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EARLY APPLICATION OF SOLAR-ELECTRIC PROPULSION TO A 1-ASTRONOMICAL-UNIT OUT-OF-THE-ECLIPTIC MISSION

by William C. Strack and Frank J. Hrach
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Cleveland, Ohio

TECHNICAL PAPER proposed for presentation at
Eighth Electric Propulsion Conference sponsored by
the American Institute of Aeronautics and Astronautics
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NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

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Abstract

Solar-electric propulsion is evaluated for an early application to an out-of-the ecliptic mission. Relatively short flight times (100-475 days) are used to assess the performance of hardware that could be built with present technology. The electric propulsion system specific mass is assumed to be 30 kilograms per kilowatt and current thruster system efficiencies (e.g., 57 percent at 2600 seconds specific impulse) are employed. Furthermore, the thrust program is simple - the thrust is constant and always directed normal to the instantaneous plane of the spacecraft orbit. The thrust is permitted to be turned off, however, and the typical trajectory is composed of several power-on and power-off constant-radius subarcs. Two currently available launch vehicles are assumed: Atlas (SLV3C)/Centaur and Titan IIIC.

The results show that a negligible performance loss is incurred by using the simple constant-radius thrust control program compared with the more complicated (variable thrust direction and solar power) variable-radius case. Also, a fixed design spacecraft with 10 kilowatts of electric power and 2600 seconds specific impulse can deliver nearly as much payload (never more than 20 percent less) to a given heliographic inclination as an entire family of designs with optimum values of power and specific impulse. This holds true for both launch vehicles.

This fixed design electric spacecraft compares favorably with an uprated version (1040 kilograms of propellant) of the Burner II chemical stage. With the Atlas/Centaur, for example, the maximum heliographic inclination attainable for 200 kilograms of net spacecraft mass is 25 degrees for the uprated Burner II and 37 degrees for the fixed design electric spacecraft. With Titan IIIC these values are 27 and 41 degrees. In these examples, the electric spacecraft requires a one-year propulsion time, and about 470 days total to reach maximum latitude compared to 91 days for the all-chemical systems.

Introduction

The purpose of an out-of-the-ecliptic mission is to gather scientific data on interplanetary fields and particles, and to observe solar activity at high solar latitudes(1). All of such data accumulated to date have been essentially within the ecliptic plane and primarily at 1 astronomical unit (AU). The past Mars and Venus probes have provided some limited data in the 0.7 to 1.5 AU range. Eventually this data base will be expanded to include a wide range of distances from the sun and inclination angles to the ecliptic plane. Near future plans, however, will be necessarily modest - especially in regard to the inclination angles due to the very high energy expenditures normally required to make plane changes. One possible way to avoid high energy expenditure is to use the gravitational field of Jupiter to make plane changes. Large inclination angles to the ecliptic are possible if a close passage is made(2). The advantage of a Jupiter gravity turn, however, is tempered by increased mission time (it takes about 500 days just to get to Jupiter) and the restricted class of orbits that the spacecraft may attain after the Jupiter encounter. A particularly desirable mission, for example, is to place a spacecraft in an inclined circular orbit at 1.0 AU, and this cannot be reasonably done by means of a Jupiter swingby. This mission has several advantages that suit it particularly well for early application: (1) data are obtained at a constant 1 AU; thus, effects due to inclination are not obscured by effects due to radius, and can be compared more meaningfully with existing data, (2) Earth-to-vehicle communication distances are comparatively small, and (3) flux to the solar panels remains constant.

An alternative to using chemically-fueled rocket propulsion to do this mission is to use low-thrust electric propulsion. Enough analytical results have been generated(3-5) to suggest that an electric powered spacecraft can compare favorably with all-chemical propulsion systems - especially at high inclination angles. The studies reported in Refs. 3 and 4 consider the case of constant power, variable radius trajectories that do not apply to solar-electric propulsion since solar power varies with radius. Reference 5 gives results for the case of thrust always directed normal to the instantaneous orbit plane in order to keep the spacecraft constrained to a constant radius of 1 AU. This is a fairly simple thrust program to implement and results in constant power output from the solar panels. This avoids the problem of matching a continuously varying power level to the thruster system. However, the study was mainly limited to short time, single burn mission profiles using an Atlas/Centaur type launch vehicle. The present study generalizes this concept to include two and three burn mission profiles and also the Titan IIIC launch vehicle. The objective is not so much to present large amounts of parametric data, as in past studies, but to determine how well an early state-of-the-art solar-electric spacecraft would perform this mission compared to all-chemical systems, and to determine reasonable values of such design variables as specific impulse, power loading, and propulsion duty cycle.

Analysis

Trajectory Assumptions

Although most of the results given in previous studies are in terms of orbital inclination to the ecliptic, it is considered preferable to present data in terms of inclination to the Sun's equator since most of the interplanetary phenomena to be measured depend on heliographic coordinates. The ecliptic plane is inclined 7.2° to the Sun's equator. Thus a launch to Earth escape velocity produces an initial heliographic inclination i_o of

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7.2°. Higher energy launches produce a hyperbolic excess velocity V_h that is most effective in changing inclination when applied at the nodes as shown in sketch A. The electric propulsion system is assumed to be turned on soon after the high thrust launch to at least escape energy; that is, in heliocentric space with velocity V_o whose magnitude is identical to the Earth's velocity V_E in order to maintain a circular orbit at 1.0 AU. From the sketch it is easy to show that the initial heliographic inclination is

$$i_o = 7.2^\circ + 2 \sin^{-1}\left(\frac{V_h}{2V_o}\right) \tag{1}$$

The Earth's gravitational effect is ignored when the spacecraft is in heliocentric space. The thrust is directed normal to the instantaneous orbit plane so that the size and shape of the orbit are not changed as the inclination is increased. This could be done without thruster gimbaling by orienting the thrust vector with the spacecraft attitude control system. Regardless of how the thrust vector is controlled, at least some attitude control is required to keep the solar panels facing the Sun. The time rate of change in inclination is

$$\dot{i} = \frac{a}{V_o} \cos u \tag{2}$$

where a is the thrust acceleration and u is the argument of latitude (sketch b). The rate decreases as the spacecraft moves away from the node and is zero at the first antinode ($u = \pi/2$) -- a position of maximum distance from the Sun's equatorial plane. Beyond the first antinode the thrust direction must be reversed in order to continue increasing orbital inclination. And again when the spacecraft reaches the second antinode ($u = 3\pi/2$), the rate of change of inclination vanishes and the thrust direction should be reversed. Hence, even for this simplified thrusting method, continuous control of the spacecraft attitude is required and occasional complete reversals of thrust direction are necessary. The thrusting program is illustrated in sketch b for a three-burn class trajectory, although one- and two-burn trajectories are also included in the study. In the majority of cases considered the thrusters are turned off near the antinodes because the ineffectiveness of thrusting there results in a payload penalty, as will be shown later.

Thrusting also changes the line of nodes (sketch b) while the orbit inclination is increasing.

The rate of change in the longitude of ascending node is

$$\dot{\Omega} = \frac{a}{V_o} \frac{\sin u}{\sin i} \tag{3}$$

The change in Ω between the first and third nodes is generally 10 to 35 degrees forward. Equations (2) and (3) as well as the other equations of motion are derived in Ref. 6.

Unless otherwise specified, the net spacecraft mass is maximized for a given final heliographic inclination by optimizing: the launch energy (equivalent to i_o), the electric thruster specific impulse, and all the thruster shutdown and restart times. The total time required to achieve a given inclination is not specified because, as will be seen later, optimal times exist within each trajectory class and these are determined by the optimization of the thruster shutdown and restart times. The electric power level is used in an iteration loop to drive the final inclination to its desired value. The Lewis N-Body computer program[7] was used to calculate the trajectories and optimize the free variables.

Chemical Systems Assumptions

The assumed launch vehicle performance is shown in Fig. 1 for the Atlas/Centaur and the Titan IIIC. The launch mass against burn-out velocity V_b at 185 kilometers altitude comes from Ref. 8. The hyperbolic excess velocity V_h is determined by the booster burnout velocity V_b and circular orbit velocity V_c:

$$V_h^2 = V_b^2 - 2V_c^2 \tag{4}$$

Equations (1) and (4) are combined with the curves in Fig. 1 to yield the relationship between launch mass and initial inclination.

