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JOVIAN PLANET MISSIONS FOR SOLAR CELL POWERED ELECTRIC PROPULSION SPACECRAFT

by Charles L. Zola

Lewis Research Center
Cleveland, Ohio

TECHNICAL PAPER proposed for presentation at Sixth Aerospace Sciences Meeting sponsored by the American Institute of Aeronautics and Astronautics
New York, New York, January 22-24, 1968

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Abstract

A preliminary analysis of multi-mission capability is made for a solar-electric spacecraft. The spacecraft has a 10 kilowatt, 345 kilogram propulsion system and uses the same thrustor specific impulse for all missions. A common launch vehicle (Atlas-Centaur) is used to start the spacecraft on a high energy Earth departure path for flyby and elliptic capture missions to Jupiter, Saturn, Uranus, and Neptune. The electric propulsion trajectories are calculated with optimal control of thrust vector direction but thrust magnitude is changed to account for solar cell power variation with distance from the Sun. Total propulsion time is limited to 800 days or less. Results show that, for all planets beyond Jupiter, high travel angle trajectories of more than one revolution about the Sun are necessary for payloads in excess of 200 kilograms. However, missions to Jupiter can use direct trajectories with propulsion times as short as 400 days.

I. Introduction

Studies of small electric propulsion probe spacecraft for exploration of the solar system date far back[1] and, over the years, have been pursued by many authors with many different methods of approach. Much recent interest [2-7] has been given to the concept of a solar-electric propulsion (SEP) spacecraft, whose primary power source is a large lightweight array of solar-photovoltaic cells. Continued progress in the development of lightweight solar cell power systems[8-12] and efficient, lightweight, long life ion thrustor systems[12-14] could lead to the first useful mission application of electric propulsion in small interplanetary spacecraft.

The author has recently completed an extensive in-house analysis of SEP missions to the Jovian planets--Jupiter, Saturn, Uranus, and Neptune. The purpose of the generalized study was to identify the best trajectories and propulsion system design parameters for an SEP spacecraft based on the Atlas-Centaur launch vehicle. The present paper is based on an observation of the previous general study that a one-design solar-electric propulsion system could deliver satisfactory payloads over a wide variety of missions.

The purpose of this paper is to present an evaluation of the multi-mission capability of the Atlas-Centaur-SEP combination with a fixed design electric propulsion system. The propulsion system has a 10 kilowatt ion thrustor array operating at a specific impulse of 4500 seconds. Payload capability of the system is evaluated for flyby and capture missions to all four Jovian planets.

II. Ground Rules

The five principal ground rules concerning the mission and vehicle parameters assumed throughout this paper are:

(1) Atlas-Centaur launch and injection at 185 kilometers (100 nautical miles) altitude to place the SEP spacecraft on a high energy Earth departure path at the start of each mission.

(2) The solar cell power system delivers 10 kilowatts (at 1 AU) to an ion thrustor array through a power conditioning and regulation system of 90 percent efficiency.

(3) Ion thrustor specific impulse is 4500 seconds with an overall thrustor efficiency of 70 percent.[13]

(4) Total electric propulsion system mass, including solar panels, power regulation, and thrustors is fixed at 345 kilograms.

(5) Maximum electric propulsion time is 800 days, regardless of the necessary trip times to the planets.

Current electric propulsion system lifetime targets are often quoted as 10,000 hours or about 400 days. Yet most outer planet missions will have trip times of several years. Later discussion will show that, for many missions, a 400 day maximum propulsion time puts severe limits on trajectory selection and payload capability for the SEP spacecraft. This paper has therefore used an upper limit of 800 days or about 20,000 hours. Thrust lifetime improvements or propulsion system designs with redundant spares may be needed to allow 800 days of propulsion.

The particular choice of power and specific impulse for the electric propulsion system was made because the generalized study showed them to be the best overall combination for the stated propulsion time limit and the Atlas-Centaur launch vehicle. These principal ground rules and other subsidiary constraints used in the SEP mission analysis for this paper are discussed in more detail in the following sections.

