Characteristics, Capabilities, and Costs of Solar Electric Spacecraft for Planetary Missions

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Introduction

SINCE 1965, when the feasibility of solar photovoltaic powered, electrically propelled spacecraft was first suggested for planetary missions,[1] the propulsion system technology has been developed to near readiness,[2–7] while further studies have been made to gain a clear perspective of the mission applicability of such spacecraft. Today there is consideration of which missions to assign to solar electric propulsion (SEP) and when.[8–11] Thus, it is useful to summarize and update the over-all picture of the characteristics, capabilities, and costs of SEP spacecraft for planetary missions in order to suggest 1) those missions for which SEP is best suited and 2) the advantages that can accrue from the multimission use of a given SEP spacecraft design. This is the objective of this survey.

The status of propulsion system technology is represented by the SERT II constant-power propulsion system now in Earth orbit[4,12–15] and by the SEPST ground test of a complete variable-power propulsion system for planetary applications.[5,7,16] Progress in both of these programs is the basis for the narrowing uncertainty in performance, weight, and cost of SEP power plants and for the belief that the propulsion system technology is sufficiently in hand that it can be counted on for missions beginning in the mid-1970s.

The various parametric studies of SEP spacecraft performance reported in the literature[17–36] have been conducted with a variety of programs, a variety of assumptions, and a variety of values for the technology parameters. In this survey, results are presented for nearly all of the missions of interest; the results having been obtained using a uniform set of assumptions and input parameters, and no more than three computer programs, which have been cross checked for consistency.

Although most SEP mission studies to date have been of the parametric performance type, at least four studies have been made of spacecraft configurational and operational feasibility. These studies, including two Mars orbiter missions,[1,37] a Jupiter flyby mission,[38] a Grand Tour mission,[39] and an asteroid fly-through mission[40–43] have all shown SEP spacecraft to be feasible and to offer over-all payload and other operational advantages compared with the conventional ballistic spacecraft. An informative summary of the tradeoffs in the SEP spacecraft design for the Jupiter flyby mission is given in Ref. 44. A summary and comparison of all SEP mission analysis reported in the literature before 1970 is presented in Ref. 45, using scaling laws to adjust for differences in performance and weight assumptions.

Basic Characteristics of a SEP Spacecraft

The application of solar electric propulsion to planetary mission spacecraft can be considered either as an additional booster stage integrated with the spacecraft rather than the launch vehicle, or as a continuous high-velocity-increment spacecraft midcourse propulsion system. The velocity increment is provided at very low acceleration, over a very long time interval, but with a specific impulse an order of magnitude higher than that of a chemical propulsion system. The net result is that the launch vehicle performance capability can be augmented significantly, providing greater payloads in most cases than the alternative of an additional high-velocity chemical stage. While substantial payload advantage is exhibited for flight times between those for zero payload and those corresponding to ballistic maximum payload (minimum energy or Hohmann), SEP payload advantages continue to increase out to flight times up to about twice Hohmann flight times.

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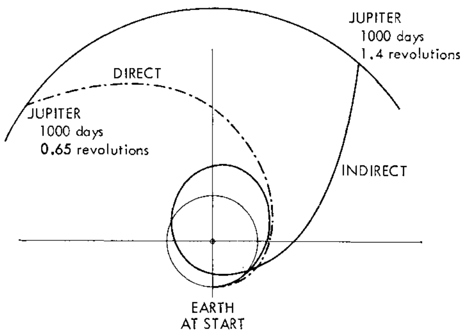

Fig. 1 1000-day solar electric propulsion missions to Jupiter.

A SEP spacecraft derives its energy, of course, from the solar flux, and hence its optimum trajectories are dictated by the variation of available solar power as a function of radial distance from the sun. When the spacecraft is going away from the sun, the available power varies about as the the reciprocal of $R^{1.7}$ rather than R^2, due to falling temperatures and consequently improving solar cell efficiency. In a flight toward the sun, the power available is limited to a factor of about 1.4 times that at 1 a.u. because of temperature limitations of the cells and array substrate. At radial positions closer than about 0.7 a.u., where the power ratio would exceed 1.4, power must be limited by tilting the array with respect to the sun-probe line, or by spacing the cells apart so as to provide greater radiation area on the array, or a combination of both.

Based on expected roll-out solar array specific mass of 15 kg/kw (Ref. 5) and a thrust subsystem specific mass of 10 kg/kw,[5] and allowing for degradation, contingencies, and peak power inside 1 a.u., it is reasonable to assume an average overall SEP power plant specific mass of about 30 kg/kw of power utilized by the power conditioning at 1 a.u. in predicting SEP spacecraft capability. At this specific mass, the optimum specific impulse generally falls between 2500 and 3500 sec, and the optimum acceleration, about $2 \times 10^{-5}g_0$, is such that several hundred days would be required for the spacecraft to spiral itself out of Earth orbit. Consequently, except for very distant missions with very high payload requirements, it is usual to design the mission to require a launch to some velocity in excess of that required for parabolic escape from the Earth. Such SEP spacecraft trajectories generally fall into two categories.

One category, called direct trajectories, involves heliocentric transfer angles typically about one-half of a revolution around the sun and are basically optimal tangent-to-tangent heliocentric transfers between planetary orbits (Fig. 1). Because of the continuous thrusting (generally for half or more of the flight time for the inner planets, less for the outer), the flight path is being continually circularized, and hence the spacecraft generally arrives at the target planet with lower relative approach velocity than does a ballistic spacecraft. For outer planet missions, this type of trajectory, which is favored for flight times less than that of a Hohmann trajectory, usually optimizes at a launch hyperbolic excess energy C_3 of about 20-50 km²/sec².

The other type of SEP spacecraft trajectory, called indirect, utilizes at least one full loop of the sun with the maximum radial position no further out than about 1.5 a.u. and dips into about 0.8 a.u. before heading outward to the target planet (Fig. 1). This type of trajectory provides additional payload at longer flight times and is generally optimized with a small hyperbolic excess energy ($C_3$0 to 10 km²/sec²). Indirect trajectories with more than one loop about the sun are also possible but are of interest only for increasing payload for very long flight times. Inward-bound indirect trajectories are also of interest for missions, such as close-in solar probes and may take several loops around the sun, depending upon the desired perihelion (Fig. 2).

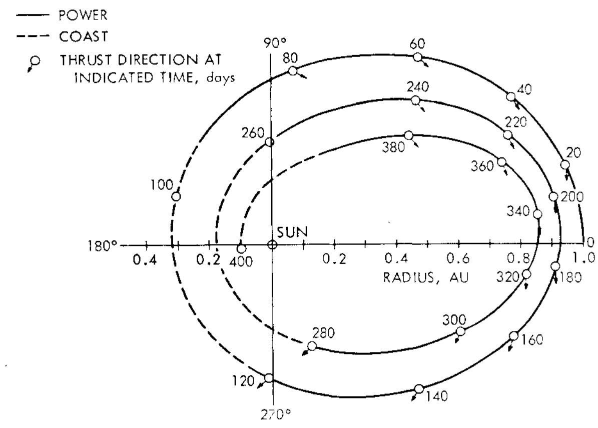

Fig. 2 400-day solar electric mission to 0.1 a.u. (from Ref. 18).

Optimized flight paths with variations permitted in the angle between the thrust vector and the sun-probe line generally require a variation of this angle over about 100° for a direct trajectory, and over about 300° for a one-loop indirect trajectory. The variation of this angle plus the variation of available power are two of the most dominant factors in the spacecraft configuration. The variable thrust angle requires that the thruster system be articulated with respect to the solar array, and that the solar array be such that the ion beam (which may diverge up to 30°) can be rotated through the plane of the array without significant beam impingement. In the interest of spacecraft design simplicity, a fixed thrust angle of about 90° with respect to the sun-probe line can be adopted instead of the optimal thrust angle program, with the penalty being about 25–30% in payload for a mission such as a Jupiter flyby.[38]

Because throttling of any one thruster beyond a range of about two or three to one (at constant specific impulse) leads to undesirable efficiency losses as well as stability problems in the plasma discharge as a result of reduced plasma density, it is clearly desirable to use a multiplicity of thrusters and power conditioners that can be successivley switched off as the available power diminishes. (The alternative of throttling at constant plasma density by varying specific impulse is also limited. Its limit is the range of voltages that the power conditioning can be designed to accommodate with high efficiency.) Through the use of a switching system with failure detection and corrective logic, these switched-off thrusters and power conditioners become available as standbys should any of the operating units fail. The switching of thrusters, however, results in shifts of the center of thrust with respect to the center of mass that are great enough to place excessive demands on the attitude control system. Thus, because of this and the shifts in center of mass resulting from propellant depletion, it becomes desirable to provide thrust vector position adjustment capability by providing for linear motion of the thruster array in two mutually perpendicular directions. By adding thruster gimballing, it is also possible to gimbal pairs of thrusters in such a way as to provide a roll torque, thus making possible complete closed-loop, three-axis stabilization during the thrusting portions of the flight.[5] For coasting portions of the flight, a conventional cold or warm gas system is also provided.

