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REACTOR REQUIREMENTS FOR FIRST GENERATION THERMIONIC
NUCLEAR ELECTRIC PROPULSION

by

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REACTOR REQUIREMENTS FOR FIRST GENERATION THERMIONIC
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by

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ABSTRACT

Data are presented to show the interaction between reactor and mission requirements for first generation unmanned thermionic nuclear electric propulsion applications. A Titan III F/Centaur launch vehicle is assumed. Reactor lifetime requirements are given in terms of payload capability, and performance for different power levels is presented as a function of mission constraints, with major emphasis on 50 and 100 kWe power plants. Significant results are that thrust times or reactor full power operating times generally can be reduced by a factor of two or three from the optimum while maintaining useful payloads, and that a single power plant is applicable to a broad range of nuclear electric solar system missions with a power level that is compatible with projected space auxiliary power requirements.

INTRODUCTION

The flight readiness of a nuclear electric propulsion system will be determined, to a large extent, by when reactor lifetimes can be expected to meet or exceed the operational thrust time requirements of the proposed missions. In turn, the demonstration of these lifetimes depends on when a power level is selected and construction of a prototype reactor begun.

The demonstration of the performance potential of nuclear electric systems has often involved a flight mode and assumptions which place rigid requirements on reactor development for near-term application. In particular, reactor lifetime has been treated as an unconstrained variable in mission optimization studies which, along with individually optimized power levels, results in enhanced performance estimates. In addition, power requirements consistent with a spiral escape mode or a nonjudicious selection of a launch vehicle have frequently conveyed the impression that prime propulsion and auxiliary power needs were quite dissimilar.⁽¹⁾ Often, these approaches involve payloads in excess of those projected as being necessary during the contemplated flight periods. The evaluation of an advanced system's capability should be made on the basis of its ability to perform a range of useful missions within technology constraints and should avoid arbitrary mission models which emphasize single points of view.

A recent study has described optimum performance and useful launch vehicle-power plant combinations for a wide range of nuclear electric solar system missions.⁽²⁾ The document describes the numerous tradeoffs possible with an electric propulsion system and allows an assessment of optimum performance with major variable constraints--namely, chemical hyperbolic injection velocity, initial acceleration level, and the thrust time of the electric vehicle. Based on launch vehicle availability and estimates of performance requirements, the selection of a Titan III F/Centaur was indicated as a representative launch configuration for unmanned planetary missions.

This paper addresses itself to the question of establishing a power level and reactor lifetime requirements for first generation nuclear electric propulsion application while considering major mission-related technology constraints in conjunction with projected auxiliary power requirements.

ASSUMPTIONS

The analysis presented here is for a two-body planar problem with circular planetary orbits. The planets considered and their semi-major axes (assumed radial distances from the sun) are given in Table 1. The Earth escape mode is chemical to some hyperbolic velocity which is treated as a free variable.

TABLE 1
PLANETARY DISTANCES

Planet	Semi-major Axis (AU)
Jupiter	5.20
Saturn	9.54
Uranus	19.18
Neptune	30.06
Pluto	39.52

Two basic types of missions were considered. The flyby intercepts the target planet in the specified time with no restriction on its terminal velocity. The capture mission assumes a zero-relative velocity with respect to the target planet at encounter and so is equivalent to a weak planetary orbit when n-body effects are taken into account. The trajectory optimization, using a variational program, involved the maximization of the vehicle's final mass with a specified power level and flight time. The central angle of travel is a free variable which provides for the best launch time in any year. Power conditioner efficiency was assumed to be 0.91. Thrustor efficiency curves are discussed in detail later.

Based on a previous analysis mentioned, two power levels were given prime emphasis.⁽²⁾ A 100 kWe plant appeared nearly optimum for the

Titan III F/Centaur launch vehicle and so would come close to duplicating the maximum performance expected, while the 50 kWe reactor is near the minimum usable size for prime propulsion on this vehicle. In addition, a 50 to 100 kWe thermionic reactor is considered desirable in order to satisfy auxiliary power requirements for a wide range of applications.⁽³⁾

Nuclear thermionic propulsion system design points are presented in Fig. 1.⁽⁴⁾ Power levels are given at the power conditioner input with 5.5 kg/kWe assumed for power conditioning, thrusters, spares, and associated structure. The U^{235} fueled system masses have been approximated as a linear function over the region of interest. The mass differences indicated for the two fuels stem primarily from criticality considerations which require the use of special design techniques at low powers with U^{235} . At higher powers the U^{235} propulsion system masses are close to those indicated by the U^{233} relationship. These design points are intended to provide reference values for the estimation of payload capability. Variations in such numbers can be reflected in performance evaluation on a one-to-one basis.

