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SOLAR-ELECTRIC PROPULSION PROBES FOR EXPLORING THE SOLAR SYSTEM

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NATIONAL AERONAUTICS AND SPACE ADMINISTRATION · WASHINGTON, D.C. · 1967

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E-4014

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SUMMARY

A limited study is made for unmanned probe spacecraft using solar-electric propulsion. Flyby and orbiter payloads of 500, 1000, and 2000 pounds are arbitrarily picked and delivered to all the planets and close-in solar orbits. A current technology solar cell powerplant of 75 lb/KW_e and existing electric ion thrustors are assumed for the electric spacecraft. Three different size chemical propulsion launch vehicles are used to boost the spacecraft to Earth escape velocity or higher at the start of each mission. A few planetary swingby cases are included for the electric spacecraft. Charts are given to show the minimum trip times for each booster/payload combination. Small chemical launch vehicles in conjunction with solar-electric spacecraft can perform many missions better than larger chemical launch systems used alone. Advantages are greatest for missions that aim to capture a payload in an orbit about the target planet.

INTRODUCTION

Recent developments in both solar-electric power systems and ion thrustors could combine to make attractive the early application of electric propulsion in interplanetary spacecraft. For continuous sunlight operation, solar cells are the most lightweight and reliable space power source available up to twenty kilowatts or more. References 1 and 2 discuss development trends in solar cells made of silicon crystal or thin film cadmium sulfide that are leading to increased efficiency, lighter weight, and higher durability in the space environment over current designs. Ion thrustors, especially the election bombardment mercury and cesium types, are demonstrating increased efficiency and operating life (refs. 3 and 4).

The purpose of this study is to present a preliminary review of the payload capability of solar electric spacecraft for unmanned interplanetary probe missions. The solar electric powerplant and mercury ion thrustor weight and performance assumptions used here have been intentionally chosen as possible with current technology.

In a recent publication (ref. 5), Strack made a study of solar-electric spacecraft for close solar probe missions. Also, Hughes Aircraft Company completed a study (ref. 6) of solar-electric Mars probe missions under a contract

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with Jet Propulsion Laboratory (JPL). Reference 7 is a similar study made by Electro-Optical Systems under an Air Force contract. The present report gives data for missions to inner and outer planets, as well as solar probe missions to as low as 0.05 AU. Although not as detailed, this study aims to cover a wider range of missions than the Strack, Hughes, and EOS studies.

For simplicity, only two basic mission-types are analyzed here: flybys, and highly elliptical planetary captures. Data is presented for missions to all planets and several low-AU solar probes. Three chemical propulsion launch booster systems are used to cover a wide range of initial weight for the electric spacecraft. For clarity, the bulk of the data is presented on simple graphs showing the minimum time at which a fixed payload can be delivered for each combination of mission and launch booster. Data is added to show the benefit of swingby maneuvers at Venus for close-in solar probes and at Jupiter for outer planet flyby probes.

APPROACH AND ASSUMPTIONS

The principal assumptions used in this study were:

(1) Planar, two-body trajectories patched at planet spheres of influence.

(2) Boost to at least escape velocity by the chemical launch vehicle.

(3) The solar power versus spacecraft-sun distance curve (fig. 1) is identical to that derived in reference 5. (It allows for panel tipping when this mode increases power output.)

(4) Variable propellant flow rate to accommodate changes in solar panel output. Thrustor specific impulse remains constant.

(5) Ion engines were assumed to have the efficiency curve shown on fig. 2. Figure 2 is based on electron bombardment ion thrustor data given in references 3 and 4.

(6) Optimal thrust direction steering and coast phases as described in ref. 5.

(7) The electric propulsion system specific mass is 75 lb/KW_e. This includes solar panels, power conditioning, controls, and thrustors (ref. 6).

(8) Tankage and propellant feed system mass equals 10% of the propellant mass (ref. 7).

(9) Total propulsion system structure mass equals 10% of the initial mass of the electric spacecraft (ref. 7).

These assumptions taken together form a reasonably conservative framework for payload calculations. It is quite likely that, for simplicity, first generation craft would use a fixed angle steering program rather than an optimum one. Such non-optimal steering programs are estimated to reduce payload by about 10%. On the other hand, the value of 75 lb/KW_e for specific powerplant mass is higher than current estimates. Such estimates have the same numerical value but include tankage, propellant feed, and some propulsion structure that are here accounted for as additional items.

Gross payload is defined as the spacecraft arrival mass minus the total propulsion system mass (panels, power conditioning and controls, tankage, plumbing, structure). For capture probes, a storable chemical braking rocket with a 300 second specific impulse places the spacecraft into a highly elliptic orbit with periapsis of 2 radii. In this case, the required chemical braking propellant and propulsion hardware is also subtracted from the arrival mass to determine gross payload. Propulsion hardware is estimated to be 20% of the braking propellant.

The three launch vehicles considered were advanced versions of (1) Atlas/Centaur, (2) Titan IIIC/Agena, and (3) Saturn IB/Centaur. Advanced booster versions were chosen because they are expected to be operational by the time early solar-electric propulsion systems become available. Each of these represents a particular size and performance class booster. The choice of booster and its burnout velocity determines the initial electric spacecraft mass as shown on figure 3. Initial spacecraft mass excludes the required payload support and protective shroud which are charged to the launch vehicle. The assumption that booster burnout velocity be at least escape velocity was made to avoid solar cell degradation due to long exposure times within the Van Allen belts. Another reason is to circumvent the panel orientation complexities associated with Earth escape spirals. If cell degradation could be eliminated and panel orientation problems minimized, sub-escape burnout velocities would offer significant payload increases at some of the higher mission times.

Several parameters are left free for optimization under the above ground rules. These are booster burnout velocity, thrustor specific impulse, initial spacecraft acceleration, and thrustor steering program. For most of the calculations, all of these parameters were in fact optimized. In a few cases, trends were interpolated or extrapolated to avoid expensive calculation.

RESULTS

Solar-electric trajectories are in general quite different from both high-thrust trajectories and nuclear-electric trajectories. Typical solar-electric trajectories for a 0.1 AU solar probe and a Jupiter flyby are diagrammed in figure 4. The optimal number of loops around the Sun is determined primarily by the mission time, the target, and type of mission (flyby or orbiter).

Payload mass as a function of trip time is shown in figure 5 for a Saturn flyby mission. The dashed curve is for the advanced SIB/Centaur booster alone (without an electric spacecraft). For trips longer than 1360 days this booster plus an electric spacecraft delivers more payload than the booster alone. The reverse is true for trips less than 1360 days. This illustrates the general rule that if the payload required is small enough, it is often better to delete the electric spacecraft, since the booster alone performs better. Dashed curves for the two smaller boosters do not appear on this figure because the mission is beyond their capability. Another general rule evident in figure 5 is that large boosters are more effective in raising payload size than reducing trip time.

Figure 6 shows the trip time required to deliver a 500 pound flyby payload to nearly all solar system targets using each of the three chemical boosters. A dashed curve represents a booster by itself, while a solid curve indicates that a solar-electric spacecraft has been added. Figures 7 through 11 present similar minimum time data for other probe missions. The main reason the data is presented in this somewhat sparse manner is to avoid confusing the reader with a deluge of information. In this way, attention can focus on only the most pertinent features of the boosted solar-electric probe mission.

In figure 6, the circles on the end of each dashed curve represent the maximum capability of the chemical boosters. Trips to targets beyond the circles are not possible with the booster alone. Kinks in the solar-electric curves reflect a change in the optimal trajectory class (i.e., the optimal number of loops about the Sun).

