved and also because Jupiter's large mass necessitates large ΔV's for establishing satellites. The mission chosen here, namely a 1.015 million n mi circular orbit around Jupiter, was chosen because this corresponds to the orbit of Jupiter's fourth moon and would therefore enable close inspection of this natural satellite which is about the same size as the Earth's moon. Furthermore, circular velocity at this distance from Jupiter is 27,200 fps which, although large, is not prohibitive.

The results of the analysis of this mission are shown in Figs. 20 through 22. With an Atlas-Centaur launch vehicle, a nuclear rocket spacecraft cannot perform the mission, and a two-stage H_2/O_2 rocket has a maximum payload of only about 200 lb, whereas ion rockets could deliver payloads of up to 5000 lb for the same trip time. Similarly, ion rockets could deliver up to about 11,000 lb using a Saturn C-1 launch vehicle as compared to about 1100 lb for the nuclear rocket and 500 lb for a two-stage H_2/O_2 rocket. However, the flight times involved are problematical with respect to presently projected electric propulsion technology. The zero-payload flight time even for α = 10 lb/kw is about 500 days; although the accelerator does not have to be operated during the coast period, presently contemplated spaceborne nuclear reactor power supplies will not be capable of shutdown and restart so that they must be operated continuously during the flight. Lifetime for the SNAP-8 power supply, however, is estimated at 420 days (10,000 hr) and unofficial reports of SPUR studies indicate a one-year lifetime for initial units although two- or three-year lifetimes might be developed for later units. A two-year lifetime for a power supply with a specific weight of 10 lb/kw would allow delivery of nearly 6000 lb payload with a Saturn C-1 launch vehicle.

The payloads shown in Figs. 20 and 21 are for a 25% power supply fraction. It is of some interest to determine whether the performance at the shorter trip times for this mission can be improved by increasing the power supply fraction. The results for a 500-day mission are shown in Fig. 22; it is seen that by doubling the power supply fraction to 50% a payload of about 3.5% of gross weight can be obtained.

Combined Use of High- and Low-Thrust Stages

The possible advantages of using high- and low-thrust propulsion in combination for space missions were originally pointed out in Ref. 12. A number of possible modes of operation were considered: low thrust followed by high thrust; low, high, and then low again; high followed by low, etc. The most practical of these modes appears to be the use of a high-thrust stage followed by a low-thrust stage. This mode of operation has been applied herein to the study of Venus satellite missions and compared with all-low-thrust and all-high-thrust operation. There are two valid comparisons which can be made: first is the determination of the range of payloads and trip times for which the combined high- and low-thrust system gives better results than either all-high-thrust or all-low-thrust. Secondly, it is valid to compare the combined system with low-thrust systems alone since electric rockets may be inherently desirable for certain missions because of the large power-generating capacity which would be available for data transmission and also because low-thrust spiral escape and capture maneuvers are desirable for mapping planetary radiation belts, magnetic fields, etc.

The detailed mode of operation studied is as follows: a velocity impulse is applied with the high-thrust rocket in order to depart from the initial low-altitude Earth orbit; the high-thrust stage is then separated and the low-thrust rocket is operated until the required velocity has been reached. Later, in the case of the planetary satellite missions, the low-thrust rocket is restarted in order to execute the capture maneuver. The question which immediately arises is that of the optimum staging. Preliminary analytical studies clearly showed that using a high-thrust stage to boost the vehicle from a circular orbit into a higher-energy elliptical orbit and then using the low-thrust rocket to escape gives results which are generally poorer for all trip times than using low thrust for the entire escape maneuver. In order for the combined system to be advantageous it is necessary that the high-thrust stage provide at least the ΔV required for escape. The optimum hyperbolic excess velocity for the high-thrust stage, $V_{\infty h}$, is in general a function of the mission and trip time as well as of the performance of the high- and low-thrust stages. If the total required hyperbolic excess velocity, V_{∞} , is very large, then the optimization of $V_{\infty h}$ for each trip time may be important. However, if V_{∞} is relatively small, as in the Venus satellite mission under consideration, then this optimization is of minor consequence, and it is sufficient to use a single value of $V_{\infty h}$ over the entire range of trip times under consideration. It was determined that the best value of $V_{\infty h}$ to use for the Venus satellite mission is in the range of approximately 7500 to 12,500 fps, since excess velocities in this range are attained with high-thrust ΔV in a low-altitude Earth orbit only a few hundred feet per second greater than that required for parabolic escape, and these excess velocities are of the same magnitude as those necessary to reach Venus. Calculations were actually carried out for values of $V_{\infty h}$ = 7940 fps and 12,350 fps. The results for the two cases were practically identical. The data presented for the combined high- and low-thrust operation in Figs. 23 through 26 are all for $V_{\infty h}$ = 7940 fps. The ion rocket stage was assumed to use a power supply weight of 25% of the gross weight of the stage, and it is assumed that the power supply has the same specific weight as for the all-low-thrust vehicle with which it is compared. Only the Saturn C-1 launch vehicle has been considered since use of a high-thrust stage with a Centaur-orbited vehicle would result in a low-thrust stage which is too small to be of interest. Figures 23 and 24 give the final results for H_2/O_2 high-thrust systems. Figure 23 compares a two-stage H_2/O_2 vehicle, an ion rocket vehicle with $\alpha = 10$ lb/kw, and a combined vehicle with an H_2/O_2 first stage and a 6955 lb gross weight ion-rocket stage. It is seen that the combined system is superior to the low-thrust system for trip times between 90 and 155 days and corresponding payloads up to about 3700 lb. However, the two-stage H_2/O_2 vehicle is best for trip times less than about 110 days and payloads less than about 2200 lb. Figure 24 gives the corresponding results for $\alpha = 20$ lb/kw. In this case the combined system is better than the low-thrust system for trip times of 150 to 240 days and payloads up to 4000 lb. The two-stage H_2/O_2 vehicle is best for all trip times up to the Hohmann transfer time of 145 days and payloads of about 2300 lb.

Comparisons using the tungsten-core nuclear rocket as the high-thrust system are shown in Figs. 25 and 26 for electric power supply specific weights of 10 and 20 lb/kw respectively. The weight of the ion-rocket stage in the combined system is 9615 lb. It is seen from these two figures that the combined system is superior to the low-thrust system for payloads up to about 5700 lb, which corresponds to trip times of up to about 175 days for α = 10 lb/kw and 265 days if $\alpha = 20$ lb/kw. However, the all-high-thrust $1\frac{1}{2}$-stage nuclear rocket is superior to either system in both cases for trip times up to and including the Hohmann transfer time of 145 days which corresponds to a payload of about 4500 lb.

Certain conclusions which are generally true for all missions employing combined systems may be drawn from the foregoing results for the Venus satellite mission. First, the better the performance of the high-thrust system and the poorer the performance of the low-thrust system, the more favorable is the combined system in comparison with all-low-thrust systems but the worse is the comparison with the all-high-thrust system. Conversely, the better the performance of the low-thrust system and the poorer the high-thrust system, the more favorably the combined system compares with all-high-thrust and the less favorably it compares with all-low-thrust. It then follows that the combined system will compare favorably with the other systems when both all-high and all-low-thrust are marginal for a mission, such as in the case of fast round-trip interplanetary missions.

In the less ambitious category of unmanned missions the greatest benefits as compared to all-low-thrust operation result for missions in which the escape spiral would be a large portion of the total trip time, whereas the advantages over all-high-thrust operation are greatest for missions which require large ΔV but only relatively short coasting times. Therefore, in the category of unmanned scientific exploration, combined thrust will probably be most advantageous for missions such as solar probes, Mercury, Mars, and Venus probes, and out-of-the-ecliptic probes and least advantageous for planetary satellite missions. The Venus satellite mission in particular involves a coast phase of not more than 145 days after Earth escape and then a low-thrust capture spiral which takes about the same length of time as would an Earth escape spiral. Thus, the Venus satellite mission is inherently one of the least attractive for use of combined high- and low-thrust systems. Nevertheless it has been shown that combined systems can provide significant increments in performance over certain ranges of payload and trip times, even for this mission.

This discussion suggests the possibility of using a high-thrust stage followed by an ion-rocket stage using the SNAP-8 system. Low-thrust systems using the SNAP-8 were found in Ref. 1 to have little application to interplanetary missions because of the long flight times involved. However, the escape spiral which required a minimum time of about 140 days would be eliminated by using a high-thrust stage, and a SNAP-8 with its 30 or 60 kw power generating capacity system might therefore become useful for deep space exploration.

Optimum Specific Impulse

Optimum specific impulse is defined as the value of specific impulse which results in maximum payload for a given trip time and given mission. For a given mission the optimum specific impulse increases with increasing payload and increases with decreasing power supply specific weight. Figures 27 through 30 give the optimum specific impulses for the Saturn probe, Venus satellite, lunar satellite, and Jupiter satellite missions. The solar probe mission is not included since it was studied on the basis of variable thrust operation.

It is seen from the results of Figs. 27 through 30 that the optimum specific impulse ranges upwards from about 4000 sec if α = 10 lb/kw and from 2540 sec (specific impulse for maximum thrust) for α = 20 lb/kw. The maximum values of specific impulse which are of interest are limited only by the maximum trip time allowable, but it does not appear in any case that specific impulses of greater than 10,000 to 12,000 sec will be of interest for these missions.

CONCLUDING REMARKS

The results have clearly indicated that the variation of payload with power supply fraction is generally slow enough so that near-optimum results can be obtained with power supply fractions between 20 and 30% over a wide range of flight times. For relatively short flight times resulting in payloads of 5 to 10% of gross weight or less, the optimum power supply weight shifts to between 35 and 50% of gross weight. This shift towards higher power supply weight takes place because it is necessary to operate at lower specific impulses for short flight times and consequently lower efficiency. However, this result must be regarded as tentative since higher efficiencies at the lower specific impulses than those assumed in this study would negate this shift. Furthermore, it is of limited interest since the curve of payload vs trip time is quite steep in the region of small payloads so that a much larger payload can be obtained with a small increase in trip time.

A partial exception to the above results is the lunar satellite mission, for which the performance is more sensitive to the power supply weight. This is primarily due to the fact that almost the entire lunar mission is carried out in the presence of strong gravitational fields as discussed earlier. Nevertheless the 20 to 30% power supply fraction rule of thumb is still valid over a significant range of trip times even in this case. Thus, for example, a 25% power supply fraction gives payloads within 15% of the optimized values for trip times between about 60 and 150 days if α = 10 lb/kw. Furthermore the payload advantage over chemical or nuclear rockets is seriously compromised for shorter trip times and increases only slightly for longer trip times.

It was pointed out that a hindrance to the feasibility of the Saturn probe and Jupiter satellite missions is the long power supply operating lifetimes required for these missions as compared to currently projected operating lifetimes. For α = 10 lb/kw the zero-payload trip time for the Saturn probe is about 450 days including about 350 days of powered time; longer trip times use lower thrust and higher specific impulse acclerators and as a result require somewhat longer powered times. Similarly, the zero-payload trip time for the Jupiter satellite mission is about 500 days. Although there is a coast period during which the accelerator does not operate, it is not presently considered feasible to remotely shut down and restart a nuclear reactor at interplanetary distances. The times involved are even greater, of course, for power supplies with higher specific weight. Thus, in order for these missions to become feasible it will be necessary to develop the capability of operating power supplies continuously over periods of up to two or three years, unless power supplies of substantially lower specific weight than those assumed herein and higher efficiencies for accelerators are developed first.

Finally, it should be noted that some of the conclusions deduced from the results of this study can be generalized somewhat. Thus the characteristic velocity requirements for a Jupiter capture mission (i.e., establishing a highly eccentric elliptical orbit around Jupiter) are approximately the same as for a Saturn probe, so that the conclusions reached herein regarding the Saturn probe mission are applicable to the Jupiter capture mission as well. Similarly, the characteristic velocity requirements for Mars and Venus satellites are similar, so that the conclusions reached herein for the Venus mission can be expected to hold for the Mars satellite mission as well.

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LIST OF SYMBOLS

q	Heat transfer rate
k	Thermal conductivity
W	Weight
t	Trip time
T	Temperature
A	Area
d	Thickness
R	Radius
V	Vaporization rate
ℓ	Heat capacity
ρ	Density
ΔV	Characteristic velocity impulse
V_∞	Hyperbolic excess velocity
$V_{\infty h}$	Hyperbolic excess velocity for combined high- and low-thrust system
X	Numerical constant
Y	Numerical constant
F	Thrust
η	Efficiency
P	Input power
I	Specific impulse
g	32.2 ft/sec^2
L	Ion engine constant

LIST OF SYMBOLS
(Contd.)

J	Low thrust performance parameter
α	Specific power

Subscripts

p	Propellant
pp	Power supply
o	Initial condition
t	Tank
i	Insulation
h	Hydrogen
L	Payload
s	Sun
v	Vaporized
f	Fixed
b	Bulk
u	Utilization

APPENDIX I

SPACECRAFT WEIGHTS

The gross weight of an upper stage is assumed to consist of the following component weights: propellant, propellant tankage, propellant insulation, residual and reserve propellant, engine, power supply. structure, and payload. Propellant is calculated as the amount needed to perform the mission only, the weight of vaporized propellant being considered as part of the insulation system weight.

For the purpose of estimating propellant tank weights the curve of Fig. 31 was used. This curve, which is calculated from Eq. (3) for propellant weights above 10,000 lb, is in good agreement with existing data for stages with propellant weights in the range of 20,000 to 30,000 lb.

$$W_t = 0.49 \frac{W_P'}{\rho_b} \tag{3}$$

In this equation propellant weight W_P' includes vaporized propellant (if any) and is expressed in pounds, while bulk density ρ_b is in lb/ft^3. For W_P' less than about 10,000 lb, Eq. (3) is not applicable because of the low surface-to-volume ratios of the tanks and also because minimum-gauge limitations prevent reduction of the tank skin thickness. At the low weights an empirical correlation was made with available data for small stages.

Residual and reserve propellant is assumed to be 2% of W_P' for the nuclear and chemical stages, but this item is assumed to be negligible in the calculation of ion-propelled stage weight. Structure accounts for 3 1/3% of the gross stage weight, while engine weight is fixed at 200 lb for the ion-propelled stage and 270 lb for the chemical stage. As explained in the text a weight of 1000 lb plus one lb per mw of power is the estimated weight of the nuclear propulsion unit, including reactor, rocket nozzle and associated equipment. Weight of the insulation system is discussed in Appendix II.

Typical weight breakdowns for several vehicles are shown in Table I.

APPENDIX II

THERMAL PROTECTION SYSTEM

When high-thrust upper stages are used to perform the Venus satellite, Jupiter satellite, and lunar missions, provision must be made for a thermal protection system to prevent excessive vaporization of the cryogenic propellants. A thorough analysis of the insulation problem is not feasible here because such a study would in turn require extensive analysis of the structural design of each vehicle. What is desired is a reasonable estimate of the amount of insulation necessary to protect a vehicle layout which would be typical of the vehicles considered in this study. In order to achieve this end the following simplifying assumptions were made:

1. Only cylindrical propellant tanks are considered, although this assumption imposes a slight weight penalty when the tanks are small.

2. Since the Sun is always the greatest external source of heat flux and a significant fraction of the total heat flux, one end of the vehicle is continuously pointed in the direction of the Sun.

3. A constant-temperature body (520 R) which represents the payload, is placed at the opposite end. The resulting vehicle layout is depicted in Fig. 32.

4. The propellant tanks are always full.

5. Propellant and tank wall temperatures are constant and equal, and all heat transferred to the propellants is accounted for by the resultant vaporization.

6. The insulation consists of successive layers of reflective foils.

7. Conduction heat leaks due to structural members, vents, pipes, etc., are neglected.

With the vehicle layout shown in Fig. 32 there are but two sources of heat flux to the propellant tanks, namely the constant-temperature bodies at each end. Radiation from the propellant tanks to space is neglected because the resultant heat flux is always at least an order of magnitude less than that from either constant-temperature body.

When both hydrogen and oxygen are transported, the oxygen tank should be adjacent to the greater source of heat, since the heat flux is proportional to the temperature difference between propellant and heat source. The hydrogen, because of its lower storage temperature (30 R), is more difficult to protect than the oxygen (140 R).

The analysis was carried out separately for nuclear and chemical upper stages. The difference between the two is that the nuclear stage requires storage of only one propellant, H_2, while both O_2 and H_2 must be stored for the chemical system. The latter is the more difficult to analyze because it involves consideration of heat transfer between propellants as well as between propellants and heat sources.

Nuclear Stage

The heat flux through superinsulation of the types considered in Ref. 15 is closely represented by a conduction equation of the form

$$q = \frac{k(T - T_P)A}{d} \tag{4}$$

where T is the temperature of the heat source and T_P that of the propellant, d is the insulation thickness, A the area over which heat is transferred and k an "effective" thermal conductivity. For hot side temperatures of several hundred degrees Rankine and cold side temperatures comparable to that of liquid H_2, the effective thermal conductivity of Fiberglass mat insulation determined in numerous experiments is 2.5×10^{-5} Btu/hr-ft R (Ref. 13). As a conservative estimate this conductivity is doubled in the calculations of this section.

In Fig. 32, the subscripts S and L refer to the constant temperature bodies on the Sun and payload ends respectively. Thus for the nuclear stage the total heat flux rate which must be absorbed by the H_2 is given by

$$q_h = 2\pi R^2 k \left(\frac{T_L - T_h}{d_L} + \frac{T_S - T_h}{d_S} \right) \tag{5}$$

and, assuming the tank is always full, the resultant vaporization rate is

$$V_h = \frac{q_h}{\ell_h W_h} \tag{6}$$

During the trip time, t, the weight of H_2 vaporized must therefore be

$$W_v = \frac{2\pi R^2 k t}{\ell_h} \left(\frac{T_L - T_h}{d_L} + \frac{T_S - T_h}{d_S} \right) \tag{7}$$

and the total weight of insulation is

$$W_i = \pi R^2 \rho_i (d_S + d_L) \quad (8)$$

The purpose of the insulation system is to maximize the weight of H_2, W_h, which is not vaporized if the initial stage weight is fixed. Therefore, since

$$W_P = W_h - W_v \quad (9)$$

and since

$$W_O = W_t + W_i + W_h + W_f \quad (10)$$

where W_t is the tank weight and W_f is the remaining fixed weight of the stage, the quantity to be maximized is

$$W_P = W_O - W_t - W_i - W_f - W_v \quad (11)$$

The tank weight is closely approximated by the empirical equation

$$W_t = .49 \frac{W_h}{\rho_h} \quad (12)$$

where ρ_h is the H_2 density in lb/ft^3. Using Eqs. (7), (8), (11), and (12), the resulting expression for W_P becomes

$$W_P = \frac{W_O - W_f}{1 + \frac{.49}{\rho_h}} - \frac{\pi R^2 \rho_i (d_L + d_S)}{1 + \frac{.49}{\rho_h}} - \frac{2\pi R^2 kt}{\ell_h}\left(\frac{T_L - T_h}{d_L} + \frac{T_S - T_h}{d_S}\right) \quad (13)$$

If the derivatives with respect to d_L and d_S are made stationary the relations

$$d_L = \sqrt{\frac{2kt\,(T_L - T_h)\left(1 + \frac{.49}{\rho_h}\right)}{\rho_i\, \ell_h}} \quad (14)$$

and

$$d_S = \sqrt{\frac{2kt\,(T_S - T_h)\left(1 + \frac{.49}{\rho_h}\right)}{\rho_i \;\, \ell_h}} \qquad (15)$$

are found.