III. Trajectories

All mission trajectories in this analysis essentially start at Earth's surface. As shown in figure 1, the Atlas-Centaur launch vehicle leaves the surface and at 185 kilometers altitude injects the SEP spacecraft at a velocity equal to or greater than escape velocity of 11,020 meters per second. The spacecraft then coasts to the sphere of influence on a hyperbolic conic section. At the sphere of influence patch point the spacecraft velocity relative to Earth, V_s, is vectorially added to Earth's heliocentric velocity, V_e, to determine the initial heliocentric velocity, V_o, of the spacecraft. It is at this point that the electric propulsion phase is assumed to start. For mission trajectories with optimum thrust vector control in the heliocentric phase, the optimum orientation of V_s is in the direction of the initial electric thrust vector.[4, 15]

The heliocentric trajectories are calculated in a simplified, two-dimensional model of the solar system. Circular coplanar orbits about the Sun have been assumed for all the planets with the important constants given in Table 1. The low thrust heliocentric trajectory calculations are made with a calculus of variations trajectory code which is equipped with a simulation of the power output profile of a typical solar cell panel as a function of distance from the Sun. The power curve used in this study, shown in Figure 2, comes directly from a previous analysis[4] which assumed an arbitrary set of solar panel design characteristics.

The ion thrustor array is assumed to operate with variable propellant flow rate (variable total thrust) to conform to the solar power profile, but specific impulse is kept at the ground rule value of 4500 seconds. In actual practice this form of thrustor operation could be closely approximated by a multi-thrustor array with periodic shut-down of individual units.

The optimum trajectory code is incorporated in a multi-variable payload-optimization code which allows the trade-off of mission trajectory and system design parameters such as spacecraft initial mass with injection velocity at 185 kilometers. For the Atlas-Centaur launch vehicle assumed in this study, a portion of the SEP spacecraft-initial-mass curve is shown in Figure 3. Injection velocities for all the SEP missions discussed in this paper are never less than escape velocity, nor are they greater than 11,700 meters per second for most cases of interest.

On arrival inside the sphere of influence of the target planet, the spacecraft is on a hyperbolic flyby path. The capture missions assume that a small storable chemical braking rocket is fired at periapsis of the encounter hyperbola. In this study, the objective of the braking rocket system is to place the payload in an elliptic parking orbit with a periapsis of two planet radii and apoapsis of 200 planet radii. The choice of a 2X200 capture ellipse at each planet is completely arbitrary since the scientific purpose of each particular mission would determine the best capture orbit. For the same trip time and planet, a capture payload is always less than the flyby case due to the propellant and hardware requirements of the braking rocket system. However, the capture mission allows repeated close encounters with the planet surface to gather more scientific data. Periods of 2X200 elliptic capture orbits are shown in Table 1 for each of the Jovian planets.

Trajectory studies to improve the payload capability of the SEP spacecraft for Jovian planet missions revealed a special class of optimal heliocentric trajectory which is uniquely beneficial to solar cell powered spacecraft. This trajectory class is characterized by a large elapsed polar angle of about one extra revolution about the Sun over the elapsed angle of the usual direct low-thrust trajectory from Earth to the target planet.[6, 7]

Typical examples of the direct and high travel angle type of optimal SEP trajectories are shown in Figure 4 with radius plotted versus travel angle. The two types of trajectories shown in the figure are taken from a 1000-day trip time Jupiter mission and a 2000-day Saturn mission but are typical of the radius-travel angle history of each trajectory type to all four Jovian planets. The direct (type A) trajectories usually take less than one revolution about the Sun. The high travel angle class of trajectories, labeled type B in Figure 4, all require about 1.5 revolutions about the Sun. For a given mission, type B trajectories require a higher effective value of low-thrust ΔV or total propulsive effort than type A trajectories. If a constant power supply were available (e.g., nuclear-electric), type B trajectories would be highly nonoptimum. But typical solar panels experience a severe drop in power output as they move away from the Sun. The B type trajectories, by staying closer to the Sun over a large fraction of the trip, derive more useful solar energy for propulsion which can offset their higher ΔV requirements. The extra time spent near the Sun on the initial loop of a type B trajectory does, however, call for longer propulsion times than are needed in type A cases. It can be seen in Figure 4 that, for the same 1000-day trip to Jupiter, the B trajectory takes almost twice the time to reach 3 AU as the direct trip. Later figures will show that, if up to 800 days of propulsion can be tolerated, type B trajectories allow payloads for SEP spacecraft that, for most missions, are much greater than missions using direct trajectories.

Another trajectory class of 2.5 revolutions about the Sun also exists but is not included in this study. The propulsion time required to complete two loops about the Sun is far beyond the 800 day ground rule of this paper. Also, the payload advantage of the 2.5 revolution type over most type B cases is too small to offset their longer propulsion time requirement.

IV. Payload

The payload calculated in this analysis might be more aptly termed a gross payload since it must be large enough to include not only scientific instrumentation but also many other subsystems such as navigation, telemetry, and environmental control equipment. An early study[16] of payload requirements for a Jupiter mission and also the Mariner spacecraft experience show that a gross payload of 200 kilograms would be sufficient for limited-objective missions having 20 to 30 kilograms of scientific instruments.