For the easier missions such as Mars and Venus flybys, electric propulsion is not required to reduce trip time. For more difficult missions, adding solar-electric propulsion reduces mission time or makes possible missions that cannot be done by the booster alone. The remaining figures are similar to figure 6. Figures 7 and 8 are for 1000 and 2000 pound flybys.

Figures 9, 10, and 11 show results for the 500, 1000, and 2000 pound orbiter-capture missions. Although capture data exists only at discrete points (the planets) on these plots, a curve is drawn through them to help identify boosters and follow trends. The capture orbit at each planet in figures 9, 10, and 11 is highly elliptical. It is assumed that the braking rocket is used at 2.0 planet radii to bring the spacecraft velocity to just below the escape value. At best, only minimal power is available from the solar panels at Saturn and the planets beyond. Therefore, for higher capture payloads, or shortest travel times for a fixed payload, the solar powerplant and thrustors are jettisoned before the braking maneuver at all planets beyond Jupiter. Lower-eccentricity capture ellipses or retaining the solar cell powerplant would require more travel time if payloads are to remain the same. Minimum times for capture missions beyond Saturn could be lower than shown in figures 9, 10, and 11. This is because extensive calculations to optimize planet approach velocities were not made.

The main points brought out by these figures are:

(1) For the easier missions such as Venus and Mars flybys, the use of a solar-electric spacecraft is unwarranted from a trip time standpoint.

(2) Solar-electric spacecraft eliminate the unaccessible regions in the mission spectrum characteristic of all-chemical systems. For the Atlas/Centaur/solar-electric system, 500 lb flyby missions become possible to distances greater than 4 AU and less than 0.4 AU, and

2000 lb capture missions are possible to Mercury, Jupiter, and Saturn. Trips beyond Saturn are also possible but the trip times are too long. The corresponding data for the larger boosters is less impressive but still significant. Even a SIB/ Centaur could not perform 2000 lb capture missions to Mercury or beyond Saturn without the aid of an electric spacecraft.

(3) There are many outward missions that can be done with either a ballistic spacecraft or a solar-electric spacecraft, but for which less trip time is required by an electric spacecraft. The 500 lb Jupiter flyby mission using a Titan IIIC/Agena illustrates the point. A 120-day trip time savings is possible in this case.

(4) Several booster-spacecraft combinations may be capable of doing the same mission. And, in general, less time is required by a sufficiently larger booster-only system than a smaller booster with a solar-electric spacecraft. When comparing systems that employ unequal booster sizes, the fundamental question is: Is the difference in booster costs offset by the time savings? There is no simple answer to this question. In some cases, nevertheless, the additional time required by a solar-electric spacecraft atop a small booster is preferable to suffering the greater costs of large boosters used alone. The 500 lb Jupiter flyby mission illustrates this point. An Atlas/Centaur/solar-electric vehicle would require 560 days, whereas the all-chemical SIB/Centaur vehicle would require 420 days. Considering that the larger booster costs roughly 5 times as much as the smaller booster, the extra 140 days required by the smaller solar-electric system seems well worth the wait. Of course, the cost of the solar-electric propulsion system needs to be included in this comparison, but so does the fact that the electric system often has a power supply available for communications and experiments that the all-chemical system does not.

Since the purpose of this study is to present performance data for solar-electric spacecraft, no comparison with chemical upper stages is intended. Using this data, any number of all-chemical systems can be compared to solar-electric systems through simple calculations.

Swingby mode.- Only a few trajectories utilizing the planetary swingby mode have been calculated for solar-electric spacecraft. Venus swingby solar probe and Jupiter swingby outer planet flyby results are shown on figure 12. The solid curve is for direct flights, and the dashed curve is for swingby flights. The Venus swingby maneuver permits about a 30% reduction in perihelion radius for a given payload and trip time. A Jupiter swingby maneuver cuts at least 600 days off the mission times for the outer planets. However, opportunities for outer planet flybys via Jupiter do not occur often. Planetary swingby maneuvers generally reduce trip time for electric spacecraft as much as for ballistic spacecraft.

Power level.- Although there is a different power level for each mission that minimizes trip time, the actual use of such power levels is unlikely.

This is because trip time is not very sensitive to the power level and because it would be less costly to develop a single electric spacecraft than to develop several spacecraft at different power levels. Ten to twenty kilowatts (electric) is suggested for both the Atlas/Centaur and the Titan IIIC/Agena launch vehicles. Between 30 and 60 KW_e is right for the SIB/Centaur. The data in the last seven figures would be changed only slightly if fixed power numbers had been assumed. It should be remembered that for most missions (between about 0.15 AU to Jupiter, or perhaps Saturn) solar-electric spacecraft arrive at their targets with useful power available for communications and experiments. This factor is of definite importance when comparing electric systems and all-chemical systems.

CONCLUSIONS

For a given booster system, the advantages of solar-electric propulsion grow with increasing mission difficulty. The more difficult missions are shown to be: flybys to far outer planets, close-in solar flybys, and almost all planet orbiting probe missions. Also, missions are made more difficult by demands for greater payload or power at the target.

Easy missions, such as small Venus and Mars flyby and capture probes, are best performed by all-chemical systems. Solar-electric spacecraft would be competitive on the easier missions if very high power levels were required at the target.

The results given here show that, in general, a solar-electric spacecraft combined with a small chemical booster provides an attractive multi-mission capability. A change to larger booster systems would only be necessary to shorten time for difficult missions. Even then, the larger booster can be used more effectively over more missions by also combining it with a solar-electric spacecraft.

For many mission and booster combinations the solar-electric spacecraft can do a mission that cannot be done by the booster alone. In other cases, the solar-electric spacecraft (a) reduces trip time, (b) provides a power supply at the target, or (c) does both.

Lewis Research Center
National Aeronautics and Space Administration
Cleveland, Ohio, June 15, 1967.

