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Solar-Cell-Powered, Electric Propulsion for Automated Space Missions

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ELECTRIC propulsion has been under serious study for the past decade, the principal motivation being potential capability and cost advantages resulting from its characteristically efficient use of propellant mass. This fuel economy derives from the use of electrical energy to drive propellant mass to exhaust velocities as much as an order of magnitude greater than that achievable in current competitive systems. Higher exhaust velocity means that a lower propellant expenditure rate is required for a given thrust level and to achieve a given mission total impulse. As a result, payload of a given launch vehicle is increased, or launch-vehicle requirement for a given mission is reduced. Moreover, electric power available during nonpropulsion periods and at destination may be profitably used. Because electric propulsion systems will operate at low thrust levels and over an appreciable fraction of the mission time, they will have trajectory-related performance characteristics that differ in many respects from those of the familiar ballistic spacecraft. Such differences may provide advantages; for example, in the case of a Mars orbiter, both approach velocity and retro-propulsion requirements are lower for the solar-electric spacecraft than the ballistic.

Electric propulsion systems have been tested in space and used operationally in an auxiliary propulsion role for satellite maneuvering. Power-source and thruster technologies have progressed to points such that a more expansive role can be considered. In 1964, it was recognized that lightweight solar arrays could be made available within some reasonable time. Electron-bombardment thruster systems have passed long life tests in the laboratory. Therefore, steps have been

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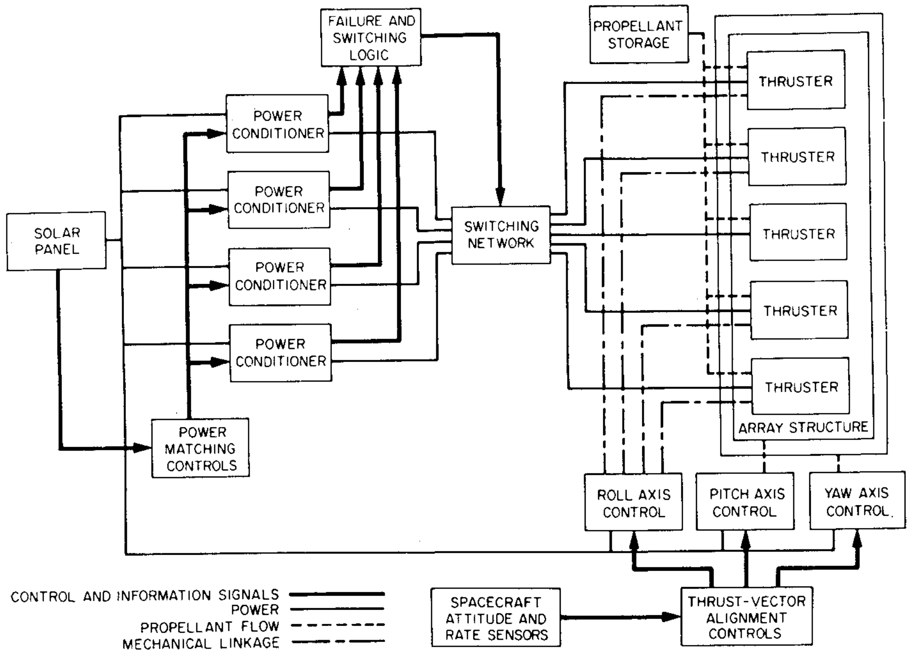

Fig. 1 Solar-powered electric propulsion system concept.

initiated to determine whether power and thruster technology could be successfully wed into practical spacecraft designs of some utility. Present NASA programs are aimed at demonstrating the practicality of such systems by the early 1970's. Assuming successful completion of this current effort, solar-powered electric propulsion mission launches in the 1975-1980 time period appear to be possible. The range of existing launch vehicles in the Atlas Centaur-Titan III class can be extended to encompass virtually all solar system targets. It is to the usefulness of such a capability that this work is addressed. The current status of applicable technology is briefly summarized, mission and systems analysis results are reviewed, and applications which appear attractive are discussed.

Technology Summary

Present solar-powered electric spacecraft concepts have not diverged substantially from those evolved in the early studies of Ref. 1. The large solar-cell array and thruster system are thoroughly integrated into the mission spacecraft. Alternative concepts accommodating the propulsion functions in a separable stagelike configuration have been suggested but not yet explored in sufficient detail to permit evaluation. A typical mission profile would call for ballistic launch to Earth escape velocity or beyond and deployment into a cruise configuration. The ion propulsion system would then be turned on and operated throughout some fraction of the transit time. The solar array provides the power required by the thruster system, which consists of the thruster array, power conditioning, control mechanisms, and propellant feed components. For good performance, an over-all specific mass α for the propulsion system, including the solar array, of approximately 30 kg/kw of electrical power from the solar array is sought.

The solar array delivers low-voltage d.c. power, the level and voltage of which varies with distance from the sun, and which must, in the power conditioner, be converted into forms suitable for use in the thruster. Based on reliability, and power profile matching considerations, a modularized approach to power conditioning, thruster, and propellant storage components is currently favored. When the thruster system is operating it may be the principal source of attitude error to the spacecraft. It is therefore important that the propulsion system compensate for thrust-vector-misalignment-induced attitude errors. Any adverse thruster system interaction problems with spacecraft structures, materials, communications, or payload equipment must be identified and solved. In some cases optimal trajectory considerations require large thrust vector excursions which significantly affect configuration. Such cases must be accommodated.