The expression for gross payload m_L of the SEP spacecraft is:

$$m_L = m_i - m_{st} - m_{ps} - m_p - m_t - m_B$$

In other words, the initial mass of the spacecraft m_i is reduced by the individual masses of the structure m_{st}, propulsion system m_{ps}, ion propellant m_p, tankage m_t, and braking rocket system m_B. The remainder is defined as payload. The braking rocket system is, of course, not present in flyby missions. Structure is assumed to be 10 percent of the initial mass of the spacecraft. Tankage and ion propellant feed system is assumed to be 10 percent of the ion propellant mass m_p. The ion propellant mass requirement varies from mission to mission but is often in the neighborhood of 30 percent of the total mass. Similarly, the mass requirement for the braking rocket system depends on the amount of ΔV required to capture the spacecraft in the 2X200 elliptical parking orbit at each planet. The braking rocket propellant is assumed to have a specific impulse of 300 seconds and the hardware requirements are assumed to be 20 percent of the propellant mass needed for the braking maneuver.

The electric propulsion system mass is fixed at 345 kilograms and consists of three major parts--solar cell panels, power regulation equipment, and ion thrustors. This study assumes a 10 kilowatt power supply to the ion thrustors through a power conditioning system of 90 percent efficiency.(3,11,12) There must actually be 10/.9 or 11.1 kilowatts of output power from the solar panels. At 22.7 kilograms (50 pounds) per kilowatt,(2,8) the solar cell panels and deployment system mass is then 253 kilograms. The remaining 92 kilograms (~200 pounds) is is assumed to be sufficient to account for the 10 kilowatt thrustor array and its associated power conditioning and regulation system.(2,3,11,12)

V. Flyby Missions

Gross payloads for SEP flyby missions to all four Jovian planets are shown in Figure 5 as a function of trip time. The full 800 days of allowed propulsion time are used for all the type B mission trajectories to each planet and also for the type A Saturn flyby mission. The type A Jupiter flyby trajectories have, however, been held to a maximum propulsion time of 400 days. It is shown in a later figure that a 400-day limit on Jupiter type A trajectories has very little effect on gross payload.

It is clear in Figure 5 that the only practical mission for type A trajectories is the Jupiter flyby at trip times shorter than 900 days. The type A Saturn flyby data is included in the figure only to show the rapid fall-off of payload if direct trips are used for SEP missions beyond Jupiter.

The drop in payload capability for Uranus and Neptune flyby missions might raise a question as to whether the strict adherance to the ground rules of this paper is too damaging to SEP performance for these missions. For this reason, the two solid square data points are shown near the Neptune curve. These points represent the flyby payloads that could be delivered to Neptune if spacecraft power, thrustor specific impulse and propulsion time are completely optimized. The possible gain in payload is 50 to 60 kilograms over the Neptune trip time range from 4000 to 7000 days. It will be shown in a later figure that most of the gross payload penalty is due to the 800-day propulsion time limitation.

VI. Capture Missions

Figure 6 shows the capture mission payloads of the SEP spacecraft for a 2X200 elliptical capture orbit about each planet. A data curve labeled SPLIT is given for each planet to designate that the payload package, with attached braking rocket, is first separated from the main spacecraft before the capture braking maneuver at 2 planet radii.

The reason for the payload separation is that chemical propulsion braking of the whole arrival mass of the SEP spacecraft results in gross payloads of 50 kilograms or less at all planets except Jupiter. However, it is probably advisable to capture the whole (curve labeled WHOLE) spacecraft at Jupiter. The payload penalty of about 100 kilograms is compensated by an expected available power output of about 700 watts from the solar array to power the scientific payload while in orbit. In contrast, the SPLIT payload must be penalized by the weight of a separate power supply. It is for this reason that, for Jupiter, a capture payload range between WHOLE and SPLIT cases is shown over the Jupiter trip time range.

The SPLIT method is the only logical capture mode for Saturn, Uranus, and Neptune. For these planets, WHOLE vehicle braking payloads are less than 50 kilograms and power output of the 10 kilowatt array is uncertain but definitely less than 200 watts. On the other hand, a 200 watt radioisotope thermal generator could be carried as part of the SPLIT payload and still leave mass for other purposes. Even then, SEP payload is very low for Uranus and Neptune captures. Larger payload margins could be obtained, but at the expense of larger launch vehicles and correspondingly larger solar-electric propulsion systems.