The resulting expressions for the insulation weight and weight of H_2 vaporized then become

$$W_i = X t^{1/2} \, W_P^{2/3} \qquad (16)$$

and

$$W_v = Y t^{1/2} \, W_h^{2/3} \qquad (17)$$

where the constants X and Y are functions of ρ_i , ρ_h , k , ℓ_h , T_L , T_S and the tank length to diameter ratio which is assumed to be 1.0. The payload temperature T_L is 520 R and the source temperature T_S depends upon the mission. As a conservative estimate the value of T_S is assumed to be the equilibrium temperature of a body at a distance from the Sun which is the closest approach distance for the mission under consideration.

Curves generated by Eqs. (16) and (17) are illustrated in Fig. 33 for the three missions which require a terminal maneuver. The insulation system weight consists of both the insulation and the weight of vaporized propellant. The trip time noted for each curve is representative of that required for the indicated mission.

Chemical Stage

The analysis for the chemical stage entails a similar optimization but is complicated by the fact that there are two propellants to protect and three surfaces to insulate. In addition the mixture ratio at the destination should be specified but the initial mixture ratio will depend upon the amounts of H_2 and O_2 vaporized during the trip and the mixture ratio desired at the destination.

Since the analysis for this case is similar in nature to that of the nuclear stage the equations will not be included here. In general the resultant insulation system weights consist largely of the insulation itself. The weight of vaporized O_2 is ordinarily greater than that of vaporized H_2, but in the case of the lunar mission this trend is reversed, apparently because of the short trip times for this mission. Even for the lunar satellite mission however, the initial mixture ratio is less than that at the destination.

In Fig. 34 typical insulation system weights are shown for chemical upper stages. As in Fig. 33 the insulation system weight includes the weight of vaporized propellant as well as the weight of the insulation itself.

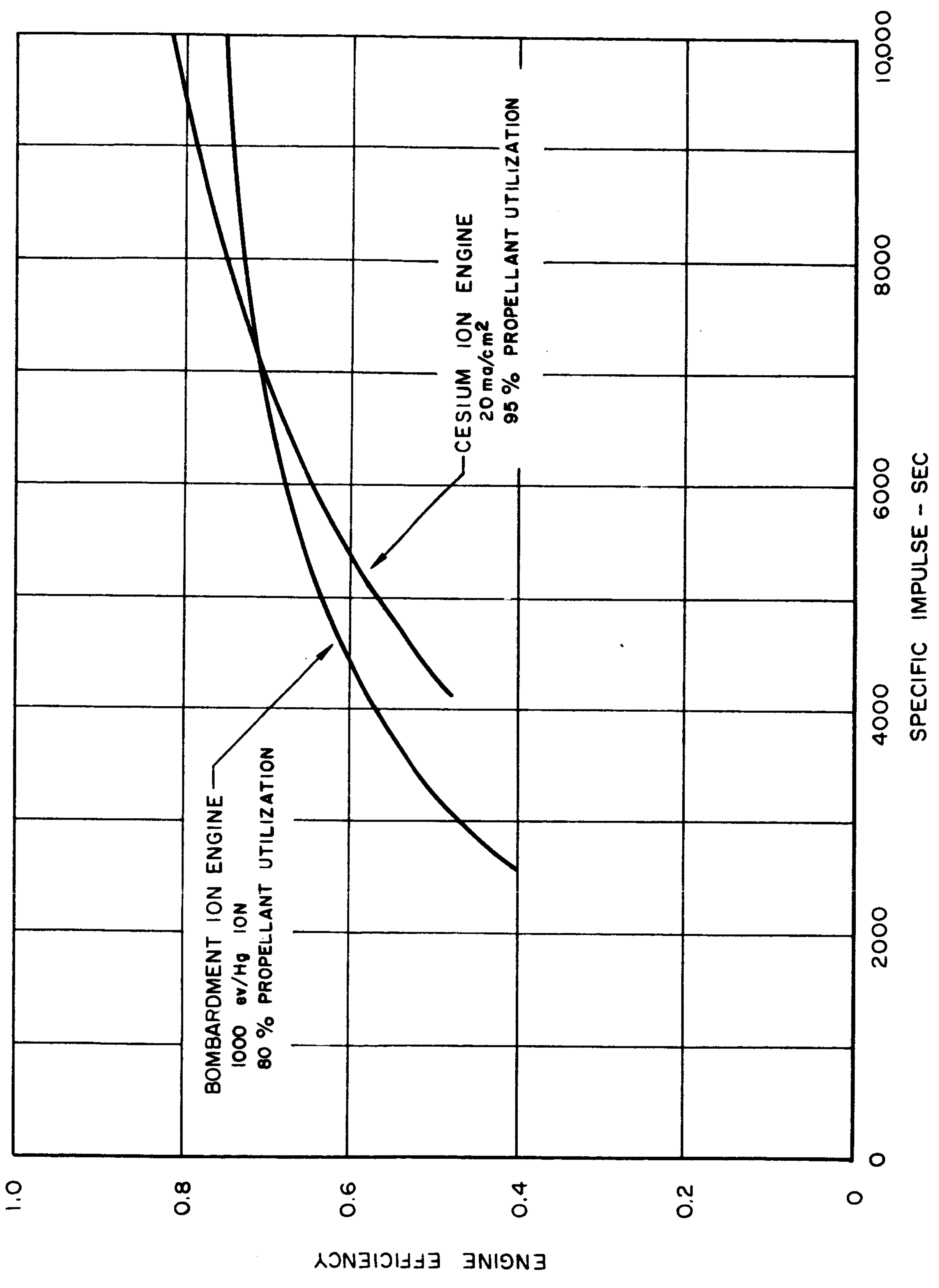
TABLE I

Typical Weight Breakdowns for Venus Satellite Mission

Saturn C-1 Launch Vehicle

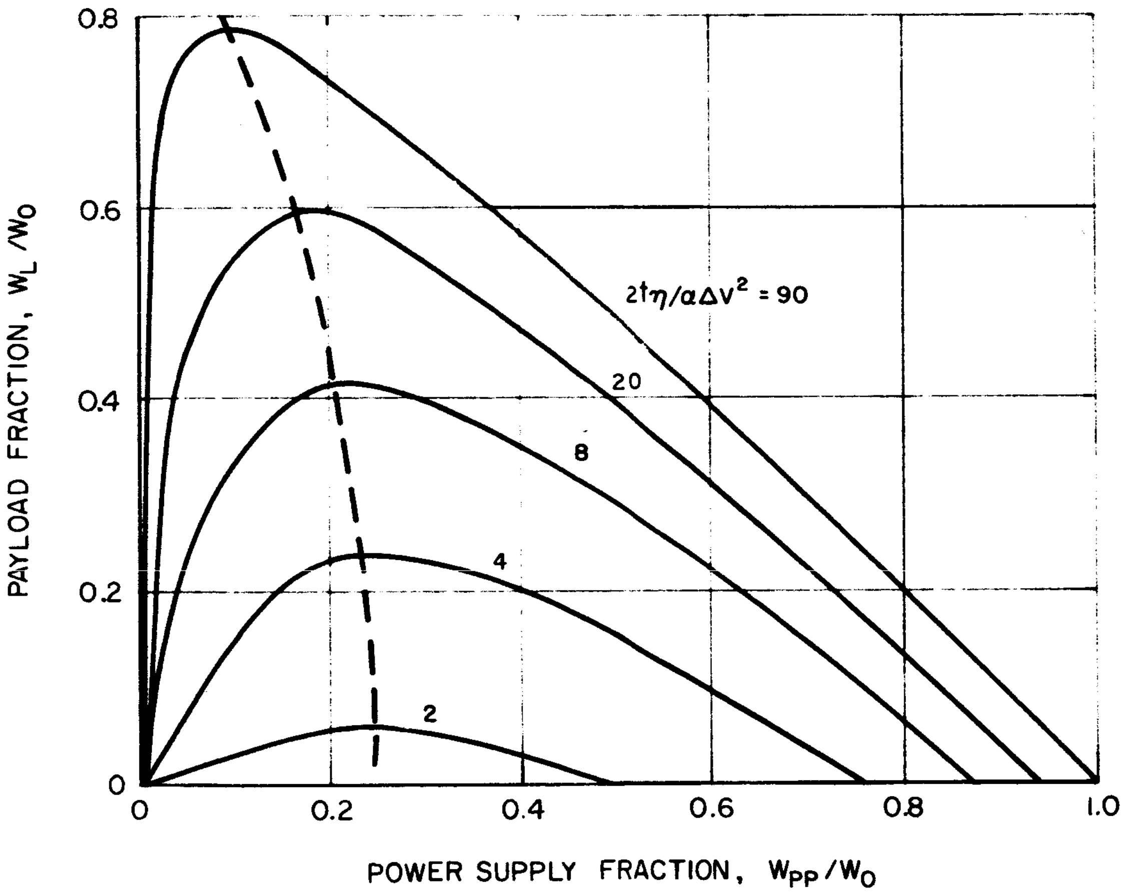
Staging	1 Ion-Propelled Stage	$1\frac{1}{2}$-Stage Nuclear	2-Stage H_2/O_2
Specific Impulse, sec	5000	800	420
Trip Time, days	165	140	140
Gross Weight	19,000	19,000	19,000
Structure Weight	595	948	826
Tankage Weight	417 (Tankage Weight and Residual & Reserve Propellant)	1104	274
Residual & Reserve Propellant		210	292
Insulation System Weight	---	398	180
Engine Weight	200	1190 Includes Reactor	320
Power Supply Weight	4750	---	---
Propellant Weight	8360	10,505	14,617
Payload Weight	4678	4645	2491