Also shown in Fig. 1 is the performance data for these two boosters with an uprated Burner II stage added on[8]. The 1040 kilogram propellant loading version of Burner II assumed here is not currently available[9]. Nevertheless, it is used in the comparison with the solar-electric system in order to compare both system types at approximately the same technology level.

Solar-Electric Spacecraft Assumptions

The electric propulsion system specific mass α is assumed to be 30 kilograms per kilowatt. This number is generally considered to represent hypothetical solar-cell powered spacecraft at the present level of technology[10, 11]. The propulsion system mass m_{ps} includes both the power and thrust subsystems as defined by the suggested nomenclature in Ref. 12. The power subsystem includes primary power, thermal control, cabling, support structure, etc. and the thruster subsystem includes thrusters, power conditioning control, cabling, support structure, etc. The tankage mass m_t is assumed to be 10 percent of the propellant mass m_p and includes tank structure, plumbing, residuals, reserves, etc. With these assumptions the net spacecraft mass m_n is

$$m_n = m_o - m_{ps} - m_p - m_t \tag{5}$$

$$= m_o - \alpha P_e - m_p - 0.1\, m_p \tag{6}$$

The net spacecraft mass includes more than just the scientific payload. It also includes equipment for guidance, attitude control, thermal control, support structures, communications, data handling and computation. In these expressions m_o is the initial spacecraft mass and P_e is the required electrical power. P_e is calculated from the useful kinetic power in the jet exhaust P_j and thrust subsystem efficiency η_{ts}:

$$P_e = \frac{P_j}{\eta_{ts}} = \frac{\frac{1}{2}\dot{m}_p V_j^2}{\eta_{ts}} = \frac{\dot{m}_p \left(g_o I_s\right)^2}{2\eta_{ts}} \tag{7}$$

where V_j is the exhaust velocity, I_s is the specific impulse, and g_o is the gravitational constant (g_o = 9.80665 m/s^2). The thrust subsystem efficiency η_{ts} is the product of the thruster efficiency and the power conditioning efficiency (assumed to be 0.88):

$$\eta_{ts} = \frac{0.88\ E_o}{1 + \left(\frac{I_o}{I_s}\right)^2} \tag{8}$$

This equation is based on an idealized thruster (assuming constant ionization power losses[13]) and has been found to correlate experimental results of real thrusters reasonably well. E_o is the asymptotic value of thruster efficiency at infinite I_s and I_o is the specific impulse for a thruster efficiency of 1/2 E_o. Eq. (8) fits the curve in Ref. 14 labeled "future 2-3 kW" by letting E_o = 0.85 and I_o = 1465 seconds. In view of current data for 30 centimeter diameter insulated grid thrusters, Ref. 14 suggests regarding this curve as 1968 "present" curve at the 2 to 3 kilowatt level. Reference 3 predicts the mid-1970's technology level with an efficiency curve that is represented by Eq. (8) by setting E_o = 0.957 and I_o = 1630 seconds. To be conservative, most of the calculations employ the 1968 "present" data but for the sake of comparison some projected mid-1970's data is also used.

Results and Discussion

Flight Time and Trajectory Classes

Typical results of net spacecraft mass as a function of flight time are presented in Fig. 2. The particular system illustrated is a Titan IIIC-launched solar-electric spacecraft that attains a heliographic inclination of 30 degrees. The solid curve represents the restricted case of all propulsion -- no coast subarcs are permitted. Generally, this curve rises rather steeply and shows the marked payload improvement possible with increased flight time. The flight times in Ref. 5 are primarily in the 80 to 100 day range which, in this case, are not attractive -- yielding only 35 kilograms of net spacecraft mass. But increasing the flight time to 275 days raises the net mass to 440 kilograms while 440 day trips provide a further increase to nearly 700 kilograms.

Note that three distinct local maxima exist that are some six months apart. These occur because the spacecraft is constrained to a continuous thrust program that is relatively ineffective every six months when the craft is near an antinode (see Eq. (2)). For example, if the powered flight time is 275 days thrusting terminates 22 days before the second antinode is reached; but for 300 day flights the thrusters operate 7 days after the second antinode. Hence the extra 25 days of the 300 day mission are spent wastefully by thrusting in the proximity of an antinode. The net result of this inefficiency is an 8 percent drop in net spacecraft mass compared to the 275 day flight.

If the no-coast constraint is removed, the existence of these three local maxima result in the definition of three distinct trajectory classes: (1) those trajectories that have a single thrust arc, (2) those that have one coast arc between two thrust arcs, and (3) those that have two coast arcs and three thrust arcs. Additional flight time would of course result in additional trajectory classes involving even more coast and thrust arcs. For conciseness the classes considered herein are simply referred to as single-burn, two-burn, and three-burn trajectories. The circled points in Fig. 2 show the performance increase due to relaxing the no coast constraint at three specific flight times. There is no benefit with the single-burn class since it is identical with the all propulsion class. The two-burn benefit is a 5 percent increase in net spacecraft mass and the three-burn benefit is 6 percent. The power levels (not shown) are essentially unchanged. Thus the performance advantage of coasting trajectories is not great. However, the thrust direction must be changed by 180 degrees at the antinodes regardless of whether coast arcs are permitted or not, and it might be necessary (although unlikely) to shut the thrusters down during this reorientation maneuver to avoid disturbance torques. Also, scientific data gathering and communication is most desirable at such times and the extra power made available by thruster shutdown might be used for these other purposes. For all of these reasons, the remaining data will be shown only for the optimum flight times with coast arcs permitted.

Performance of Electric and Chemically Powered Spacecraft

The potential performance of a solar-electric system is compared to the all-chemical systems in Fig. 3. Part a is for the Atlas/Centaur while part b is for the Titan IIIC launch vehicle. Net spacecraft mass is plotted as a function of final heliographic inclination for the launch vehicle by itself (dotted curve), the launch vehicle with the uprated version of the Burner II stage added (solid curve), and the launch vehicle with fully optimized solar-electric spacecraft (the spacecraft design changes along each curve) added (dashed curves). There are three curves for the electric spacecraft -- one for 1-burn trajectories, one for 2-burn trajectories, and one for 3-burn trajectories. The time required to reach an antinode following the final thruster shutdown is also noted for each curve. This time is treated as the mission time (i.e., rather than the time to attain a specified inclination) since the scientific data to be collected is of most interest at the antinodes. Actually, the mission time for the solar-electric case varies slightly with inclination, but since the variation is only several days an average time is quoted.

The values of net spacecraft mass of main interest lie approximately between 200 and 400 kilograms. These estimates come from related mission spacecraft such as the 400 kilogram Mariner 7 and the proposed 210 kilogram spacecraft for project HELIOS. The HELIOS mission[15] is a 0.3 AU solar probe with some 50 kilograms of scientific experiments aboard. Both launch vehicles are limited to heliographic inclinations of 19 degrees if no upper stage propulsion is used. Adding an uprated Burner II stage would raise this limit to about 25 degrees for the Atlas/Centaur or 27 degrees for the Titan IIIC for 200 kilograms of net spacecraft mass. If hypothetical and fully optimized solar-electric systems are substituted for the Burner II, the performance is improved only if 2- or 3-burn trajectories are used. The 2-burn trajectories would allow 32 degrees for 200 kilograms net spacecraft mass using the Atlas/Centaur or 36 degrees using the Titan IIIC. The 3-burn trajectories would permit 39 degrees using the Atlas/Centaur or 43 degrees with the Titan IIIC (not shown). Roughly speaking then, an uprated Burner II looks attractive in the 20-25 degree range while solar-electric spacecraft look attractive in the 25-43 degree range.

It must be emphasized that the solar-electric data in Fig. 3 represents a whole family of spacecraft, optimized with regard to specific impulse, installed electric power, launch velocity, and coast arc timing. The data, therefore, do not reflect the performance of a single spacecraft design. Such a single design would have fixed values of specific impulse and electric power, although launch velocity and coast arc timing could still be optimized. The actual performance of such a single design is presented later in this report and affords a fairer comparison of electric and chemical propulsion.