VII. Propulsion Time

Figure 7 is given here to show the effect of propulsion time on gross payload for the 1000-day Jupiter and 5000-day Neptune flyby missions. The previous payload figures showed that trip times to the outer planets are very long. Many important components of the SEP spacecraft will need a high life expectancy if the probe mission is to succeed. A major requirement is a long life ion thrustor system. However, due to the rapid fall off of solar power, most of the propulsion work of the ion thrustors is complete by the time the spacecraft reaches 3 AU. It is for this reason that an upper limit on propulsion time can be set at 800 days. For the 1000-day Jupiter flyby, curves are shown in Figure 7 for both the A and B type mission trajectories. A propulsion time of 800 days is optimum for the B type Jupiter case, and if necessary, can be cut to as low as 550 days before the payload is the same as the A type Jupiter flyby. If very short propulsion times are required, type A mission trajectories are the best choice for trip times beyond 900 days. For all type A Jupiter missions, a 400-day propulsion time limit has little or no effect on gross payload.

The 5000-day Neptune flyby curve is also shown in Figure 7 because it is an extreme example of the payload reduction caused by an 800 day limit on propulsion time. The payload lost is about 60 kilograms or 20 percent of the maximum value at 1400 days of propulsion time. Similar curves for Uranus would show a payload loss of no more than 50 kilograms and, for Saturn, no more than 30 kilograms.

It should be noted that the propulsion time limits discussed in this paper are not necessarily the required operating lifetimes of the ion thrustors. Figure 8 is given to show the propulsion power variation along typical type A and B trajectories for the SEP missions discussed in this paper. The type A is a 1000-day trip to Jupiter with a 400-day propulsion time limit, and the type B case is a 2000-day Saturn trip with an 800-day propulsion time. Such curves result from the combined effects of solar power variation with radius and radius with time along each trajectory type. For a multi-thrustor electric spacecraft, the power variation means that for most of the propulsion time, the required number of operating thrustors is less than the number on board. The shape of either the A or B type power history could be used to advantage in achieving, with very few spares, a high reliability thrustor system beyond the expected life of individual thrustors.

Figure 9 shows the mass breakdown of SEP spacecraft for three typical Jupiter missions taken from the data of this study. The two 1000-day Jupiter flyby examples contrast the type A, 400-day propulsion time, and type B, 800-day propulsion time cases. The major difference between the type A and type B examples is in the ion propellant section. The type A direct mission, to compensate for the rapid solar power decrease, requires an injection velocity of nearly 11,600 meters per second and therefore has a relatively small initial mass of 805 kilograms. But, also due to the high injection velocity, the ion propellant requirement of this spacecraft is more than 200 kilograms less than needed for the SEP mission using a type B trajectory. The type B Jupiter flyby requires an injection velocity of only about 11,200 meters per second and therefore starts the mission with about 275 kilograms more total mass than its type A counterpart. Additional structure, propellant, and tankage requirements total about 235 kilograms more than the type A spacecraft and therefore reduce the payload difference to only 40 kilograms.

The bottom schematic in figure 9 represents an SEP spacecraft breakdown for a 1400-day trip time Jupiter capture mission. The propellant, tankage, and structure requirements are similar to the 1000-day type B flyby example. The SEP 1400-day mission, however, requires an injection velocity only slightly over escape and therefore has a high initial mass of 1225 kilograms. Gross payload in less, however, than the 1000-day flyby mission due to the 163 kilogram braking rocket system needed to capture the whole spacecraft in the 2X200 elliptic orbit.

VIII. Concluding Remarks

This paper has attempted to show that a multi-mission interplanetary probe could be based on a single launch vehicle (the fairly small Atlas-Centaur) and a fixed-design solar-electric propulsion system. At least 200 kilograms of gross payload can be delivered on most flyby and capture missions to all four Jovian planets. Reasonable payload levels require up to 800 days of propulsion time and very long trip times. The trip times range from 2 to 4 years for Jupiter missions up to 12 to 18 years for the Uranus and Neptune missions. Such long trip times are the most detrimental aspect in the consideration of outer planet probe missions using the launch vehicle and SEP system performance assumed for this paper. Larger payloads and shorter flight times would require a higher performance launch vehicle, much lighter components in the electric propulsion system, and the possible use of Jupiter swingby trajectories.