ION ENGINE EFFICIENCY



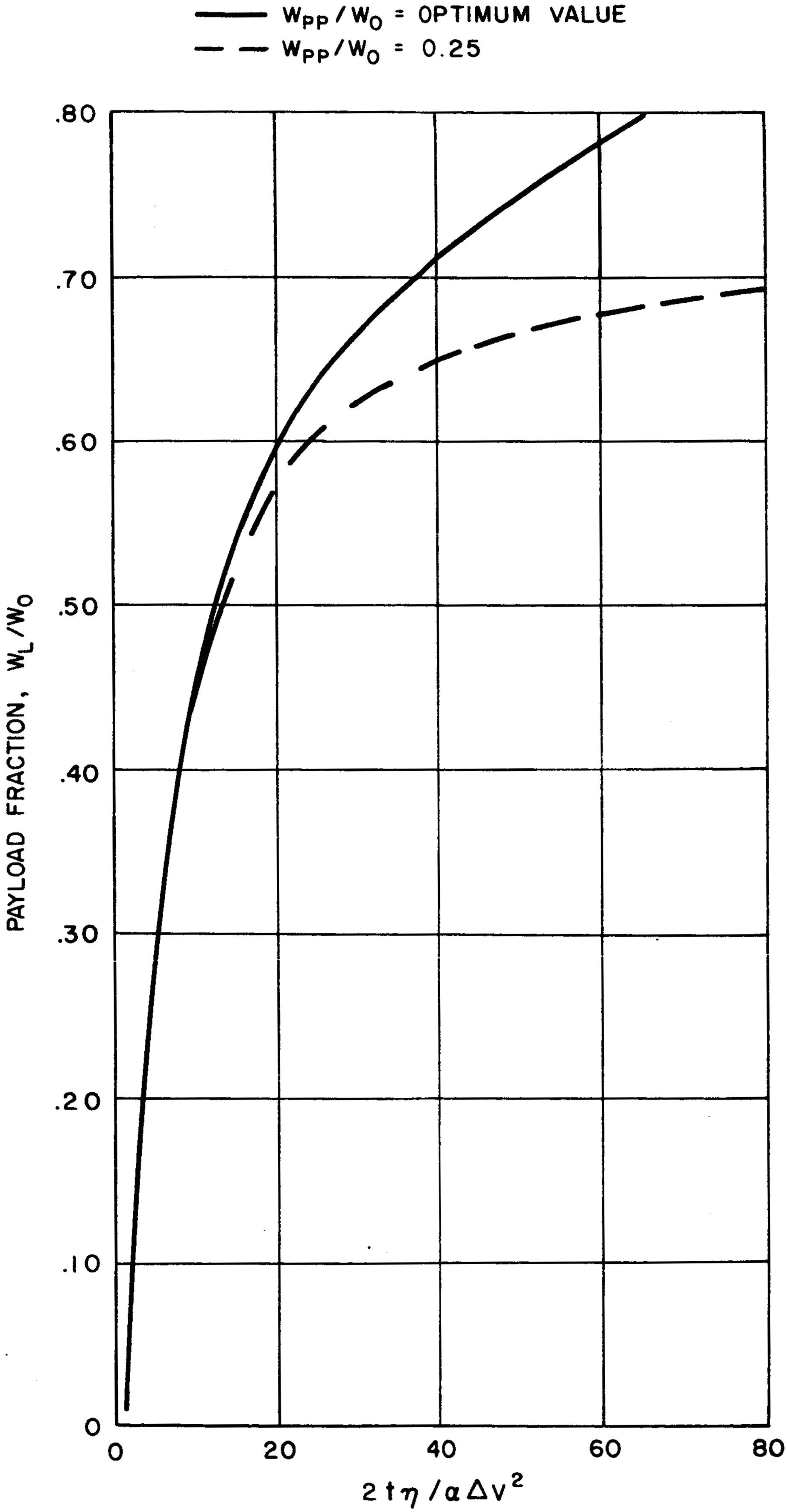
EFFECT OF POWER SUPPLY FRACTION ON ELECTRIC PROPULSION PERFORMANCE

FIELD-FREE SPACE, CONSTANT EFFICIENCY



IDEAL PAYLOAD FRACTIONS FOR MISSIONS IN FIELD-FREE SPACE

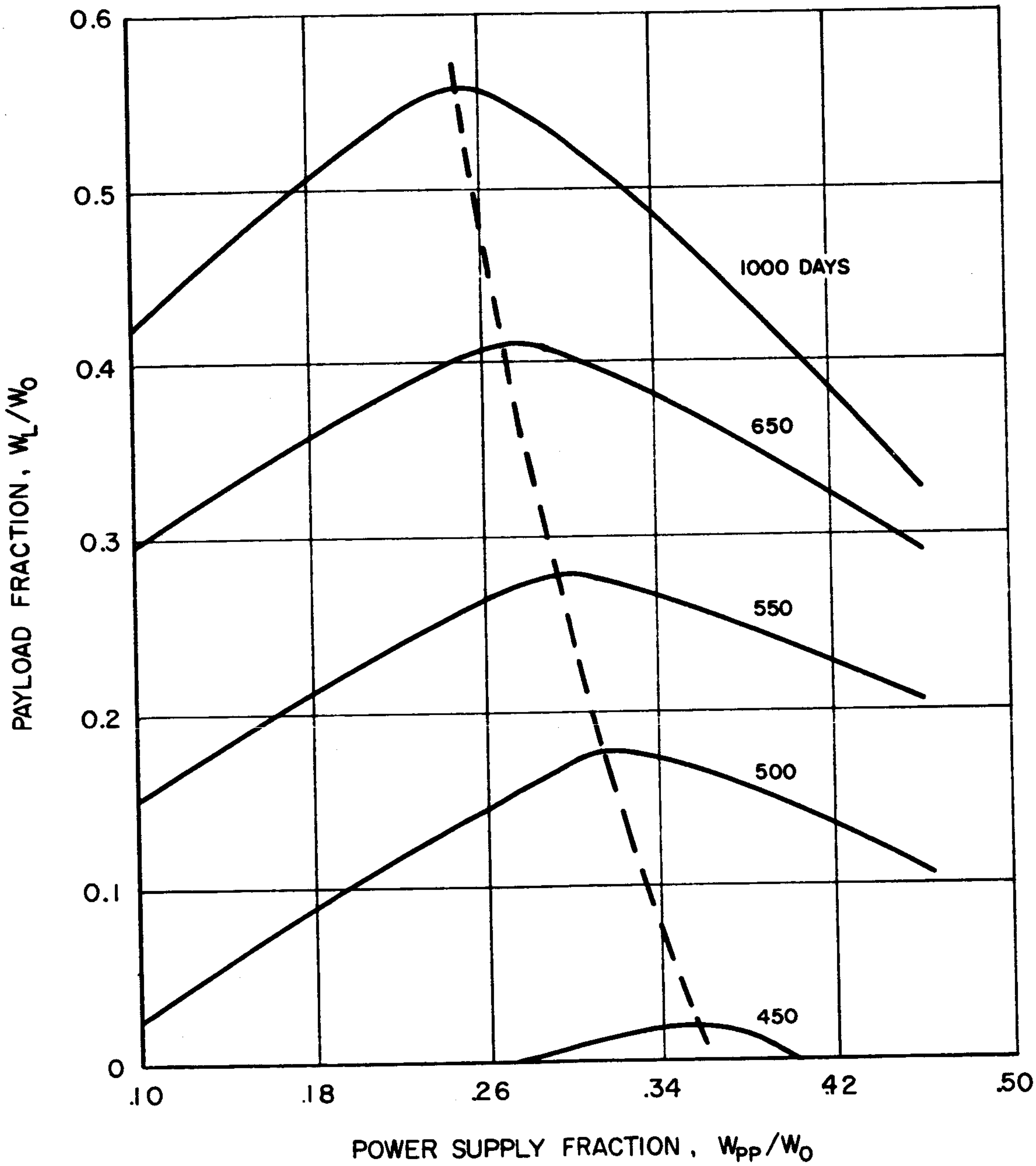
CONSTANT EFFICIENCY



EFFECT OF POWER SUPPLY FRACTION ON PAYLOAD FRACTION

SATURN PROBE MISSION

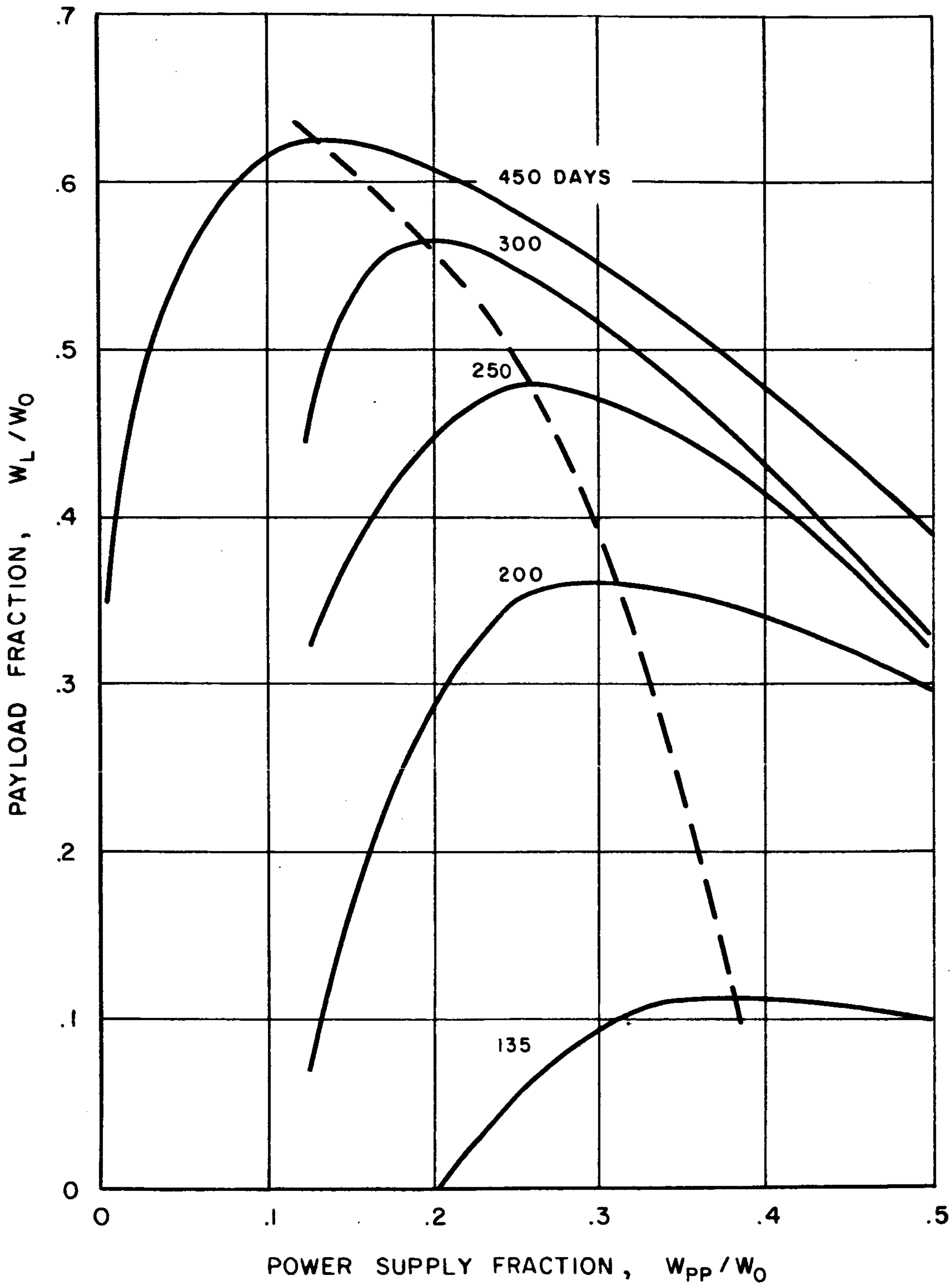
α = 10 LB/KW



EFFECT OF POWER SUPPLY FRACTION ON PAYLOAD FRACTION

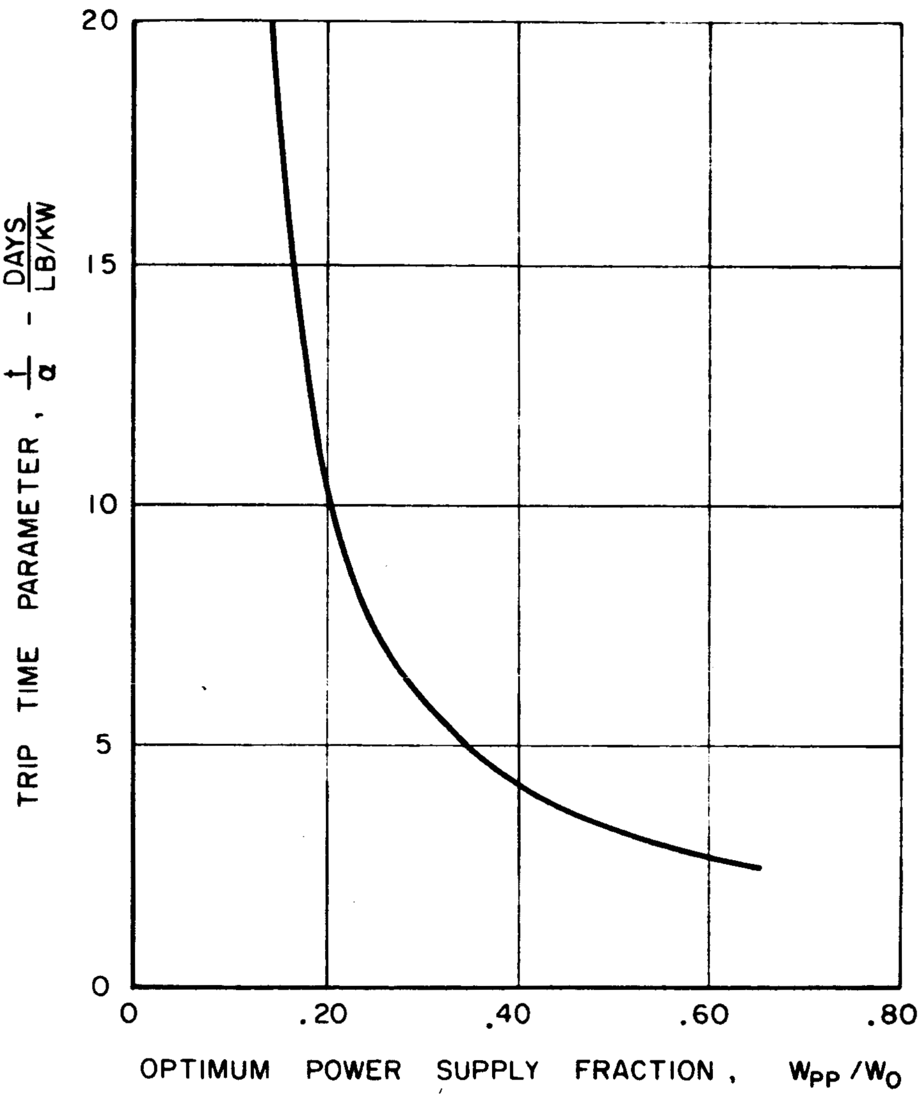
VENUS SATELLITE MISSION

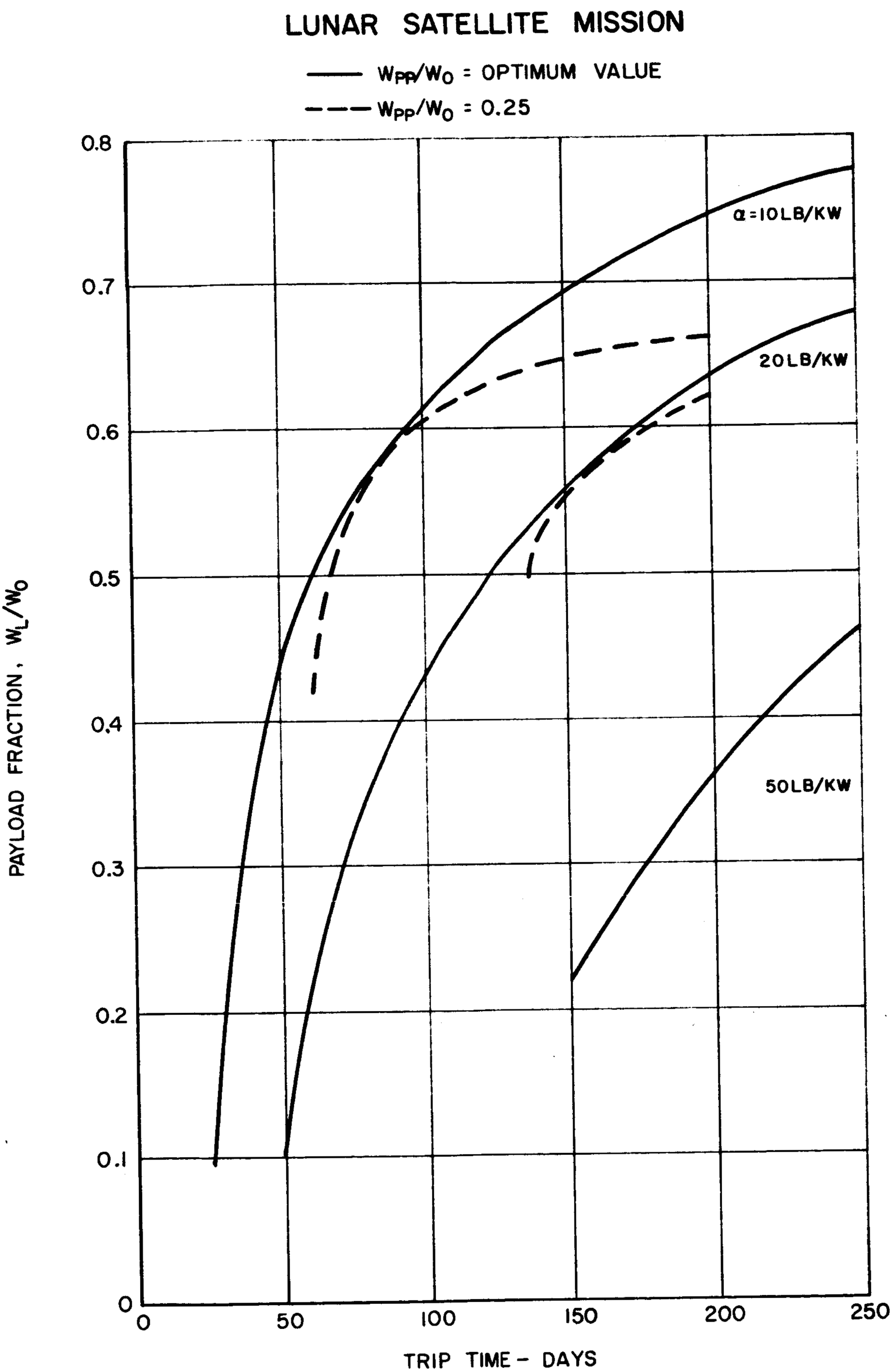
α = 10 LB/KW



LUNAR SATELLITE MISSION

OPTIMUM POWER SUPPLY WEIGHT



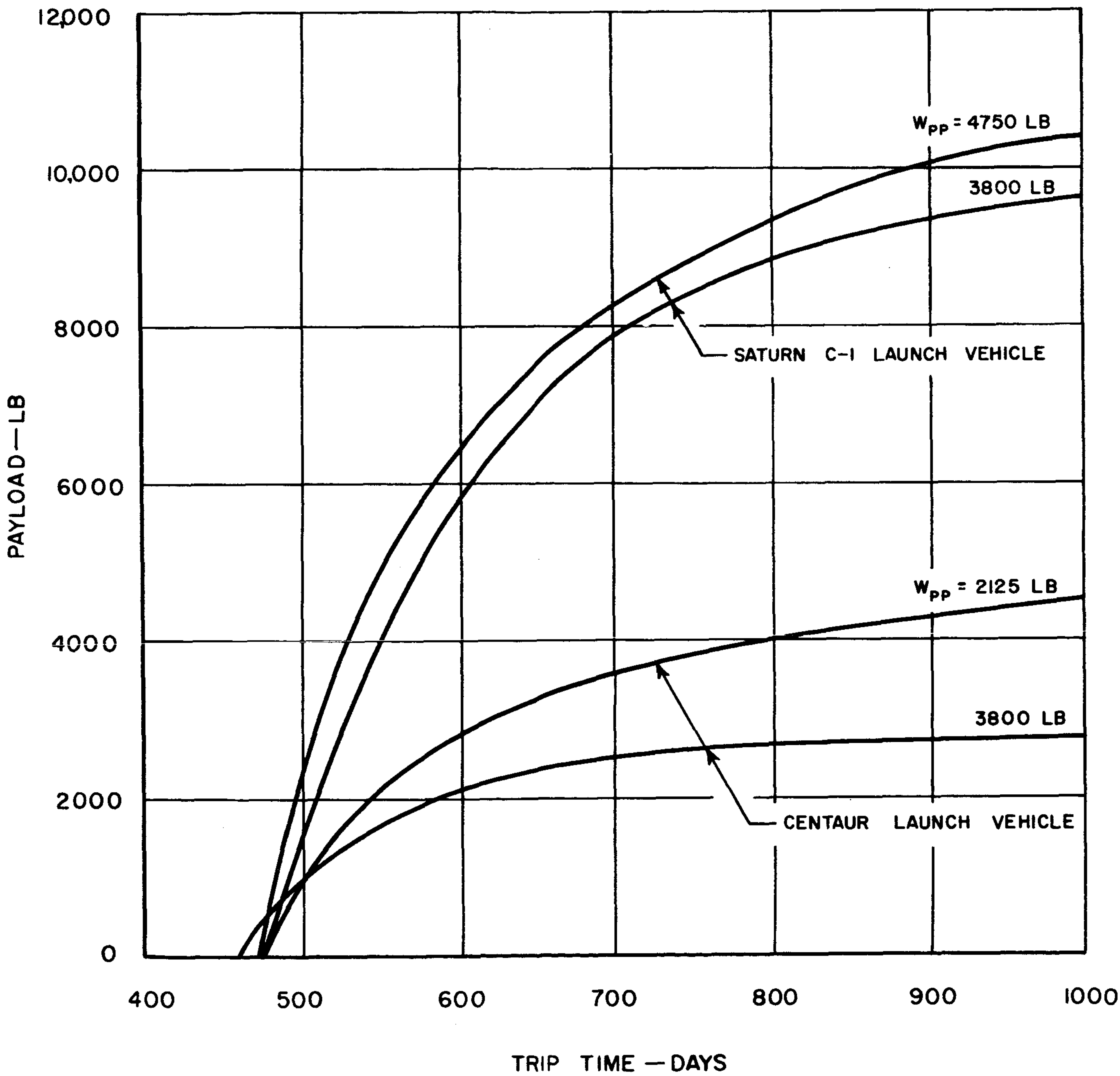


LUNAR SATELLITE MISSION
W_{PP}/W_O = OPTIMUM VALUE
W_{PP}/W_O = 0.25
α = 10 LB/KW
20 LB/KW
50 LB/KW
PAYLOAD FRACTION, W_L/W_O
TRIP TIME - DAYS
0
0.1
0.2
0.3
0.4
0.5
0.6
0.7
0.8
0
50
100
150
200
250