The program on which the feasibility of obtaining the desired system performance rests is composed of two major elements: laboratory demonstration of appropriate solar array and engine system technology and oribital flight test of an ion engine system on the SERT II spacecraft scheduled for launch in the fall of 1969. These efforts are sponsored by the NASA Office of Advanced Research and Technology and conducted mainly by the Jet Propulsion Laboratory (JPL) and NASA Lewis Research Center.

Solar Array

Stimulated by possible electric propulsion application, an effort to determine the feasibility of a 50-kw fold-out solar array with a specific mass α_w at 1 a.u. of 23 kg/kw has been in progress since 1964. Ratcheson[2] concluded early in the program that such an array was feasible, provided that 8-mil silicon cells could be mounted on an epoxy fiberglas tape substrate and supported by stiff lightweight beryllium structures. Progress since that time (see, e.g., Ketchum et al.[3,4]) has served to establish a firm technology base. A recent analysis[5] for a 14-kw array gave $\alpha_w = 22$ kg/kw and showed that an 18% allowance for proton degradation and measurement uncertainty should be assessed in order to provide moderately conservative performance. A more advanced fold-out approach based upon large-scale aluminum electroforming has been reported on by Carlson.[6] Such a concept would use thin (4-mil) solar cells with integral cover glass and could provide power at 12 kg/kw.

In addition to the "traditional" fold-out concepts, there are an impressively large number of current efforts on deployable arrays of the window-shade or roll-up type. In this approach solar cells are mounted on thin plastic sheets and rolled up on a drum from which they are deployed once in space. Parallel experimental efforts at Ryan,[7] General Electric,[8] and Fairchild[9] have all concluded that $\alpha_w \leq 15$ kg/kw can be expected from such an approach. Berry et al.[10] concluded that the next step should be qualification of a 5-kw array with $\alpha_w \leq 18$ kg/kw using conventional 2-cm × 2-cm silicon cells.

Based on the foregoing achievements, an α_w allocation of ~20 kg/kw is considered conservative for performance estimation purposes.

Thruster System

A block diagram of the present concept of a solar-powered electric propulsion system is presented in Fig. 1, taken from Ref. 11 (which, along with Ref. 12, provides excellent background material on this subject). Choice of thruster is perhaps the single most important determinant in specifying the remainder of the propulsion system. If present trends continue, there would appear to be little doubt that the mercury electron-bombardment ion engine will prevail as the prime candidate for application.

Electron bombardment thrusters have been tested in the laboratory for periods of continuous operation up to a year and a mercury-fueled thruster of this type has been chosen for the SERT II flight. A brief description of its operation[13] should serve to illustrate important parameters of this class of thruster. Figure 2 is an artist's concept of a cross section through the SERT II engine. Liquid mercury is vaporized by heating, passed through an isolator that serves to electrically decouple the engine from the propellant supply, and enters a distributor manifold. About 20% of the flow passes through the "hollow cathode" and the keeper electrode, which supports a discharge supplying ~2 amp of electrons to the main discharge chamber. (Other cathode types have also been evaluated and operated for long times; however, the hollow cathode is the presently preferred approach.) The remainder of the flow is injected into the chamber through the distributor. A magnetic field in the discharge chamber tends to constrain electrons to cyclic motion about the field lines and enhances the collisional ionization process. The baffle also enhances the ionization process by controlling discharge voltage or the flow of electrons from hollow cathode to anode. Ions formed in the chamber drift to the screen and accelerator grid where an applied field accelerates them through concentric holes. This beam of positive ions is neutralized downstream of the grids by electrons injected into the stream by the plasma bridge neutralizer. The entire thruster is surrounded by a ground screen which prevents space plasma electrons from being drawn into the thruster surface.

This thruster is approximately 15 cm in diameter and is designed to operate at 1 kw. As we shall discuss later, module sizes of from 1 to 3 kw are consistent with anticipated requirements, and, as a consequence, several thrusters larger than the SERT II thruster have also been tested (see, e.g., Refs. 13 and 14).

Because the power generation system represents the major fraction of propulsion system weight and volume, the most significant figure of merit for a thruster is its efficiency in converting electric power into thrust power. Figure 3 shows the dependence of this efficiency η upon specific impulse and operating power of the engine. This η must be penalized by power conditioning inefficiency and inefficiencies that may be involved in throttling thrusters to match the mission power profile. The lower curve displays the performance achieved

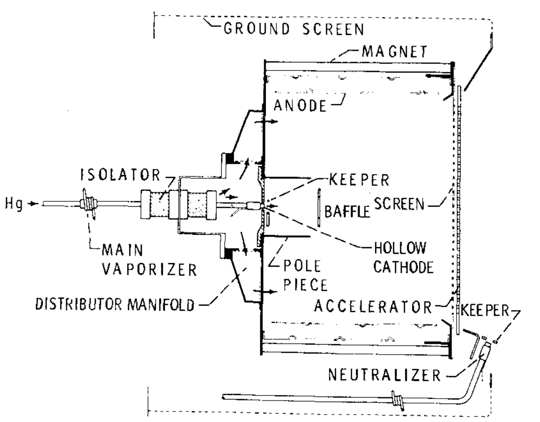

Fig. 2 Sectional view of SERT II thruster.