One other factor affecting the spacecraft design is the necessity for the power conditioning to dispose of up to 10% of the available power (efficiency loss) by self-radiation. This requires the use of about $2\frac{1}{2}$ ft² of external spacecraft surface per kw of installed power.

The remaining significant spacecraft configurational problems are centered around the integration of the science experiments and optical sensors with the thrust subsystem. Careful consideration must be given to fields generated by the system and to beam interaction such as light contamination or particle coating of optical sensors. Fortunately, most of these problems are alleviated for planet-centered science experiments because the thrusting will have ended long before planetary encounter. Even for interplanetary science experiments that operate during thrusting, most of these interactions can be solved by judicious application of shielding and positioning, according to current studies of such problems.[46–51] The ability to control the SEP spacecraft potential with respect to the space plasma by adjusting the thruster neutralizer bias has been shown to be particularly beneficial to the measurement of charged particles in interplanetary space.[52,53] Four examples of conceptual spacecraft designs are described below to illustrate special characteristics of SEP spacecraft. Since the spacecraft were designed for a variety of launch vehicles at not necessarily optimized conditions, these capabilities should not be considered as part of the unified capabilities presented in the following section.

An example of a fixed-thrust-angle SEP spacecraft with multiple fold-out panels designed for a Mars orbiter-lander mission is shown in Fig. 3.[53] Note that the fixed thrust angle with respect to the sun-probe line is about 90°, and that the beam is directed toward the empty quadrant between adjacent arms of the solar array. Launched to a C_3 of about 7 km^2/sec^2 by a Titan IIIM launch vehicle, this 9.6-kw SEP spacecraft was designed to place about 1150-kg net spacecraft mass into a 5000-km circular orbit of Mars in 1973 with about 290 days of flight time.

An example of a steerable-thrust-angle SEP spacecraft with multiple fold-out panels designed for a Jupiter flyby mission is shown in Fig. 4.[38] Although designed for a direct trajectory, the spacecraft design in which the whole central cube rotates on an axis between pairs of solar array arms, would also be appropriate for the larger variation of thrust vector angle required of an indirect trajectory. Launched to a C_3 of about 3 km^2/sec^2 by an Atlas/Centaur launch vehicle, this SEP spacecraft with 14 kw (at 1 a.u.) of installed power was designed to deliver about 300 kg of net spacecraft mass to the vicinity of Jupiter in 1975 with about 880 days of flight time.

An example of a fixed-thrust-angle SEP spacecraft with a single fold, fold-out array designed for the so-called Grand Tour Mission of 1977 is shown in Fig. 5.[39] This type of array has the advantage of greater structural rigidity and a higher resonant frequency. The launch vehicle selected, the Titan IIIC/Burner(2336), was found to be adequate for the design's net spacecraft mass of about 580 kg using a 12.8 kw (at 1 a.u.) solar array and a SEP thrust subsystem of 10.3 kw (about half of optimum) with a fixed-thrust angle. Because of tight requirements on the approach corridor at Jupiter for the swingby, very severe restrictions were placed on the optimum SEP trajectory and flight time. With injection at C_3 of 35 km^2/sec^2, the flight time to Jupiter was 725 days, to Saturn 1450 days, to Uranus 3000 days, and to Neptune 4300 days. As for all outer-planet missions, thrusting was stopped before the approach to Jupiter because of the very low level of power available beyond about 4.8 a.u. Had the same launch vehicle been selected for the SEP spacecraft design as for the ballistic spacecraft (*i.e.*, the Titan IIID/Centaur) the net spacecraft could have been doubled or the flight time to Neptune reduced by about 700 days. Nevertheless, the ballistic approach was recommended[39] for this mission because 1) the ballistic spacecraft mass capability was judged to be adequate, 2) there was reluctance to utilize a new technology on such a unique mission unless it was absolutely necessary, and 3) there was a small cost disadvantage when as few as two flights would have to bear the full development costs of the SEP.

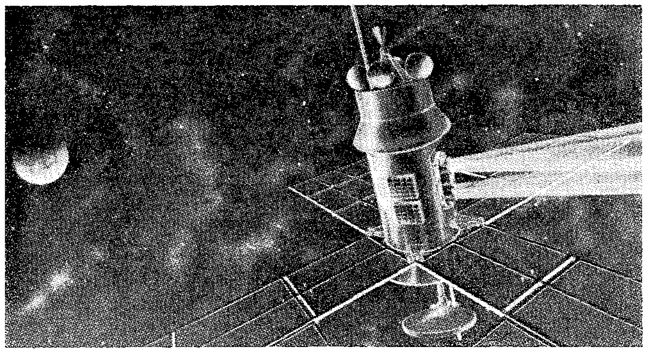

Fig. 3 Titan IIIM-launched 9.6-kw SEP spacecraft in transit to Mars.

Finally, in Fig. 6, an example is shown of a fixed-thrust-angle SEP spacecraft with a roll-out solar array designed for an extended flight through the asteroid belt (out to 3.2 a.u.).[40] It was found that this modest energy mission could be performed adequately with a SEP spacecraft of only 6 kw at 1 a.u. (installed power) using an Atlas SLV3C/Centaur launch vehicle. The net spacecraft mass exclusive of SEP power plant was 440 kg, while the injection energy C_3 was selected at 16 km^2/sec^2. The result of the brief study of this mission was that it might be appropriate for the flight demonstration of a SEP planetary spacecraft and be capable of returning valuable science data on particles and fields and the meteoroid hazard in the asteroid belt.[40] Additional studies of this mission produced other fixed-thrust-angle spacecraft in which full hemispherical clearance of the thrust beam was provided in order to avoid any deposition problems.[42,43]

Capabilities of SEP Spacecraft for Planetary Missions

Additional characteristics of SEP spacecraft are best discussed after presenting a picture of SEP spacecraft mission capabilities. Discussion of mission capabilities is divided into four categories: 1) outer-planet flyby missions; 2) outer-planet orbiter missions; 3) inner-planet orbiter missions; and 4) area missions, which include flights out of the ecliptic and solar and asteroid probes. Flybys of inner planets are not included because SEP offers no enhancement of capability other than to reduce approach velocity, which is usually of little value to a flyby mission.

The trajectory optimization program from which capabilities for all missions in the first three categories, except Mercury orbiters, were derived was that of Analytical Mechanics Associates of Seabrook, Md. This program, developed from

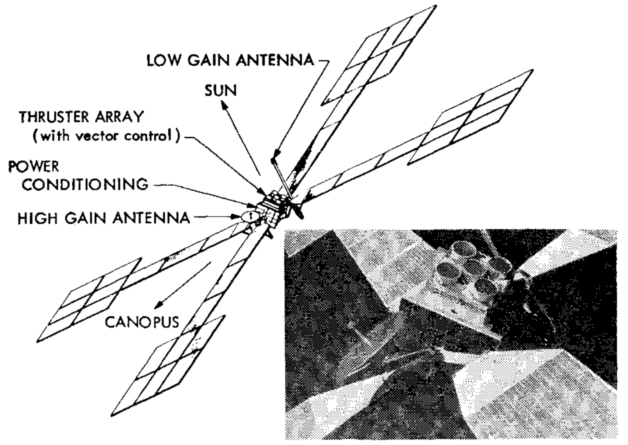

Fig. 4 Atlas/Centaur-launched 14-kw SEP spacecraft Jupiter flyby.