At this time, estimates of payload requirements are somewhat arbitrary and controversial stemming from an absence of specific mission commitments.^(5,6) However, our past planetary program can provide a guide for the establishment of minimum requirements for future missions. The Mariner Mars and Venus spacecraft weighed between 200 and 400 kg. Lunar Orbiter was about 400 kg, and the Surveyor spacecraft on the Moon has a mass under 300 kg of which there is a 50 kg package of science. An extrapolation on these spacecraft masses for general solar system exploration, reflecting a capability to the farthest planets, seems appropriate. It was assumed that 1000 and 500 kg are acceptable payloads for any capture or flyby mission, respectively, considered here, a large fraction of which would probably be science. Flight times were arbitrarily limited to $6\frac{1}{2}$ years at Pluto without considering any planetary swing-by assists.

The term payload is defined as the vehicle's final mass at the target minus the entire propulsion system.

RESULTS

All performance data is presented in terms of the total final mass of the spacecraft at the target including propulsion system, structure, and scientific payload. The results are therefore independent of propulsion system mass assumptions which have the usual uncertainties associated with the development of an advanced system. In this form, limits on propulsion system mass can be determined for a given payload and flight time. Or, the payload capability of a specified propulsion system can be found. The lines labeled U^{235} and U^{233} , which are shown in later figures, are intended as reference points for the mass-power relationship given in Fig. 1. All data assumes the use of a Titan III F/Centaur launch vehicle.

OPTIMUM PERFORMANCE

Initial effort was directed toward the computation of optimum performance to the planets for 50 and 100 kWe power plants. Thrust times were not constrained. Figures 2 and 3 summarize these results. Performance for both the U^{235} and U^{233} systems appears adequate at this point, considering the assumed payload requirements. Significant is the fact that the 50 kWe system holds up well over all the missions. Comparison of the 100 kWe data with optimum power results is very close, as expected.⁽²⁾ This insensitivity of payload to power variations around the optimum is a significant characteristic of nuclear electric systems (to be discussed later).

THRUST TIMES

Approximate thrust times for these missions are summarized in Fig. 4. The actual times are fairly insensitive to the target planet and power level. The optimum thrust times of 15,000 to 30,000 hours corresponding to flight times in excess of 1000 days should not be construed as being essential for such missions. This data represents optimum fixed power

performance, but excluding reactor technology considerations, and can be misleading.

The development goal of a 10,000-hour thermionic fuel element lifetime has generally been assumed analogous to a statement of minimum reactor lifetime. This number has been questioned as being unrealistic in light of the thrust times just discussed, or the results of similar optimization studies.

The results show, however, Figs. 5 and 6, that the imposition on a 10,000-hour thrust constraint produces a payload capability across the solar system still within the projected requirements and on some missions is nearly indistinguishable from the unlimited thrust case. The effect of major thrust time reductions on the 50 kWe system now becomes prominent and indicate the use of U^{233} on some missions. Nevertheless, the 10,000-hour constraint is generally a mild perturbation on performance capability--certainly for the major planets which presently hold the greatest interest for future exploration. In the case of Jupiter, there is practically no change in performance which stems, primarily, from the short optimum thrust times associated with this body. Similar features are noted for Saturn. The relative insensitivity of performance to thrust time is realized by the significant tradeoff capability of the system parameters, and indications were that these could be pushed still farther.

The Jupiter flyby in Fig. 7 withstands thrust time reductions under 2500 hours retaining considerable performance. The Jupiter capture in Fig. 8 for a 100 kWe system still shows comparatively good performance around 5000 hours. The 50 kWe system on this same mission, Fig. 9, causes performance to fall off quickly and now necessitates longer flight times to attain acceptable payloads with reduced thrust times. The use of U^{235} fuel here further increases flight times in order to attain acceptable performance. A Saturn flyby in Fig. 10 holds up very well under thrust time reductions to 5000 hours. The Saturn capture, Fig. 11, can be performed with 10,000 hours of thrust time. In addition, a sizable decrease in flight time can be realized if a U^{233} system is assumed. These data show the acceptability of bringing mission thrust requirements into line with projected technology capabilities.

LAUNCH WINDOWS

The effect of thrust time reductions on launch window availability is an important consideration. Typically, low thrust vehicles have large comparatively flat windows. However, the data presented here were generated for an optimum central angle of travel, and so it is necessary to view performance across the window and with reduced thrust times. As shown in Fig. 12, an 800-day Jupiter capture generates a window covering a span of two-thirds of any year. A thrust time reduction to 5000 hours would provide a four-month window with a U^{233} system but would require a longer flight time to attain similar performance using U^{235} . For missions to planets farther out in the solar system, the launch window constraints become more relaxed. For instance, a Uranus flyby in Fig. 13 produces a nearly flat window illustrating the fact that the concept of a launch window loses significance for flights to distant planets with electric propulsion.