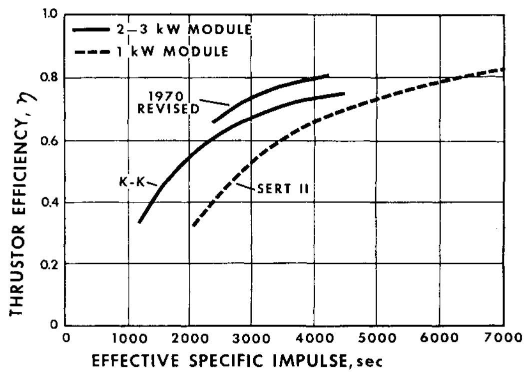

Fig. 3 Thruster efficiency variation with specific impulse and power level.

with the SERT II 1-kw thruster.[15,16] The upper curve is 1970 predicted performance of a 2- to 3-kw thruster that takes advantage of technology developed in the SERT program.[11] The intermediate K-K curve represents a performance prediction[12] of the 2 to 3-kw module that was made about a year ago, and this performance has since been exceeded in the laboratory. Curve K-K was used in the JPL Jupiter flyby study[5] and is regarded as being consistent with the attainment of long life.[13]

The power conditioning units (Fig. 1) contain the various power supplies and control loops required for thruster operation. They accept input power at voltages which vary by a factor of two or more due to temperature effects on solar arrays. They transform this power into forms suitable for use in the thrusters, and with present concepts provide for stable operation over a factor of two in power. The best approach to the thruster throttling has not yet been determined, although at least two methods for doing so have been explored. A power conditioning system has been built and operated for 500 hr in a closed-loop mode in the laboratory with a thruster at a single operating point.[1] The operating point will of course vary throughout a mission with the available or programed power profile. Therefore, a modified version of this power conditioning equipment is being used in open-loop, power-matched operation in order to identify control requirements. The information and experience generated in this testing will then be applied in complete closed-loop system tests.

Other elements of the system will also be included in closed-loop tests aimed at demonstrating over-all system feasibility. These include thrust vector alignment control, failure logic, switching, etc., as shown in Fig. 1. Practicality of these individual elements has been shown via reduction to design, or construction. For example,[5,11] propellant tankage with a weight of 3% of mercury propellant loading has been successfully tested, and a lightweight actuator suitable for the gimbal-translator thrust vector alignment system is in test. What remains to be shown via successful closed-loop system operation is aggregate feasibility.

The objectives of current programs are to achieve such demonstrations as may be required within an over-all thruster system specific mass α_{ts} of 10 kg/kw. Based on present hardware specifications, this α_{ts} would be distributed approximately as follows: power conditioning, controls switching and logic (92% efficient), 5 kg/kw; thruster (including redundancy), 2 kg/kw; thrust vector control, fixed weight, and contingency, 3 kg/kw.

The specific mass will vary with power level due principally to change in engine module size and the amorization of fixed masses. It also appears possible to hold the efficiency penalty associated with engine throttling over a 2:1 power range to ~0.05. However, this area is still under active research and the throttling methods finally selected on the basis of over-all control system characteristics may have some-

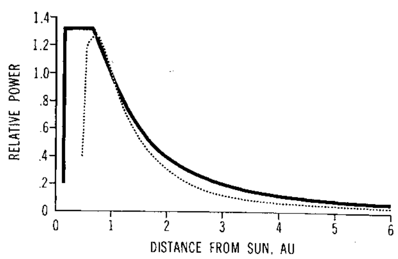

Fig. 4 Solar array power relative to that at 1 a.u. vs sun distance. Dashed curve based on Sandstrom[22]; solid curve from Strack.[23]

what greater penalties. These mass estimates are consistent with hardware in the power range of from 5 to 20 kw using a thruster module in the 2- to 2.5-kw range. At lower powers, a 1-kw thruster module may be more appropriate, and because of fixed losses both efficiency and specific mass would be adversely influenced.

Determination of the appropriate level of modularization is only one facet of the over-all propulsion system design problem that necessarily involves tradeoffs among many parameters, some of which are poorly specified. In order to alleviate this problem the development of a computer program with the objective of optimizing the design of modular propulsion systems was commissioned.[17] The method chosen uses relative reliability and mass as criteria and appears to offer significant promise. A first application of this methodology to a spacecraft design problem is presently in process.[18]

In addition to the present ground-based programs, it is anticipated that the SERT II flight test will make major contributions to the development of mission technology. Plans call for a 1969 Thorad-Agena launch into a near polar sunlit orbit. The gravity-gradient-stabilized Agena equipped with a 1.5-kw solar array will serve as the basic spacecraft. Two 1-kw mercury bombardment engines, fired sequentially, will be used in satisfying the prime experiment objectives of verification of long-term engine performance in the space environment. Experiments aimed at assessing possible electromagnetic interference with communications and propellant deposition effects are also planned. In addition, this program will serve to demonstrate solutions to thruster system testing, storage and handling, and some spacecraft integration problems. SERT II will attempt a direct in situ measurement of propellant deposition on solar arrays located near the ion beam. A program to evaluate beam interactions with spacecraft structures is also underway in the laboratory.[19] In this effort, the effects of ion and atom impingement on surfaces and materials have been modeled and verification experiments identified and designed.

An attempt to assess the potential problem or interference between the propulsion system and science measurements is also in process.[20] The first phase of this work has determined that while optical, neutral-particle, and high-energy-charged-particle measurements appear unaffected, the bulk of particle and field measurements are subject to possible problems if the thruster is simultaneously operating. The principal sources of interference effects appear to be contaminant magnetic fields from array or thruster currents and spacecraft equilibration potential relative to space plasma. The latter may be alleviated via neutralizer bias; the former, by magnetic decomtamination design criteria. Study of this important area is continuing, but as yet no demonstrated and unresolvable interaction areas which would compromise science experiments exist.

There are a large number of components involved in solar electric propulsion systems. Prior to committing a mission to this technology, each of these components and their collective behavior should be both acceptable and well understood. Present technology programs with these objectives have encountered no fundamental problems, and indicate that a level of capability of some interest for flight applications should be available in the early 1970's. This level will exist without ecourse to more radical technological advances such as the "thin-film" solar array or single insulating grid thruster, which may be viewed alternatively as contingency or growth capability.