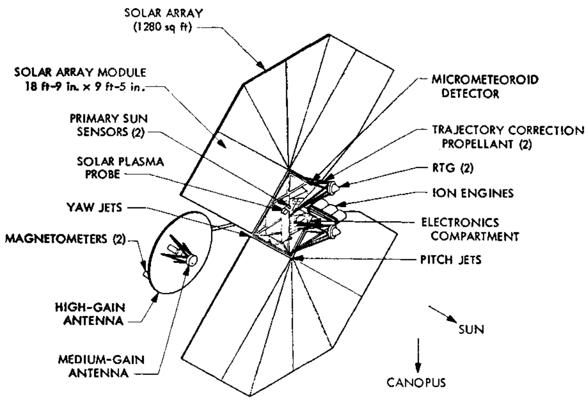

Fig. 5 Titan III-launched 12.8-kw SEP spacecraft multi-planet mission.

the Princeton TOPCAT Program, is described below. These results were also spot-checked by Carl Sauer, of the Jet Propulsion Laboratory, using his integrating, trajectory-optimizing low-thrust program, with agreement found to be within a very few percent in each case checked. These data[29] represent the first available integrated set of results obtained for a wide range of missions and launch vehicles by a single program and successfully cross-checked by another completely independent low-thrust trajectory program.

The Analytical Mechanics Associates' low-thrust trajectory and vehicle optimization program is an integrating program based on the calculus of variations. A two-body heliocentric gravitational force field is assumed throughout the trajectory. The equations of motion and the Euler-LaGrange equations are numerically integrated using a backward difference prediction-correction technique with a Runge-Kutta starter. The data were obtained for open angle transfers between circular coplanar orbits. Although not utilized for the results presented, the program can include both tabular and analytic planetary ephemeris. The initial and final spacecraft positions are taken to be coincident with the launch and target planes, respectively, while initial and final velocities are obtained by adding to the planets' velocities the appropriate hyperbolic excess velocities. This procedure is in effect a zeroth-order asymptotic matching procedure that has been found to yield results within a percent or so of a higher-order asymptotic matching program.

The general mission profile assumed consists of the high-thrust launch maneuver; the low-thrust heliocentric phase; one or more coasting phases; and for the orbiter missions, a high-thrust chemical retro maneuver. Both the launch and retro maneuvers are treated analytically, and the times required to make them neglected. A curve fit of the launch

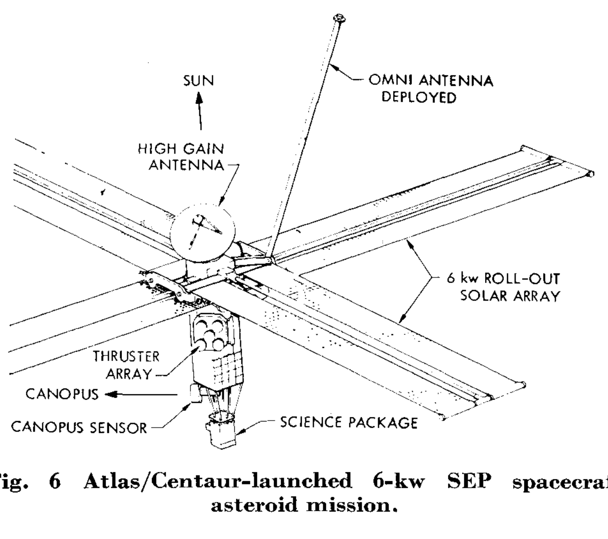

Fig. 6 Atlas/Centaur-launched 6-kw SEP spacecraft asteroid mission.

vehicle capability as a function of launch energy is incorporated into the program for a number of launch vehicles, the source being the *1969 NASA-OSSA Launch Vehicle Estimating Factors Handbook*.[54] For the purpose of computing the retro velocity required, it is assumed that the spacecraft is inserted into the final orbit at its pericenter point.

The power of the solar-electric system is assumed to vary with distance from the sun according to a 5th-order polynomial curve fitted to tabular solar array performance provided as input. The exhaust velocity of the thrust subsystem is optimized and held constant for each flight path, consequently the thrust varies with distance proportionately with the power. The solar arrays are always assumed to be oriented normal to the sun-spacecraft line, whereas the thrust vector angle is permitted to optimally vary in the plane of the ecliptic. The power level at 1 a.u., the launch and arrival excess velocities, and the duration of thrusting for a given total flight time are optimized to yield maximum net spacecraft mass.

Outer-Planet Flyby Missions

Presented in Fig. 7 is mission capability of the Titan IIIX(1205)/Centaur with solar electric propulsion spacecraft for performing flyby missions of the outer planets Jupiter, Saturn, Uranus, and Neptune. Pluto is not included because capability is extremely launch-year dependent. The SEP capability is compared with an all-chemical launch vehicle-ballistic spacecraft capability, where the launch vehicle considered is the Titan IIIX(1205)/Centaur/Burner-(2336). The capabilities presented are based upon calculations which assume the planets to be in circular orbits at their average distance from the sun, the planetary orbits to be coplanar, and the launch window to be of zero length. The consequences of each of these assumptions to the relative capabilities of SEP and ballistic spacecraft are discussed below. The measure of comparative capability in each case is the net spacecraft mass, which includes structure and other necessary engineering subsystems, such as experiment and housekeeping power, guidance and control, communications, temperature control, data handling, etc., in addition to the science instrument payload. It does not include any mass of power for electric propulsion, mass of electric thrusters, propellant, and tankage. For Jupiter, where some 5% of the solar power upon leaving 1 a.u. is still available for experiment and housekeeping functions, this basis of comparison is unfair to SEP to the extent that such available power could actually be put to use, thereby alleviating the allocation of mass for that purpose out of the net spacecraft mass. For planets beyond Jupiter, the solar power available is so low that this is no longer a factor.

In the SEP curve for Jupiter, a sharp change of slope is noted at a flight time of about 1000 days. At flight times less than 1000 days, direct trajectories (about $\frac{1}{2}$ solar revolution) provide the greater net spacecraft masses. Beyond 1000 days, the net spacecraft mass capability from such trajectories begins to level off, and it becomes more favorable to go to indirect trajectories (about $1\frac{1}{2}$ revolutions). Payload capabilities from such trajectories continue to increase out to about twice Hohmann ballistic flight time (900 days). Still greater net spacecraft mass capability could be provided by going to trajectories which take two loops around the sun. Note that the net spacecraft mass advantage of SEP over ballistic indicated is about 50% at 900 days. Beyond that flight time, no further improvement in the ballistic capability is possible, of course, whereas the SEP capability continues to increase to 130% above the ballistic maximum, as the flight time increases out to about 2000 days.

A similar pattern is exhibited for Saturn, Uranus, and Neptune. At Saturn the SEP net spacecraft mass is 170% above the ballistic direct flight capability at Hohmann flight time. Alternatively, the SEP capability could be used to reduce flight time to only 40% of Hohmann flight time at spacecraft mass equal to that of a Hohmann ballistic spacecraft. It should be noted that the ballistic net spacecraft capability can be enhanced by 80% by use of a Jupiter swingby if one is willing to accept the severe restrictions on launch dates. The position of Jupiter during the 1980's is such as to preclude such swingbys. For SEP spacecraft, swingbys of Jupiter can also enhance net mass. Typical increases in SEP net spacecraft mass due to 1977-launched Jupiter swingbys are 20% at Saturn for 1500 days, 40% at Uranus for 2500 days, and 40% at Neptune for 3500 days.[23,26] In addition, Grand Tour missions of the four planets can also be flown with SEP with the net spacecraft mass advantage over the ballistic about 100%.

If one were to select 500 kg of net spacecraft mass as all that is required, say for a Mariner class flyby, then compared with ballistic spacecraft, Fig. 7 indicates that SEP spacecraft could reduce the flight time by about 50 days at Jupiter, about 250 days at Saturn, about 900 days at Uranus, and over 2500 days at Neptune for the launch vehicles assumed.

As indicated in Fig. 7, the launch vehicle assumed for the SEP spacecraft was the Titan IIIX(1205)/Centaur. Detailed optimizations were also performed at Analytical Mechanics Associates for five other launch vehicles. The results of Figs. 7, 10, 12, and 13 can be scaled to these other launch vehicles by multiplying net spacecraft mass and power by the factors indicated in Table 1. The factors, derived from launch vehicle capability at Earth escape velocity, nearly exactly scale the results for indirect trajectories, which nearly always optimize at a launch velocity very close to escape. For direct trajectories (shorter flight times) the optimizations occur at velocities modestly above escape velocity, with the result that the scaling factor is only approximate but generally within 10% (Table 1).