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TABLE I - ASSUMED PHYSICAL CONSTANTS OF THE PLANETS

	Sun	Earth	Jupiter	Saturn	Uranus	Neptune
Gravitational constant, μ, m^3/s^2	1.3245×10^{20}	3.9860×10^{14}	1.3003×10^{17}	3.8745×10^{16}	5.9461×10^{15}	7.0566×10^{15}
Heliocentric orbit radius, meters	--	1.4950×10^{11}	7.778×10^{11}	14.261×10^{11}	28.691×10^{11}	44.956×10^{11}
Heliocentric orbit velocity, m/s	--	29765	13050	9640	6780	5470
Planet radius, meters	--	6.3712×10^{6}	6.9892×10^{7}	5.7532×10^{7}	2.3701×10^{7}	2.1535×10^{7}
Sphere of influence radius, number of planet radii	--	150	690	950	2180	3970
Velocity of circular orbit at 2 planet radii, m/s	--	5593	30500	18350	11200	12800
Period of 2X200 capture ellipse days	--	--	120	164	110	80

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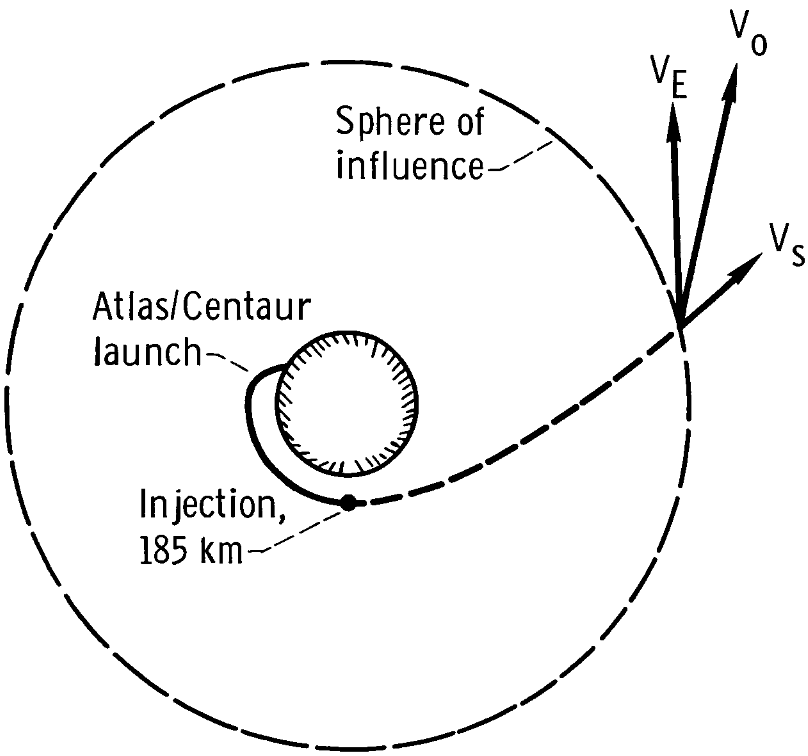

Figure 1. - Spacecraft launch and earth departure.

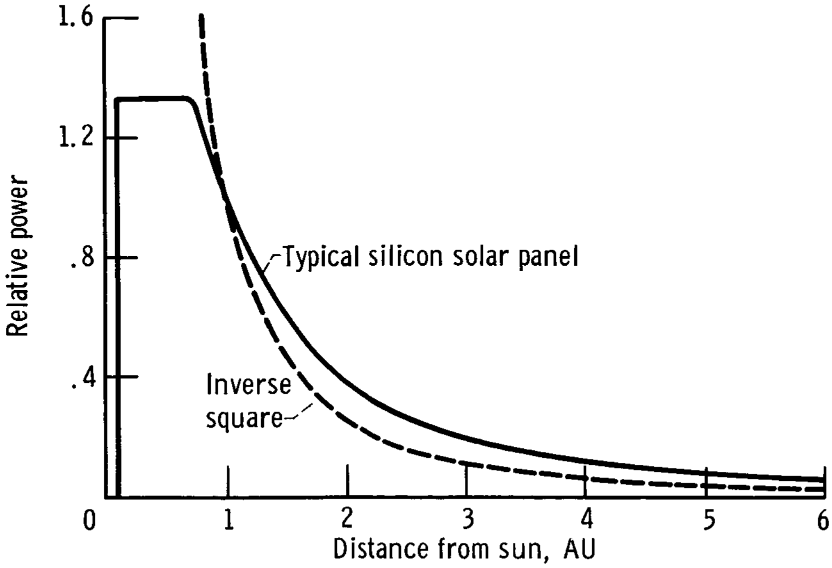

Figure 2. - Power variation of solar cell panel.