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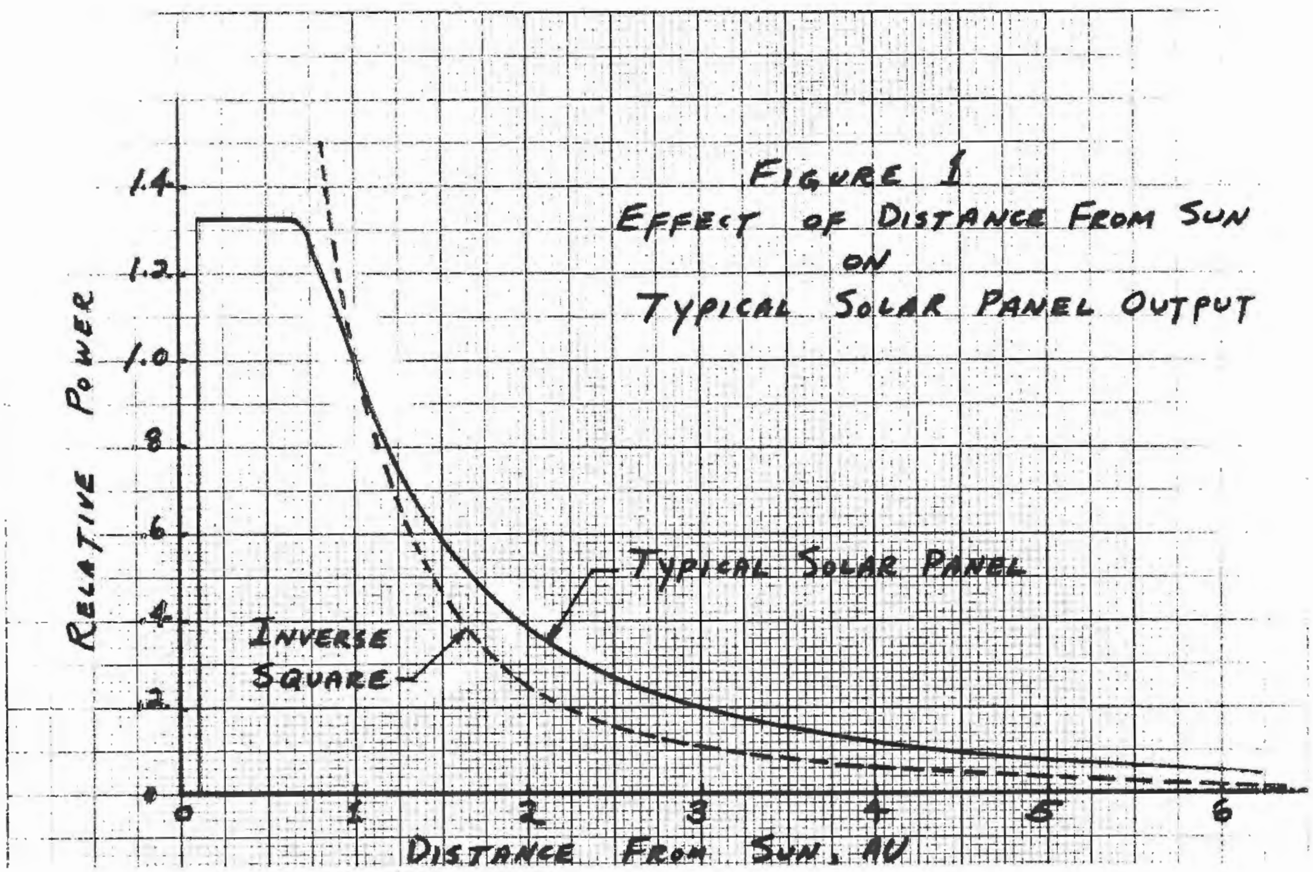
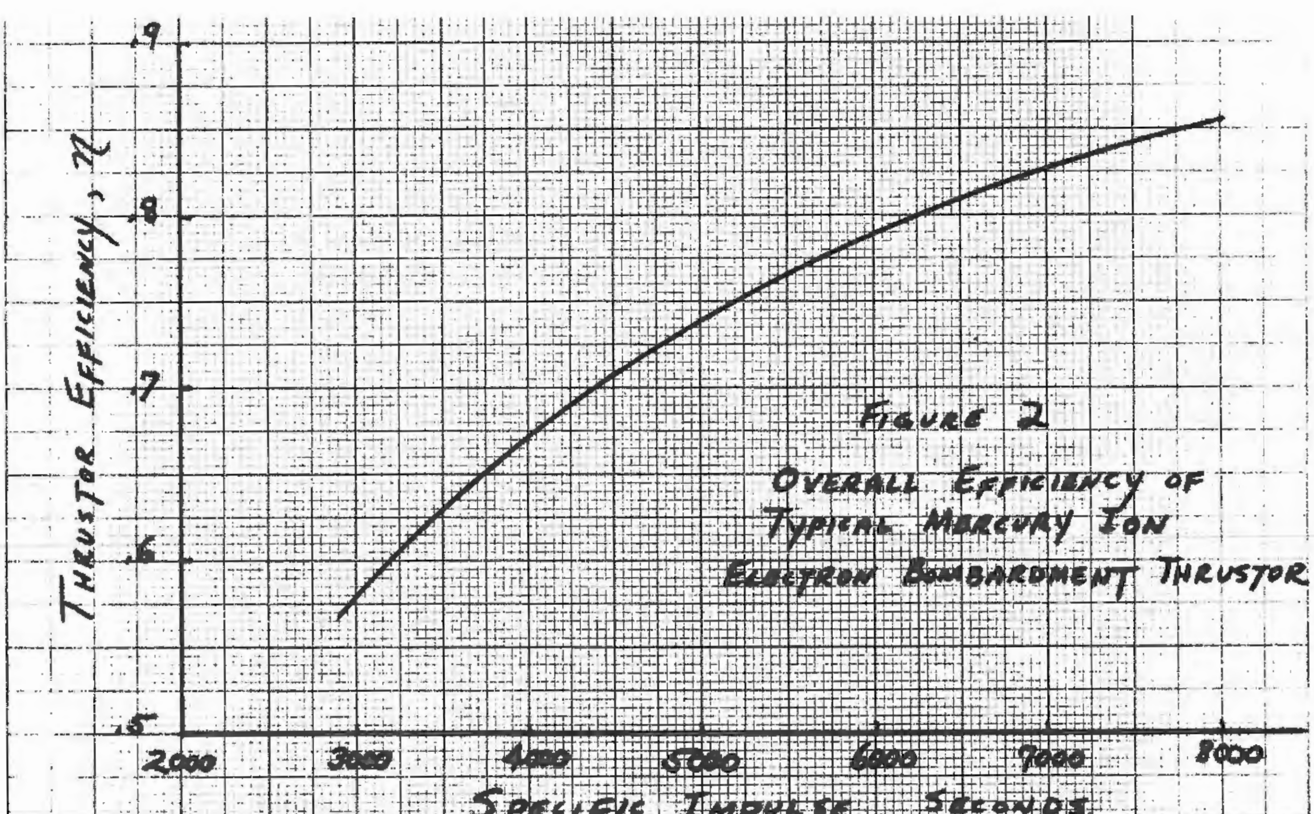


FIGURE 1
EFFECT OF DISTANCE FROM SUN
ON
TYPICAL SOLAR PANEL OUTPUT
RELATIVE POWER
1.4
1.2
1.0
.8
.6
.4
.2
0
TYPICAL SOLAR PANEL
INVERSE SQUARE
0
1
2
3
4
5
6
DISTANCE FROM SUN, AU


THRUSTOR EFFICIENCY, η
.9
.8
.7
.6
.5
FIGURE 2
OVERALL EFFICIENCY OF
TYPICAL MERCURY ION
ELECTRON BOMBARDMENT THRUSTOR
2000
3000
4000
5000
6000
7000
8000
SPECIFIC IMPULSE, SECONDS