As mentioned previously, the SEP results presented pertain to a circular orbit, coplanar solar system, without provision for launch windows. The consequence of reducing the launch window to zero is to make the comparison between SEP and ballistic capability relatively better for the ballistic. The penalty for a 30-day launch window for a ballistic flight is a requirement for a higher launch velocity by about 60 to 200 m/sec for the inner planets, and 600 to 900 m/sec for the outer planets. Approach velocities are increased by the 30-day launch window by 300–1200 m/sec for the inner planets, and by 0–900 m/sec for the outer planets. The consequences of these added velocity requirements are typically penalties of about 20–25% for a Mars orbiter[25] or a Jupiter flyby, and 30–35% for an outer-planet orbiter.[54] The net spacecraft mass degradation of a SEP spacecraft as a result of using a fixed set of spacecraft parameters over a 30-day launch period is generally much less. In Ref. 25 it was shown that the penalty is only 8% for a Mars orbiter, and in a study of the Grand Tour the penalty was shown to be only 10%. The reason for the lower penalty for launch window for SEP spacecraft is the combination of 1) the capability of SEP trajectories to be modified, and 2) the higher specific impulse. On the other hand, the need for lengthy launch windows is diminishing as a consequence of recent experience in operations of launch vehicles and spacecraft.

The analysis of SEP spacecraft and ballistic spacecraft capability, both on a circular orbit coplanar solar system basis, probably also favors the ballistic spacecraft in a relative

Table 1 Launch vehicle scaling factors for SEP spacecraft

Launch vehicle	Spacecraft mass scaling factor
SIC/SIVB/Centaur	4.16
Titan IIIX(1207)/Centaur	1.41
Titan IIIC	0.41
Atlas SLV3X/Centaur	0.39
Titan IIIX/Centaur	0.28

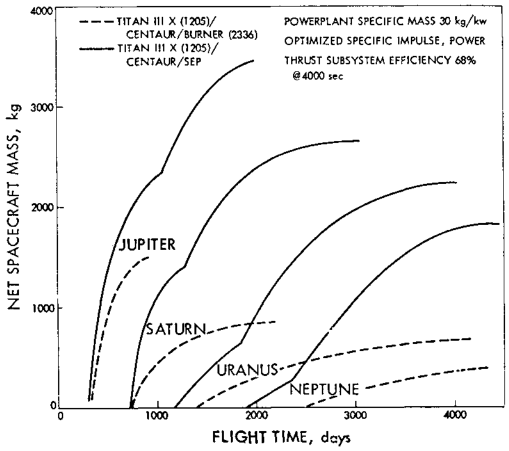

Fig. 7 Outer-planet flyby missions.

sense unless one is willing to restrict ballistic spacecraft launches to the most opportune year An example of the year-to-year variations in SEP and ballistic spacecraft capability for performing Mars orbiter missions is given in Fig. 8. The SEP net spacecraft mass capability, derived from data Ref. 53, is based on a constant fixed (and thus nonoptimum) powerplant of 9.6 kw and specific impulse of 4000 sec. Even less year-to-year variation is to be expected for a SEP spacecraft optimized for each launch opportunity. The ballistic spacecraft capability (with zero launch window) was obtained from Ref. 54.

Finally, in discussing the SEP net spacecraft mass capability for the outer-planet flyby missions, it is instructive to look at the distribution of mass into propellant, powerplant, and net spacecraft as a function of flight time for both direct and indirect trajectories, as shown in Fig. 9 for a Saturn flyby. Particularly worthy of note is that for direct trajectories, as flight times are reduced, there is less dependence on the SEP for velocity and more on the launch vehicle. SEP powerplants tend toward a zero power level and the launch vehicle launch energy C_3 tends toward that required of a ballistic spacecraft. At longer flight times both the net spacecraft mass and powerplant mass reach a maximum after which they diminish. Near that maximum, about 30% of the initial mass is devoted to powerplant for an optimal design. In contrast, indirect trajectories nearly always optimize at a launch energy C_3 quite near zero (*i.e.*, bare escape), while both powerplant and propellant masses decrease for longer flight times, reaching an asymptote where about 25% of initial mass is devoted to each. The cross-over point, where net spacecraft mass for an indirect trajectory equalled that for direct trajectory, was about 1300 days for the Saturn flyby example.

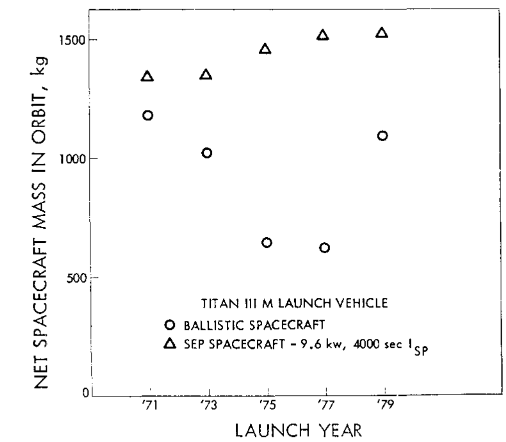

Fig. 8 Mars orbiter missions, 2.5*R* circular orbit, variation with launch year.

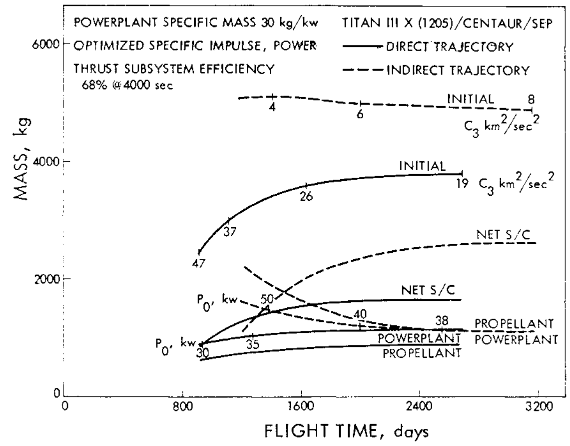

Fig. 9 Saturn flyby missions mass distribution.

Outer-Planet Orbiter Missions

Presented in Fig. 10 is mission capability of the Titan IIIX(1205)/Centaur for performing orbiter missions of the outer planets, both with solar electric propulsion and ballistically with the Burner II. The planetary orbit selected for the comparison is that of 2-planetary-radii periapsis and 38-planetary-radii apoapsis, an eccentricity of 0.9. This is a compromise between a near-circular mapping orbit and a bare capture. Actually, the periapsis at Jupiter may have to be about 4 to 6 radii to avoid the severe trapped radiation hazard, and greater than about 3 radii at Saturn to avoid the planet's rings. The capability for performing these missions would be a bit lower than that shown if the apoapsis remained fixed and the periapsis were raised.

The pattern of capability of SEP spacecraft as compared with the ballistic is seen to be similar to that for the flybys previously discussed. Again, the discontinuities in the slopes of the SEP curves result from conversion from direct trajectories (at lower flight times) to indirect trajectories for longer times. Peaks in the capability of one-loop indirect trajectories are noted for Jupiter and Saturn both at about the same net spacecraft mass, about 1800 kg.

At Jupiter, with this circular coplanar no-launch-window basis of comparison, the SEP provides about 50% more net spacecraft mass for Hohmann flight time, but up to 120% more mass at times up to twice ballistic Hohmann flight time.

If 750 kg were selected as the minimal orbiter of interest, only at Jupiter would the ballistic approach be adequate without going to the SIC/SIVB/Centaur launch vehicle or adding a high-energy chemical kick stage to the Titan/Centaur at substantially increased launch vehicle costs.

Again, it should be noted that the capability data are presented for flights that do not require the severe launch date restrictions of a Jupiter swingby. Jupiter swingbys are of doubtful value for orbiter missions because of the substantial increase in target planet approach velocity.