USE OF SAME POWER SUPPLY IN TWO VEHICLES

SATURN PROBE MISSION

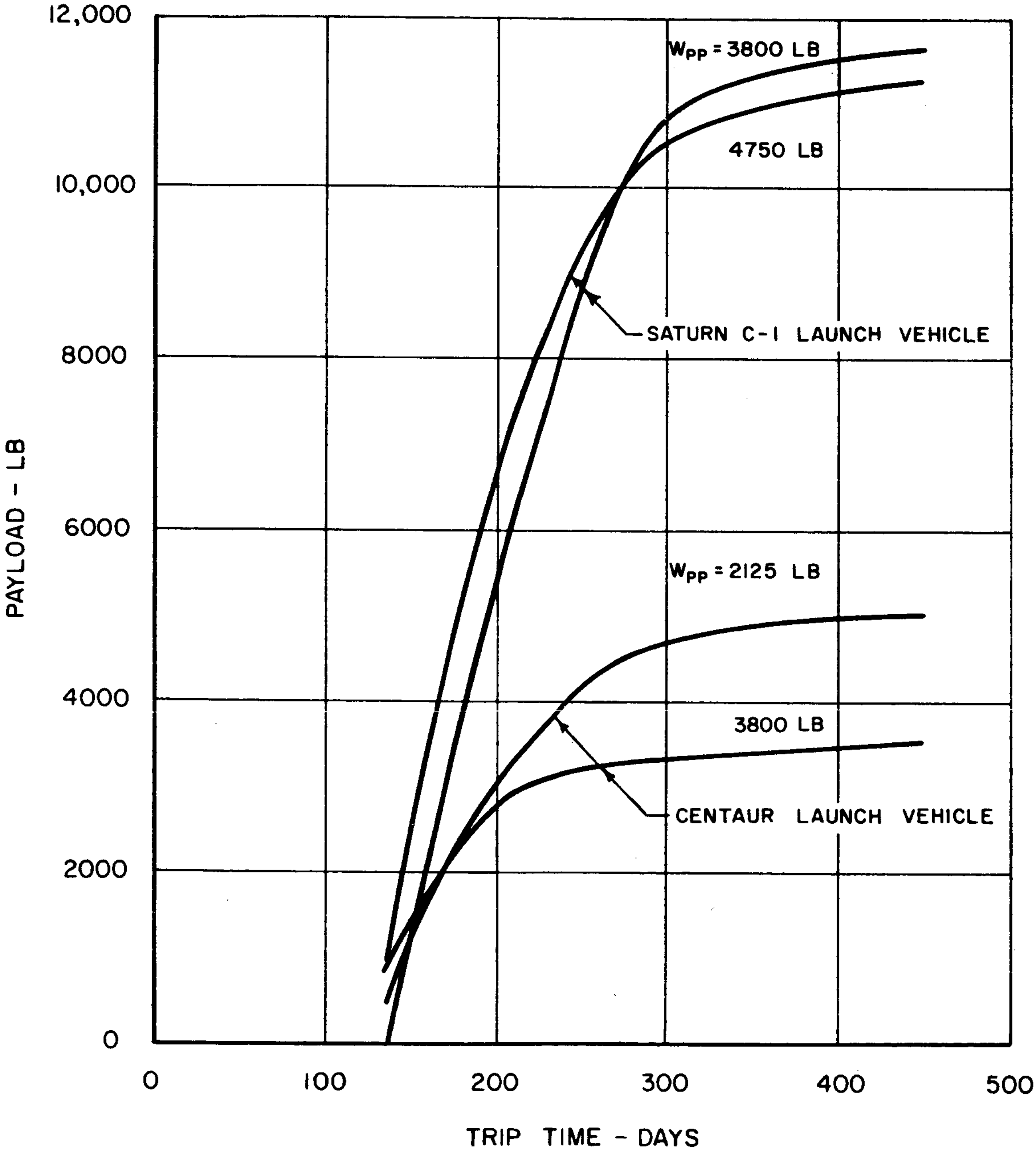
α = 10 LB/KW



USE OF SAME POWER SUPPLY IN TWO VEHICLES

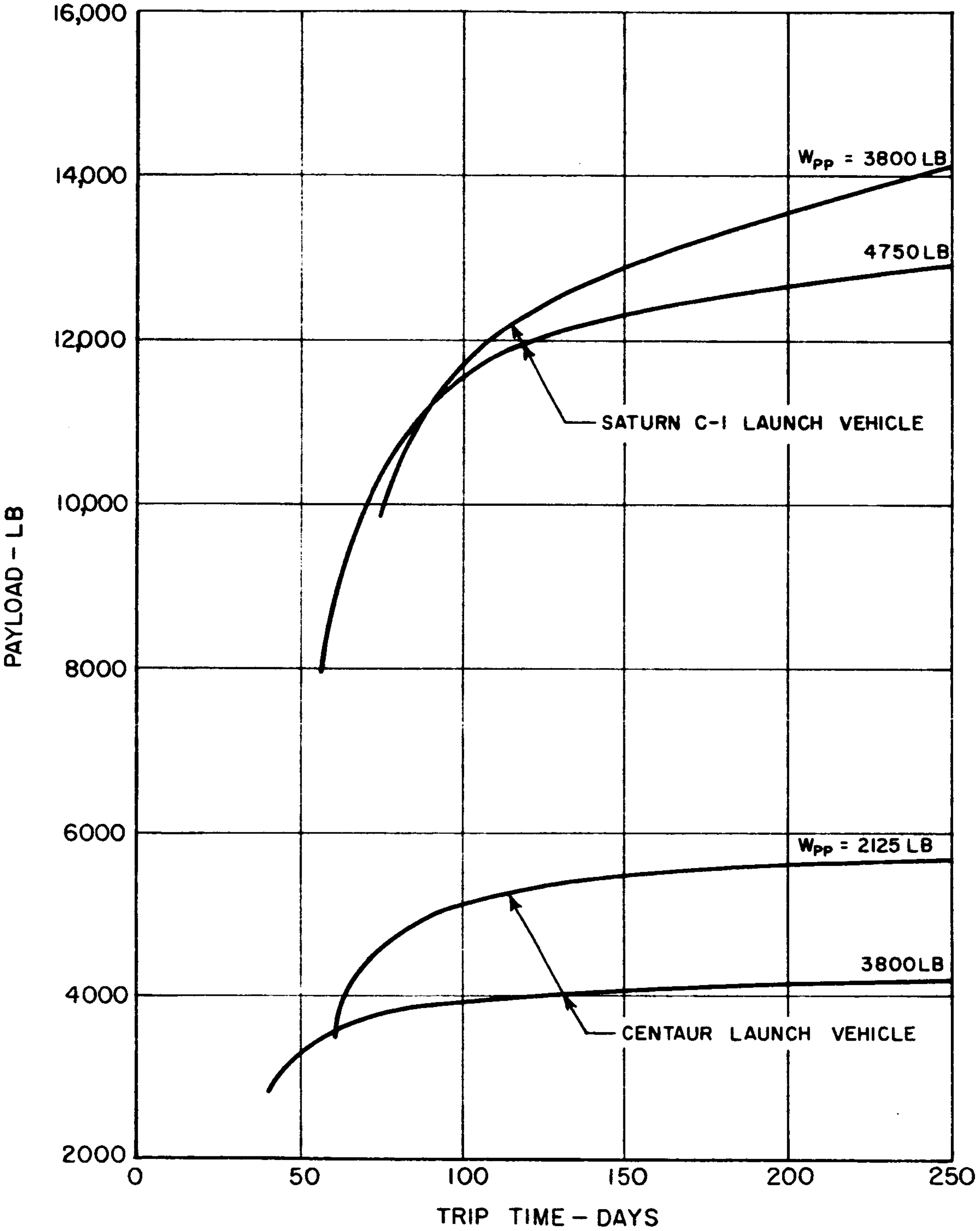
VENUS SATELLITE MISSION

α = 10 LB/KW



USE OF SAME POWER SUPPLY IN TWO VEHICLES

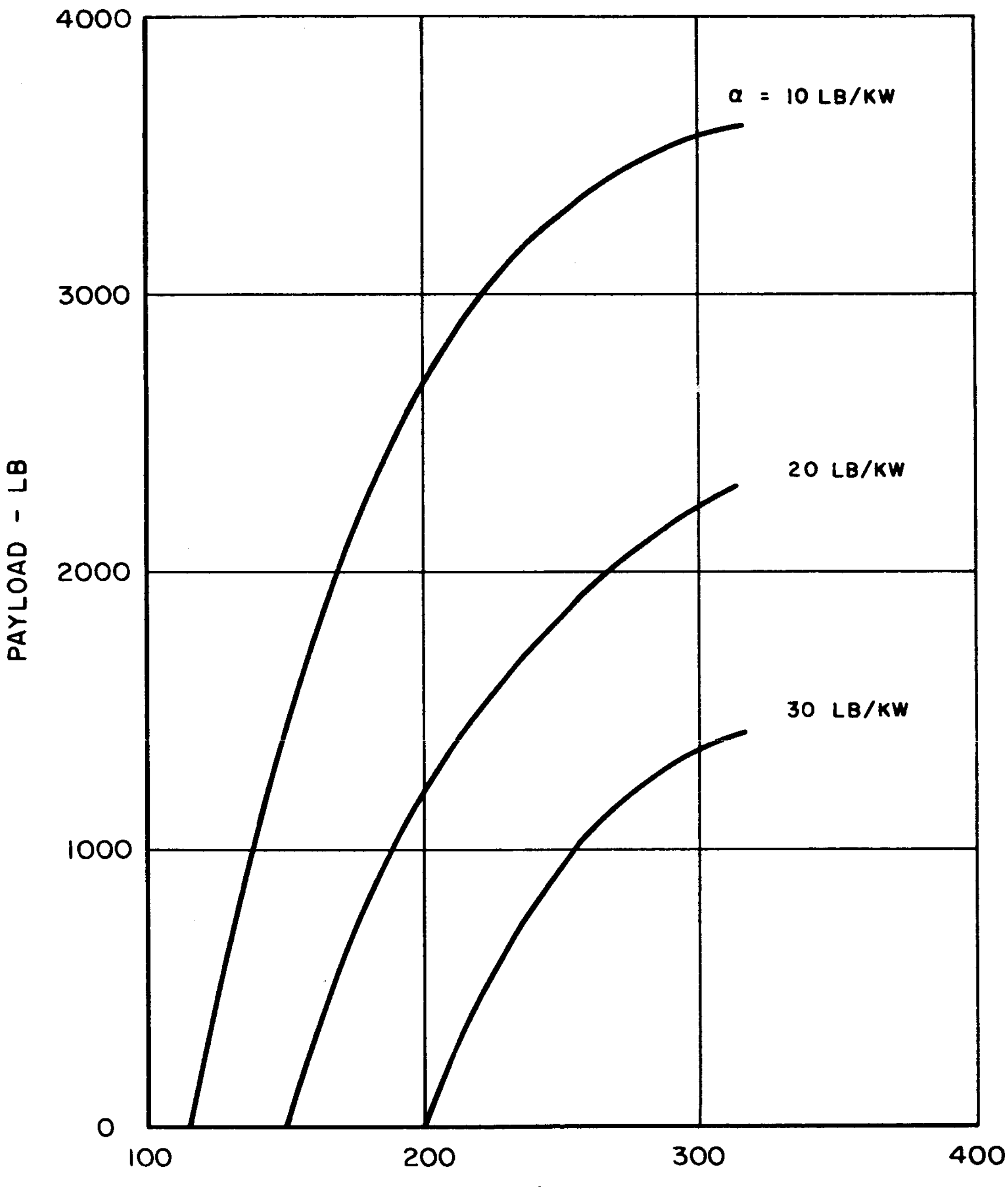
LUNAR SATELLITE MISSION



SOLAR PROBE MISSION

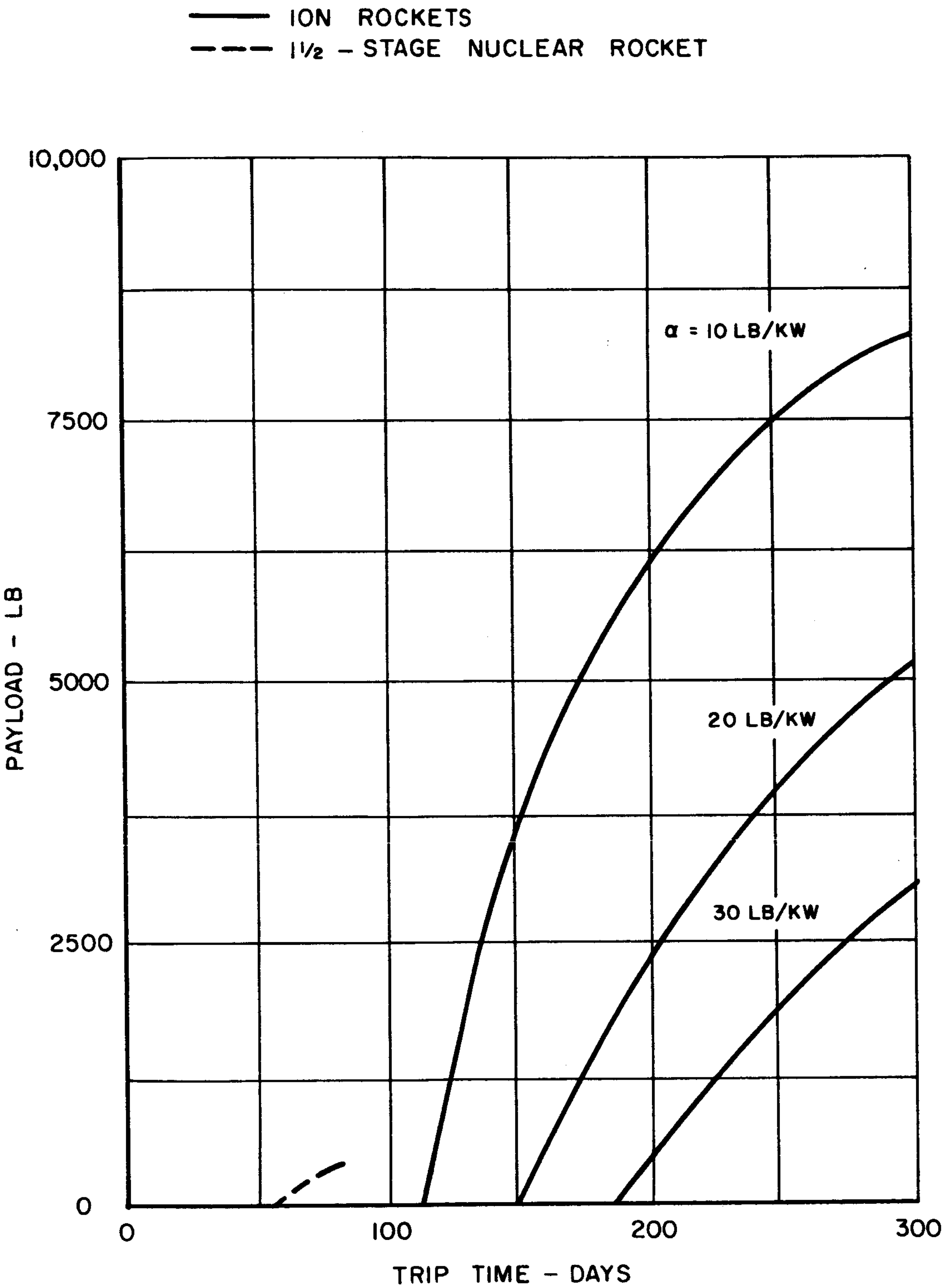
ATLAS—CENTAUR LAUNCH VEHICLE

POWER SUPPLY WEIGHT = 2125 LB



SOLAR PROBE MISSION

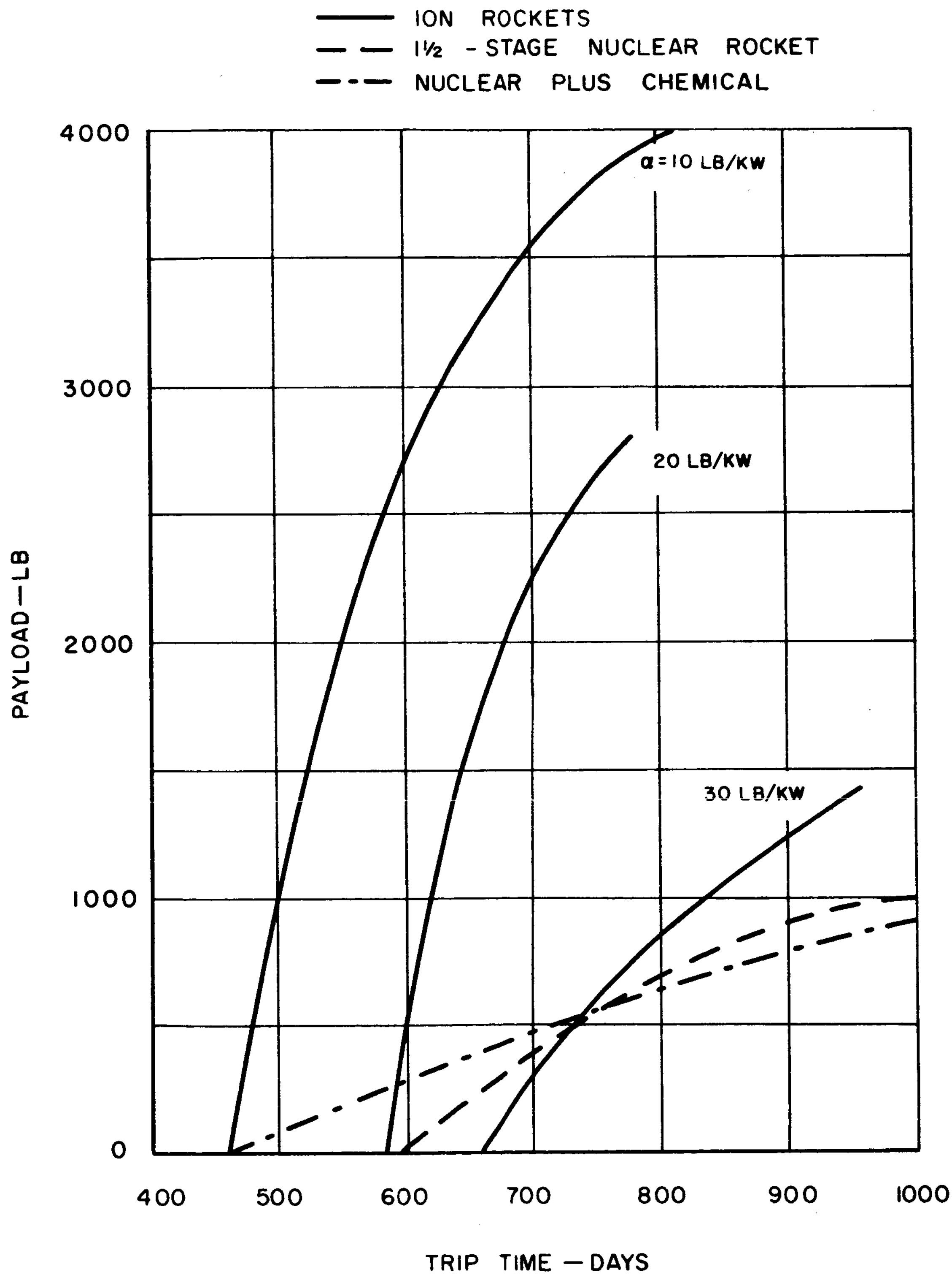
SATURN C-1 LAUNCH VEHICLE
POWER SUPPLY WEIGHT = 4750 LB