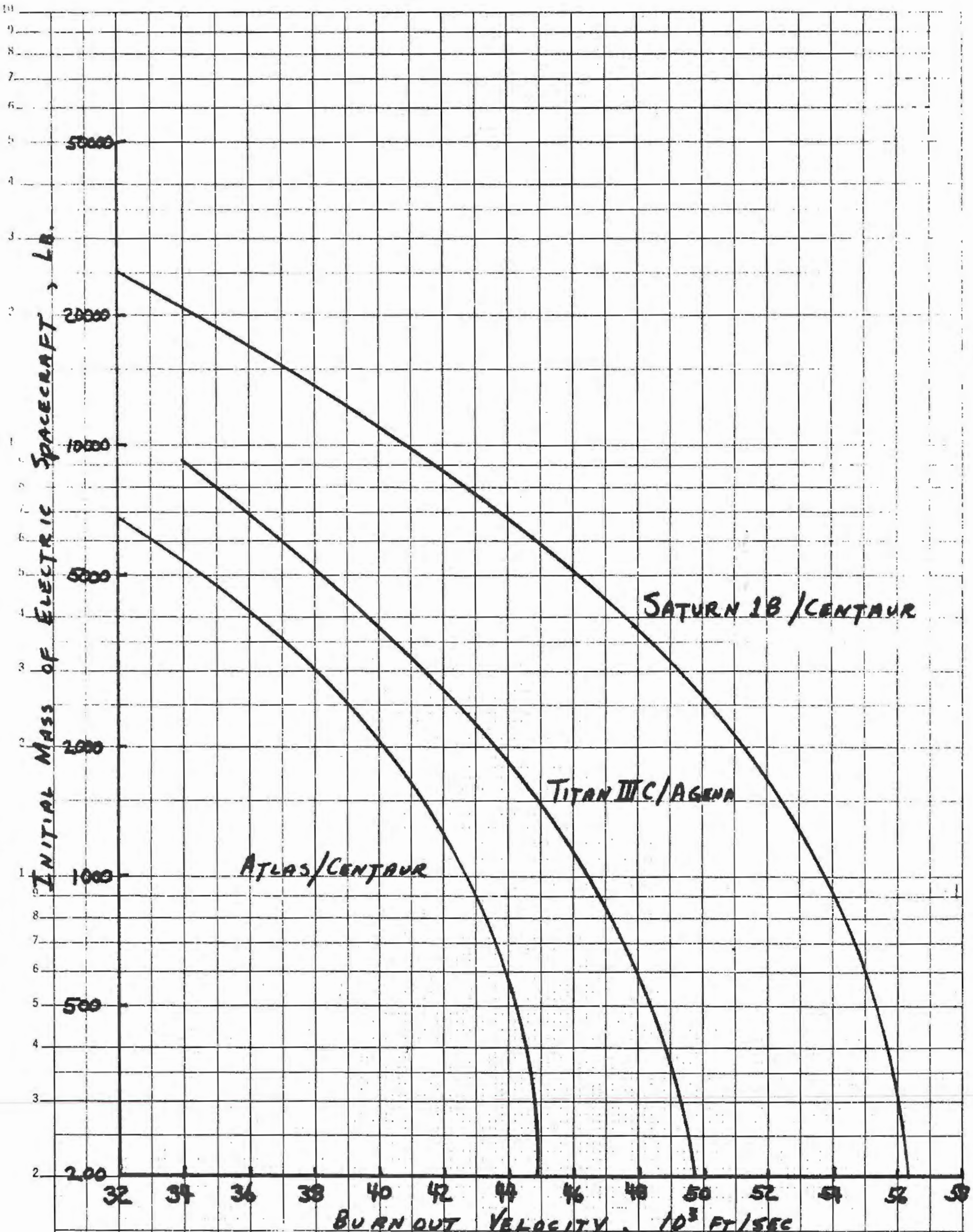


FIGURE 3. PERFORMANCE OF ADVANCED BOOSTERS USED FOR ELECTRIC SPACECRAFT

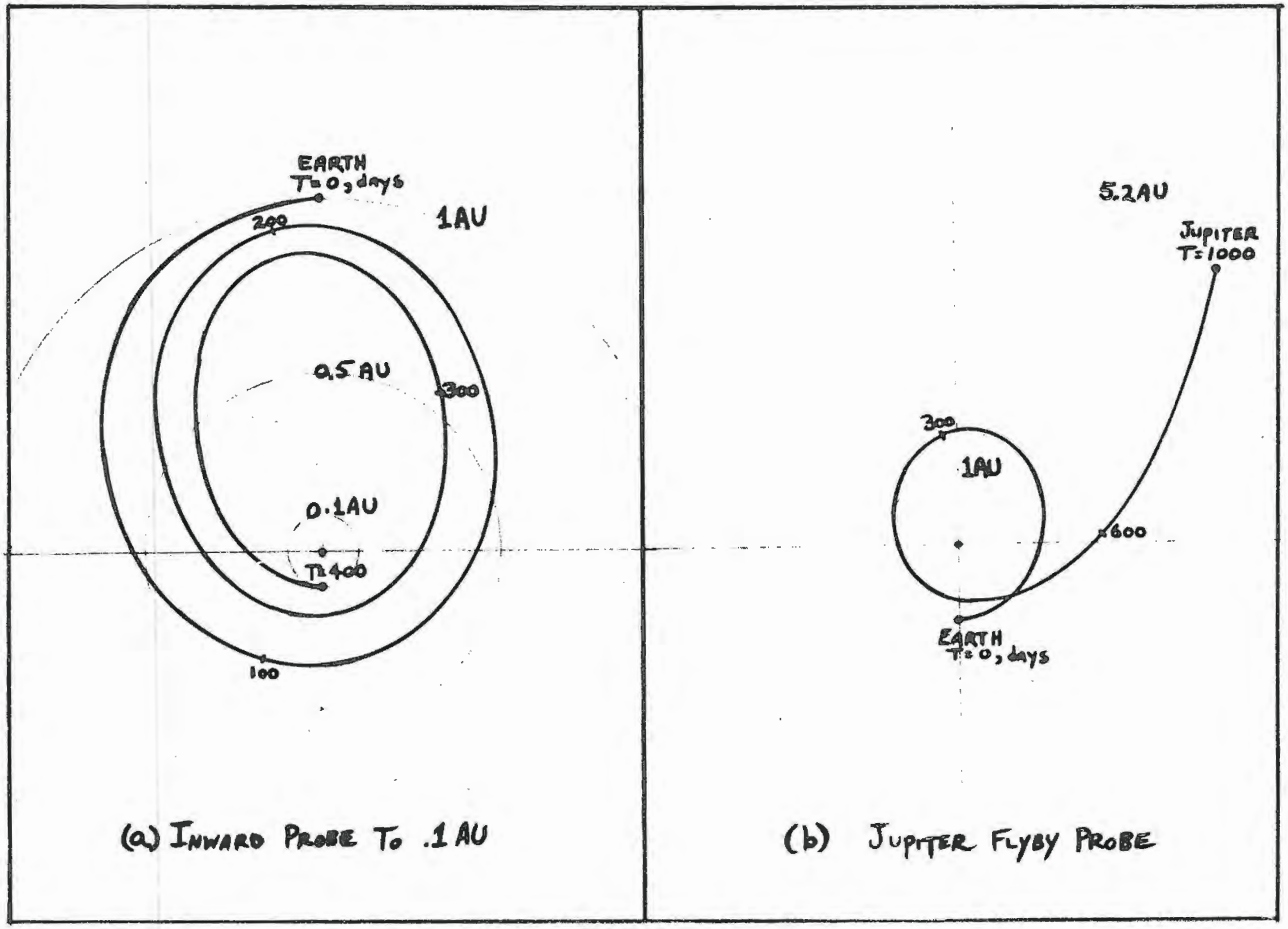