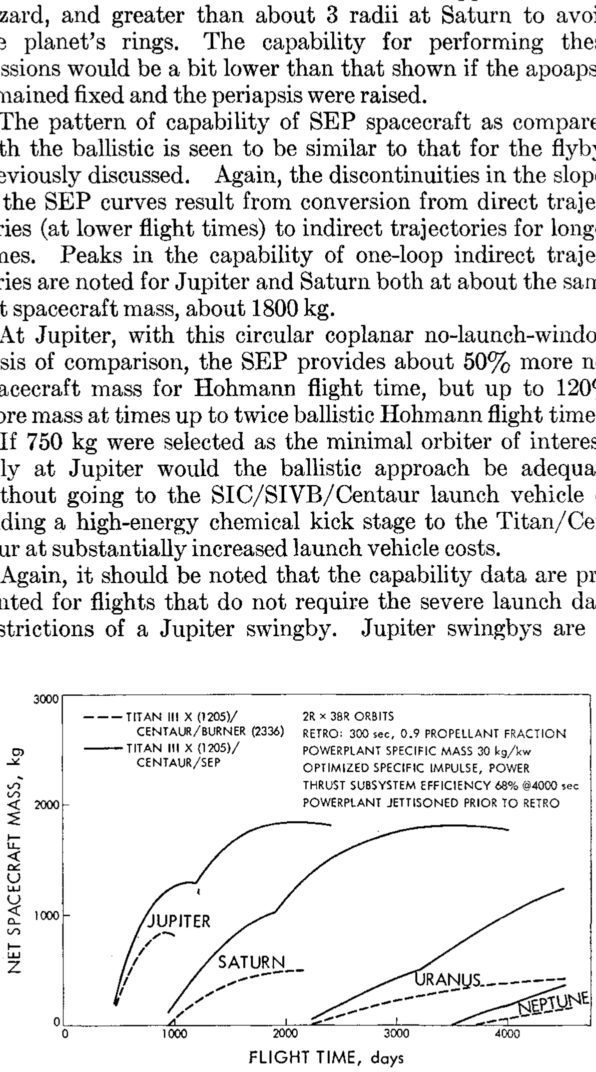

Fig. 10 Outer-planet orbiter missions.

The ability of SEP optimum orbiter trajectories to reduce target planet approach velocities is illustrated in Fig. 11. At shorter flight times the optimum approach velocity of a SEP spacecraft is nearly the same as that of a ballistic spacecraft, but as the Hohmann flight time is approached, significantly lower approach velocities are optimal for SEP spacecraft flying direct trajectories. Indirect trajectories achieve even lower approach velocity at longer flight times. This has a strong influence on the net spacecraft mass capability of SEP spacecraft at such flight times, as can be seen.

Inner-Planet Orbiters

The capability of SEP spacecraft to perform orbiter missions of Mars and Venus is compared with that of ballistic spacecraft in Fig. 12. Note that the Burner is not used for these missions because at the low launch energy required, it offers no payload enhancement. The trajectories computed at Analytical Mechanics Associates were optimized for $2R \times 38R$ orbits of Mars and Venus. By using the approach velocities obtained for these missions, capability for tighter, more circular orbits was derived without reoptimization. It is believed that the penalty for this nonoptimal condition is quite small. In the computation of the net spacecraft mass for these missions, the mass of the solar array was included because of its utility for performing the mission of the spacecraft such as visual or radar mapping.

It is most significant to note that the direct trajectory missions, i.e., to the left of the discontinuity of slope in each of the Mars and Venus curves, tend toward lower and lower power levels at shorter flight times. At approximately 150 days for Venus, and 240 days for Mars, the optimum power level goes to zero and hence the spacecraft becomes ballistic. These curves meet like this because both solar and SEP trajectories were computed by the same program using the same solar system model, i.e., circular orbit, coplanar, and zero launch window. The benefit of SEP for Venus gained by flying longer flight times is almost negligible, whereas for Mars, increases of 15–35% are gained (depending upon orbit desired) by increasing flight time by about 200 days. This marginal advantage appears to disagree with other comparisons available in the literature.[25,37] In Ref. 23, the net spacecraft capability gained by flying a SEP spacecraft of optimum power on a direct trajectory of about 216 days to Mars, launched in May 1971, was about 20% greater than a ballistic spacecraft without a launch window and about 65% greater with a 32-day launch window. It is evident that the difference between these results and those of Fig. 12 is a direct consequence of significant launch window effects, as well as year-to-year variations in the comparative capability of SEP spacecraft. In Ref. 25, the surprisingly large advantage shown for SEP over ballistic flight is due in part to assuming a very conservative retro design, in part to requiring a 30-day launch window of the ballistic spacecraft, in part to making the comparison for a particularly bad ballistic launch year, and in part to crediting the whole SEP powerplant (thrusters and all) as part of the net spacecraft mass because of its use in lowering the orbit. For those missions where substantial solar power is desired in orbit for mapping missions, and where the thrust subsystem is utilized for orbit adjustments, the added cost and complexity of a SEP spacecraft over a ballistic spacecraft may prove to be worthwhile even if the payload advantage is otherwise moderate.

The inner-planet orbiter mission, where a vey large payload advantage of SEP can be shown, even on the basis for comparison used in this survey, is the Mercury orbiter mission. The SEP capability shown for both loose and tight capture orbits is substantial for flight times over about 400 days.[27] No ballistic capability is indicated for comparison because the assumed ballistic launch vehicle, the Titan IIIX(1205)/

Centaur/Burner can deliver negligible payload for this mission. The Saturn V/Centaur can be shown capable of only about 100 kg at the ballistic Hohmann flight time of about 100 days without the addition of a two-stage retro. The difficulty with a ballistic flight is the very high out-of-ecliptic-plane component and high approach velocities, and hence correspondingly high retro velocities, required for orbiter missions. SEP flights to Mercury, on the other hand, arrive in plane with very low approach velocity and hence require little retro mass.

Area Missions

In the category herein called area missions are flights out of the ecliptic, in toward the sun, and to the asteroids.

Capability for SEP flights up to 700 days to 45°, 60°, and 75° out of the plane of the ecliptic at 1 a.u. derived from data of Ref. 28, is presented in Fig. 13. There is reason to believe that useful payloads out to 90° are also possible at longer flight times. By comparison, single points are shown for ballistic flights to 45°, 60°, and 75° out of the ecliptic at 1 a.u., requiring about 800 days longer than the SEP spacecraft as a consequence of a required swingby of Jupiter to get the launch energy down low enough to get a useful payload. Perhaps even more significant to the comparative capabilities is the long observational time of the SEP spacecraft at the desired celestial latitude, and the observational time at lower angles while the spacecraft is rising in celestial latitude angle. A further dividend is the substantial power available yet not counted here as part of the net spacecraft mass. More recent data for a smaller launch vehicle (Titan IIIC/Burner), for lower payloads, and lower angles of inclination are presented in Ref. 33.

A similar comparative situation occurs for solar probes, of which the 0.1 a.u. probe shown in Fig. 13 is representative. For this mission, about 500 days longer is required for a ballistic spacecraft because of the necessity to go out to Jupiter in order to swing by and pick up energy to reduce required launch energy. The SEP net spacecraft mass data presented were derived from Ref. 18, in which a flat solar power profile was assumed in to 0.1 a.u. Although the solar array temperature control problem will probably prevent maintaining full power that close to the sun, the furthest in toward the sun that the 400 day 0.1 a.u. trajectory requires powered flight is about 0.3 a.u. Again, the observational characteristics of the SEP trajectory are far superior to those of the ballistic swingby trajectory. The SEP trajectory provides 20-day observation periods near 0.3 a.u. after 100 days, and near 0.2 a.u. after 260 days, on the way to 0.1 a.u. at 400 days. Observation periods of about 10 days between 0.2 and 0.1 a.u. are repetitively available thereafter about every 130 days. In contrast, a ballistic Jupiter swingby spacecraft has but a very few observational days in the vicinity of 0.1 a.u. about every $2\frac{1}{2}$ years or so.

Finally, in the category of area missions, are flights to the asteroids. For asteroid belt flythroughs out to a 3.5-a.u. apohelion, Fig. 13 shows only a slight capability advantage of SEP spacecraft at the expense of about 150 extra days of flight time. Since such a mission requires comparatively little energy, such a marginal capability advantage is to be expected. Probably only for a mission whose purpose is to sample the asteroid belt over an extended period of time does the use of SEP offer much attraction.[40–43] For such a mission it is evident that the Atlas/Centaur with SEP would offer sufficient net spacecraft mass. An actual circular orbit rendezvous at 2.77-a.u. (about the middle of the asteroid belt, and the mean distance of Ceres from the sun) presents quite a different comparative picture. Here the retro mass required by the ballistic trajectory is quite substantial because of the high approach velocity to the 2.77-a.u. orbital position, and the absence of a planetary mass to aid in establishing the orbit. By comparison, the SEP spacecraft is able to establish its circular orbit without a chemical retro at all, although

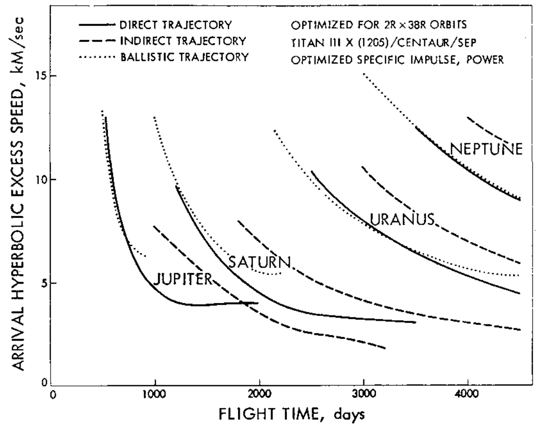

Fig. 11 Outer-planet orbiter missions, arrival hyperbolic excess speed.

use of a small chemical retro does result in a few percent higher net spacecraft mass. The ballistic spacecraft trajectory reaches its Hohmann flight time at only 470 days with a maximum capability about 300 kg, whereas the SEP spacecraft will provide several times that net mass at somewhat longer flight times.