Mission Analysis

As has been seen, propulsion system technology inputs change with time and system choice, and, as a consequence, many alternatives are offered the mission analyst. There are also various launch vehicles to choose from for which performance data[21] are available. Solar-electric propulsion, deep-space mission profiles studied to date make use of the launch vehicle to inject the electric spacecraft into an Earth departure coast trajectory having, at least, escape energy. Low-thrust Earth escape spirals have not yet been thoroughly evaluated for solar-electric mission profiles. Such trajectories may call for great complexity in configuration of spacecraft and/or significant trip time penalties.

Another important initial consideration is the choice of power profile, i.e., the variation of solar-cell power output with solar distance R_s. Two examples are given in Fig. 4. The curves share two characteristics: 1) an increase of power output of at least 25% at $R_s < 1$ a.u. and 2) a rapid decrease at $R_s > 1$ a.u. to $\sim$50% power at Mars and less than 10% at Jupiter. Although any power profile is dominated by the inverse-square variation of solar flux density with R_s, it is also affected by solar-cell characteristics and the solar-panel design. Because the energy conversion process in solar-cells decreases as cell temperature increases, there is an upper limit on the gain in output power at $R_s < 1$ a.u. In Fig. 4, the flat portion of the solid curve at its maximum output results from tilting the solar panel away from the normal to the sun line to control the temperature. Decreasing solar flux beyond 1 a.u. decreases the available power but increases efficiency. The gain in efficiency with decreasing temperature beyond 1 a.u. is less in the dashed curve[22] than was assumed for the solid curve.[23]

For missions beyond 1 a.u., this power profile shortens the time available for effective propulsion and causes a steady drop in power to the thrusters. The reverse is true on inbound missions. This power variation strongly influences trajectory selection. Figure 5[23] shows two typical inbound trajectories for a 0.1 a.u. solar probe using electric propulsion: one is best for a spacecraft having a constant source of electric power, the other is best when solar power is used. In the former case, it is better to first take the spacecraft to $R_s \simeq 2$ a.u., then descend to 0.1 a.u. In the solar-powered case, the strong dependence of power and thrust on R_s leads to a trajectory looping the sun, in ever-smaller ellipses as the perihelion decreases to 0.1 a.u.

In outer planet missions the influence of solar-electric power variation on optimal trajectory shape appears in the so-called direct and indirect classes[5,32,34] of mission trajectories, as shown in Fig. 6 for a typical Jupiter flyby mission. The direct trajectory has been so named due to the steady increase of solar distance while the electric propulsion system is operating. The indirect trajectory is characterized by an initial loop about the sun in the vicinity of 1 a.u. followed by a high-energy direct-type path outward to the target planet. Although this loop around the sun provides more solar energy input for the propulsion system, it also requires more pro-

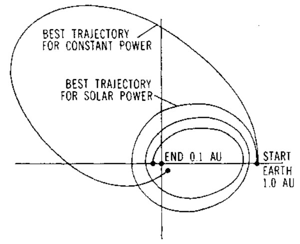

Fig. 5 500-day electric propulsion missions to 0.1 a.u.

pulsive effort and longer propulsion time for a given total mission time. In many cases, propulsion requirements of indirect trajectories increase as mission trip times are reduced so that direct trajectories become better choices for solar-electric systems.

Review of Previous Studies

Table 1 summarizes a representative cross section of missions studied for solar-electric propulsion. Gross payload is defined here as the total spacecraft mass minus structure,* tankage, propellant, and propulsion system masses. The large differences in power and payloads shown are functions of both mission and launch vehicles studied, the latter ranging in size from Thorad-Agena to Saturn IB/Centaur.

Geoconcentric Missions

The literature on geocentric solar electric missions is somewhat limited although current interest is increasing. Ramler[25] discussed techniques for maximizing altitude gain or time in continuous sunlight for a spacecraft with a solar-electric propulsion system. Meissinger et al.[26,27] proposed a 3-kw, 300-kg, solar-electric propulsion spacecraft as an economical follow-on to SERT II and outlined an interesting mission to explore the interaction of the tail of Earth's magnetic field with the solar proton wind to as far as 200 Earth radii. The spacecraft's gross payload of ~100 kg would include 25 kg of scientific instrumentation. The mission would include stationkeeping or maneuvering for 100–200 days and would require ~100 days of propulsion time accumulated over several thrust periods.

Others[28,29] have described the application of solar-electric propulsion to raise a communication satellite to synchronous orbit altitude; e.g., an 8-kw, 1000-kg, TV relay satellite can raise itself to synchronous orbit from an initial parking orbit of 200–300 km. This allows the use of a much smaller chemical launch vehicle than would be required of an all-ballistic mission. The solar-electric mission takes ~6 months of continuous propulsion and calls for 300 kg of the satellite to be devoted to mercury propellant and ion thrusters. Further studies of the satellite raising mission should include an expanded evaluation of tradeoff effects in the areas of system design and comparisons with alternate methods.

Table 1 Survey of solar-electric propulsion mission studies

Mission	P_e at 1 a.u., kw	Spacecraft mass, 100 kg	Gross payload, 100 kg	Ref.
Geocentric				
SERT II	1	...	...	15
Geomagnetic tail	2–3	3	1	27
Satellite raising	7–12	10	4–7	28, 19
Inner planet				
Mars orbiter	5–50	10–70	3–25	24, 30, 31
Mercury orbiter	10–50	10–70	2–10	30
Solar monitor	2–3	3	1	26
Out of ecliptic (30°)	2–3	3	1	26
Solar probe (0.1–0.3 a.u.)	2–40	3–37	1–10	23, 26
Outer Planets				
Asteroid flyby, rendezvous	2–>50	3–70	1–25	26, 34
Jupiter, Saturn, etc.	9–50	10–70	2–20	5, 30, 32, 34
Grand tour	10–30	10–20	2–8	33

* Structure may equally well be included by definition in gross payload as it is in many of the studies.