SATURN PROBE MISSION

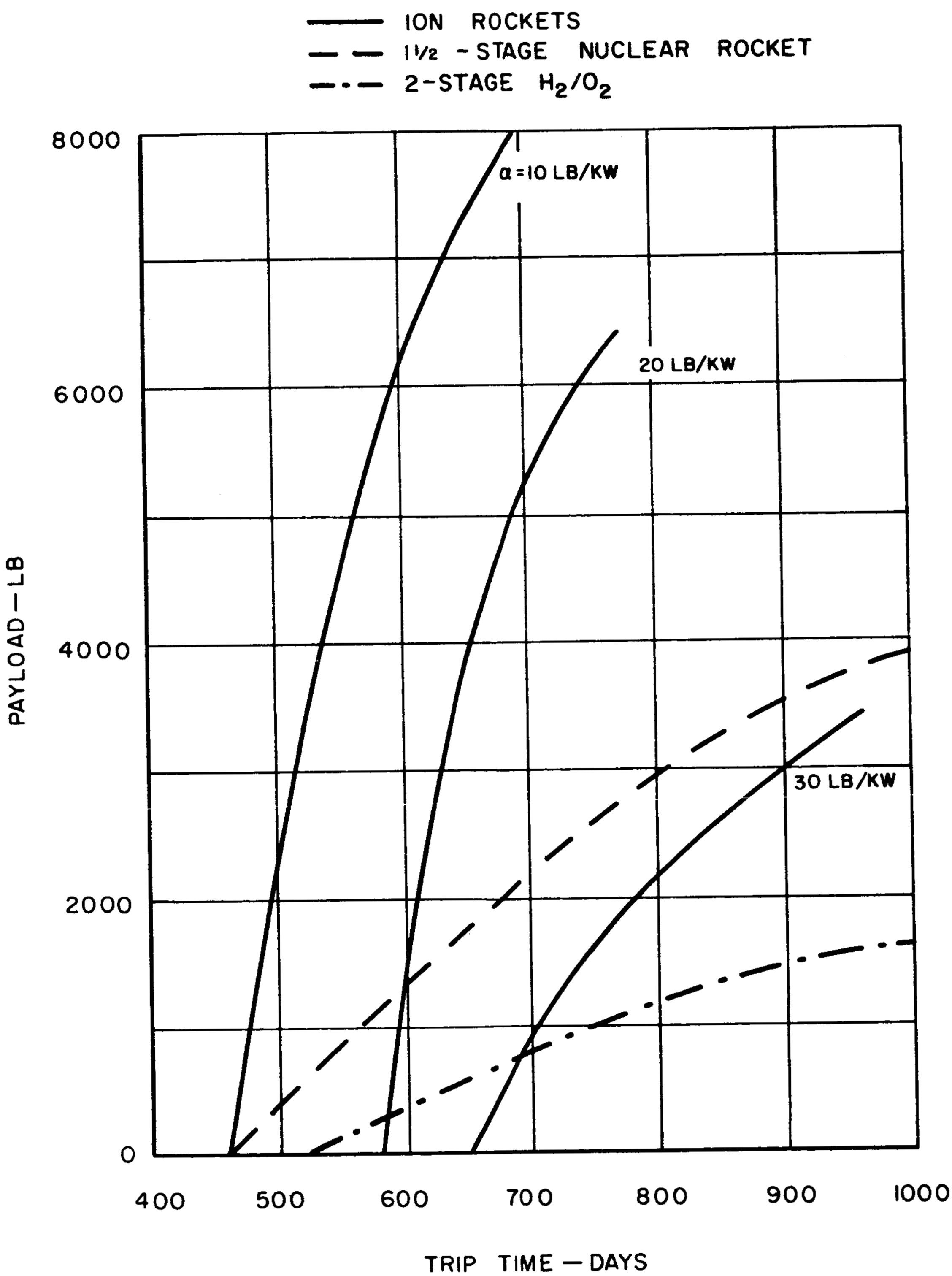
ATLAS—CENTAUR LAUNCH VEHICLE

POWER SUPPLY WEIGHT = 2125 LB



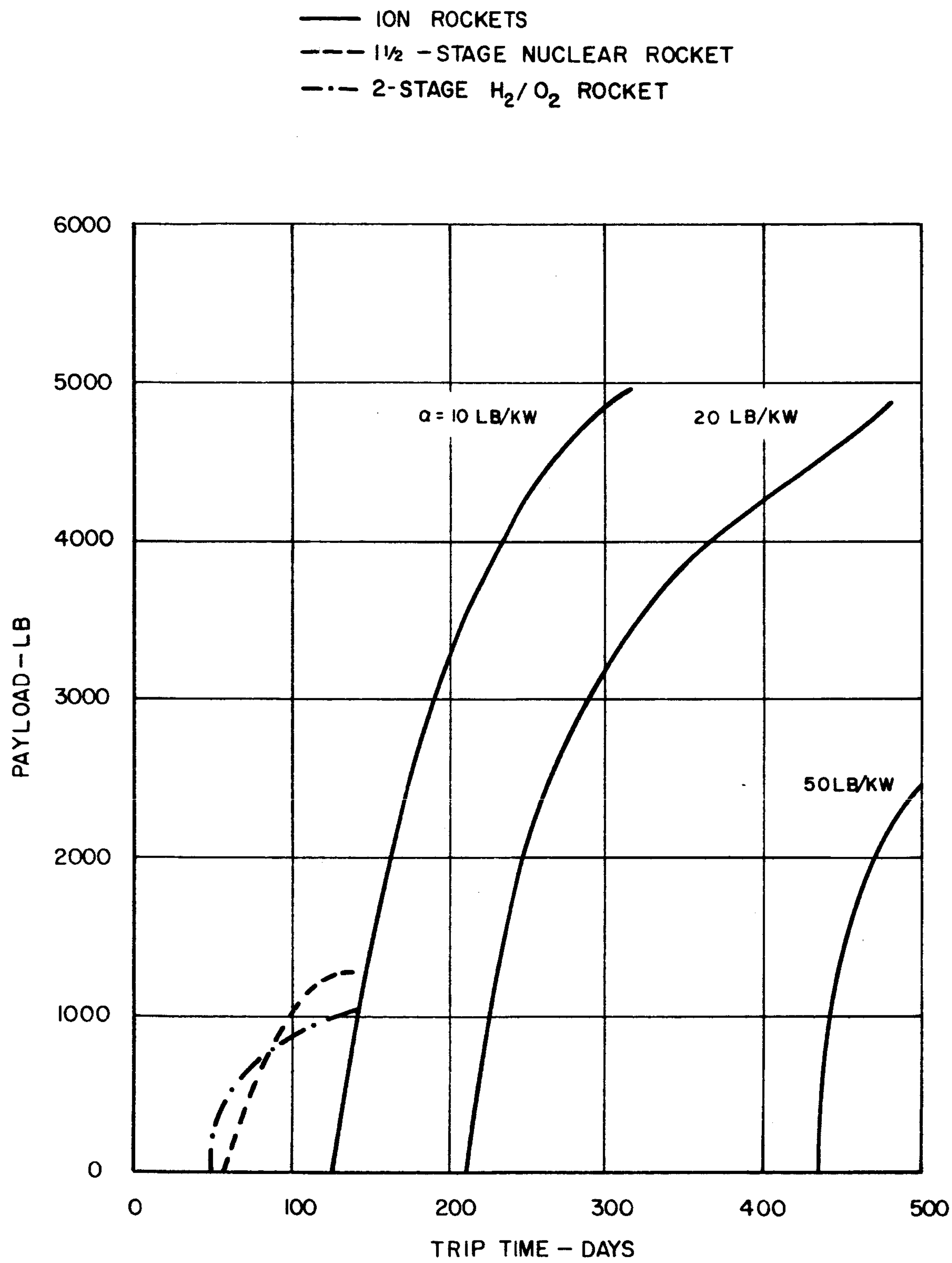
SATURN PROBE MISSION

SATURN C-1 LAUNCH VEHICLE
POWER SUPPLY WEIGHT = 4750 LB



VENUS SATELLITE MISSION

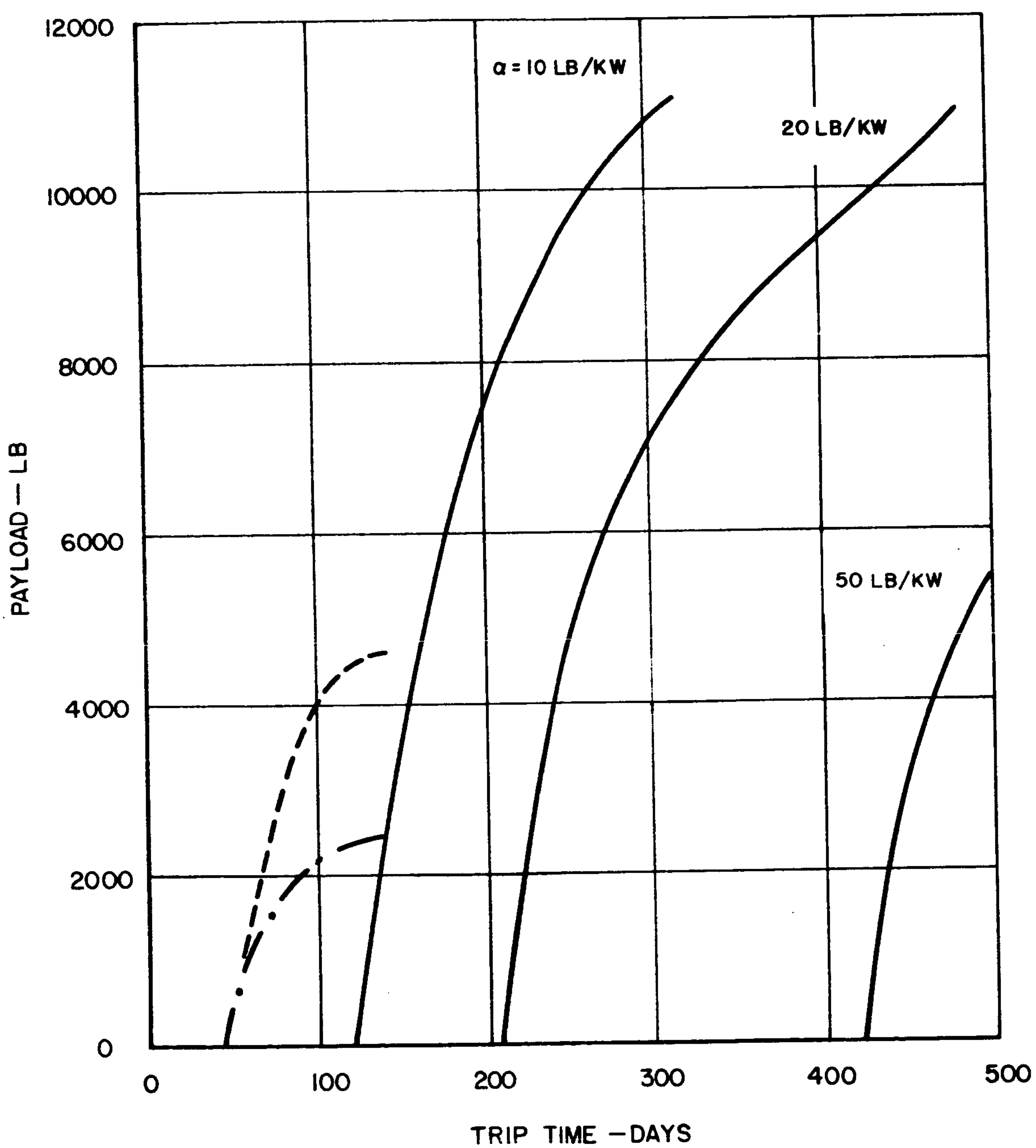
ATLAS-CENTAUR LAUNCH VEHICLE
POWER SUPPLY WEIGHT = 2125 LBS



VENUS SATELLITE MISSION

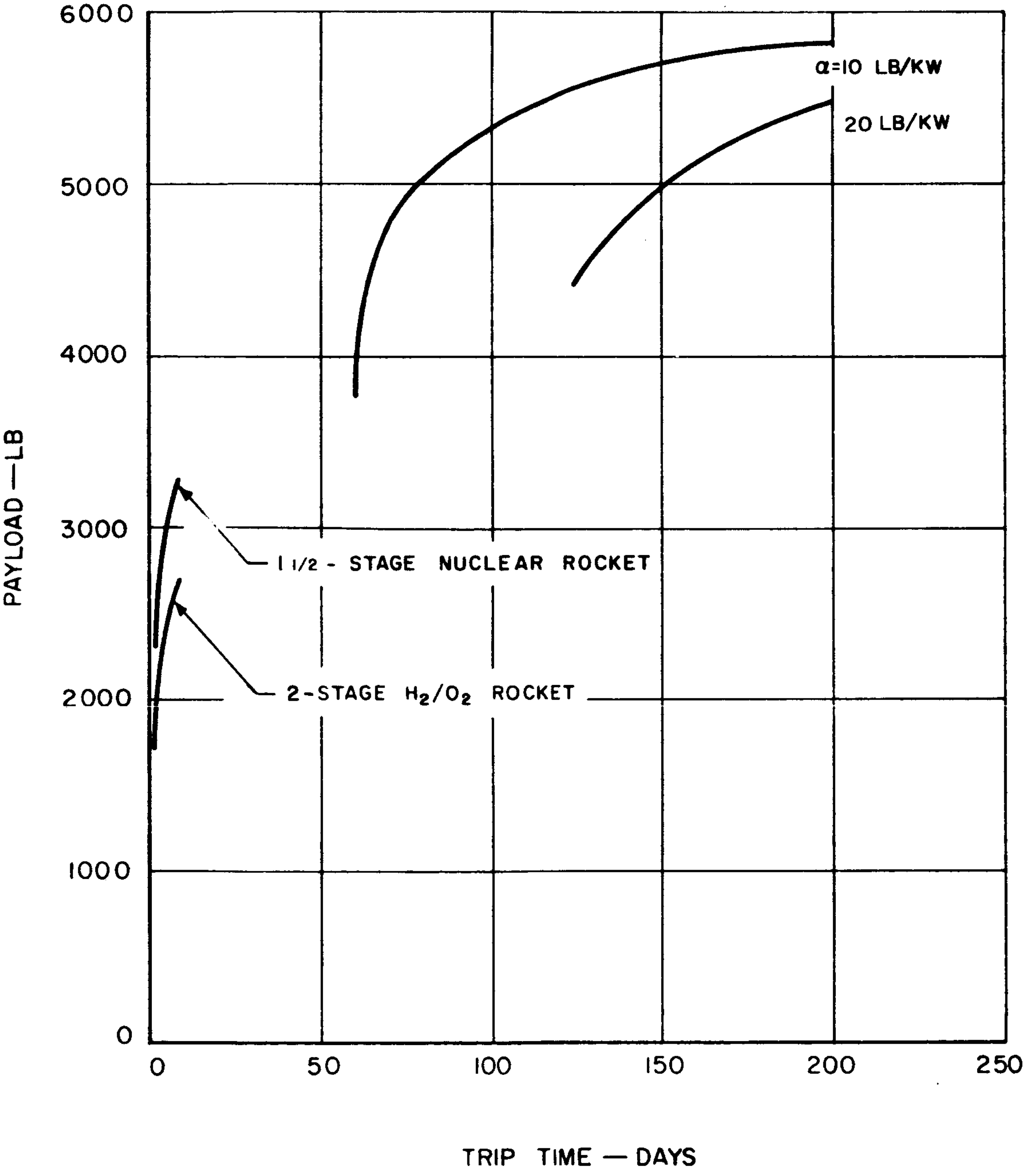
SATURN C-I LAUNCH VEHICLE
POWER SUPPLY WEIGHT = 4750 LBS

——— ION ROCKETS
- - - 1½ - STAGE NUCLEAR ROCKET
—·— 2-STAGE H_2/O_2 ROCKET



LUNAR SATELLITE MISSION

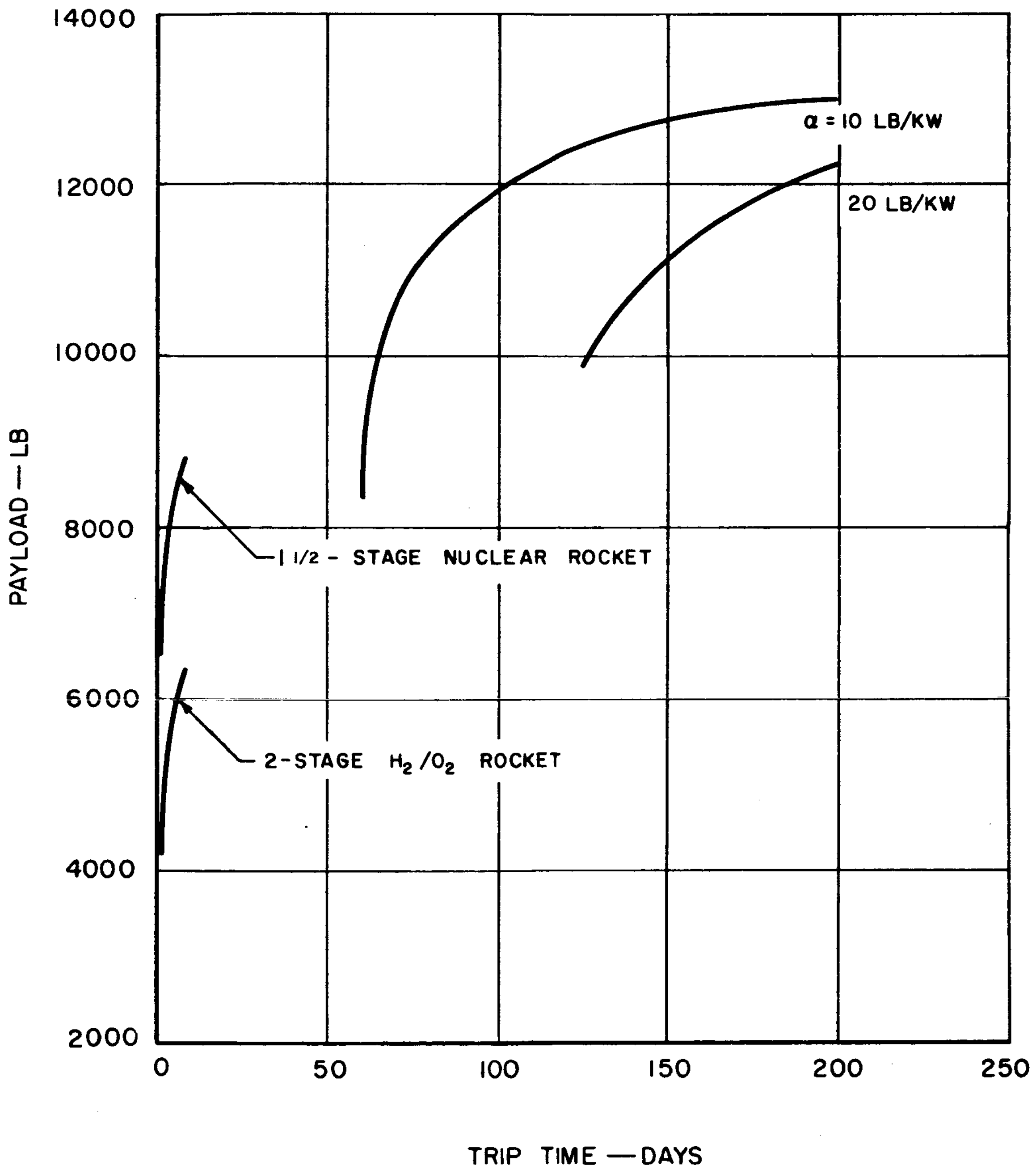
ATLAS—CENTAUR LAUNCH VEHICLE
POWER SUPPLY WEIGHT = 2125 LB