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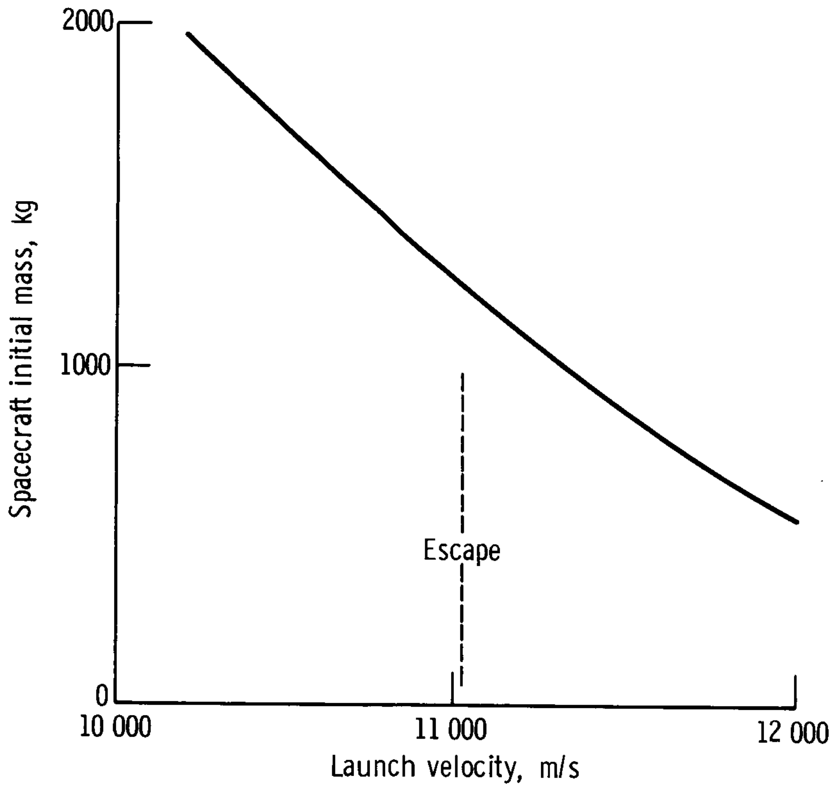

Figure 3. - Electric spacecraft initial mass. Atlas-Centaur launch vehicle at 185 kilometers.

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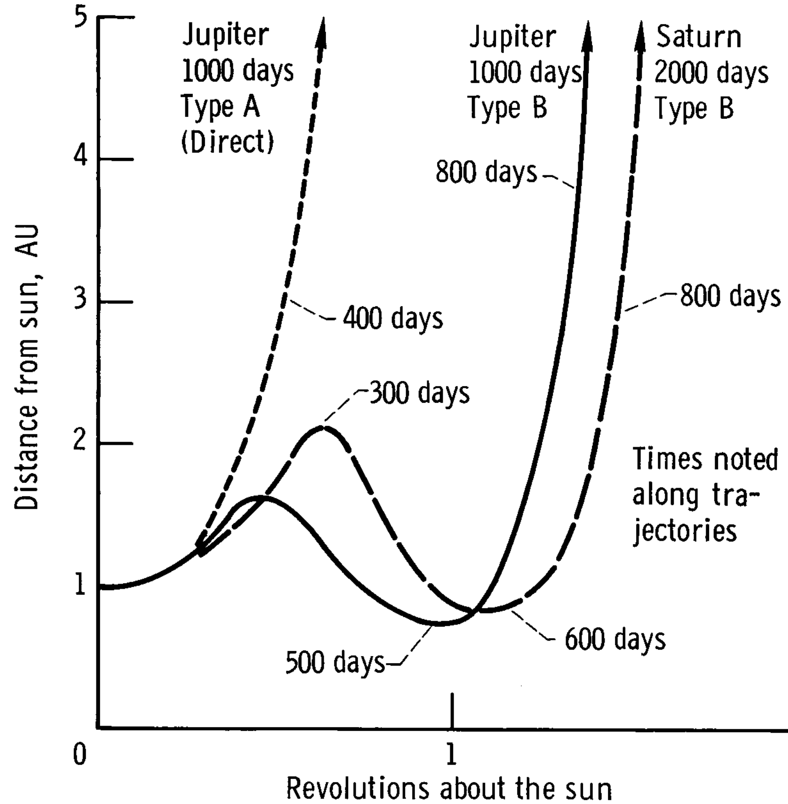

Figure 4. - Optimum trajectories for solar-electric propulsion.