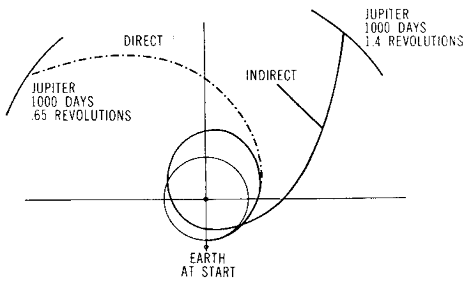

Fig. 6 1000-day solar-electric propulsion missions to Jupiter.

Inner and Outer Planet Missions

Three inner planet missions were proposed by Meissinger[26] for the small, first-generation spacecraft (Table 1). The solar monitor mission would place the spacecraft in orbit about the sun at 1 a.u. but lagging about 90° behind Earth. Since the sun rotates on its axis once every 28 days, observations made by the spacecraft could be relayed to Earth about 7 days before they would be visible to Earth-based observatories. In the other inner planet missions the 3-kw spacecraft could take its 25 kg of science instruments to 30° out of the ecliptic plane and to ~0.3 a.u. from the sun. Its outer planet mission would be to probe the asteroid belt. Asteroid rendezvous missions to 2, 3, 3.5, and 4 a.u. were examined in Ref. 34 for larger launch vehicles.

Solar probe missions with approaches as close as 0.05 a.u. were evaluated in a substantial study by Strack[23] for 19- to 35-kw spacecraft with total masses of 1000 and 3000 kg. This study illustrated the use of multirevolution trajectories for solar-electric and ballistic missions.

Mars and Mercury orbiter missions were included in a broad-brush survey[30] of payload capability for combinations of solar-electric spacecraft and three advanced launch vehicles. Flyby and orbiter missions were studied for targets throughout the solar system. Solar-electric and ballistic mission results were compared on the basis of minimum trip time for arbitrarily fixed payload mass. Sauer[24] presented tradeoffs among trajectory requirements and optimum design parameters for the solar-electric propulsion system for Mars orbiter missions. A Mars orbiter mission was discussed earlier by Molitor et al.[31] based on a lower technology level for solar-electric propulsion systems and trajectory analysis.

One advantage of solar-electric propulsion on Mars and Mercury orbiter missions is that approach speeds relative to the planet are much lower than are possible on equivalent ballistic missions, so that retropropulsion system mass requirements are smaller. Launch-vehicle-injected mass would be halved for given values of payload and available power in Mars orbit.[24,31] It has been noted[30] that for Mercury, ballistic trajectory approach speeds are so high that ballistic orbiter missions are nearly impossible.

A study by JPL[5] used the Jupiter flyby mission to illustrate the depth and detail that are now possible in analyzing operational and design problems for solar-electric spacecraft. Many examples of propulsion system design tradeoffs with trajectory and mission considerations were discussed at length. Zola[32] and Flandro[33] reported less detailed, broader studies of solar-electric missions to Jupiter and beyond. Zola evaluated the multimission capability of a fixed-design propulsion system for flybys and elliptic orbit captures to Jupiter, Saturn, Uranus, and Neptune. It was noted that propulsion times of 800 days or less are sufficient for all outer planet missions, since the effectiveness of the thrusters decreases as the solar power decreases. Flandro analyzed the combination of the Jupiter gravity swingby technique with solar-electric propulsion on the Earth-Jupiter trajectory leg. Solar-electric Jupiter swingby mission payloads were evaluated for flybys of all the outer planets, including the 1977

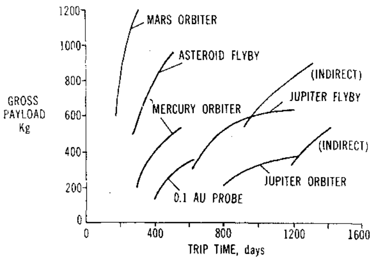

Fig. 7 Variation of payload with trip time for Titan III C launched solar-powered electric propulsion missions.

"grand tour" mission to Jupiter, Saturn, Uranus, and Neptune.

Current Technology Capability

This section is addressed to missions that appear feasible for solar-electric propulsion with technology expected to be available in the early 1970's. Gross payloads are evaluated for a solar-electric spacecraft having a propulsion system specific mass of 34 kg/kw, consistent with the estimates and degradation factors discussed earlier, and ion thrusters with efficiencies noted in the K-K curve of Fig. 3. Some missions and payloads of Table 1 are adjusted to the Titan III C as the common launch vehicle for the solar-electric spacecraft and displayed in Fig. 7. The missions are: 1) Mars orbiter, 2) Mercury orbiter, 3) Jupiter flyby and orbiter, 4) solar probe to 0.1 a.u., and 5) asteroid flyby. In all cases the electric spacecraft total mass does not exceed 2200 kg (Titan III C escape capability) but is as low as 1200 kg for missions that require a high launch velocity. Thruster specific impulse I_s is in the 3000–5000 sec range for all the missions shown in Fig. 7, and installed solar-cell power at 1 a.u. ranges up to 24 kw. Propulsion times for these missions optimize at 50 to 80% of the trip time.