The improved performance of the 2- and 3-burn solar-electric propulsion systems comes at the price of increased mission time. Actually this penalty is not overbearing since for this mission the spacecraft can gather important data all along its transfer trajectory. This is illustrated in Fig. 4 where the distance from the Sun's equatorial plane is plotted against time for a 200 kilogram spacecraft launched by Atlas/Centaur. The all-chemical system rises continuously to reach its maximum distance of 0.42 AU in 91 days. The distance from the Sun's equatorial plane would then continue in a sine-wave pattern, reaching 0.42 AU below the equatorial plane six months later and then returning to the maximum latitude point above the plane six months after that.

The electric spacecraft generates a pattern similar to a sine-wave but with increasing amplitude during the propulsive periods. It reaches 0.31 AU at the first antinode in 91 days, 0.46 AU below the equatorial plane six months later, and 0.63 AU above the plane six months after that. It would reach the all-chemical limit of 0.42 AU in 261 days. The total electric propulsion time is 365 days (8800 hours) which is also a typical thruster lifetime estimate for near-term mission applications. It is also important to note that the 25 degree heliographic inclination achieved by the all-chemical spacecraft would be attained by the electric spacecraft if its propulsion system functioned for only 187 days (4500 hours) of operation. Thus, since at the time of this writing the SERT II mission[16] has already demonstrated flight-rated thruster subsystem lifetimes of four months, one can be reasonably certain that even a near term electrically propelled spacecraft would succeed in reaching at least the all-chemical propulsion inclination limit, if not considerably more.

All of the data shown in Figs. 3 and 4 represent systems optimized to deliver maximum payload. The corresponding values of the electric power level and thruster specific impulse are presented next along with the effect of using nonoptimum values. The sensitivity data are given with the underlying idea of fixing the spacecraft design.

Electric Power Requirements

In Fig. 5 the optimum electric power level is plotted for both launch vehicles as a function of final heliographic inclination. The inclination values that correspond to 200 and 400 kilograms of net spacecraft mass (from Fig. 3) are noted on each curve. These net mass values bracket the range of primary interest and together with 2- and 3-burn trajectories lead to optimum power levels that are surprisingly constant. For example, the best power using the Atlas/Centaur varies only between 10 and 11 kilowatts for 200 to 400 kilograms of net spacecraft mass. This occurs between 26 and 32 degrees for 2-burn trajectories and between 31 and 39 degrees for 3-burn trajectories (from Fig. 3). The optimum power using the Titan IIIC varies between 16 and 20 kilowatts for the same spacecraft size range. Thus, the optimum electric power for the Titan IIIC launched spacecraft is 1.7 times the optimum power for the Atlas/Centaur launched spacecraft -- roughly the same ratio as the launch vehicle capabilities near escape speed.

These power levels are optimum in regard to payload capability only. Since solar-electric spacecraft are relatively expensive (e.g., silicon solar cell arrays cost about $300 per watt[17]), the complete system is likely to be more cost effective at reduced power levels if the associated payload penalty is not too large. The lower part of Fig. 6 shows two typical tradeoff curves of net spacecraft mass against installed electric power. It is immediately apparent that the electric spacecraft performance is attractive over a rather broad range of power level. Consider first the Titan IIIC launched spacecraft mission to 40 degrees heliographic inclination with 3-burn class trajectories. The optimum power level of 19 kilowatts yields nearly 300 kilograms. At 15 kilowatts the net spacecraft mass drops 4 percent to 285 kilograms, and at 10 kilowatts it drops 19 percent to 240 kilograms. This figure also shows that the optimum specific impulse decreases from 3400 to 2600 seconds when the power is reduced from 19 to 10 kilowatts. And the optimum initial spacecraft mass decreases 33 percent - from 1660 to 1120 kilograms. The specific impulse variation is not particularly important, but the 45 percent reduction in electric power and the 33 percent reduction in initial spacecraft mass might very well be worth the 19 percent net spacecraft mass penalty.

Similar tradeoff results are obtained for Atlas/Centaur launched spacecraft to 30 degrees using 2-burn trajectories. However, in view of the fact that 10 kilowatts is about optimum for this launch vehicle, as well as being a good compromise choice for the Titan IIIC, it might be wise to consider a standard 10 kilowatt design that would nicely match both boosters. Furthermore, this idea is reinforced by the previous result that the range of optimum power levels is quite broad. This implies that the 10 kilowatt power level is a reasonably good choice over the entire spectrum of interesting missions (defined earlier as those missions that can be accomplished with 200 to 400 kg of net spacecraft mass). More evidence for this conclusion will be given after a good compromise value of the specific impulse is also obtained.

Specific Impulse Requirements

The optimum specific impulse values are given in Fig. 7. The results are very nearly independent of the launch vehicle and are therefore plotted as a single set of curves. The average value in the 30 to 40 degree range is 2900 seconds for 2-burn trajectories and 3650 seconds for 3-burn trajectories. As in the case of electric power, however, net spacecraft mass is not particularly sensitive to specific impulse as is shown in Fig. 8. For the same two example missions discussed previously, specific impulse values between 2500 and 4700 seconds result in no more than a 15 percent penalty in net spacecraft mass. The figure also shows that significantly less electric power is required if the specific impulse is lowered from its optimum value. In order to pick a good compromise value of specific impulse for a fixed spacecraft design, note first that if the 10 kW constraint is imposed the performance of the Titan IIIC case is affected much more than that of the Atlas/Centaur. Therefore, using the optimum specific impulse for the 10 kW Titan IIIC case would result in a good overall compromise provided that the performance of the Atlas/Centaur is not seriously affected. From Fig. 6, this value of specific impulse is 2600 seconds. Figure 8 shows that for this specific impulse the optimum power level for the Atlas/Centaur case is 8.7 kW -- which is close enough to the 10 kW constraint value to suggest that 2600 seconds is indeed a good compromise value for all cases of major interest.

A Fixed Spacecraft Design

The results of the previous two sections suggest that the use of 10 kilowatts of electric power and 2600 seconds specific impulse for a fixed design spacecraft might result in a good overall power and payload tradeoff over the whole mission spectrum of interest. That this is indeed true is shown in Fig. 9 where both the family of optimum designs and the single fixed design are compared. The fixed design performance penalty ranges from negligible to a maximum of 20 percent at 45 degrees using the Titan IIIC.

The performance of this single electric spacecraft design is compared to all-chemical systems in Fig. 10. This figure is the same as Fig. 3 except that it concerns one spacecraft design instead of a family of optimum designs. Comparing these two figures shows that imposing the single design constraint on the electric system does not materially affect the comparison with all-chemical systems. The single design electric propulsion system can deliver far more net spacecraft mass than the uprated Burner II and also extends the maximum inclination limit (for 200 kg of net mass) from 25 to 37 degrees using Atlas/Centaur and 27 to 41 degrees using Titan IIIC.

Whether or not one would actually use a single fixed design electric spacecraft for various missions remains an open question. What is illustrated here is that if such a spacecraft did exist it would be quite versatile indeed. Extending this concept to include completely different missions (e.g., close solar probes) is the next logical step toward the evolution of a multimission solar-electric spacecraft. Some analysis work has already been done in this area. In Ref. 18, for example, a 6.5 kilowatt, 3500 second specific impulse design is analyzed in depth for four different missions. From Ref. 19 a 10 kilowatt, 3250 second specific impulse design appears to be attractive for four missions. It is clear that if a number of different missions are considered the power and specific impulse values found to be attractive here for the extraecliptic mission would probably be replaced with a new set that would be an appropriate compromise for many dissimilar missions. Further work in this area is recommended and should include a wider range of missions and launch vehicles.