FIGURE 4 - TYPICAL TRAJECTORIES FOR SOLAR-ELECTRIC PROPULSION PROBES

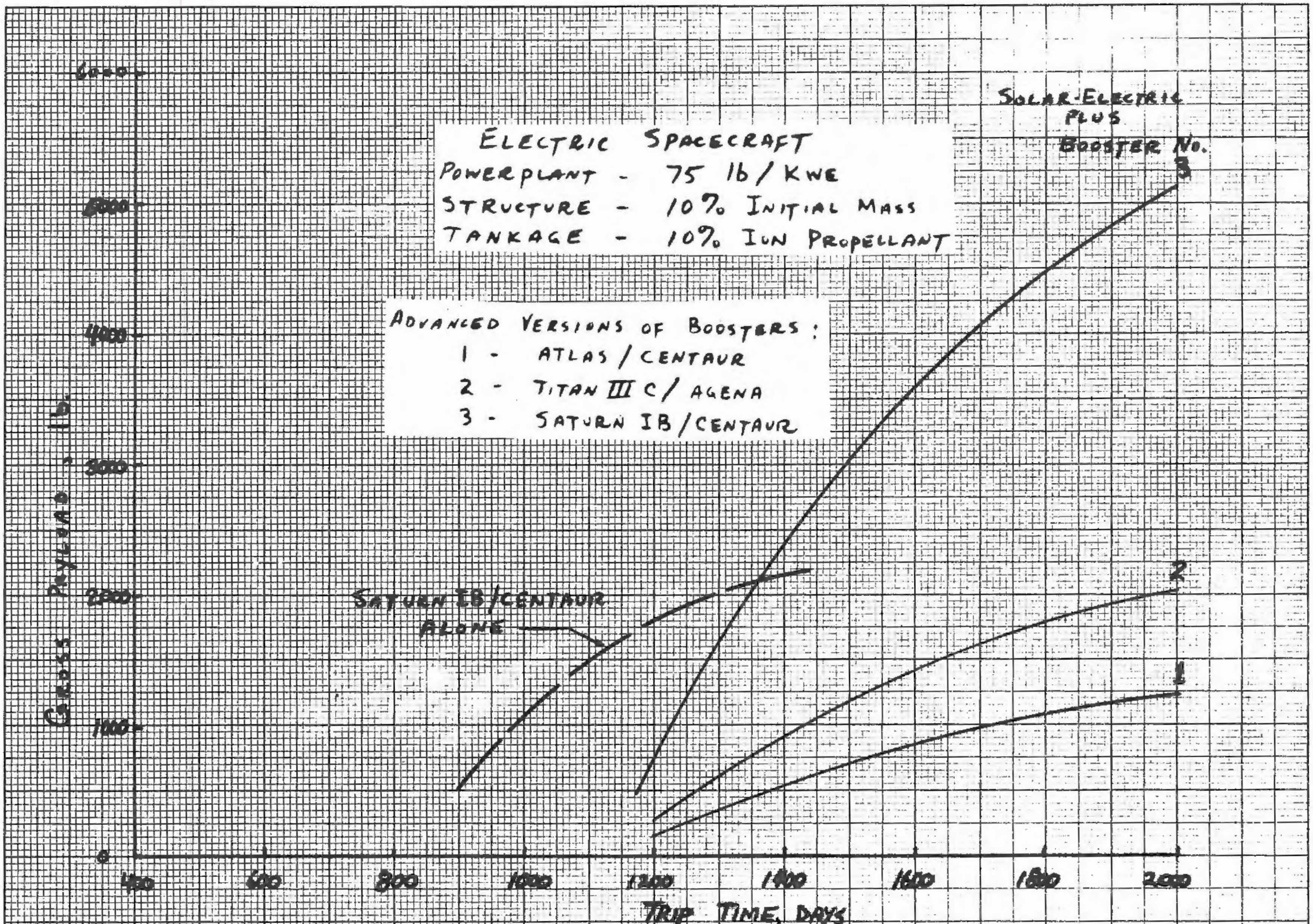


FIGURE 5 - PAYLOAD DELIVERED TO SATURN BY SOLAR-ELECTRIC FLYBY SPACECRAFT