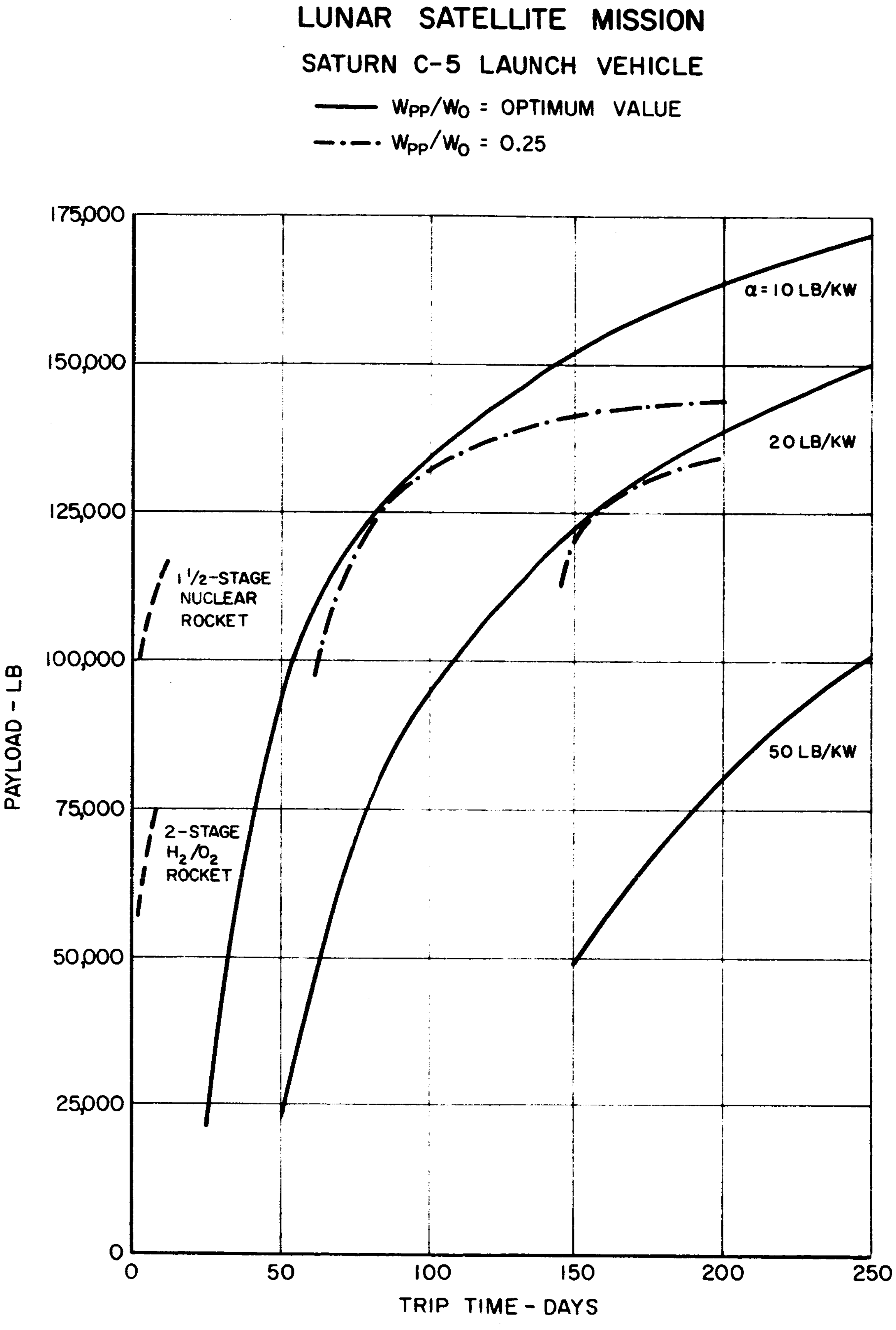


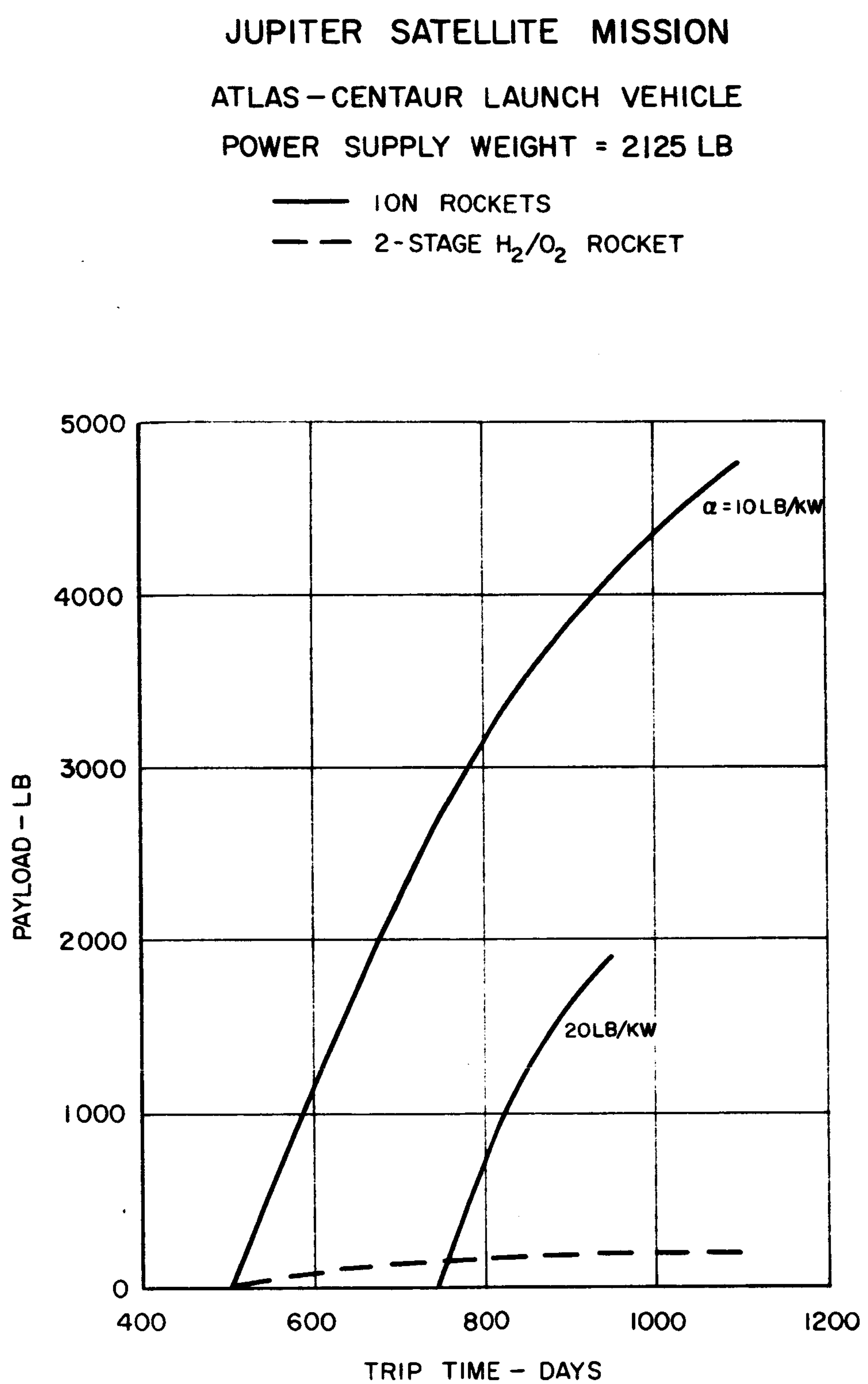
LUNAR SATELLITE MISSION

SATURN C-I LAUNCH VEHICLE

POWER SUPPLY WEIGHT = 4750 LB

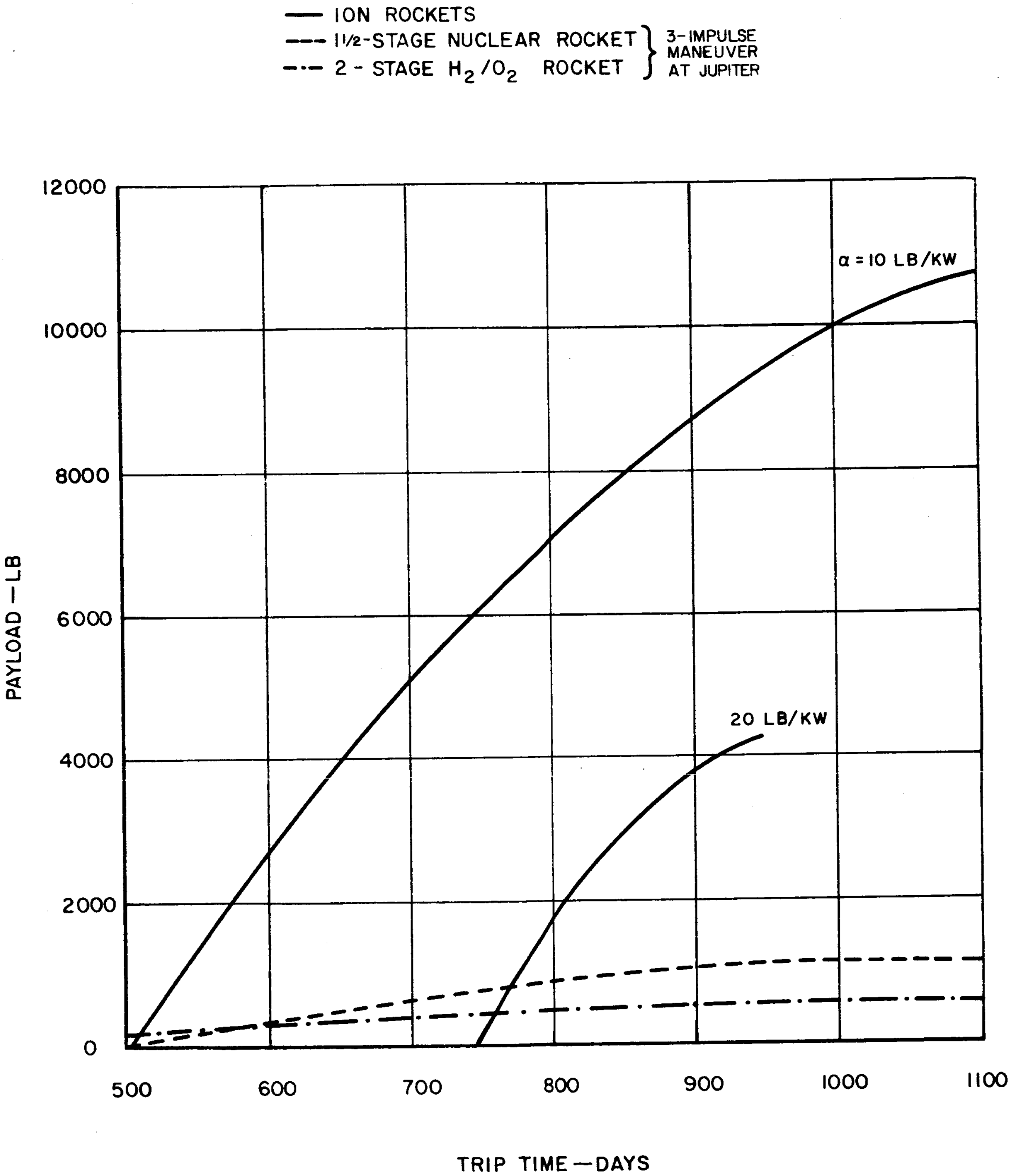



LUNAR SATELLITE MISSION
SATURN C-5 LAUNCH VEHICLE
W_{PP}/W_O = OPTIMUM VALUE
W_{PP}/W_O = 0.25
α = 10 LB/KW
20 LB/KW
50 LB/KW
1 1/2-STAGE NUCLEAR ROCKET
2-STAGE H_2/O_2 ROCKET
PAYLOAD - LB
TRIP TIME - DAYS
0
25,000
50,000
75,000
100,000
125,000
150,000
175,000
0
50
100
150
200
250


JUPITER SATELLITE MISSION
ATLAS-CENTAUR LAUNCH VEHICLE
POWER SUPPLY WEIGHT = 2125 LB
ION ROCKETS
2-STAGE H_2/O_2 ROCKET
PAYLOAD - LB
TRIP TIME - DAYS
α = 10 LB/KW
20 LB/KW
0
1000
2000
3000
4000
5000
400
600
800
1000
1200