State-of-the-Art Effects

Electric propulsion assumptions. - All of the preceding results are for state-of-the-art inputs assumed to be current. The complete propulsion system specific mass is assumed to be 30 kilograms per kilowatt, the tankage fraction is 10 percent, the thruster efficiency curve reflects current designs, and a simple, nonoptimum thrust control is employed. The effect of altering these particular state-of-the-art assumptions is shown in Fig. 11 for a 3-burn Atlas/Centaur launched spacecraft. The solid curve repeats earlier data and reflects the current state-of-the-art assumptions just specified. All other curves are to be compared to it. The dashed curve is for estimated mid-1970's technology thrusters[3] -- about 6 percent more efficient -- and 3 percent tankage. The performance gain is generally rather modest -- allowing an additional 2 degrees of inclination, for example, at the net spacecraft size of 200 kilograms. The two dotted curves show the effect of assuming the propulsion system specific mass to be 25 and 35 kilograms per kilowatt instead of 30 kilograms per kilowatt. Again the performance change is not large -- between one and two degrees of inclination. The predicted mid-1970's technology gain would offset a specific mass increase of around 5 kilograms per kilowatt. On the other hand, gains from both improved thrusters and decreased specific mass are additive.

Optimal thrust control. - Several sample cases of optimal thrust control were generated with the computer code described in Ref. 20. In this case the spacecraft can no longer be constrained to a radius of 1.0 AU because the thrust is not required to be normal to the orbit plane. The payload gains over the fixed thrust program are so negligible (e.g., less than 1 percent) that a comparison curve could not be drawn on Fig. 11 distinct from the reference curve. This is a particularly important result for early application missions since the possible penalties for strictly optimal thrust programming (increased subsystem weight, cost, and less reliability) are avoided. For solar-electric propulsion systems, the optimal trajectories are almost identical with the constrained trajectories - the spacecraft stays at essentially 1.0 AU rather than drifting outward to Mars orbit as in the case of the constant power trajectories reported in Refs. 3 and 4. This difference arises because as the distance from the Sun increases the solar panel output decreases in approximately an inverse square relationship.

Launch vehicle performance. - The Atlas (SLV3C)/Centaur and Titan IIIC boosters are assumed in this study because they already exist and will probably be available in the mid-1970's. At the time of this writing, the only booster larger than these (excluding Saturn V) that seems certain to exist by 1975 is the unbuilt Titan IIID/Centaur. It is the planned launch vehicle for the 1975 Viking mission to Mars and has considerably higher performance[8] than either the Atlas/Centaur or Titan IIIC. The performance growth potential using the Titan IIID/Centaur for the out-of-the-ecliptic mission is illustrated in the table below. The solar-electric data is for the 10 kilowatt, 2600-second specific impulse fixed design spacecraft.

System	Heliographic Inclination for 200 kg of Net Spacecraft Mass, degrees
Atlas/Centaur	19
Atlas/Centaur/ uprated BII	25
Atlas/Centaur/ solar-electric*	37
Titan IIIC	19
Titan IIIC/ uprated BII	27
Titan IIIC/ solar-electric*	41
Titan IIID/Centaur	29
Titan IIID/Centaur/ uprated BII	34
Titan IIID/Centaur/ solar-electric*	51

The performance of all three Titan IIID/Centaur combinations is considerably better than the corresponding Titan IIIC combinations. The net result is a 34 degree limit for the all-chemical system compared to 51 degrees for the solar-electric system, assuming 200 kilograms of net spacecraft mass and 3-burn low-thrust trajectories. It is also significant that the Atlas/Centaur/solar-electric system achieves 3 degrees of inclination more than the Titan IIID/Centaur/uprated Burner II. Thus, a significantly cheaper launch vehicle could be utilized for the electric spacecraft than for the Burner II and still deliver more performance. This is an important factor to account for when comparing differences in upper stage costs. The higher cost of electrically propelled spacecraft compared to Burner II, for example, is offset by: (1) much better performance using the same booster, or (2) reduced launch vehicle costs by using a smaller, cheaper booster.

Concluding Remarks

What is shown in this study is that current electric propulsion technology could produce a spacecraft that yields important performance advantages compared to all-chemical systems. How one weighs the advantages of higher performance and smaller launch vehicle requirements with the disadvantages of higher space vehicle cost and longer flight time is not dealt with here. To avoid costly development of systems that are best suited to isolated cases only, considerations such as these should, in fact, be viewed from an overall space program standpoint rather than for a single mission. It is clear, nonetheless, that a 1 AU out-of-the-ecliptic mission is particularly well-suited to solar-electric propulsion and the relatively simple normal-to-the-orbit thrust control requirement enhances its prospects for early application. Simplified navigation and trajectory requirements are other desirable features of this mission. Considering all of these factors together leads to the suggestion of doing this mission with a first generation solar-electric spacecraft in order to flight-test new hardware as well as collect scientific information.

* Using 3-burn class low-thrust trajectories (average propulsion time is 360 days, average time to attain final inclination is 442 days, and average mission time is 467 days.)

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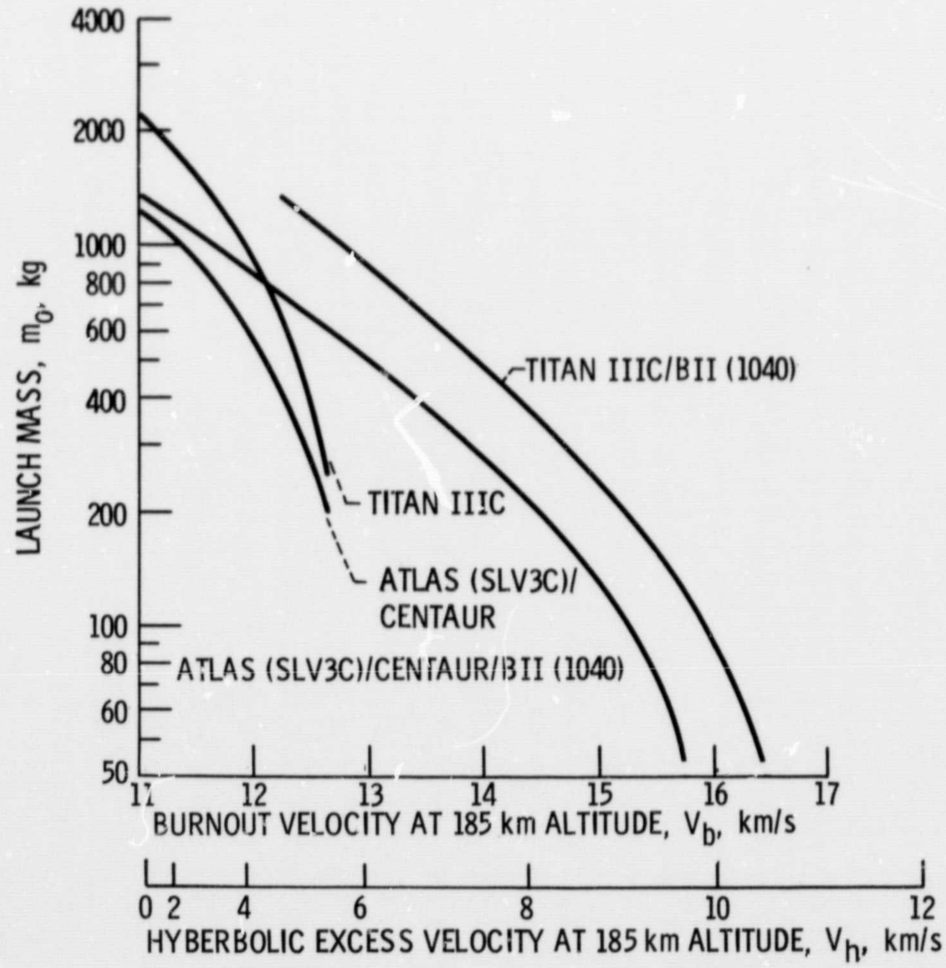

Figure 1. - Launch vehicle performance.