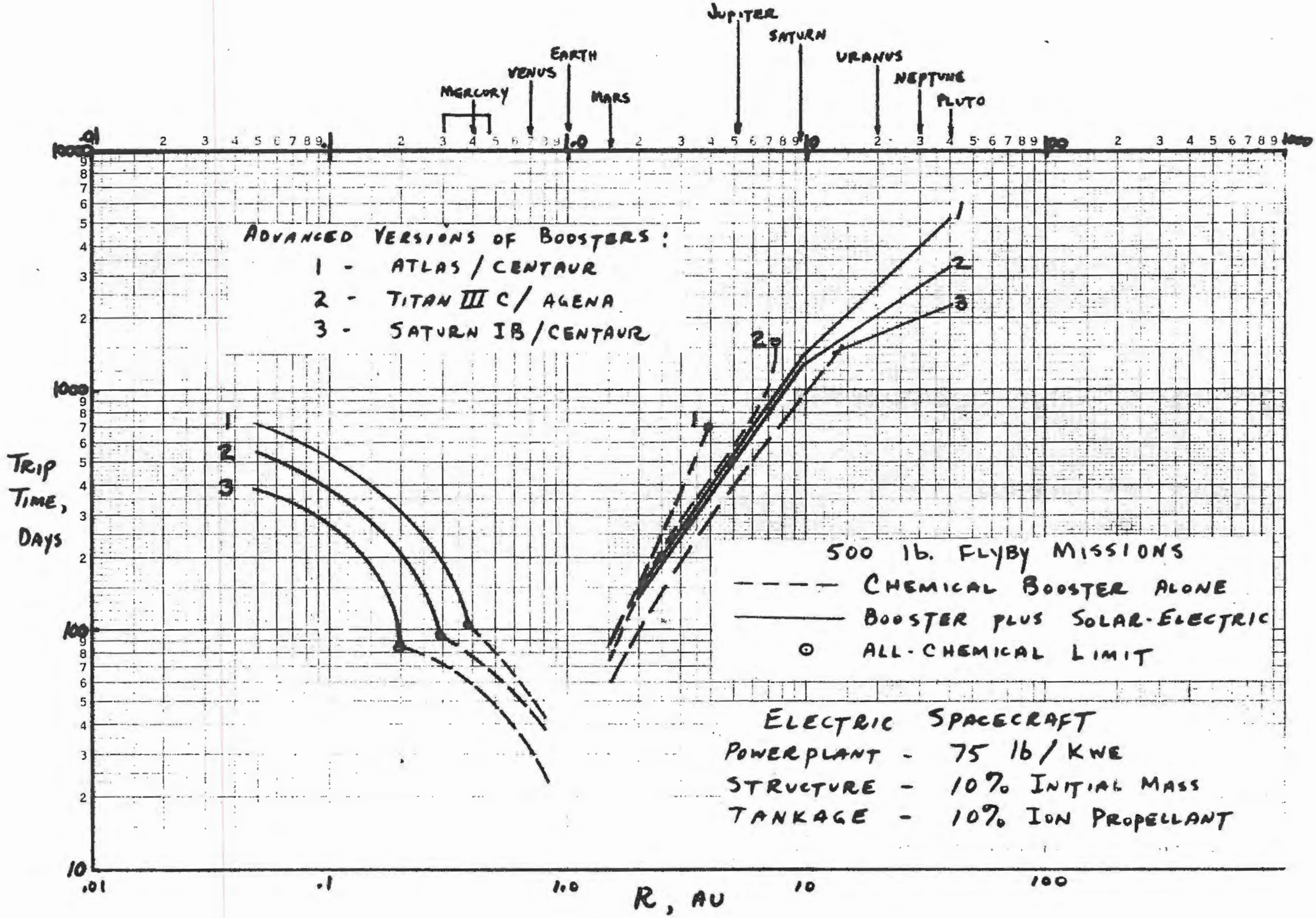


Figure 6 - Minimum Trip Time for 500 lb. Flyby Missions

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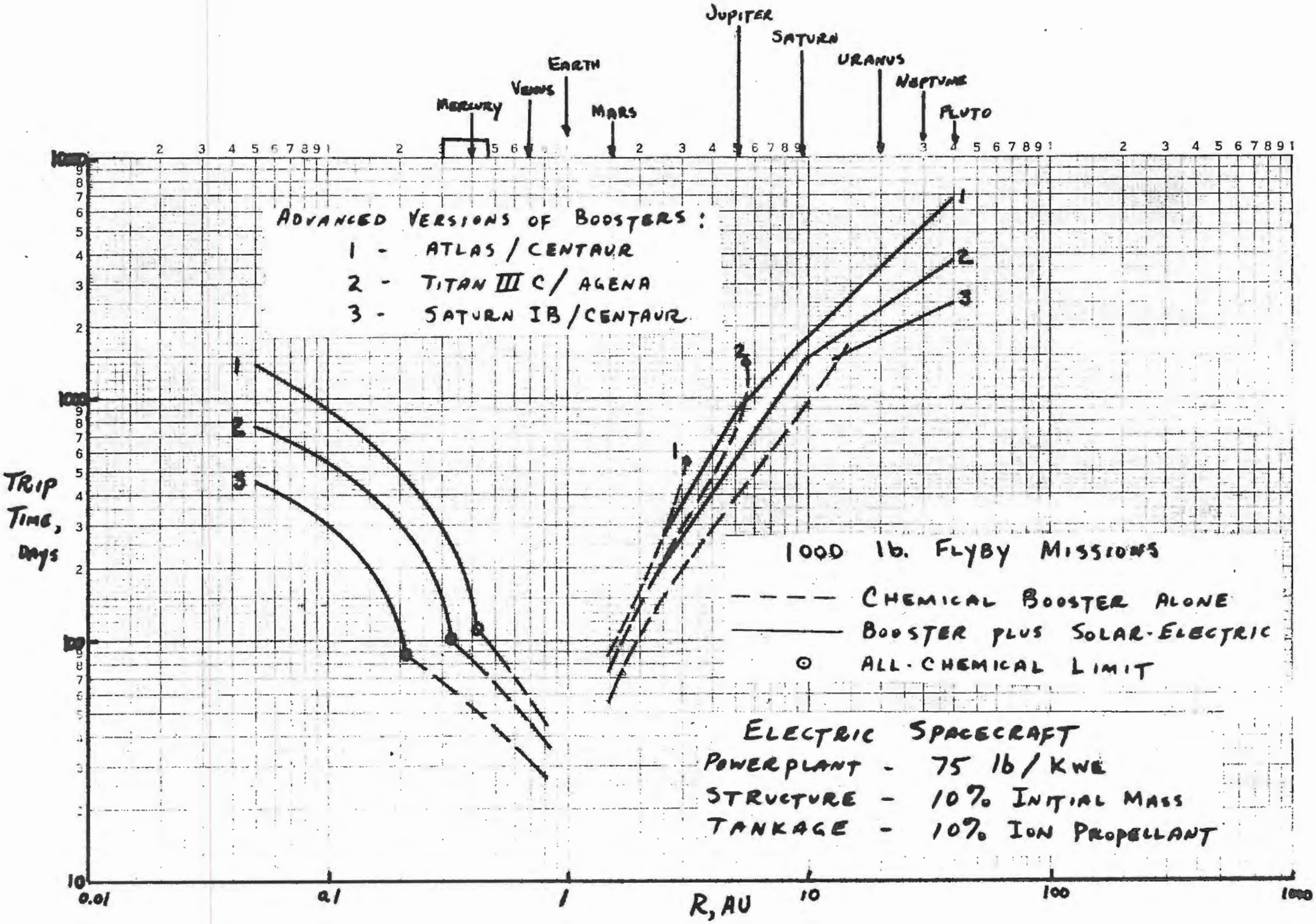