JUPITER SATELLITE MISSION

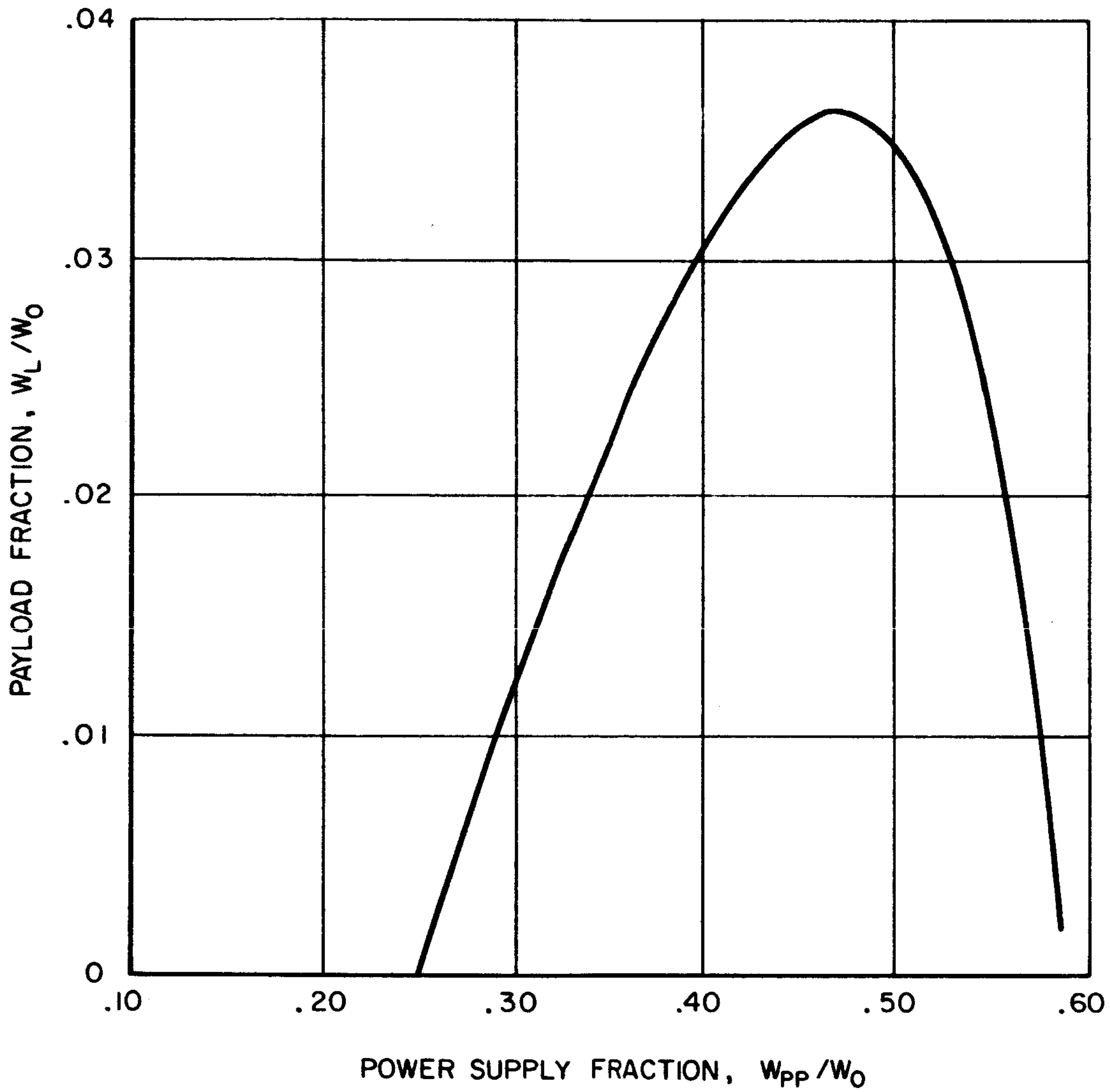
SATURN C-1 LAUNCH VEHICLE
POWER SUPPLY WEIGHT = 4750 LBS



500 - DAY JUPITER SATELLITE MISSION

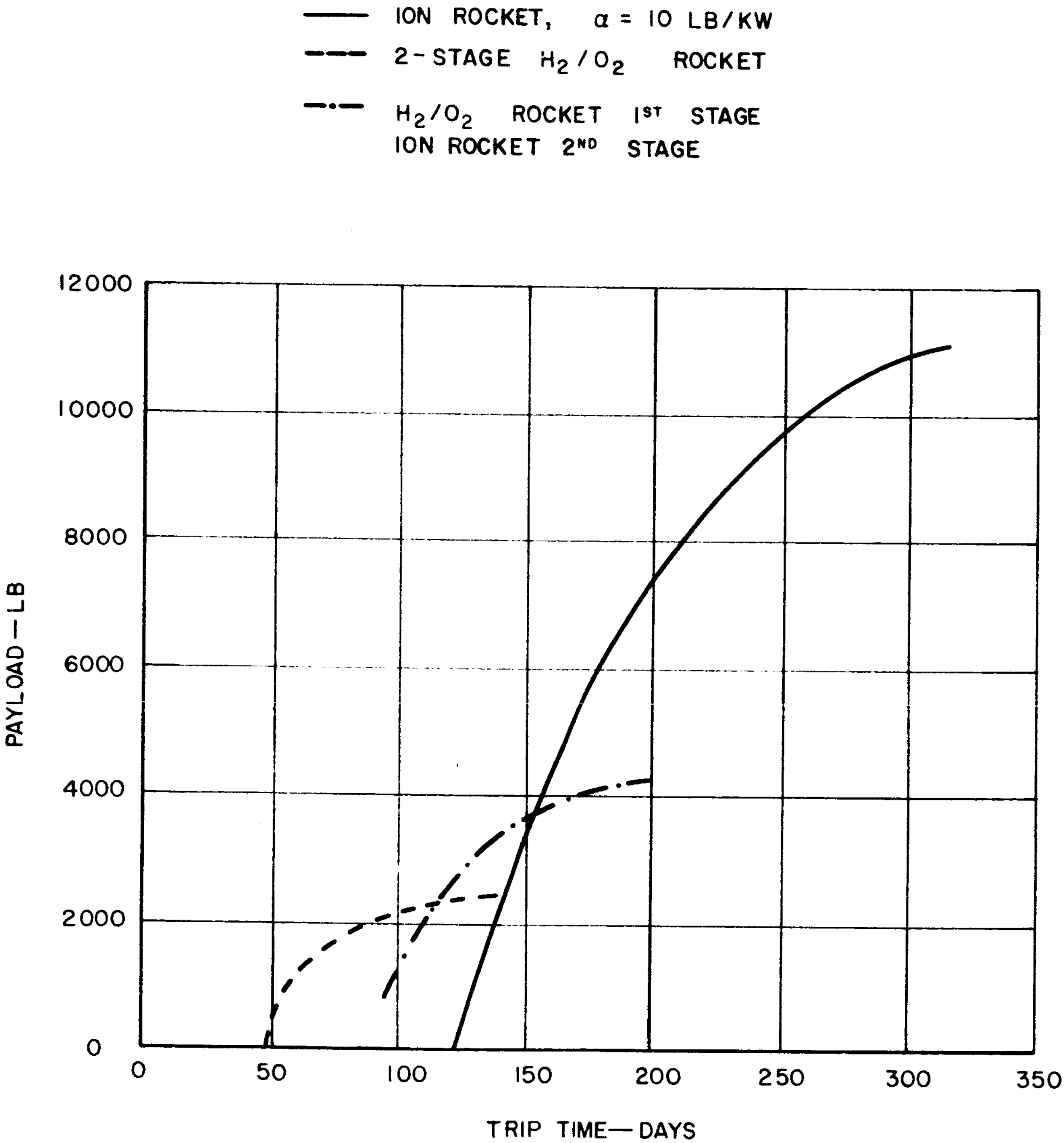
PAYLOAD FRACTION VERSUS POWER SUPPLY FRACTION

α = 10 LB/KW



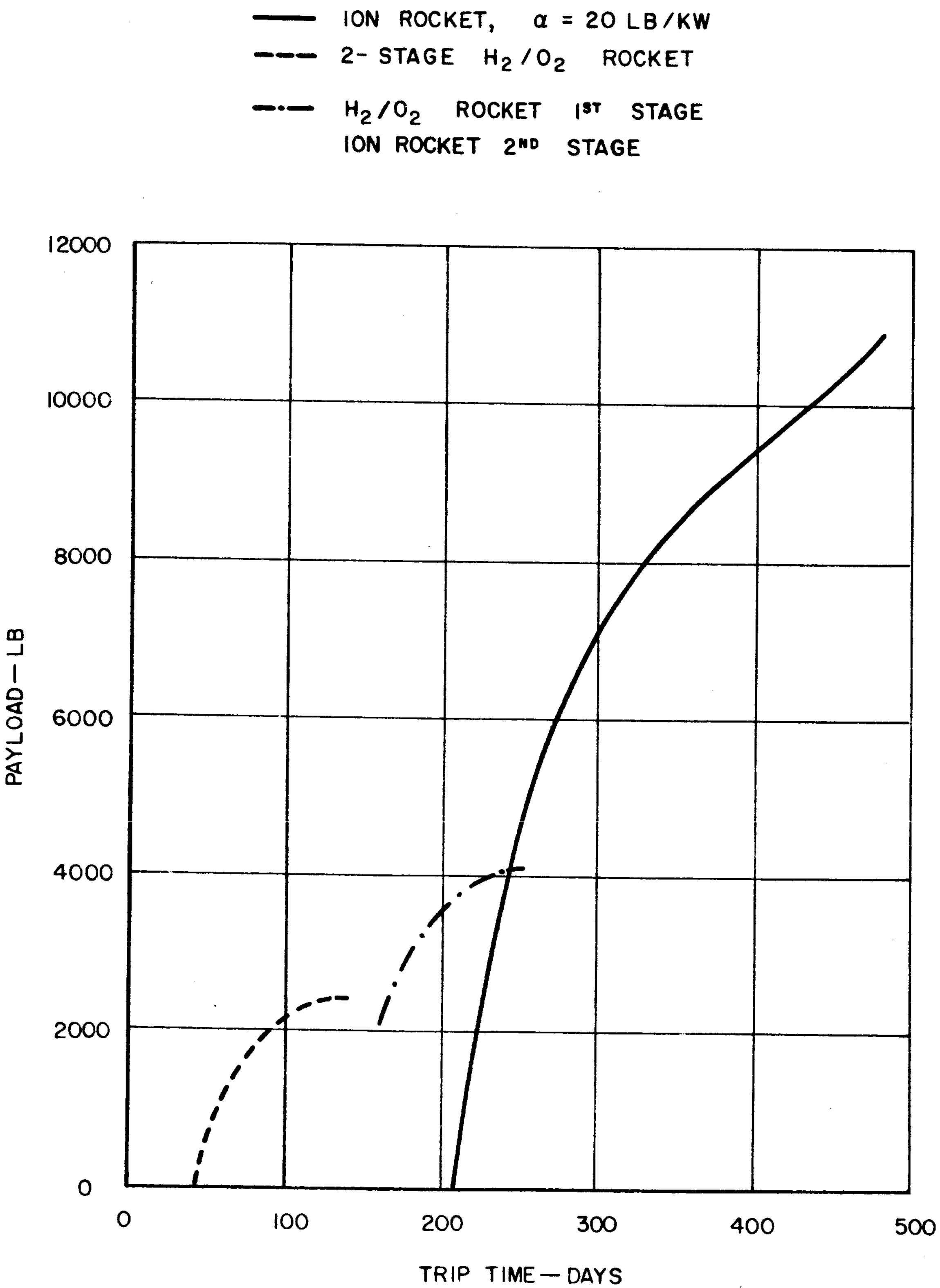
COMBINED USE OF HIGH AND LOW THRUST FOR VENUS SATELLITE MISSION

SATURN C-I LAUNCH VEHICLE



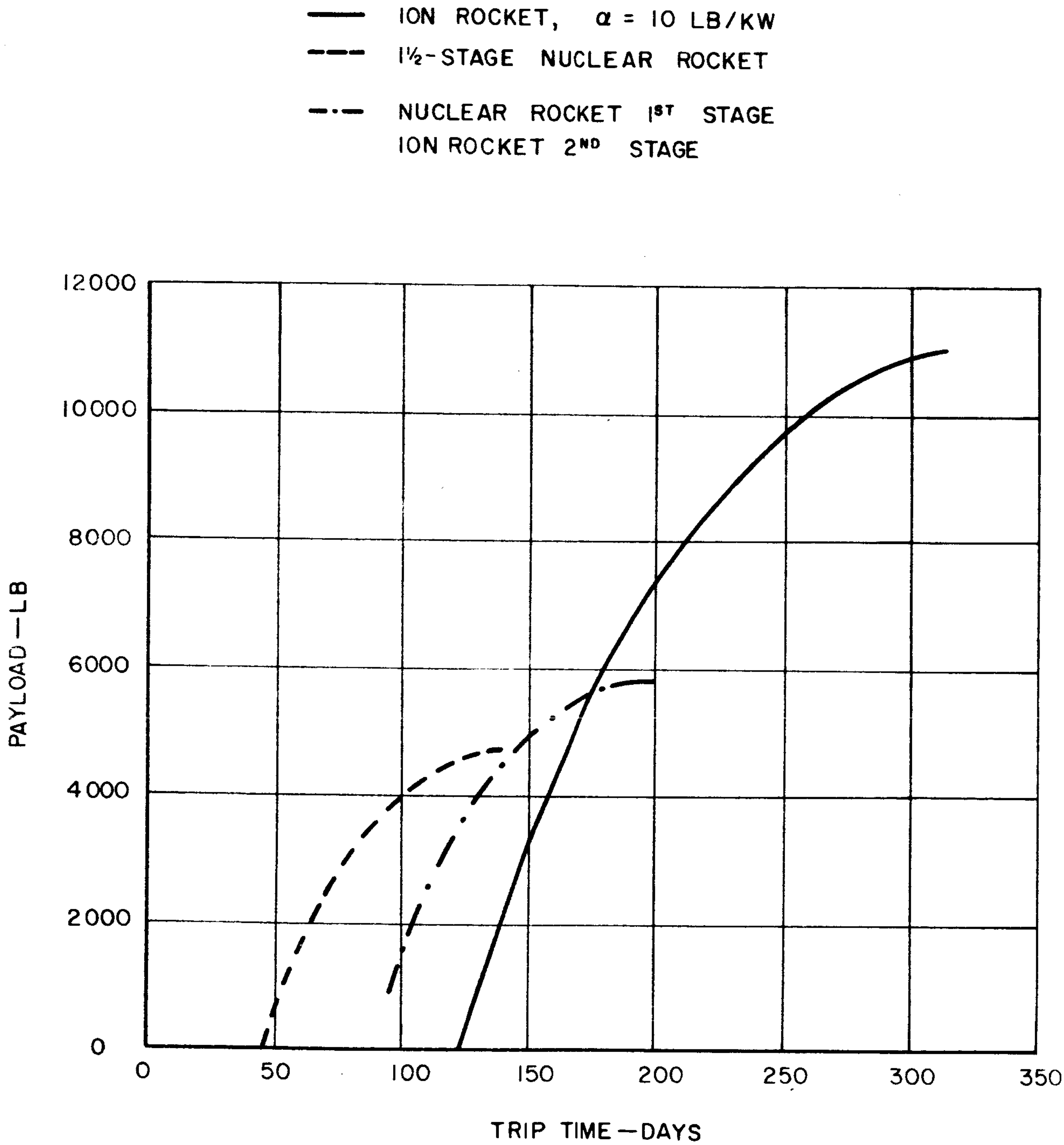
COMBINED USE OF HIGH AND LOW THRUST FOR VENUS SATELLITE MISSION

SATURN C-1 LAUNCH VEHICLE



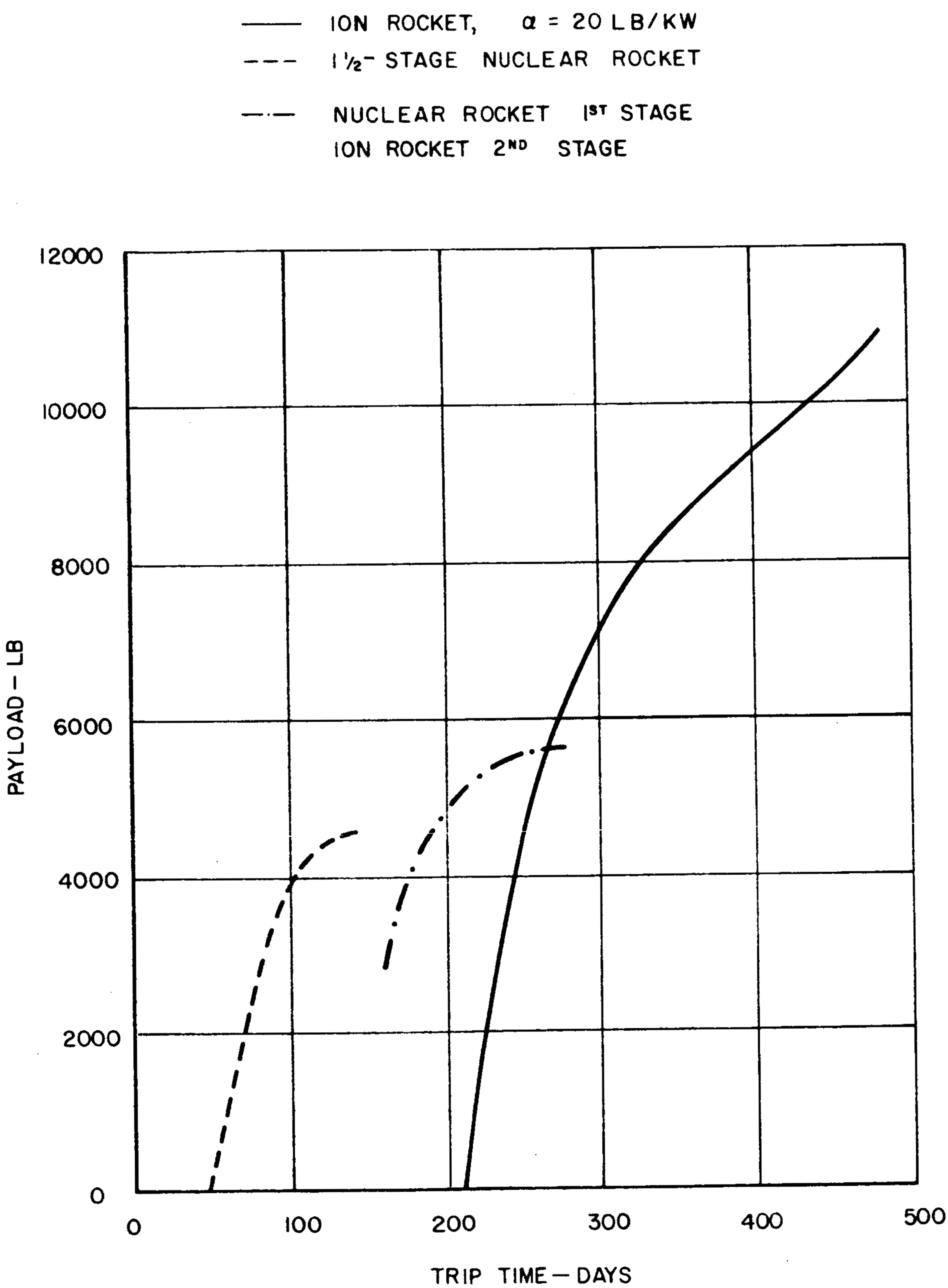
COMBINED USE OF HIGH AND LOW THRUST FOR VENUS SATELLITE MISSION

SATURN C-1 LAUNCH VEHICLE



COMBINED USE OF HIGH AND LOW THRUST FOR VENUS SATELLITE MISSION

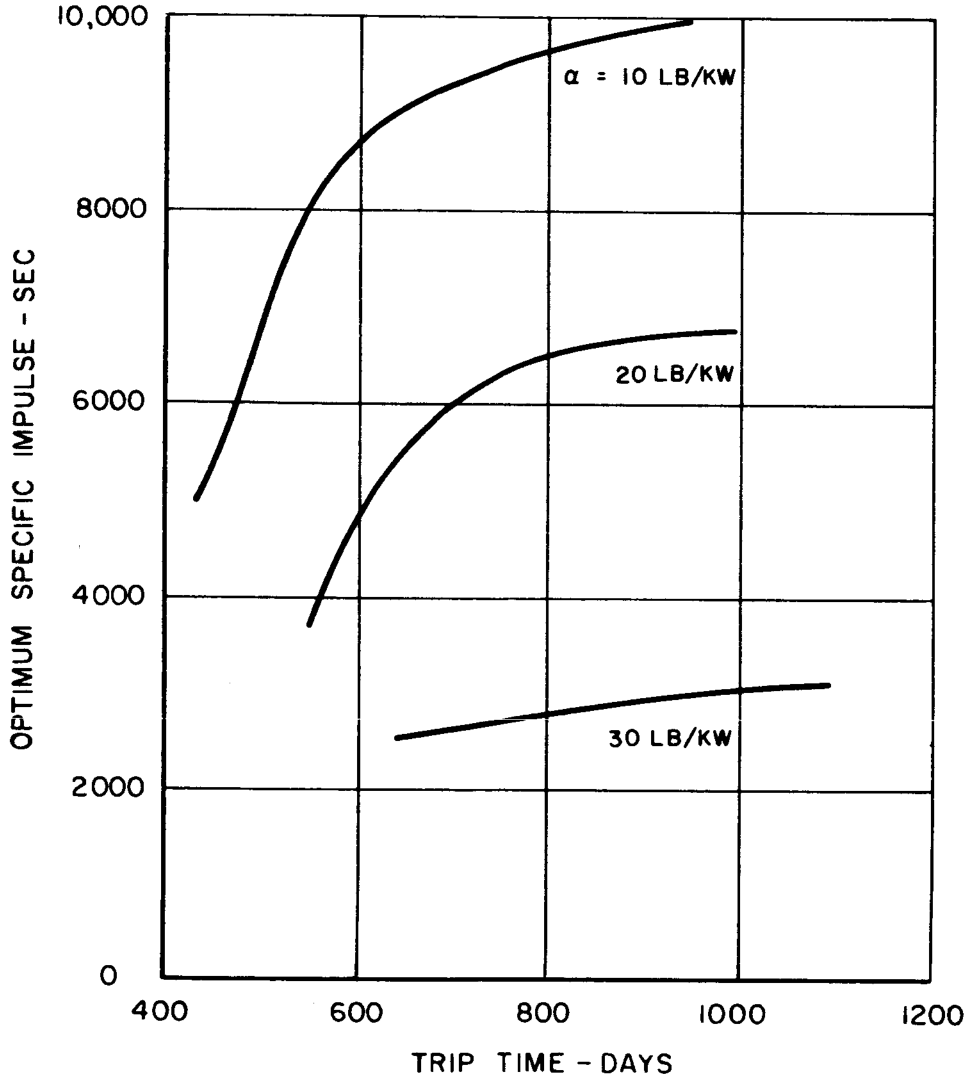