Except for a 400-day trip-time difference, the 0.1 a.u. solar probe data of Fig. 7 is similar in payload level to the Jupiter orbiter. The Jupiter orbiter payload is for a capture in a highly eccentric, elliptic parking orbit having a period of about 50 days and a distance of closest approach to the surface of one Jupiter radius (71,400 km). Unlike Mercury and Mars orbiter missions, Jupiter orbiter retropropulsion requirements are so large that elliptic capture orbits must be used. The higher gross payload capability of missions using indirect trajectories can be seen in the Jupiter flyby and orbiter data at trip times above 1000 and 1300 days, respectively. Indirect trajectories have higher approach speeds relative to Jupiter than direct trajectories. Therefore, indirect orbiter missions have higher retropropulsion requirements than direct orbiters. For this reason, the payload advantages of indirect over direct trajectories for orbiter missions occur at relatively higher trip times than for flyby missions.

As was pointed out, the optimum installed solar-electric power level for maximum gross payload can go above 20 kw for spacecraft launched by Titan III C. However, it may be advantageous or required to consider lower solar power levels. Improved efficiencies of ion thrusters at low I_s and the ability to vary the launch speed provided by the booster have been found to give a high degree of flexibility in power level without seriously affecting gross payload.

Variations in gross payload with solar-electric propulsion system power level are shown in Fig. 8 for a few of the missions of Fig. 7. The payload is not greatly affected by power level; installed power can be cut to 50% of optimum in all cases, except the Mercury orbiter, at a small cost in payload. The Mercury orbiter curve is quite broad but is more sensitive than most missions to power reductions greater than 40%. The optimum power level Mars orbiter mission, at 7 kw can be reduced to 3 kw for a payload decrease of less than 10%.

Spacecraft power level reflects the thrust and I_s at which the electric thruster system is operated. Lower I_s decreases the required power, but the increased propellant requirement can bring on a rapid reduction in payload. In general, increasing the launch velocity given the spacecraft by the launch vehicle reduces the low-thrust portion of the total mission requirements. Owing to this effect, the necessary acceleration level (thrust/mass) of the electric propulsion spacecraft decreases. But, for a given launch vehicle, spacecraft mass also decreases with increased launch velocity. Therefore, higher launch speed has a twofold effect on reducing the required power and thrust level, since both mass and thrust/mass decrease. Furthermore, decreasing the propulsion requirement on the spacecraft by increasing launch velocity allows the use of lower electric thruster I_s with little or no gain in propellant requirement. The net result is that lower power requirements can be achieved at a minimum of payload decrease by a proper tradeoff of launch velocity and I_s.

For all the cases given in Fig. 8, the lower power levels are the result of optimum combinations of increased launch speed and decreased I_s. The Mercury orbiter is an example of a mission that benefits less from increased launch speed. This is because the electric propulsion system is needed to provide a trajectory with a low approach speed at Mercury. Therefore, power reduction depends only on lower I_s, resulting in a more rapid decrease of payload. For this particular Mercury orbiter mission, $I_s \simeq 5000$ sec at the 20-kw level but drops to $\sim$2400 sec for an installed power of 10 kw. Gross payload capability is equal to or not much less than optimum in the 7- to 10-kw range for the other missions shown in Fig. 8. At these low powers, the I_s is 3000–4000 sec for all cases except the Jupiter flyby, for which it decreases to 2400 sec at 7 kw.

Systems Analysis and Spacecraft Design

Integration of solar electric propulsion systems hardware into an interplanetary spacecraft design has not been attempted; thus detailed analyses are required to assess accurately the real payload potential. Broad-brush mission studies done using theoretically derived values of the specific mass of the thruster system, thruster and power conditioning efficiency, and solar panel characteristics, indicate an excellent payload potential for the assumed range of parameter values. The next step in proving system feasibility is the detailed design of a spacecraft using hardware technology consistent with the selected mission study. Compromises in the system capability due to mission constraints such as launch windows and arrival conditions and due to hardware constraints such as temperature control, power sharing or configuration must be assessed to obtain parameters sufficiently realistic to proceed with program decisions. This section deals with the design compromises characteristic of a solar-electric spacecraft to be used as a probe. Specific references are to a Jupiter flyby mission study completed at JPL.[5]

The design of a solar-electric spacecraft is carried out in two steps. The first step is a broad mission sweep which isolates the general launch era and the approximate spacecraft electric propulsion parameters. The computer program selects the launch energy, total spacecraft mass, gross power, engine I_s, fuel loading, and thrust profile to maximize the spacecraft gross payload. The parameters that most heavily influence the allocation of weight between the propulsion system and

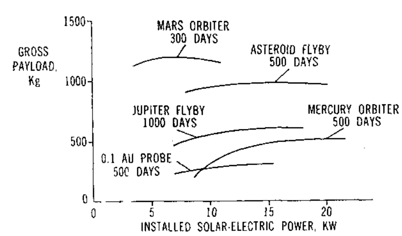

Fig. 8 Variation of payload with power at 1 a.u. for Titan III C launched solar-powered electric propulsion missions.

the payload are the propulsion system specific mass ($\alpha = \alpha_w + \alpha_{ts}$) and efficiency η. These two parameters relate the physical mass of the propulsion system to the optimum spacecraft acceleration. Since α is only an approximate scaling factor, spacecraft design details and weight requirements must be worked out to verify the assumed value. It has been found that the values of an optimum set of parameters are not particularly sensitive to α; changes in α on the order of 20% are not significant in the optimum choice of I_s or system power P_e as long as the total spacecraft mass m_o is held constant. Thus, changes in weight among the subsystems may be handled by bookkeeping rather than reoptimizing the trajectory; the resulting inaccuracy of representation of the optimum spacecraft is small relative to the approximations inherent in the trajectory program. In essence, this means that once the approximate set of propulsion parameters have been chosen (fuel weight, total power, I_s) and m_0 has been fixed (fixing injection velocity for a given launch date), a change in either solar array (α_w) or thruster system specific mass (α_{ts}) results in a pound-for-pound trade between payload and the respective system.