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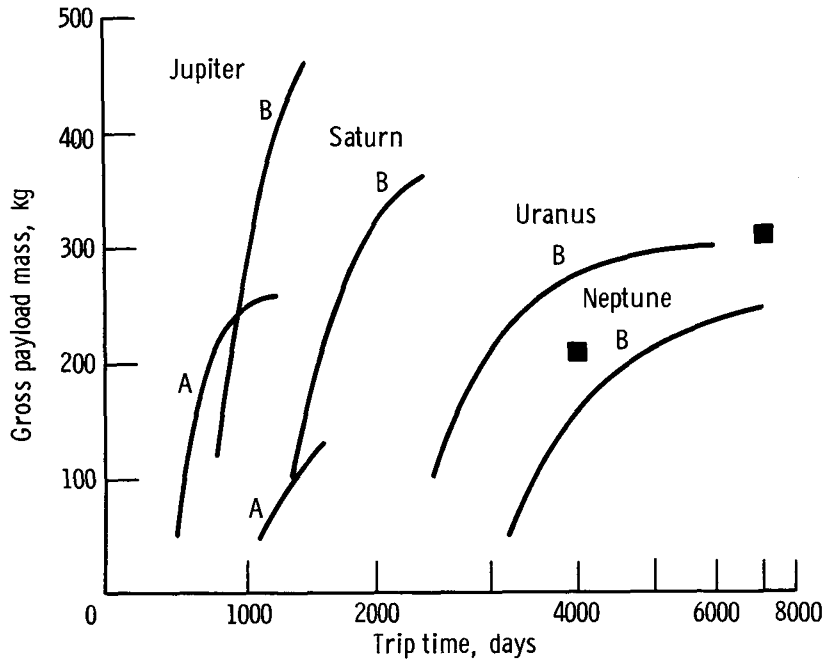

Figure 5. - Solar-electric propulsion flyby payloads for Atlas-Centaur launch vehicle. Power, 10 kilowatts; I_S, 4500 seconds; maximum propulsion time, 800 days.

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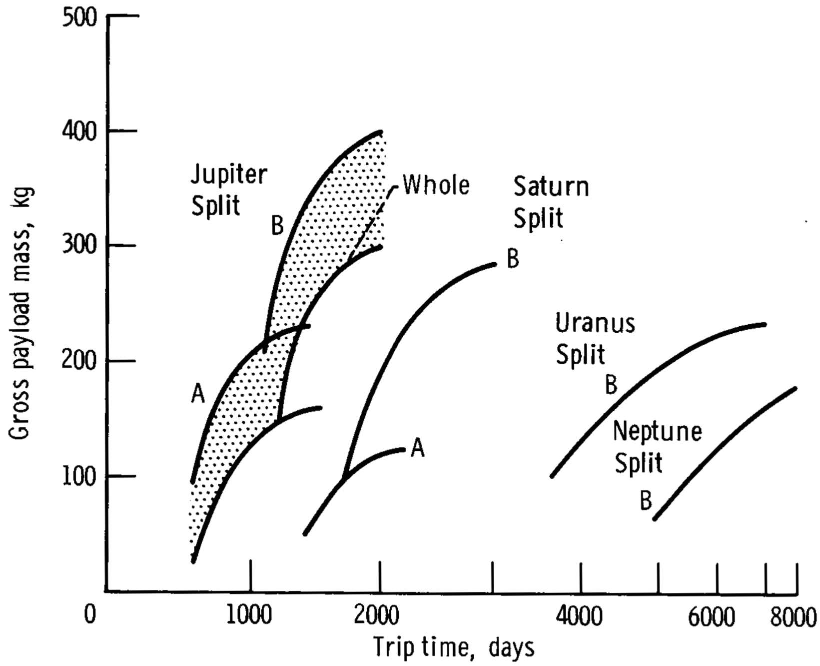

Figure 6. - Solar-electric propulsion capture payloads for the Atlas-Centaur launch vehicle. Power, 10 kilowatts; I_S, 4500 seconds; maximum propulsion time, 800 days; capture ellipse, 2x200.