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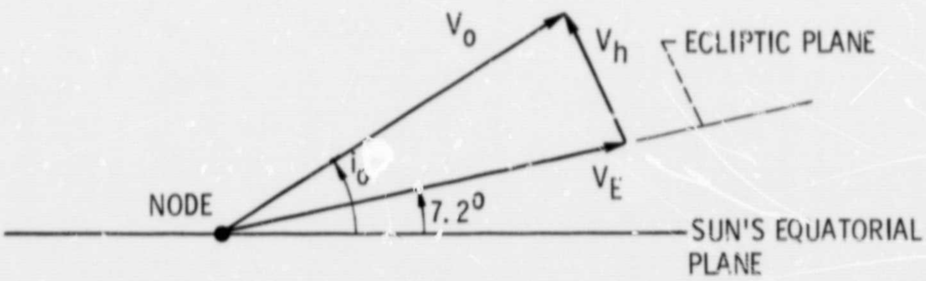
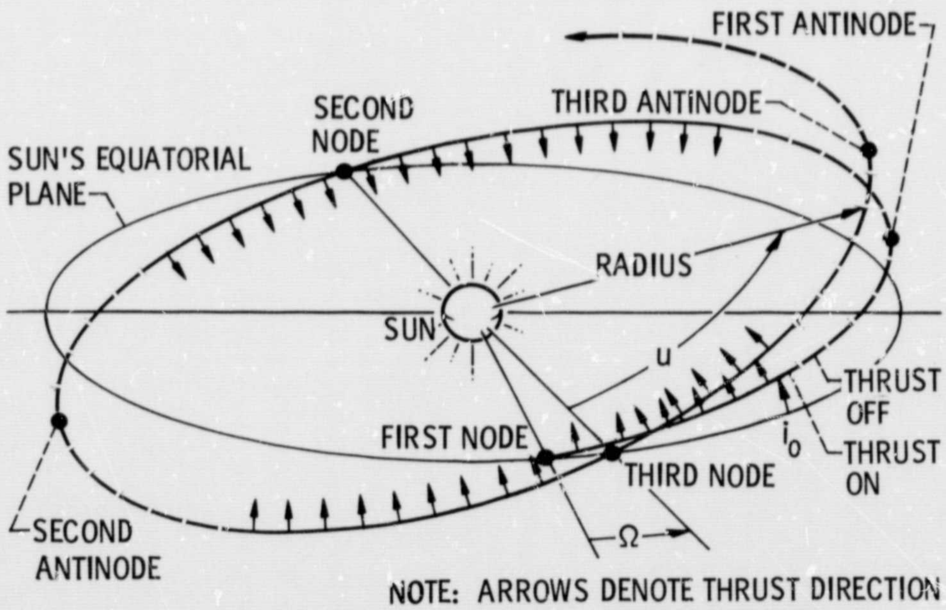


(a)



NOTE: ARROWS DENOTE THRUST DIRECTION (NORMAL TO ORBIT PLANE)

(b)

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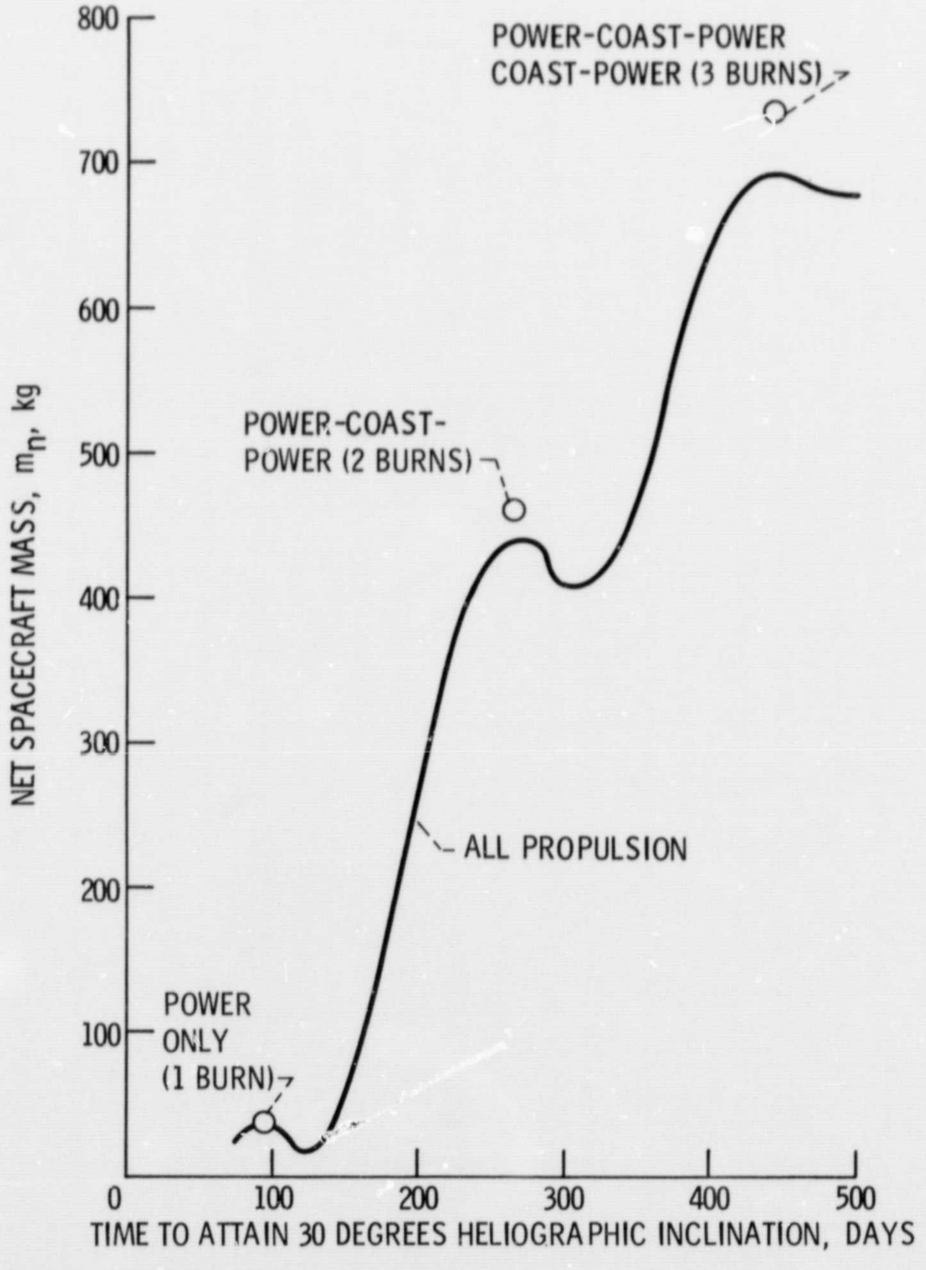

Figure 2. - Effect of flight time on performance of solar-electric spacecraft using Titan III C launch vehicle. Also shows effect of all-propulsion constraint. System variables optimized.