SATURN C-I LAUNCH VEHICLE



OPTIMUM SPECIFIC IMPULSE

SATURN PROBE MISSION

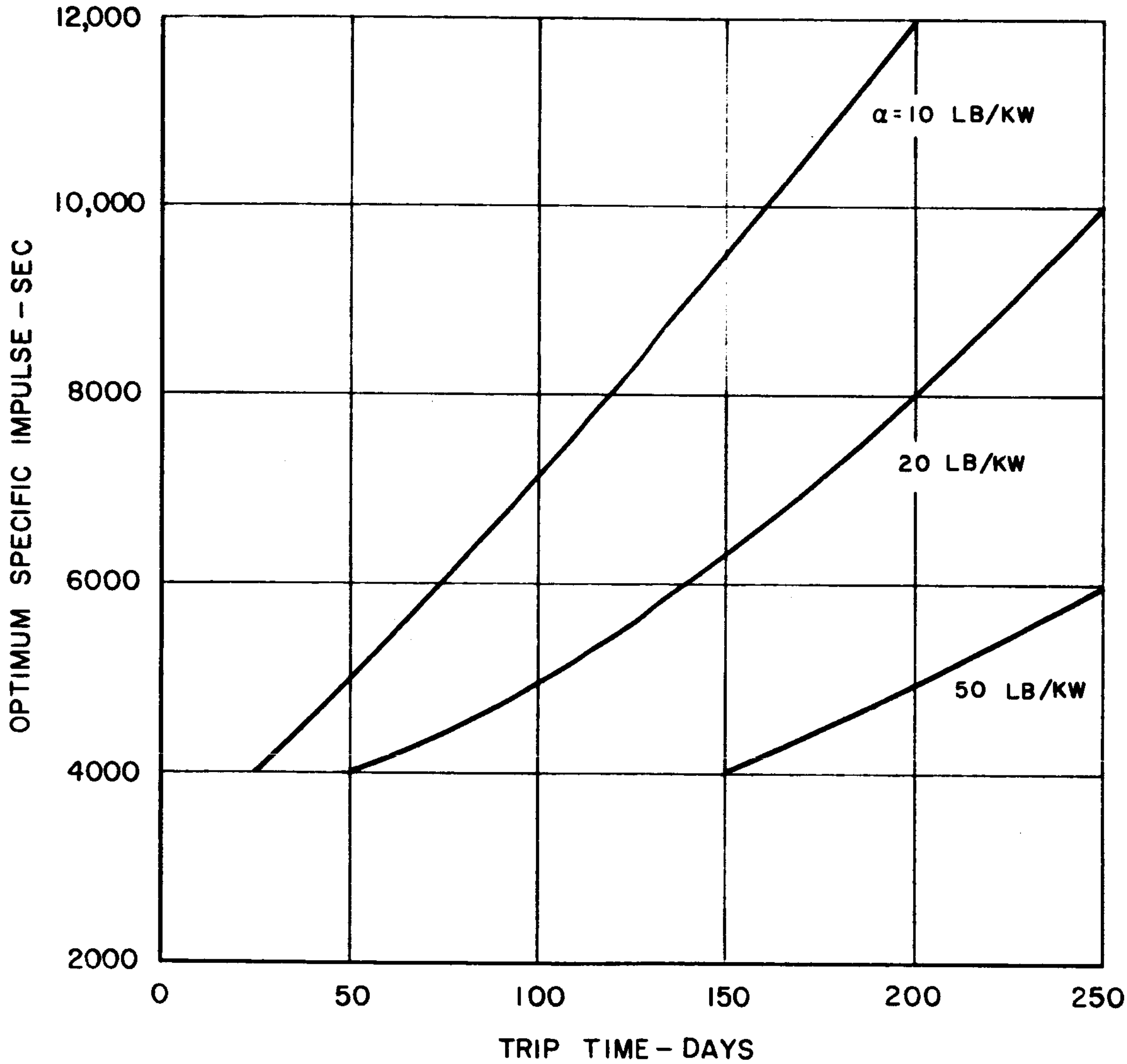
$W_{PP}/W_0 = 0.25$



OPTIMUM SPECIFIC IMPULSE

LUNAR SATELLITE MISSION

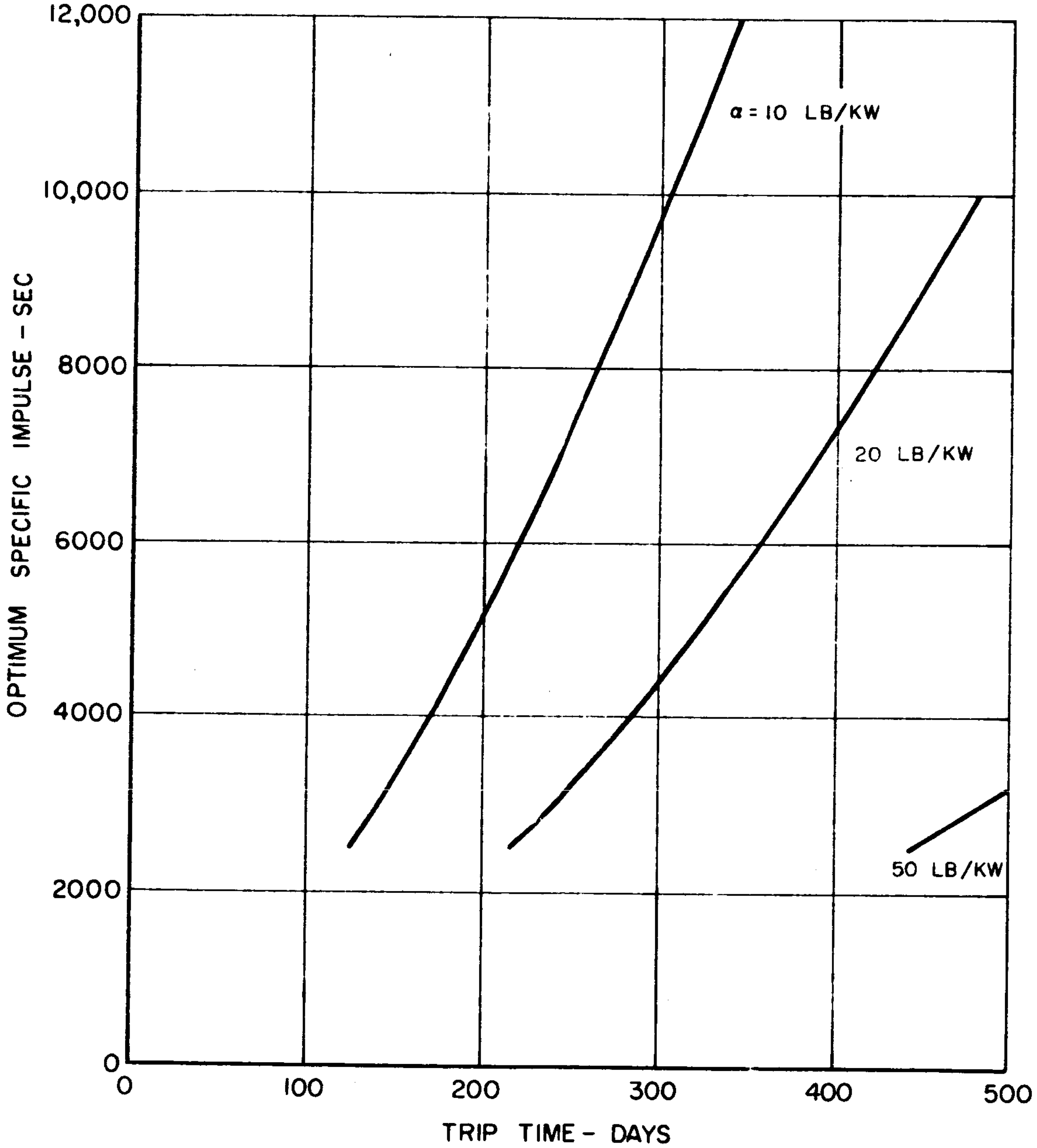
OPTIMIZED POWER SUPPLY WEIGHT



OPTIMUM SPECIFIC IMPULSE

VENUS SATELLITE MISSION

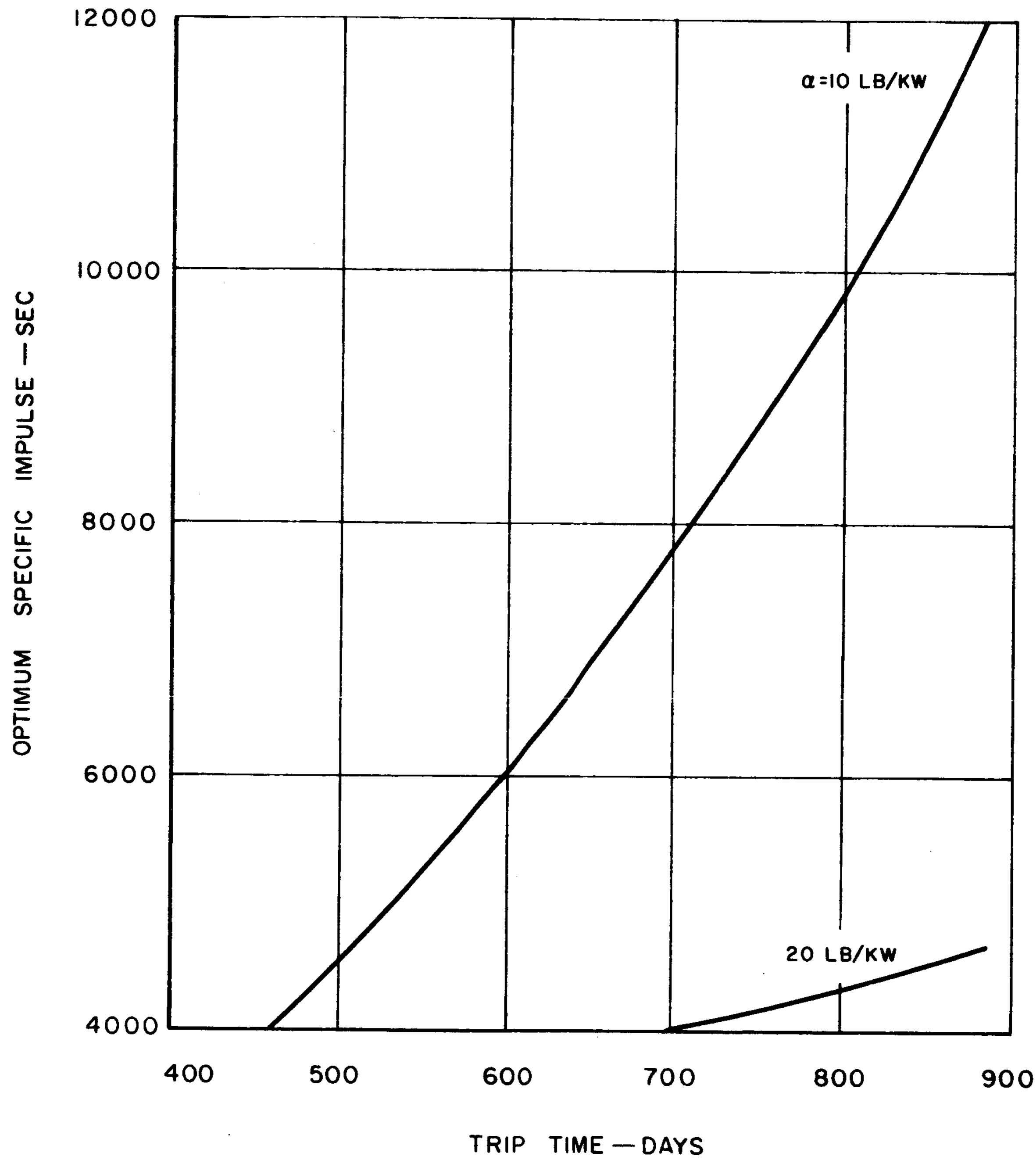
$W_{PP}/W_0 = 0.25$



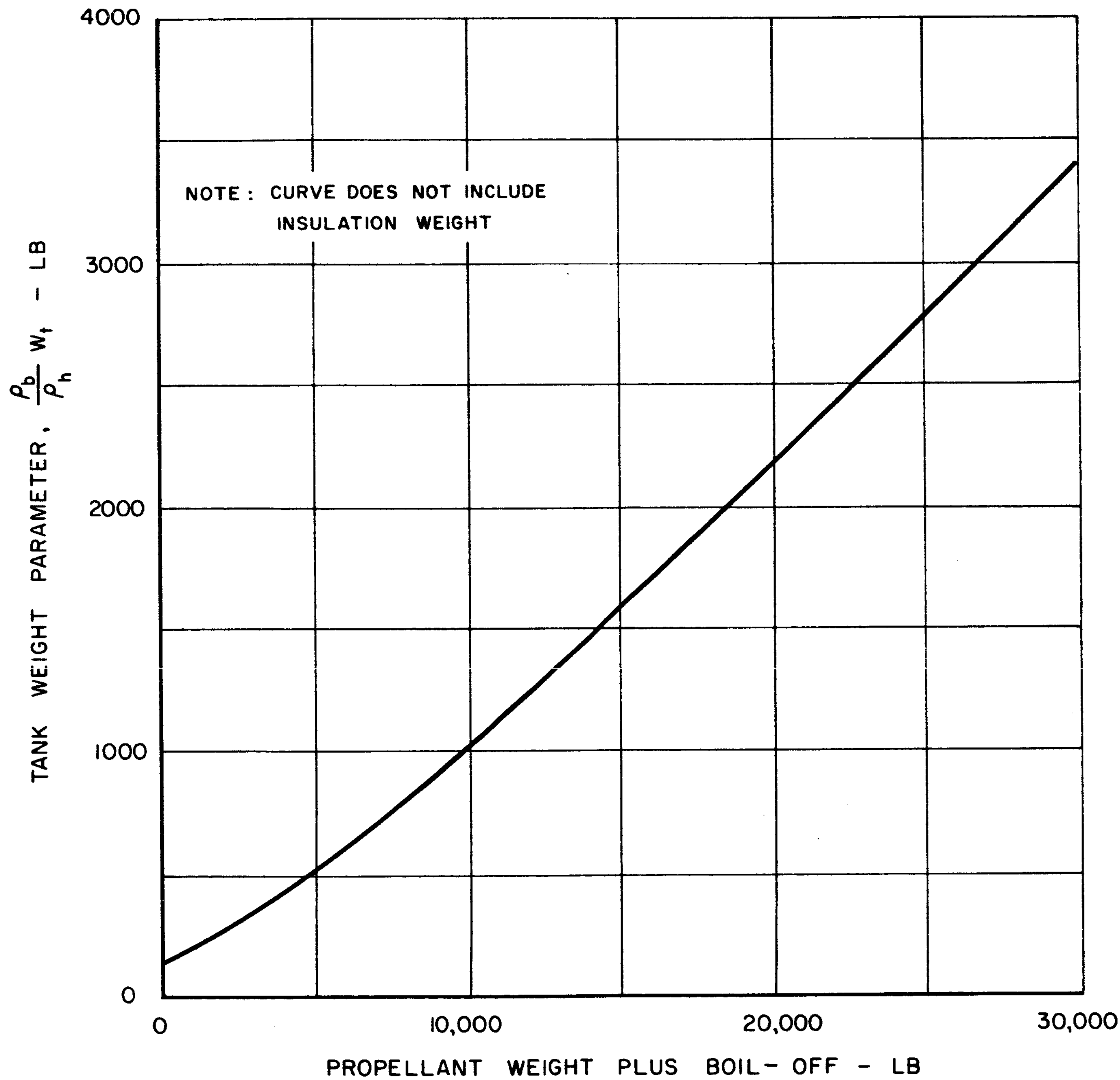
OPTIMUM SPECIFIC IMPULSE

JUPITER SATELLITE MISSION

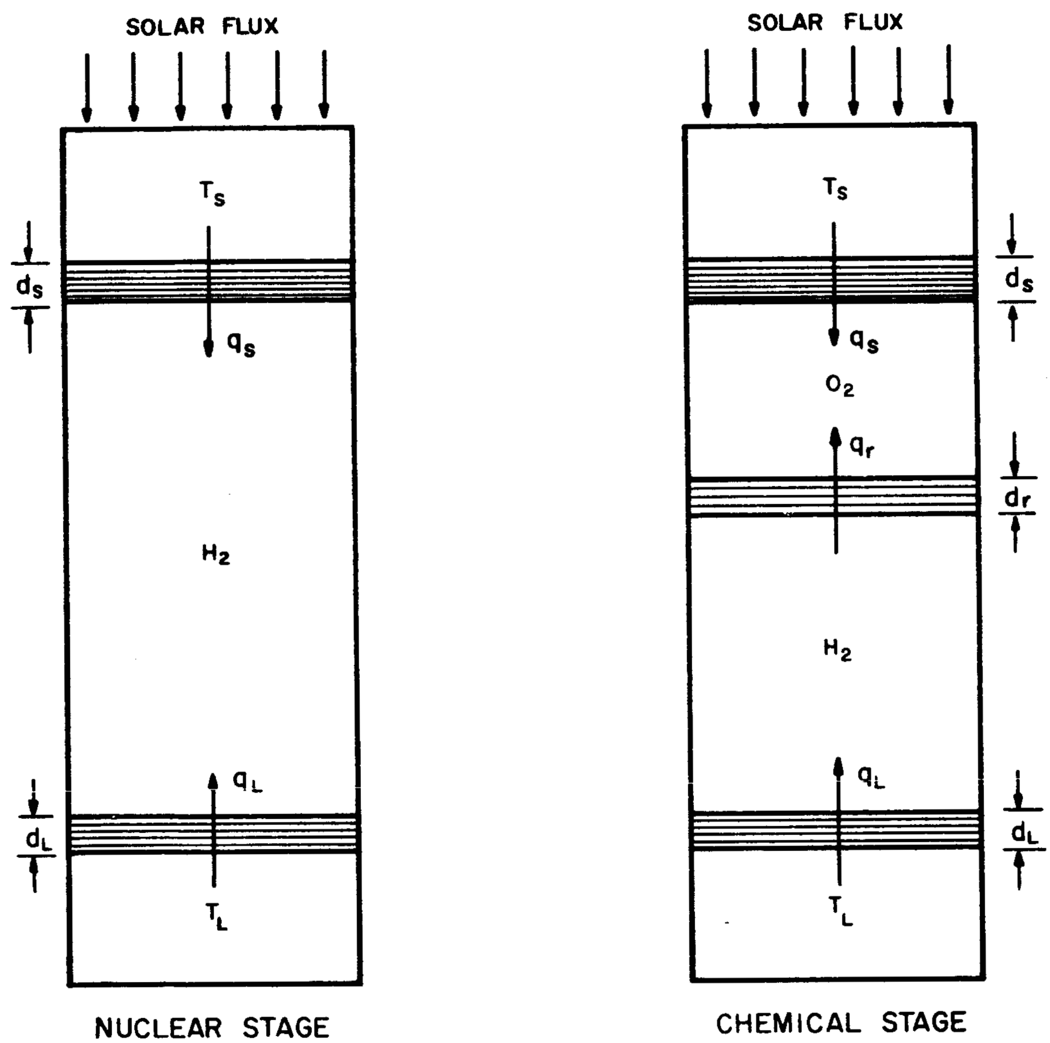
$W_{PP}/W_0 = 0.25$



TANKAGE WEIGHT

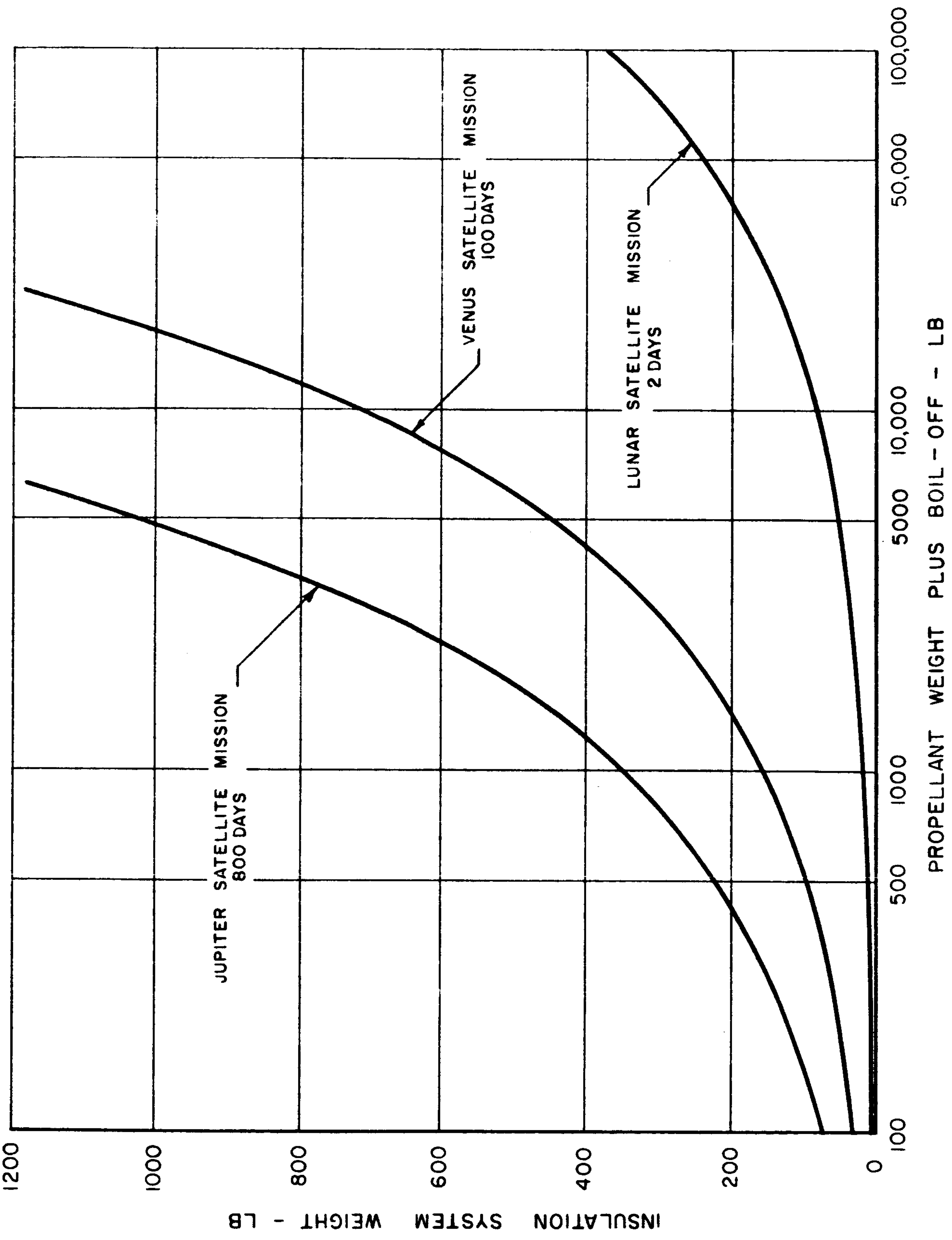


VEHICLE LAYOUT FOR INSULATION SYSTEM ANALYSIS



TYPICAL INSULATION SYSTEM WEIGHTS

NUCLEAR STAGE
H_2 PROPELLANT



TYPICAL INSULATION SYSTEM WEIGHTS

CHEMICAL STAGE
H_2/O_2 PROPELLANT