This loose coupling between the two phases of spacecraft and mission design is both an advantage and a disadvantage. Over-all parameters such as P_e, m_o, and preferred I_s may be chosen with some authority without resorting to detailed design study. However, the question of realistic payload is not settled until the mission and spacecraft peculiarities are resolved by the second step, actual or conceptual hardware configuration.

Rather than discussing a particular spacecraft design (which represents one set of solutions to the engineering problems), let us consider some of the difficulties in designing solar-electric spacecraft for planetary missions, using as a reference the historic Sun-Canopus, 3-axis-stabilized spacecraft with an Earth-pointing, high-gain antenna.

Problems Due to Long-Duration Propulsion

Thrust vector control, both in direction and magnitude, is required. The exact method of control is open to argument, but it seems that the most straightforward method is to preprogram the control in an onboard function-generator according to nominal preflight calculations.

The *mass change* during flight due to propellant expulsion is significant. Since attitude-control problems are minimized by minimizing center of gravity migration, propellant tank location must be carefully evaluated.

Attitude control for a spacecraft having large, flexible solar panels, potential migration of large masses, and the continuous force of the thrusters and the solar wind, is a significantly different problem from that for a ballistic spacecraft. Considering the solar and thruster forces as perturbations correctable via a standard mass-expulsion attitude-control system results in prohibitively large system weights. The alternatives are to use the solar or thruster forces as attitude control forces. The thruster forces are approximately two orders of magnitude larger than the available solar forces, thus it seems inevitable that the thrusters be used for attitude control during the powered phases. This implies that another system must be included for initial acquisition of the celestial reference and for control during the nonpropelled phases. It appears that application of existing technology is adequate for this task.

A navigation system for accurate detection of the spacecraft position and velocity, calculation of the required future control, and onboard storage of this new program are required. The spacecraft must have at least a Doppler transponder, and perhaps an accurate range-measuring device. In addition, the capability to calculate the new thrust program must be a part of the mission operations. This calculation is most logically done on ground-based computers since the calculations are complex and the data on which the calculations are based is collected on the ground. Since the corrections are calculated on the ground, they must be transmitted to the spacecraft much the same as the midcourse corrections of the Ranger and Mariner spacecraft. The period of active spacecraft navigation extends throughout the propulsion phase. Preliminary studies indicate that a limited number of stored program corrections will be required, perhaps as few as two, one at the start of the propulsion phase, and one near the end.

Problems Due to a Large Solar Array

Existing *launch vehicles* are diameter-limiting, the most common diameter being 10 ft. The solar-cell panels must be rolled or folded, and the shroud length must in many circumstances be increased. The existing shroud families, such as for the Titan, place an upper limit due to packaging of around 15–20 kw for fold-up arrays. The additional shroud weight decreases the available m_o in the ratio of $\sim$1 kg per 10 kg of shroud.

Dynamic response of the multihinged panels is cause for concern. Its prediction is an extremely complex analytical or testing problem, which requires extensive research to assure that there are no problems such as dynamic interaction with the attitude control system both during deployment and acquisition and during the propelled phase. Preliminary simplified analyses indicate that damping must be added to the panel support structure, increasing α_w.

Field-of-view requirements affect the spacecraft configuration. Sun orientation is not a problem since the panels are orthogonal to the sun line. Canopus sensors are affected by reflected light, and since accurate orientation is required for navigation, it is necessary to provide clearance around the Canopus line of sight. The thrust-vector program has its simplest form when referenced to the ecliptic plane; thus, one of the spacecraft reference axes should be the ecliptic. Canopus is not directly at the south ecliptic pole, so, as the spacecraft revolves about the sun with one axis oriented toward the sun, the Canopus line of sight cuts out a cone. These considerations indicate that a large solar panel would be most unwelcome in the area orthogonal to the ecliptic plane. The Earth and Jupiter, for example, are nearly coplanar, so the majority of the thrusting is done in the ecliptic. The thrust vector swings around the spacecraft in approximately the ecliptic plane indicating that solar panels cannot be located in this area either. Thus, with the Sun-Canopus orientation, there are three constraints which must be satisfied to locate the solar panels. They are thrust-vector clearance, Canopus-sensor clearance, and the requirement to remain nearly orthogonal to the sun line. One preliminary design which has been proposed places four panels $\pm 45°$ from the ecliptic plane. If the required panel area can be restricted by lowering the required power, it is possible to place a panel in the Canopus direction yet make it short enough to see over using a tower-like structure.

Auxiliary power is more economically derived from the solar array than from a separate supply for spacecraft designed for missions as far from the sun as Jupiter. A separate power conditioning unit is required, because the nature of the loads is distinctly different. The spacecraft subsystems require essentially constant power; thus as the spacecraft goes farther from the sun, more panel area is devoted to powering this load.

Problems Due to Trajectory

Because of *power variation* along the trajectory, the propulsion system must accommodate wide ranges of input power and voltage, and power-matching techniques (such as thruster throttling and multiple thrusters discussed earlier) must be employed. Because *temperature control* problems are severely aggravated when the spacecraft passes closer to the sun than the orbit of Venus, a potential area of high-payload trajectories (passages nearer the sun) is severely compromised.