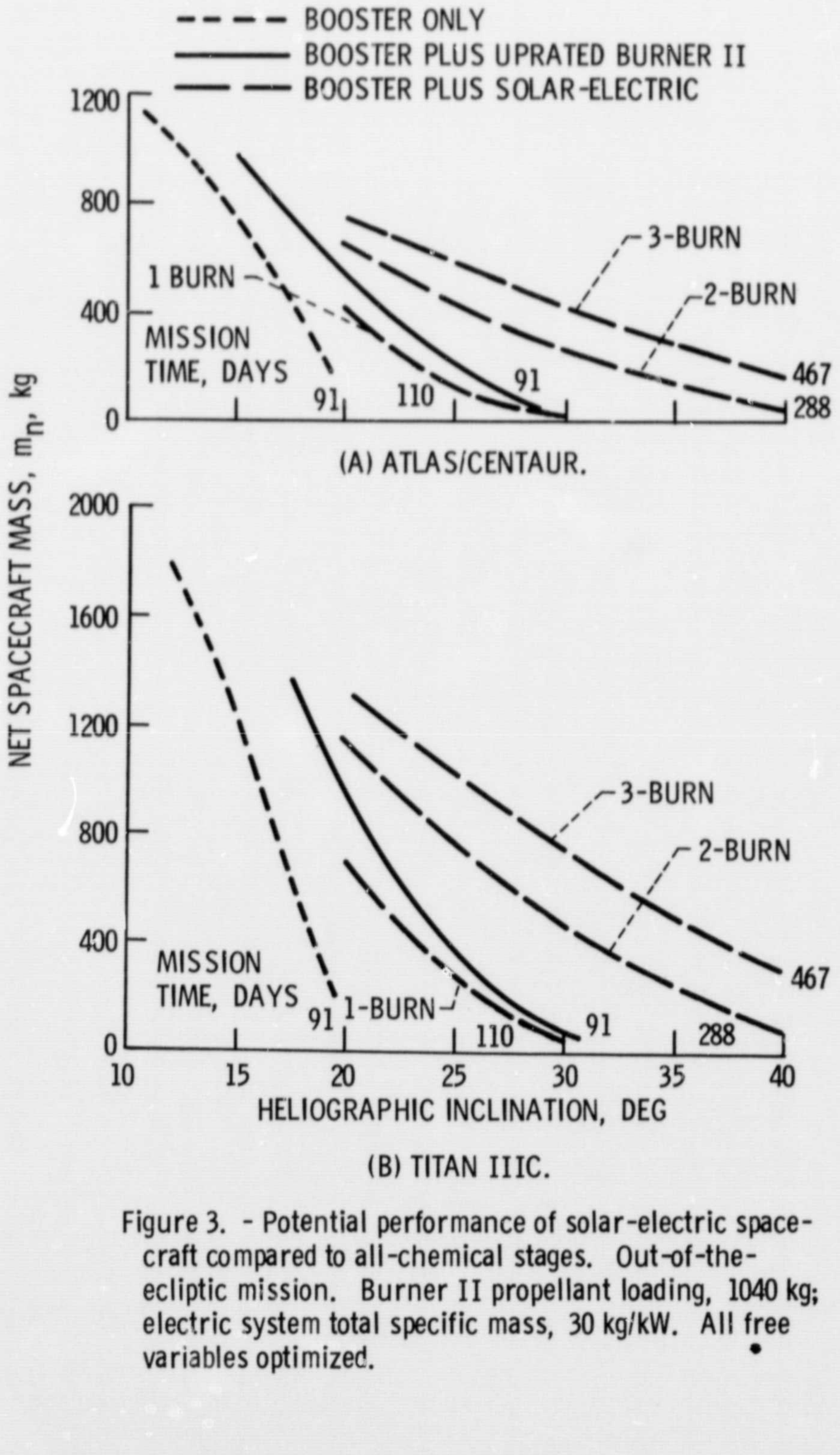

Figure 3. - Potential performance of solar-electric spacecraft compared to all-chemical stages. Out-of-the-ecliptic mission. Burner II propellant loading, 1040 kg; electric system total specific mass, 30 kg/kW. All free variables optimized.

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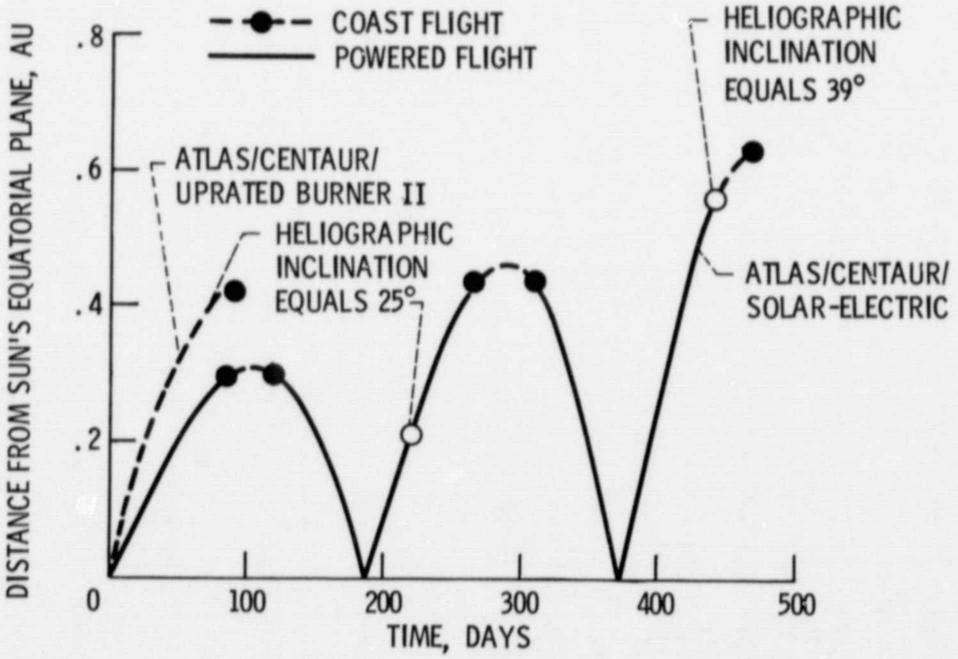

Figure 4. - Time history of distance from Sun's equatorial plane. Net spacecraft mass, 200 kg. All free variables optimized.

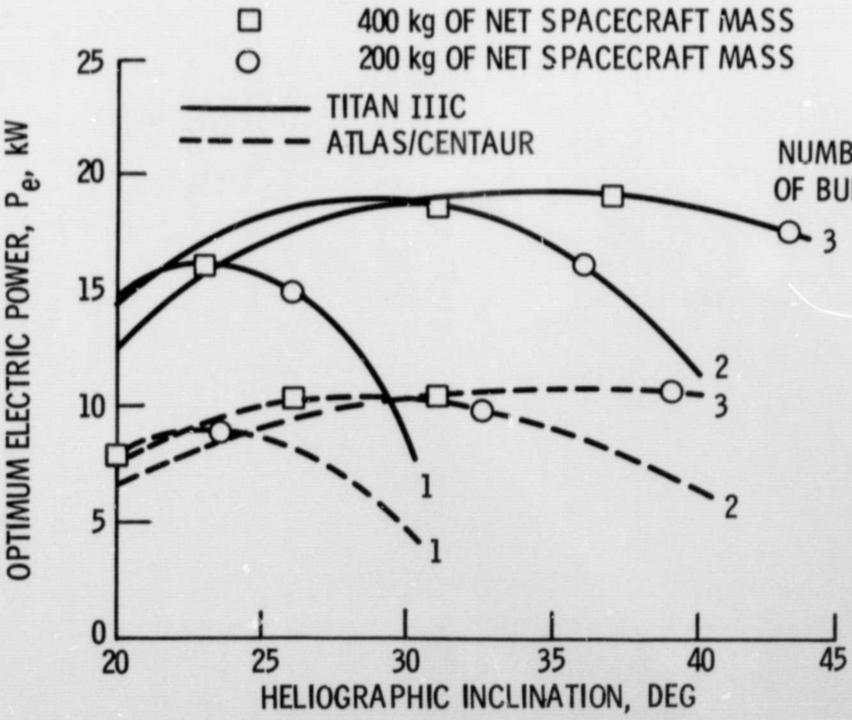

Figure 5. - The optimum electric power requirements for out-of-the-ecliptic missions using electric propulsion. All free variables optimized.