FIGURE 7 - MINIMUM TRIP TIME FOR 1000 lb. FLYBY MISSIONS

4-1-67

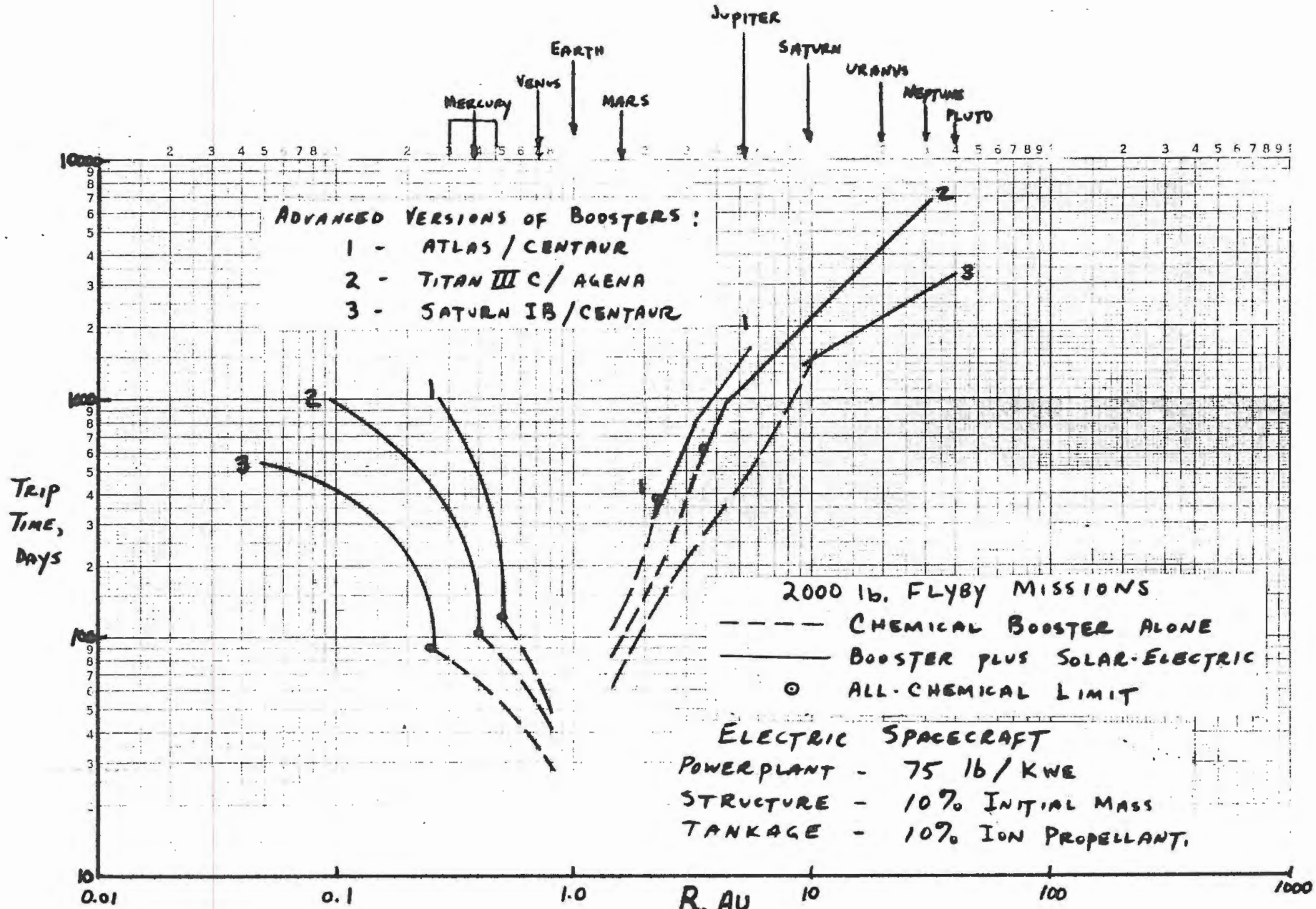

FIGURE 8 - MINIMUM TRIP TIME FOR 2000 lb. FLYBY MISSIONS

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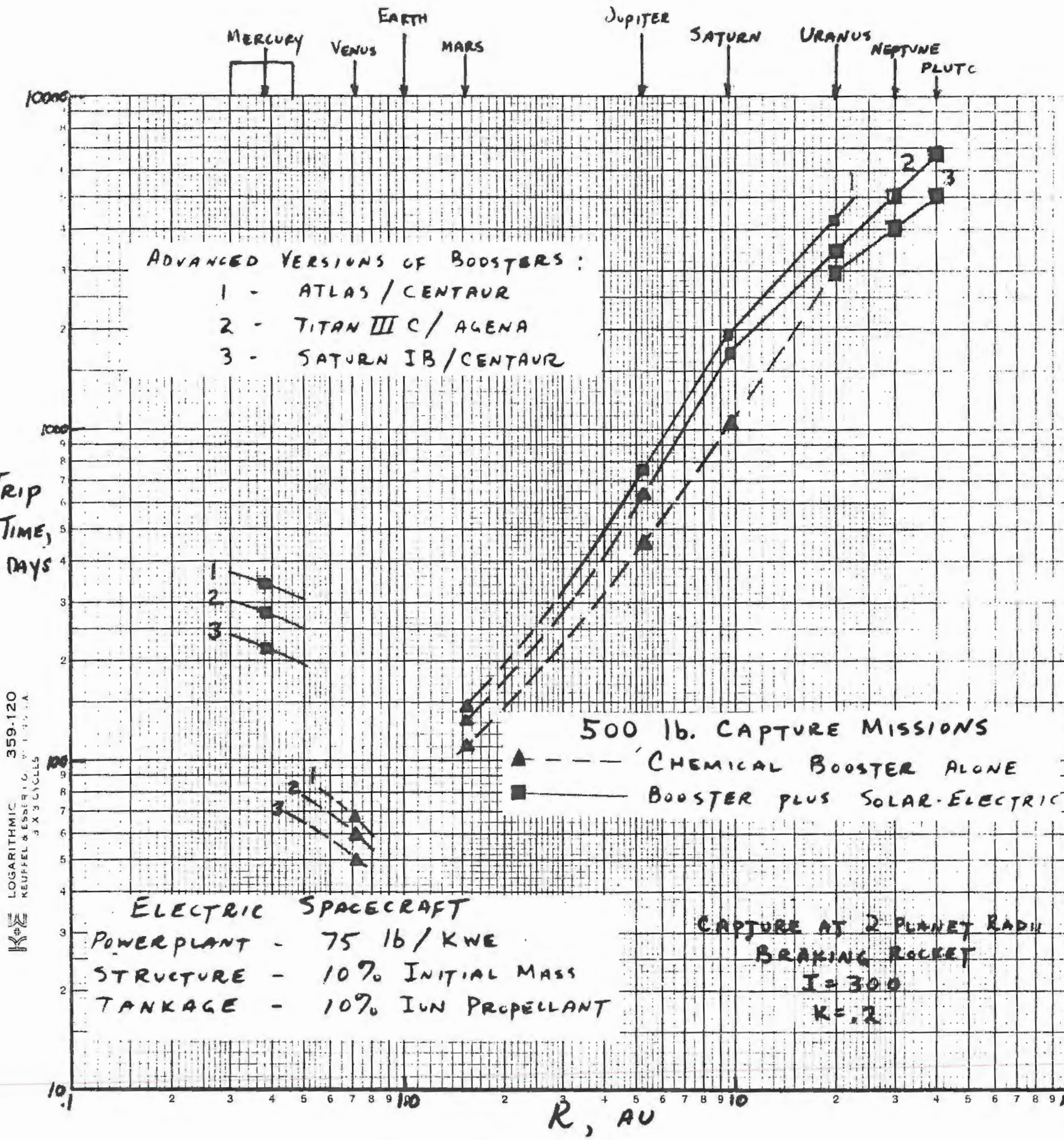