Additional Characteristics of SEP Spacecraft

Trajectory Characteristics

The factor that has made the calculation of optimized SEP trajectories a difficult task, their almost unlimited variability, is actually one of the favorable characteristics of SEP spacecraft. This is manifest in being able to readily trade spacecraft mass with flight time almost up to launch by adjusting the trajectory parameters and minor adjustment of the propellant load. In fact, the flight path can be changed while in flight to adjust to unforeseen occurrences such as imperfect prediction of powerplant performance, or a degrading solar flare, or to uncertain target position such as that of a comet. The full possibilities of exploiting the additional degrees of freedom of SEP trajectories have not yet been explored.

It is evident from Figs. 7, 10, 12, and 13 that families of SEP spacecraft trajectories exhibit what is roughly equivalent to a Hohmann minimum energy flight time for a given trajectory mode. Beyond that time it is no longer possible to improve SEP net spacecraft mass capability except by modifying the trajectory mode to include additional loops of the sun. At Jupiter that flight time, for one loop indirect trajectories, was found to be about 2000 days for both flybys and orbiters. At Saturn it was 3000 days for flybys and 3500

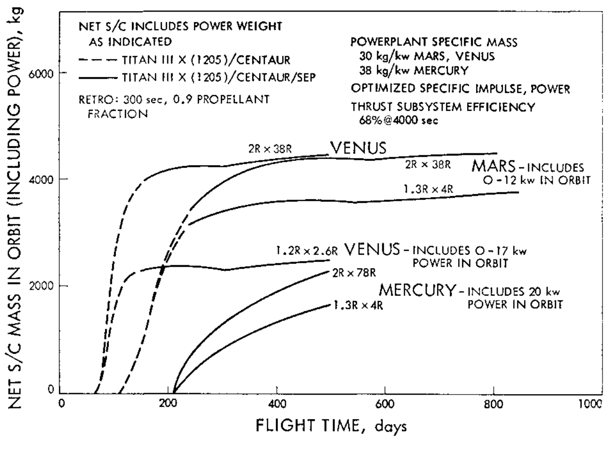

Fig. 12 Inner-planet orbiter missions.

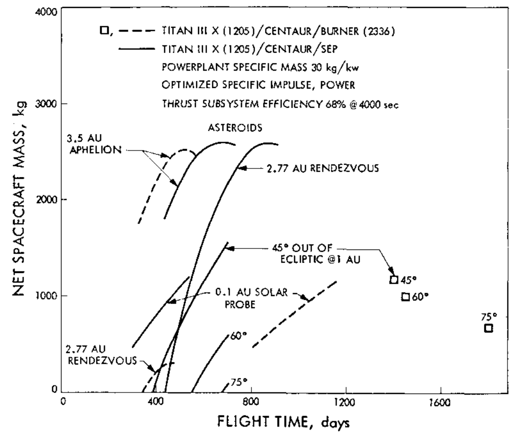

Fig. 13 Area missions.

days for orbiters. For Uranus and Neptune flybys it was 4000 and 4500 days, respectively, whereas for orbiters it was considerably longer.

Nonoptimum Power and Specific Impulse

A very significant factor in selecting the design-point power level of a SEP spacecraft is the fact that the net spacecraft (payload) falls off less than linearly with power levels less than optimum. This characteristic is shown for a Neptune flyby mission in Fig. 14, where operation at 80% of optimum power sacrifices but 10% in net spacecraft mass, and at 66% of optimum power sacrifices but 20%. The falloff of net spacecraft mass for this Neptune flyby mission is more rapid than for many other missions where the ballistic (zero power) capability is higher than the 14% for this mission. For such missions it is not unusual to sacrifice but 10–15% in net spacecraft mass in reducing power to half of optimum.[30] The value of being able to use less than optimum power with such small sacrifice becomes apparent when considering a single spacecraft or powerplant for a wide range of missions, and in view of the fact that the cost and complexity of the solar-electric powerplant is nearly proportional to the power level. It is important to note that the optimum power is primarily dependent upon the launch vehicle mass injection capability near Earth escape velocity. For nearly all of the Titan IIIX(1205)/Centaur launched SEP missions of Figs. 7, 10, 12, and 13, optimum power fell in the narrow range between 35 and 55 kw. The exceptions were at very short flight times for direct missions where the optimum power tends toward zero, and the orbiters of Mars and Venus where the low-energy requirement favored low power also. It can also be seen in Fig. 14 that the way in which less-than-optimum power can be

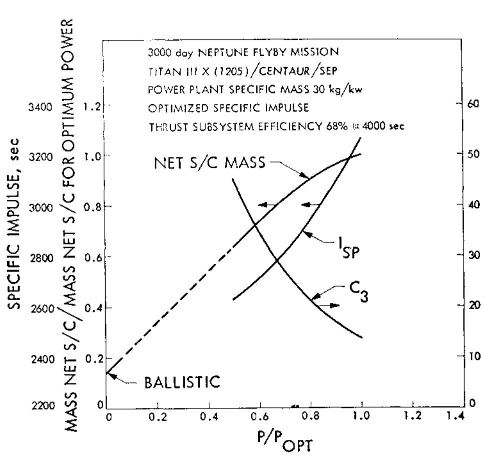

Fig. 14 Effect of using nonoptimum SEP spacecraft power.

utilized, of course, is by depending more heavily on the launch vehicle, i.e., higher values of C_3. The move toward lower power also results in lower optimum values of specific impulse. This, however, is not an advantage, in that efficiency decreases and development problems increase with lower specific impulse, exactly opposite to experience with chemical rocket engines.

In this regard, it is fortunate, however, that the variation of net spacecraft mass with specific impulse off of optimum is slight at higher than optimum specific impulse, typically about 5% less net spacecraft for 30% higher than optimum specific impulse (see Fig. 15). The falloff is much more precipitous at lower than optimum specific impulse, especially down toward 2500 sec or lower where the thrust subsystem efficiency is falling rapidly. The optimum specific impulse is primarily tied to the power plant specific mass, increasing as the specific mass is reduced. The optimum specific impulse for most of the missions and flight times presented fall within the range from about 2500 to 4000 sec. The importance of this characteristic becomes evident when the multimission capability of a given SEP power plant is considered.

Power Availability

Another of the characteristics of SEP spacecraft that should be noted is the availability of power at levels that generally exceed the present requirements of science experiments for all missions out to and including Jupiter. Only for the Mars, Venus, and Mercury orbiter mission where high-power mapping is a logical mission objective, has any of this power been credited to the net spacecraft mass (Fig. 12) and then only up to the amount believed to be useful. Extra power availability may well impact the nature of the science experiments that are feasible and may reduce mass requirements for other engineering subsystems, such as data storage and processing, because higher communication bit rates are possible. For the inner-planet orbiter missions the power can also be used at times for orbit trim and for changing of orbital altitude with very precise control. In Ref. 37, it was shown that the Mars mapping SEP spacecraft of about 2200 kg could be lowered from an orbit of about 4 planet radii to 1.24 radii in 40 days using only about 150 kg of mercury propellant.