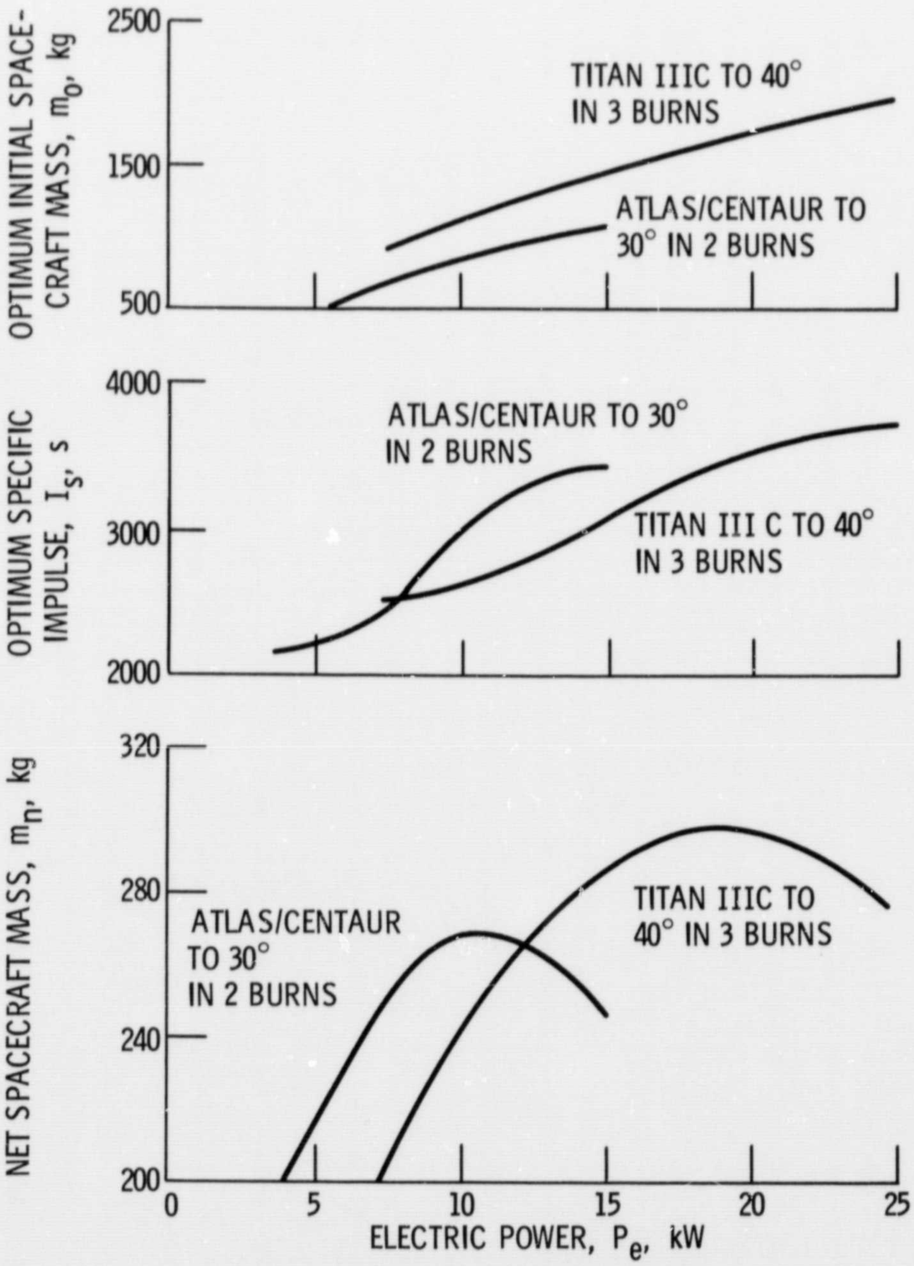

Figure 6. - The effect of using nonoptimum electric power level for out-of-the ecliptic missions.

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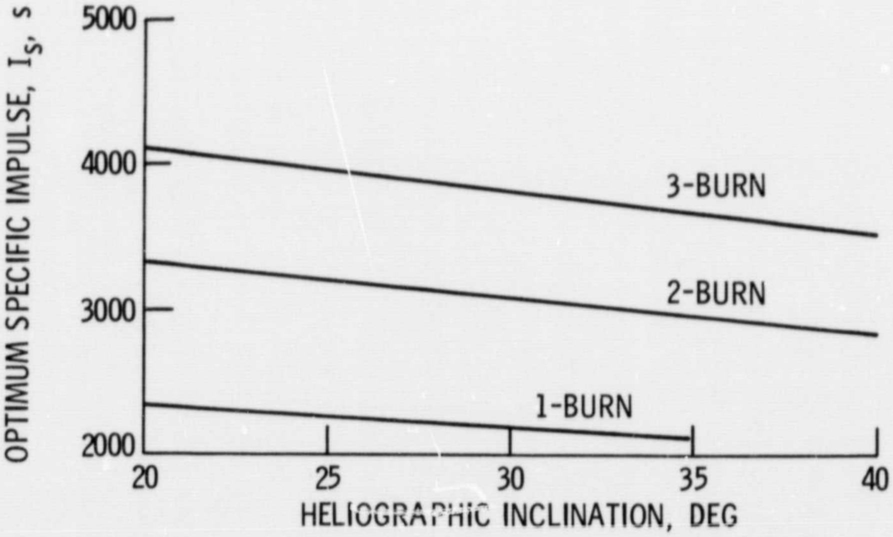


Figure 7. - The optimum specific impulse requirements for out-of-the-ecliptic missions using electric propulsion. Atlas/Centaur and Titan IIIC launch vehicles. All free variables optimized.

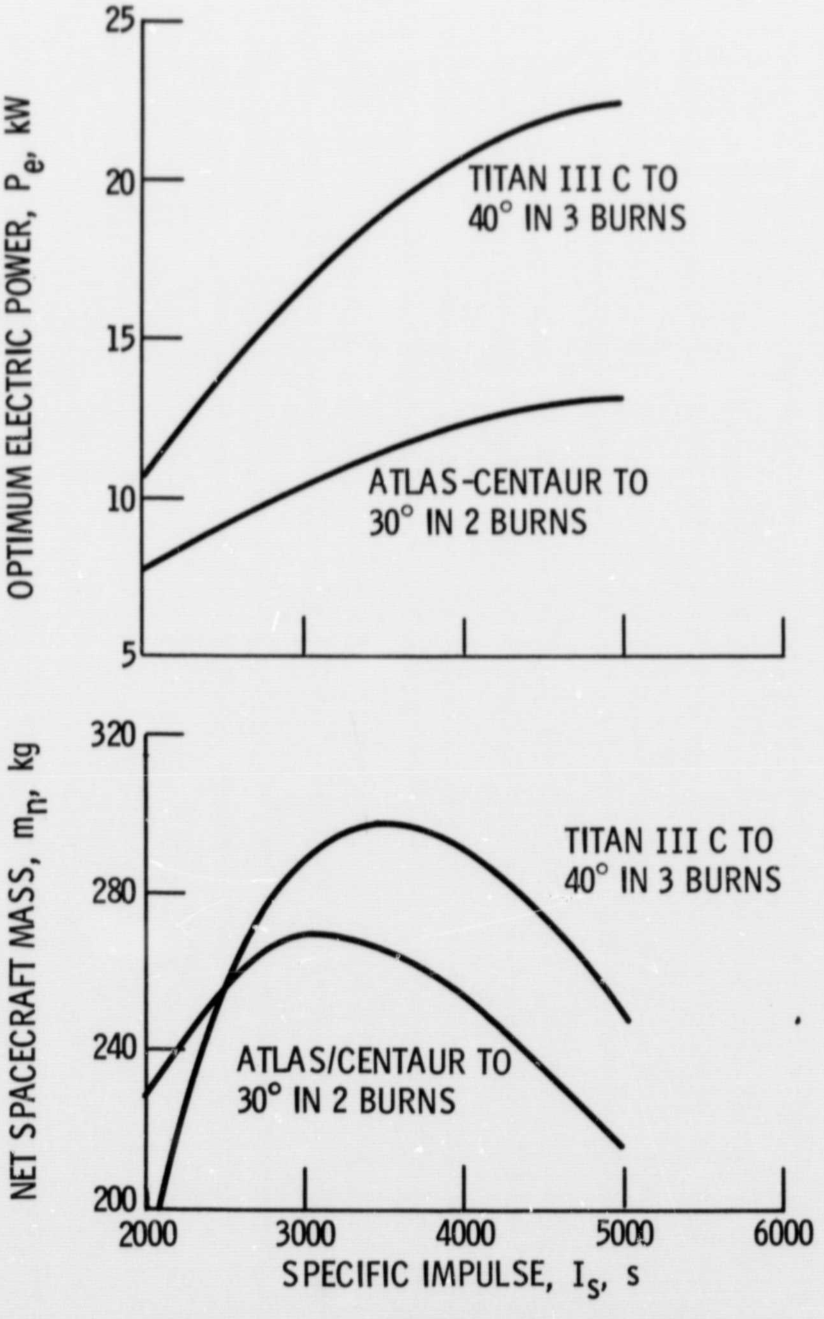


Figure 8. - The effect of using nonoptimum specific impulse for out-of-the-ecliptic missions.