Spacecraft design alternatives arise if the payload of an optimized solar electric spacecraft (one which has the full set of propulsion and trajectory parameters freely chosen to maximize payload) exceeds that required to satisfy the mission objectives. In this situation, constraints may be placed on the propulsion or trajectory parameters to make the mission design more acceptable; e.g., the flight time may be shortened, P_e may be lowered, or the I_s may be moved from it optimum value to improve reliability, simplify fabrication, and/or lower cost. If sufficient (25–30%) excess payload is available, it may become feasible to employ a single, fixed thrust direction, thereby reducing mechanical complexity and improving the configuration to the extent that the whole band in the ecliptic plane need no longer be reserved for thrust vector clearance. The bulk of the trajectory results to date indicate that there is a class of trajectories which will arrive at the target using a single nominal thrust direction. In most cases studied, this direction is orthogonal to the sun line, thrusting in the direction of motion for outer planet flyby, and counter to it for inner planet flyby. If excess payload still exists after fixing the thrust direction, attention may then be directed towards changes to reduce the power requirement, since this will both reduce the cost of the spacecraft and improve the configuration.

Since these changes to the spacecraft lower its performance, the optimium combination of chemical and electrical stage will tend to emphasize the chemical stages more. Therefore, low-power, fixed-thrust-angle electric spacecraft trajectories are characterized by higher injection energies and smaller total injected mass. For example, the Atlas-Centaur has sufficient capability to perform a Jupiter mission with a fully optimized solar electric spacecraft, yet the Atlas-Centaur/Burner II or the Titan III-C has enough excess capability to fly a much simplified spacecraft. It is possible that the minimum overall cost, minimum risk mission will be one which uses the larger chemical boost vehicle and the simplified electric spacecraft.

Unique Solar Electric Design Characteristics

The flexibility of a mission design using solar electric spacecraft occurs and may be used in two general design areas. The first area is over-all mission design. A large number of combinations of potential electric propulsion parameters will meet a given set of mission objectives. Although this inherent flexibility makes the mission design difficult, it also assures that coming close to some "perfect" set of parameters is sufficient. This means that there is some room for error in the hardware performance predictions, and a design margin exists which will be of great use in assuring program success.

The second area is that of application of a specific spacecraft design to a given mission. Once the spacecraft parameters have been chosen and metal has been cut, many system compromises still can be made without endangering mission success. For example, one can compensate for overweight systems by redesigning the trajectory.

Conclusion

Solar-powered electric propulsion presents space program planners with a challenge. As this is a new and radically different propulsion system, a number of new planning and implementation problems are encountered. The capabilities of such systems often cannot be presented in a way familiar to the user. Many solar-electric missions have a ballistic counterpart involving a larger booster or additional stages. Therefore, economic comparisons on both a total program and individual project basis are indicated. Such tradeoffs are limited by the lack of cost data for electric propulsion systems based on experience. Also, distinction between the functions of the launch vehicle and spacecraft is not as clear as in missions for ballistic systems. This problem, along with the large number of subsystems making up the spacecraft itself, may require some revision of existing policies for program management and organization.

Electric propulsion is significantly more than an "add-on" system to a ballistic spacecraft. Intimate interactions between the propulsion system and other spacecraft functions create intricate design problems. Yet it appears that these problems having engineering solutions. System analyses and spacecraft design considerations have shown that any necessary engineering compromises do not have unduly severe effects on system performance. Detailed design studies indicate that practical spacecraft hardware can be built. Both flight and ground programs are directed toward a state of technology consistent with mission capability in the 1975–80 time period.

Mission analyses indicate potential advantages of a solar-electric spacecraft due to both multimission applicability and extending the utility of existing launch vehicles. Solar-electric propulsion also provides the growth potential inherent in an infant technology and the ability to carry out otherwise economically impermissible missions such as Mercury orbiter, extra ecliptic, or asteroid rendezvous probes. The risk due to program commitments in this new area of technology may be offset by advantages in flexibility, performance, and cost.

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Development of Meteoroid Protection for Extravehicular-Activity Space Suits

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The space suit used during extravehicular activities requires meteoroid protection to prevent perforation of the pressure bladder. The development of protective coverlayers for the Gemini and lunar-surface extravehicular activity (EVA) suits is described. The meteoroid mass, which must be absorbed in the coverlayer to satisfy the mission exposure time and the reliability requirements, was simulated by a hypervelocity glass projectile of equal penetrating energy. Textiles or "soft goods" that were compatible with suit fabrication techniques were impacted with the laboratory projectile to determine response to impact and resistance to penetration. An effective projectile breakup was achieved with nylon cloth; therefore, the bumper concept that was developed previously for metallic sheets was applied to the suit protection. Two configurations were developed, one of which used nylon felt and the other used neoprene as the primary energy absorber. The configurations provide an acceptable probability of no bladder perforations during a 24-hr exposure with an areal density less than 0.10 g/cm².

Introduction

EXTRAVEHICULAR activity (EVA) exposes an astronaut's space suit to the meteoroid environment for short periods. To prevent meteoroid penetration of the pressure bladder and subsequent decompression of the space suit, protection must be provided to absorb meteoroid impacts. Several techniques and methods for preventing puncture of the bladder were considered. Parameters such as weight, astronaut mobility in a pressurized suit, thermal protection, ease of fabrication, astronaut comfort, and schedules eliminated metallics and required that the protective coverlayer be compatible with the intravehicular suit. The short leadtime that was available for design of a Gemini protective coverlayer and the lack of information concerning hypervelocity impact into textile materials (soft goods) precluded extensive analytical studies to determine optimum materials and layup. Therefore, an engineering-design approach was taken, and the experimental impact tests were confined to textiles that were used or approved for fabrication of intravehicular space suits. The acceptable materials from the preliminary tests then were evaluated ex-

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