Spacecraft Design and Operation

As we reach beyond our nearby neighbors Mars and Venus, it must be expected that the missions will grow more difficult and challenging and will naturally lead to requirements for more complex spacecraft than we have flown thus far. The level of complexity and need for new technology content of ballistic spacecraft will increase dramatically for these missions to include such things as high-energy kick stages, multiple zero-g starts, long-term storage of cryogenic propellants, isotope power supplies, approach guidance etc., to name but a few. Some of the characteristic features of SEP spacecraft have been mentioned such as stowing, deploying, and operating large flexible solar arrays that must be kept pointing toward the sun; rotation of the spacecraft body with respect to the solar array to provide variable thrust vector angle; adjustable thrust vector position to account for shifts in center of gravity and to provide 3-axis stabilization; variable power and thrust; switching off of engines as power declines; continuous operation of the propulsion system, and continuous guidance and control, both for hundreds of days. In general, it is expected that a SEP spacecraft will be functionally and design-wise more complex than its ballistic counterpart. This is the natural price of greater capability. However, the relative difference in complexity of the two types of spacecraft will decrease as the more difficult missions impose increasingly more demanding functional requirements. It should be noted that all of the elements of complexity of a SEP spacecraft mentioned are clearly not needed for the earlier, less demanding of the future missions. These elements could be added in stages for successively more demanding missions such that the discontinuities in technology, after the initial step of applying SEP to missions, could be fairly minimal and well-ordered. The use of fixed thrust vector angle and less power than optimum are likely to effect the most significant reduction in complexity and cost. To date, no SEP spacecraft has been designed for planetary missions. There have been only conceptual design studies thus far.[1,37–39,42,43] Thus it is likely that as additional design talent is brought to bear and actual detailed designs of SEP spacecraft are performed 1) some of the requirements that lead to SEP spacecraft complexity will be eased by demanding less than optimum performance capability, and 2) presently conceived solutions to some SEP spacecraft functional requirements will be simplified through further design and development.

Costs of SEP Spacecraft

Probably the one factor most important to the decision as to whether a new concept should be employed is its comparative cost with the existing method of performing similar tasks. This necessitates the comparison of projected costs with present actual costs, a most difficult comparison at best, but one that cannot be avoided.

It seems appropriate to compare the cost of SEP with chemical propulsion for planetary spacecraft only on an over-all program basis, rather than on any artificially constrained portion of it. Total costs should include those for: launch vehicle hardware and operation, and any development and integration; spacecraft design, hardware, and integration; science payload; and mission operations. A comprehensive discussion of each of these is beyond the scope of this survey; however, the cost impact of utilizing SEP as compared with a ballistic spacecraft in each of these areas is noted and summed.

The basic cost of the SEP powerplant (thrust and power subsystems) is most directly tied to the cost of the solar array, which is expected to be the dominant cost. This cost is quite linear with power level, whereas the cost of the thrust subsystem may be fairly insensitive to power level. The net result is a conservatively projected power plant cost per kilowatt of about $800,000 near 5 kw, reaching an asymptote of about $500,000 near 25 kw. This is the cost per kilowatt purchased, not flown. Thus, for example, if the program philosophy demands three fully operational units be purchased per flight, one as a spare, one as a never-to-be-flown fully operational test unit, and one for flight, then these figures must be tripled. In practice, utilizing the modular power and thrust subsystem approach, complete spares or test units would not be required, reducing the ratio of kw-bought to kw-flown to less than 2. The so-called nonrecurring or development costs are estimated to be between 2 and 3 times the per-kilowatt-purchased cost. As a consequence of the SEP characteristics for operation at off-optimum power and specific impulse described, it should be possible to limit this development cost to a once-only cost for a wide range of missions launched on a given launch vehicle.

The impact on the cost of the spacecraft design, development, fabrication, and integration (exclusive of the SEP subsystem cost) resulting from the use of SEP is much harder to get a handle on than the cost of the powerplant. One estimate of this cost was about 10–15% of the spacecraft program cost for a two-flight program.

Because of the variability of estimates of future rates of launch, the costs of launch vehicles for future programs are difficult to pin down. Clauser[55] has listed the present costs per launch of the Titan III and Saturn V as about 40 and 200 million dollars, respectively. Assume for our purposes that the "learning curve" process will drop those figures to about 20 and 120 million dollars, respectively. Assume also that the costs per launch of all of the members of the Titan family (including those with the Centaur, and/or the Burner, and/or solid strap-ons) are the same within a million dollars or so. Based on these costs, it is reasonable to assume the Atlas/Centaur to cost about half of the Titan family, i.e., 10 million, and the SIC/SIVB/Centaur somewhere about midway between the projected Titan and Saturn V costs, say about 70 million.

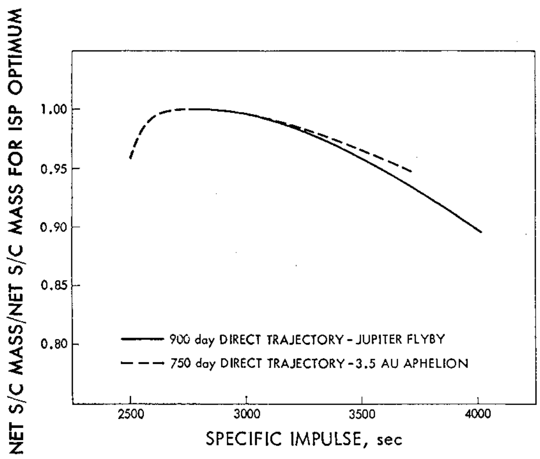

Fig. 15 Effect of nonoptimum specific impulse on net spacecraft mass.

Consider as an example a mission where an Atlas/Centaur/SEP spacecraft can accomplish a mission that would require a Titan IIIX(1205)/Centaur/Burner for a ballistic spacecraft. The reduction in launch vehicle costs for a two-flight program would be about 20 million dollars, assuming no development or integration costs would have to be incurred for the larger launch vehicle. The optimum power of the SEP powerplant would be of the order of 12 kw. Thus, at about half of optimum the power level would be about 6 kw. If three SEP powerplants were purchased for this two-flight program, the recurring cost would be about 15 million dollars, the powerplant development costs about 10–15 million dollars, and the spacecraft-program cost increment due to SEP about 10–15 million dollars. Thus, considering incremental increases in science payload, mission operations, and program management costs, which could add up to 5 million dollars more, the SEP spacecraft approach is conservatively estimated to be about 20–30 million dollars more than the ballistic spacecraft approach for this program, perhaps representing a 10–15% higher over-all program cost. For subsequent flights of essentially the same SEP spacecraft, the SEP power plant and spacecraft development costs would no longer apply, and thus the SEP costs would be about equal or a little less. Should the first SEP spacecraft program be compared against a ballistic spacecraft alternative that required the financing of the development or integration of a major new stage, such as a Centaur or kick stage, the extra costs resulting therefrom could shift the cost comparison in favor of the SEP by a substantial amount.

From this comparison it is clear that a reduction in launch vehicle requirement from a larger to a smaller member of the Titan family by use of SEP would be unlikely to result in lower over-all costs due to the small spread in the costs of launch vehicles within this family.

On the other hand, should the use of SEP on a Titan/Centaur provide capability for doing a mission requiring a SIC/SIVB/Centaur or a Saturn V, the over-all cost comparison is likely to be substantially in favor of the SEP. Here the optimum SEP power level might be as high as 50 kw. With operation at half optimum, the SEP powerplant development cost would be about 25–35 million, and the recurring costs of three purchased power plants for a two-flight program about 35–40 million dollars, the spacecraft program cost increment due to SEP about 20–30 million dollars, and the increment in the science payload, mission operations, and program management would be about 5 million dollars. The sum of these cost elements is quite similar to that projected for a Mars orbiter SEP spacecraft of about the same power level.[1] On a first program use of this SEP spacecraft, the total costs of the SEP approach would possibly be fairly close to those of the ballistic approach using a SIC/SIVB/Centaur. For subsequent programs using that SEP spacecraft, or any mission where the ballistic approach would require the Saturn V, substantial saving would accrue from the SEP approach.

It can also be shown that when SEP is used, for example, to double the payload that can be delivered with a given launch vehicle, the cost per pound of net spacecraft mass is 30% or so less using the SEP approach as compared with the ballistic approach, based on the kind of SEP costing inputs mentioned.

A Phased Sequence of SEP Spacecraft Application

As has been indicated, much of the potential cost advantage to be accrued through the use of SEP can only be realized through multimission use of the two principle SEP subsystems, the solar-power and thrust subsystems, and the spacecraft as well. Thus it is of interest to examine potential SEP missions to see which can benefit from the use of the same SEP powerplant and the same basic SEP spacecraft, accepting less than optimum performance by operating at common off-optimum conditions. While the results presented in Figs. 7, 10, 12, and 13 are for optimum power and specific impulse, some general indication of off-optimum operation penalties has been gained from performance and spacecraft conceptual design studies of a few missions[38,42,56] (Figs. 14 and 15). In addition, approximate allowance for real ephemeris and finite launch-window constraints can be made. From these considerations, approximate flight times can be derived for individual missions that fall within a mission set that might share a common SEP powerplant and spacecraft. By such a process, a first cut at a phased sequence of SEP missions is outlined in which the technology, complexity, and sophistication is increased in stages. Additional mission analysis and spacecraft conceptual design will have to be performed to confirm the off-optimum performance estimates made and to determine if all of the functional requirements for the several missions can be satisfied with the common spacecraft.