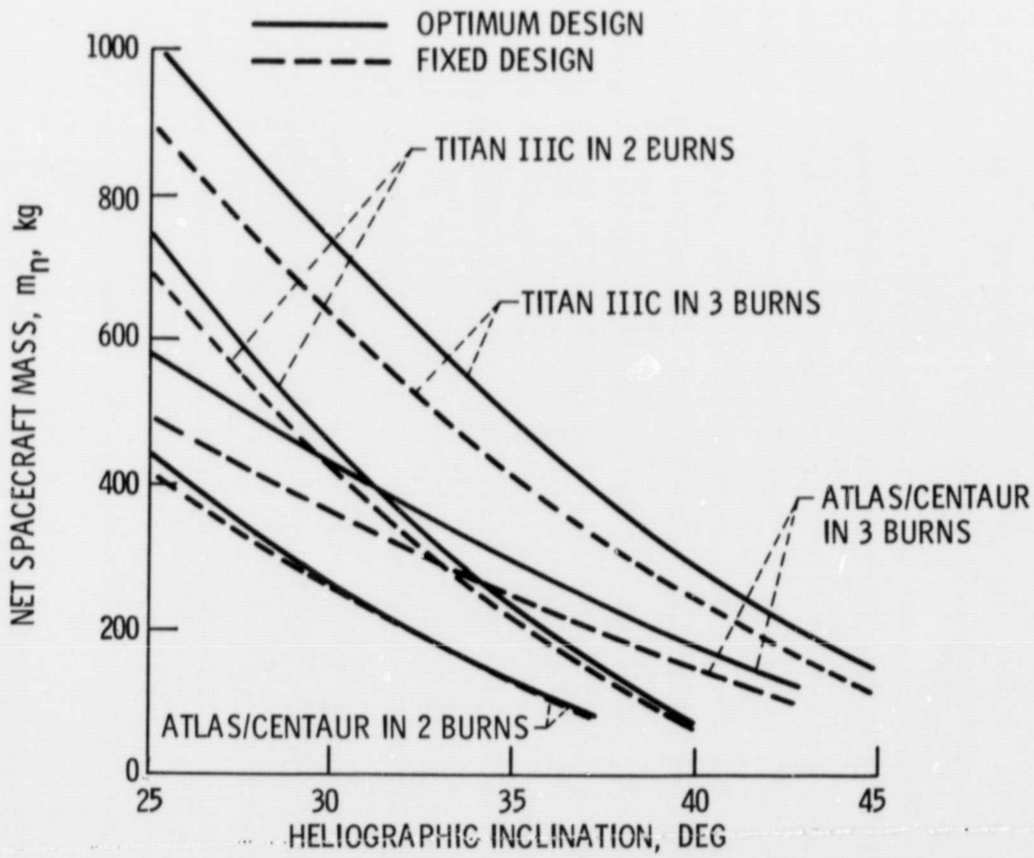
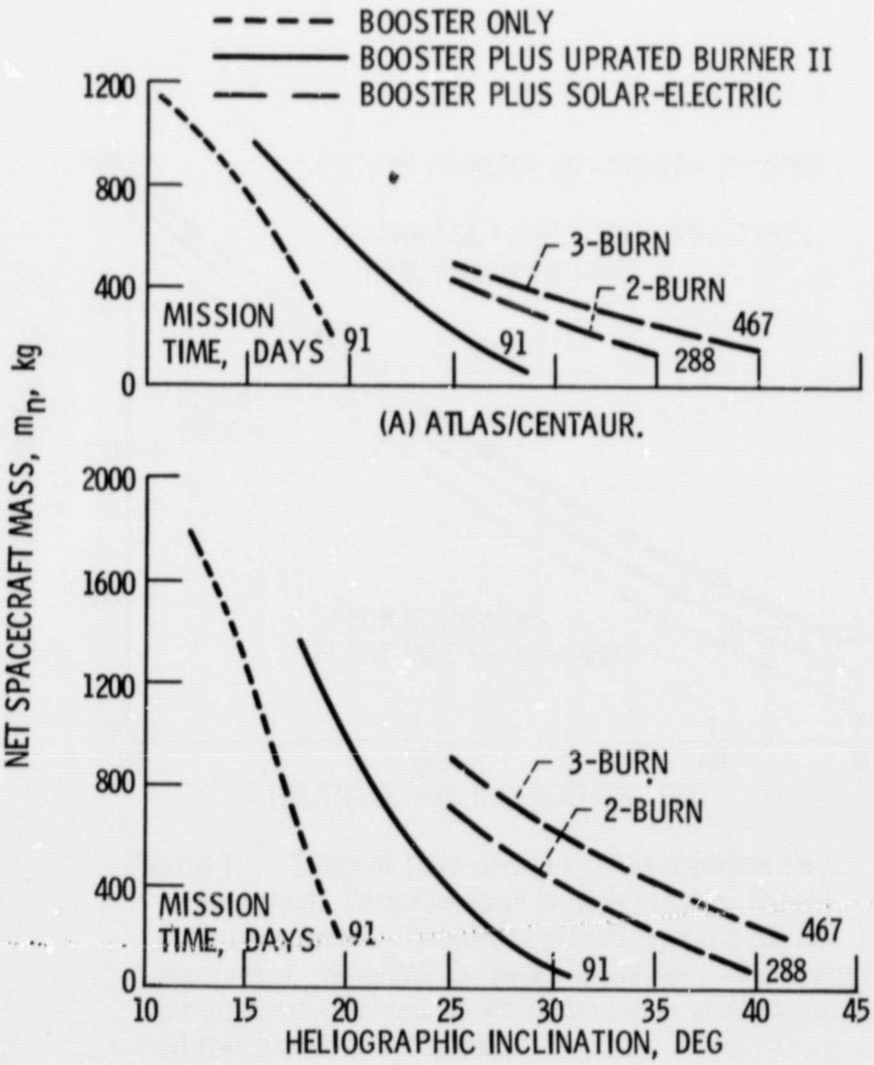


Figure 9. - Comparison of fixed spacecraft design with family of optimum designs for out-of-the-ecliptic missions. Fixed design power level, 10 kW; specific impulse, 2600 seconds.



(B) TITAN III C.

Figure 10. - Performance of single 10 kilowatt, 2600 second specific impulse solar-electric design compared to ALL-Chemical stages. Burner II propellant loading, 1040 kg; electric system total specific mass, 30 kg/kW; launch velocity and coast are timing optimized.

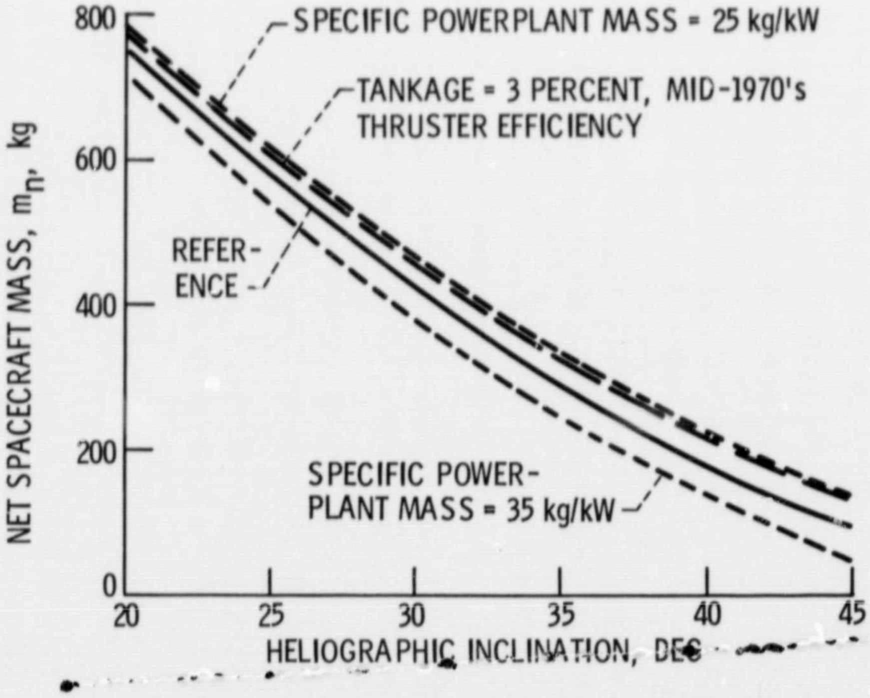


Figure 11. - Effect of state-of-the-art assumptions on performance. Atlas/Centaur launch vehicle, three-burn trajectories. Reference curve: specific powerplant mass, 30 kg/kW; tankage, 10 percent; 1968 thruster efficiency; thrust control, normal to orbit plane; all free variables optimized.