FIGURE 9 - MINIMUM TRIP TIME FOR 500 lb. CAPTURE MISSIONS

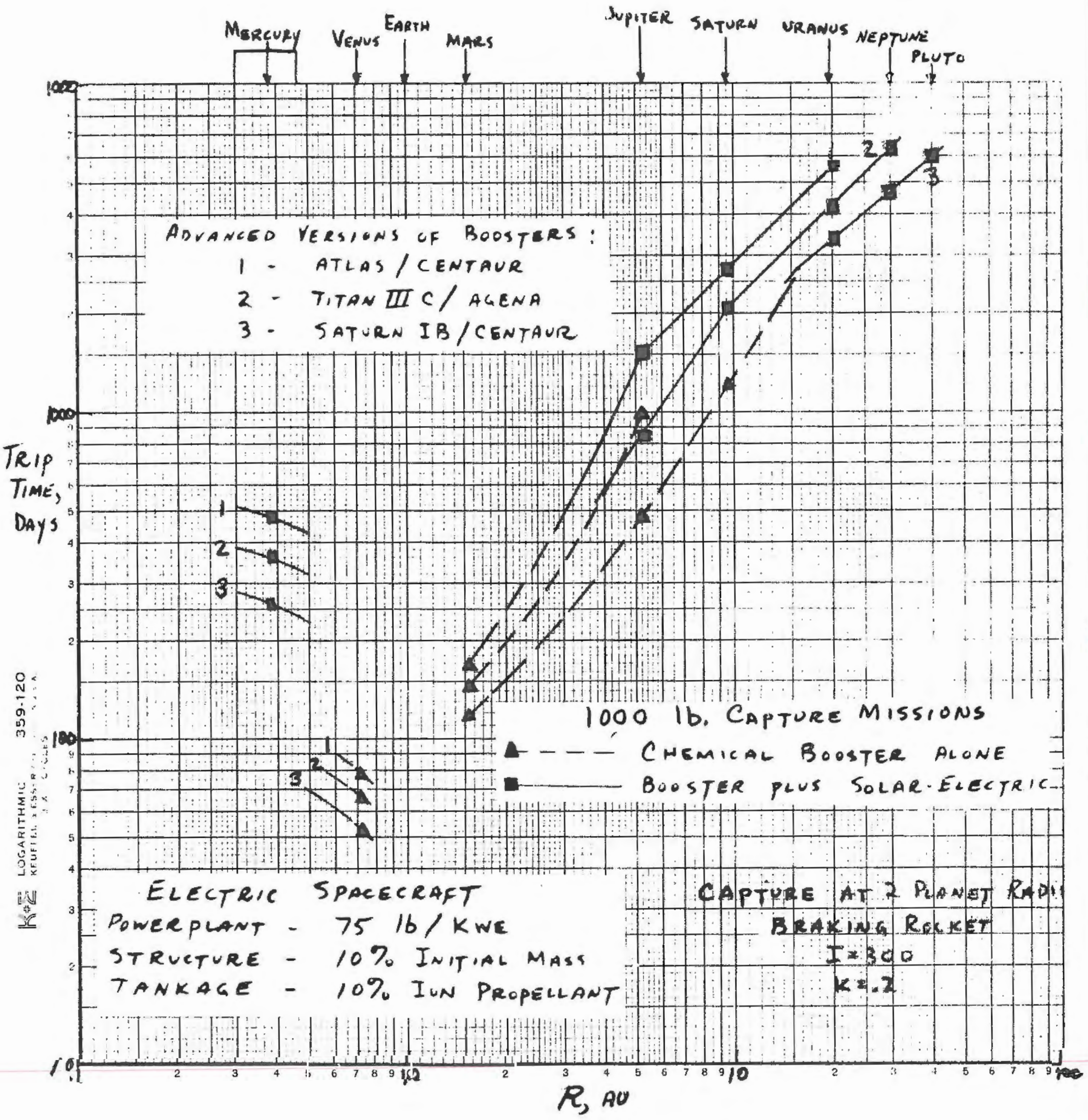

Figure 10 - Minimum Trip Time For 1000 lb. Capture Missions

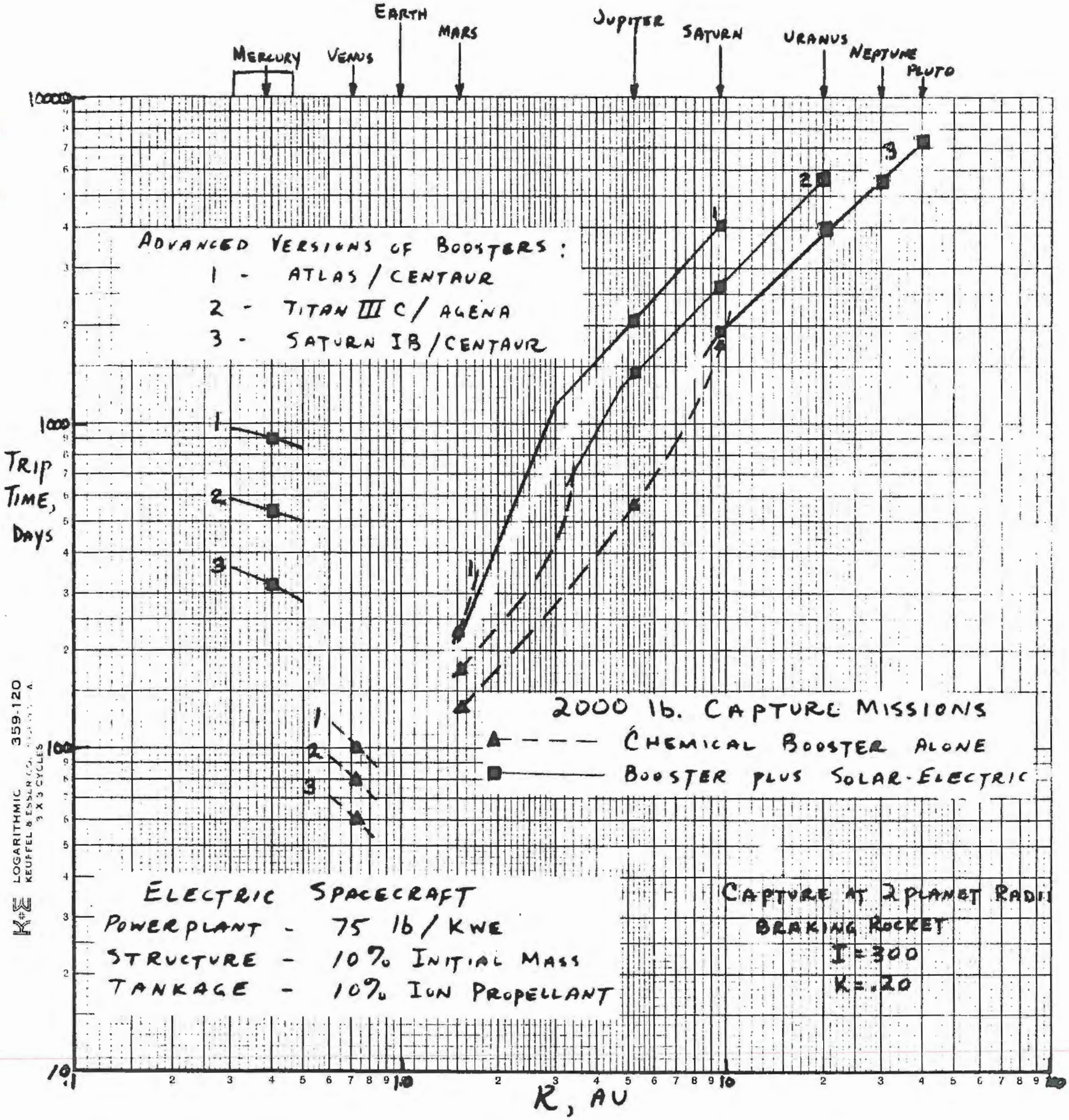

FIGURE 11 - MINIMUM TRIP TIME FOR 2000 lb. CAPTURE MISSION

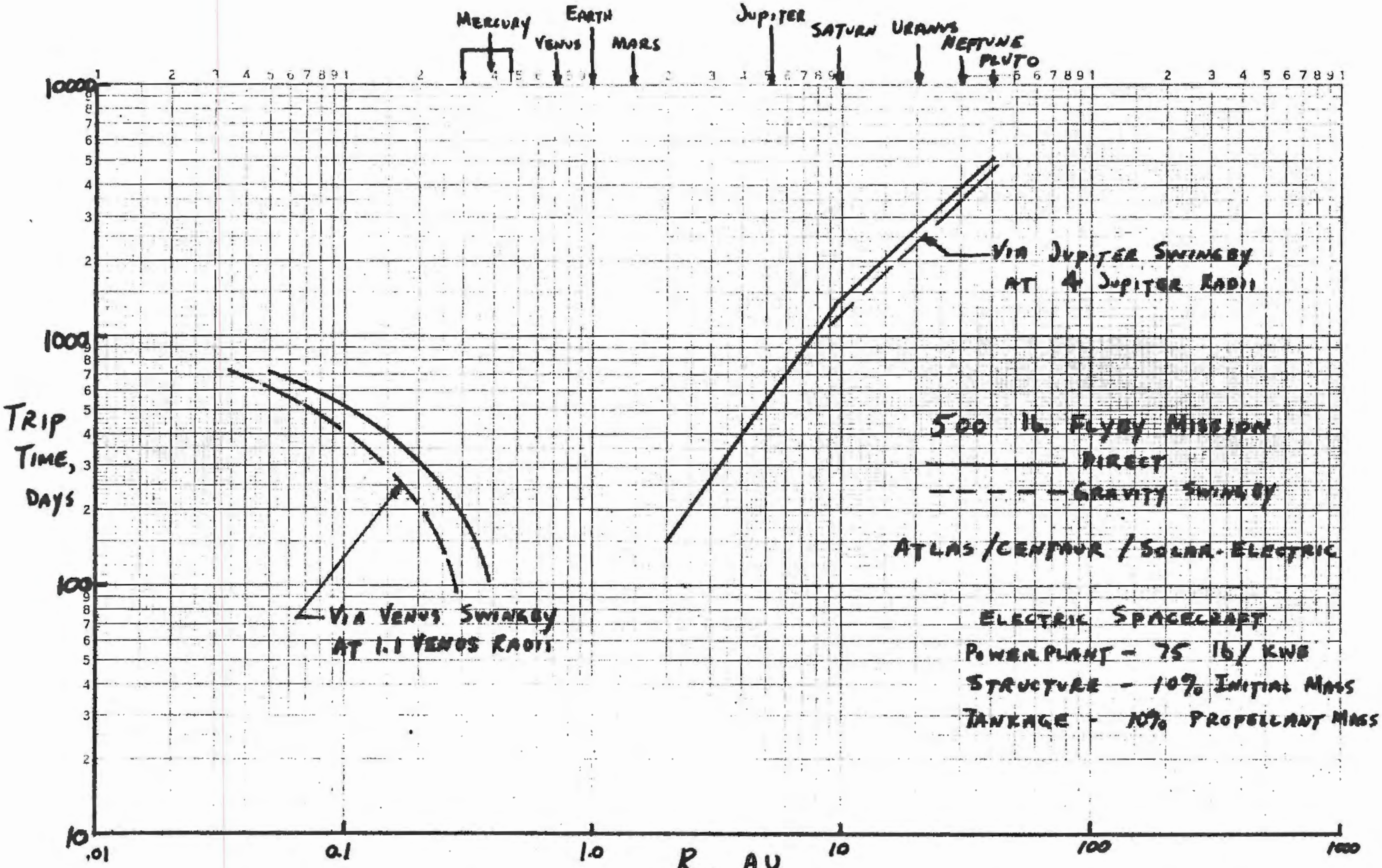

FIGURE 12 - EFFECT OF GRAVITY SWINGBY FOR SOLAR-ELECTRIC SPACECRAFT

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