In a first phase it is suggested that a 6–10 kw, 3000–4000 sec specific impulse SEP powerplant be developed and used on a fixed-thrust-angle spacecraft of 450–500 kg net mass (payload) launched on an Atlas SLV3C/Centaur vehicle. This combination could fly a 3.5-a.u.-aphelion asteroid mission (with 900 days beyond 2 a.u.) in about 700 days to aphelion. This mission might be desirable as an early mission due to its tolerance to premature termination of thrusting, and because of the significance of the meteoroid hazard (its prime science objective) to future flight beyond Mars. As alternative early missions, the same SEP spacecraft could be put into a 2.77-a.u. circular orbit (center of the asteroid belt) in about 850 days or could be flown to 45° out of the plane of the ecliptic at 1 a.u. in about 900 days. All three of these missions could be flown in the mid-1970s (1975–1977), based on the presently expected pace of solar array, thrust, and software subsystem technologies. To make the developed subsystem elements have even broader utility than the missions mentioned, it is reasonable to consider building this first-phase powerplant out of solar array and thruster modules that could form the building blocks for a higher-power second phase.

A second phase SEP powerplant should be of some 15–20 kw, with 3000–4000 sec specific impulse. The solar array should be developed to take maximum temperatures of up to about 200°C in order to permit its use for Mercury and solar probe missions. The powerplant would be incorporated on a spacecraft that can be tilted around a fixed thrust axis arranged 90° to the sun-probe line. By tilting the spacecraft around this axis by as much as 70° to the normal, it should be possible to restrict array temperature to 200°C and maintain solar power as far in as about 0.3 a.u. at a power level about 1.4 times that at 1 a.u. Although only limited analyses of missions such as these are available, the results suggest that such a spacecraft, with net spacecraft mass (payload) up to 1000 kg could be launched by a Titan IIIX(1205)/Centaur and could 1) fly to 60° out of the ecliptic in about 1000 days, 2) fly in to 0.1 a.u. in about 600 days, and 3) orbit mercury in a $1.3R \times 4R$ orbit in about 650 days. As a second SEP spacecraft applications phase, these missions could be flown in the latter portion of the 1970's (1978–1980).

As an alternative, or concurrently with the second phase, one might consider mapping orbiters of Mars and Venus as a phase 2A. The powerplant would also be of 15–20 kw (with array retraction capability), and 3000–4000 sec specific impulse as before. The prime difference would be that the solar array would not have to be developed for the higher operating temperature but could be designed to the more usual 140°C. Should, somehow, the weight or performance penalty for the higher temperature operation be negligible, this distinction would, of course, vanish. The other important difference would be in the spacecraft design, which would probably want to have variable thrust angle capability in order to provide approximately 2000 kg of net spacecraft mass (including about 400 kg of power) arranged for planet-mapping missions, and capable of SEP-powered orbit adjustment.[37] This spacecraft, launched by the Titan IIIX(1205)/Centaur, could orbit Mars ($1.3R \times 4R$) in about 260 days with about 10 kw in orbit, and could orbit Venus ($1.2R \times 2.6R$) in about 200 days with about 25 kw in orbit. Some 10–20% of the 2000 kg could be sacrificed for circular orbits, if desired, by using larger retros. Based on an estimate of the pace at which the technology and spacecraft could be developed, missions such as these could be flown either in the latter part of the 1970's (1978–1980), or following phase 2 described previously.

A third phase of SEP spacecraft applications might include flybys of the outer planets in the first half of the 1980's (1980–1985). Such a program of flybys might be required to provide additional details to fill out that portion of the outer-planet science objectives not completed during the two currently considered Grand Tour missions of the late 1970's. After these two missions there is a very long interval before Jupiter swingby ballistic missions are again possible. Again, a 15–20 kw, 3000–4000 sec specific impulse powerplant, similar to that of phase 2A, would be suitable. The Titan IIIX-(1205)/Centaur could provide 1000 kg net spacecraft missions to Jupiter in about 500 days, Saturn in about 1100 days, Uranus in about 2200 days, and Neptune in about 3100 days.

Finally, a fourth phase of SEP spacecraft applications might include moderate-capture orbiters ($2R \times 38R$) of the outer planets Jupiter, Saturn, and Uranus. The same powerplant, spacecraft (with chemical retro added), and launch vehicle as used for phase 3 might again be suitable. Such missions, about 900 days to Jupiter, about 2000 days to Saturn, and about 4300 days to Uranus, could be scheduled for launches in the mid-1980's or earlier, should the phase 3 flybys be unnecessary due to completion of outer-planet science objectives by the Grand Tour flights. At a modest sacrifice in net spacecraft mass or flight time, higher periapsis orbits or more circular orbits can be provided.

Thus, the use of SEP spacecraft might well extend over a decade or more and be the forerunner of nuclear-electric spacecraft which could eventually provide the capability to explore the whole solar system with payloads of substantial power and mass.

Summary—Conclusions

This paper has presented the current estimate of the characteristics, capabilities, and costs of Solar Electric Propulsion

(SEP) for automated missions to the planets, and has suggested a first-cut phased sequence for such missions in which an attempt was made to maximize common usage of developed SEP subsystems and SEP spacecraft. In such multimission use it is believed that the use of SEP will not only increase spacecraft payload or alternatively reduce flight time, but will also effect substantial program cost savings based on the economics of an over-all mission set when compared with the alternative of ballistic spacecraft.

The basic characteristics of a SEP spacecraft discussed included the fact that the SEP is a high velocity-increment stage providing the velocity at high specific impulse and hence low propellant mass. The impact of the varying solar power as a function of distance from the sun was discussed as it relates to the nature of optimum trajectories and to the required throttling of the thrust subsystem. It was shown that special functions of a SEP thrust subsystem such as translators for thrust vector position control can also be utilized for attitude control, just as residual power from the SEP power plant can be used for science experiments. The use of a variable thrust-vector-angle with respect to the solar array, needed for those missions where optimum payload is needed or desired, was shown to influence the spacecraft design and its complexity to a substantial extent. The interactions of the SEP powerplant with science experiments has been investigated and shown to be of moderate consequence for planetary missions by virtue of the thrusting operation usually having been completed long before planetary encounter. Even for interplanetary science experiments and communications, which must be operative during thrusting, it has been shown that compatibility with the SEP powerplant can be achieved with little penalty.

It was shown that the performance penalty for operating at power levels down to half of optimum, and specific impulses up to 30% or so above optimum can be quite modest under certain circumstances. This was shown to contribute toward making it possible to utilize a given SEP spacecraft and powerplant for several missions and in that way effect substantial cost savings over a mission set.

The capabilities of SEP spacecraft were compared with those of ballistic spacecraft on a circular, coplanar solar system model without penalty for finite length launch window. Compared on a real ephemeris, 30-day launch window basis, the comparative advantage of SEP spacecraft over ballistic spacecraft becomes significantly greater. It was shown that the SEP could provide substantial spacecraft net mass for flyby and orbiter missions of each of the outer planets, ranging from 50 to 200% greater than ballistic spacecraft in equal or shorter times than would be flown ballistically. The SEP comparative advantage was especially significant for flybys of Uranus and Neptune, and orbiters of Saturn and Uranus where ballistic capability launched by any of the Titan family launch vehicles is probably inadequate.

It was shown that Titan-launched SEP spacecraft could provide a capability for orbiters of Mercury not even possible using the Saturn V except with the addition of a high-energy two-stage retro. For orbiters of Mars and Venus the basic SEP performance advantage was found to be modest, but the dual use of the solar arrays for SEP and high-power mapping might make even these missions attractive.

Very large reductions in flight time were found possible using SEP for close solar probes and out of the ecliptic shots as compared with ballistic spacecraft because of the ballistic spacecraft having to swing by Jupiter to pick up enough energy. The comparative advantage of SEP for flying through the Asteroids was found to be small if any, while the capability for establishing a circular orbit rendezvous near the center of the Asteroid belt was found to be very much greater than possible for a ballistic spacecraft.

Based on the characteristics, capabilities, and costs thus far determined for SEP spacecraft for planetary missions, it is apparent that there are sufficient advantages over other alternatives that this approach should figure prominently in the future mission planning.

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