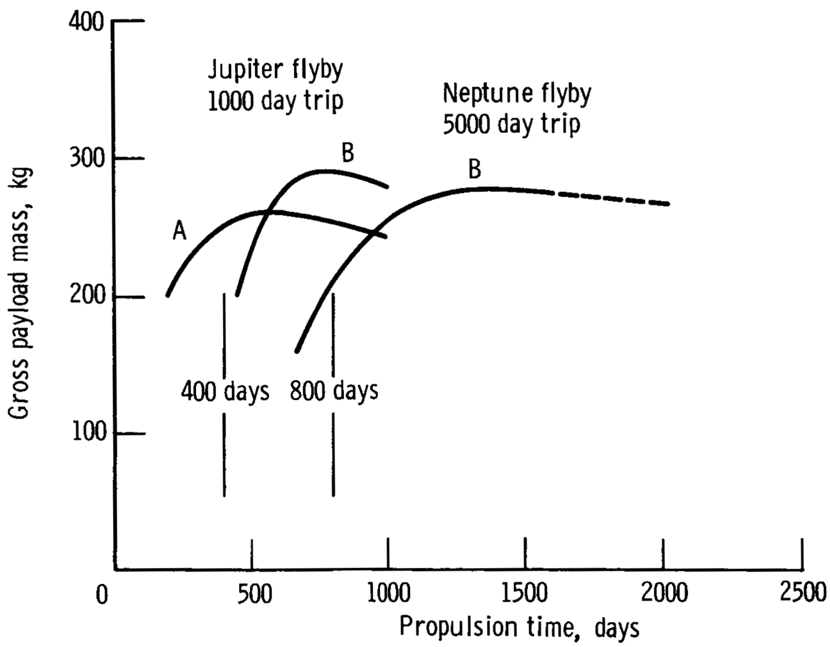

Figure 7. - Effect of propulsion time for the solar-electric propulsion Atlas-Centaur launch vehicle. Power, 10 kilowatts; I_S, 4500 seconds.

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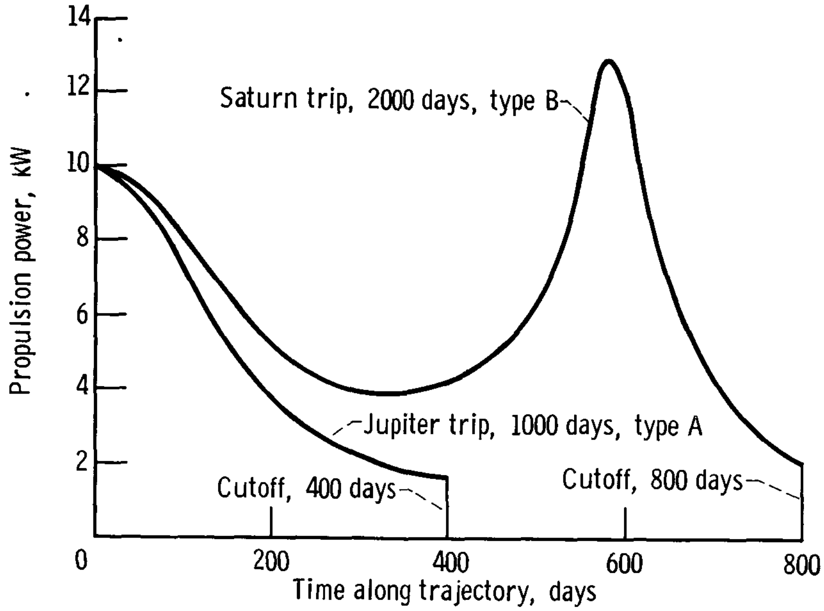

Figure 8. - Typical propulsion power histories for type A and type B trajectories.

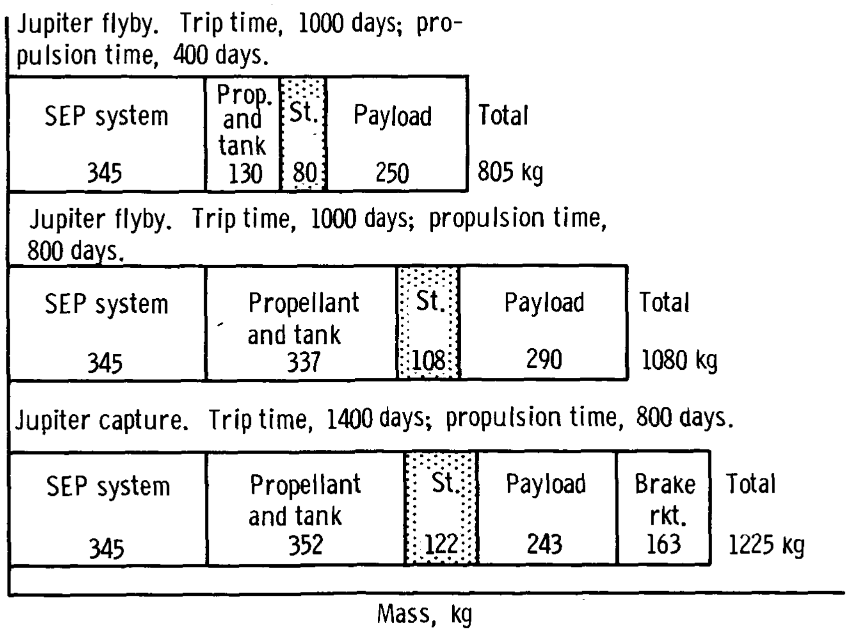

Figure 9. - Total mass and component mass typical SEP spacecraft. Power, 10 kilowatts; I_s, 4500 seconds. Atlas-Centaur launch vehicle.