REACTOR POWER LEVELS

The final mass of a vehicle as a function of reactor power on a 600-day Jupiter capture mission is given in Fig. 14. On the same plot are shown propulsion system mass curves for the U^{233} and U^{235} fueled reactors obtained from the data in Fig. 1. The difference between these two functions is the payload. When the slope of the propulsion system mass function equals that of the final mass curve, the optimum power is defined. Most significant is the fact that these slopes are practically the same over a wide range of powers about the optimum. As the power level is increased beyond the optimum, the increased propulsion system mass is nearly offset by the reduced propellant mass resulting from higher exhaust velocities. Therefore, there is no significant performance disadvantage in going to either side of the optimum power. An increase in the slope of the propulsion system mass function would indicate lower powers for the same mission, while a decrease in this slope implies a higher optimum point. Designs in the power region where an all U^{235} system is no longer criticality limited should cause the assumed U^{235} power-mass function to bend over and more closely approach the U^{233} relationship. This would suggest slightly higher powers for U^{235} , in this transition region, than for a U^{233} system.

As Fig. 15 shows, the change in flight time to 800 days is insignificant in determining an "optimum" power level. Making the mission easier just shifts the mass curve up without changing its shape. This is the prime mechanism for increasing payloads. Similar data is shown in Figs. 16 and 17 for 1200- and 1600-day Saturn captures. Also of great importance, as illustrated in Figs. 14 through 17, is the fact that thrust time reductions to 5000 hours do not significantly change the shape of the final mass curve. This implies that there is not much to be gained by increasing power levels for reduced thrust time missions.

A 1600-day Uranus flyby in Fig. 18 generates the same picture. As indicated in Fig. 19, a composite of 10,000-hour performance from Figs. 14 to 18, not only is the optimum power level insensitive to the mission selected, but a wide range of powers about the optimum generate essentially the same performance. In fact, a 100 kWe system's performance is nearly indistinguishable from the "optimum system" around 140 kWe.

LAUNCH VEHICLE

If the launch vehicle were changed from a Titan III F/Centaur to a Titan III D/Centaur, Fig. 20, a significant shift in the optimum power point is noted, Fig. 21. As expected, the smaller vehicle calls for a smaller power plant, and the latitude in the selection of the point decreases. Smaller vehicles have less tradeoff capability than do large ones and consequently are more sensitive to system variations.

An approximate final mass curve could be constructed for any launch vehicle having a similar payload-velocity excess relationship by estimating the injection velocity at a given power level and scaling the final mass curves and the power level by the ratio of the launch vehicle masses at that point. This, in effect, produces the same acceleration profile for both vehicles and, therefore, duplicates the mission. For instance, a 260/SIVB/Centaur launch vehicle, having twice the escape mass of the Titan III F/Centaur, would generate final mass curves indicating optimum power levels approximately twice those of the smaller vehicle. So, while the mission or mission time does not appreciably affect the power level selection, the launch vehicle does.

THRUSTOR EFFICIENCY

Thrustor performance is of major importance in electric propulsion mission analysis and particularly so for early reactor application. Figure 22 shows the electrical and mass efficiencies that were assumed in the analysis. Common to all plasma thrustors, at the present, is a rapid drop-off of efficiency at low exhaust velocities. It is informative to project on payload improvement potentially available from these devices or to assess performance degradation using the present technology that has been assumed. Figure 23 illustrates the payload gain on an 800-day Jupiter capture mission when the thrustor electrical and mass efficiencies are assumed to be 0.9 and 0.95, respectively, over the entire exhaust velocity range. The most dramatic changes occur at the short thrust times where it is necessary to lower the exhaust velocity in order to increase the vehicle acceleration level. At longer thrust times and out to the optimum time, the payload differences are less significant. The optimization tends to select operating points away from the inefficient exhaust velocity region. Only a requirement for increased acceleration, like with a limited thrust time or a very short mission time, will cause the exhaust velocity to be driven into this inefficient region. As shown in Fig. 24, the improved efficiency curves for a fixed thrust time generate a higher operating exhaust velocity than do the nominal efficiencies. In effect, the higher efficiencies produce the equivalent of a larger power plant for a fixed thrust time which then converts this "excess" power into a higher exhaust velocity. There is considerable activity being directed toward the efficiency improvement of ion thrustors at low exhaust velocities. Such technology advances will further enhance the role that first generation reactor systems can play.

CONCLUSIONS

The establishment of payload requirements is, perhaps, the prime factor in selecting a launch vehicle; and the launch vehicle, in turn, essentially dictates a power level. For a given launch vehicle, the optimum power is fairly insensitive to changes in mission and flight time. Furthermore, reactor lifetimes (thrust times) do not significantly influence power selection. This relative insensitivity of power to large variations in major mission and reactor parameters indicates the use of a single power plant for a wide range of application.

A nominal 100 kWe thermionic reactor is almost a no-compromise plant on the Titan III F/Centaur launch vehicle. Many interesting missions could be performed with somewhat less than this power.

The prime mechanism for improving payload with a given propulsion system is the increase of the flight time to the target. Usable propulsion system masses for a given launch vehicle are largely determined by what are considered acceptable flight times.

Results indicate that a U^{233} fueled reactor would be essential for most missions with thrust times limited to near 5000 hours. Also, significant decreases in flight time could be realized with such a high performance system.

Thrust times of 20,000 to 30,000 hours are not necessary in order to perform useful missions. Most Jupiter missions can be performed with slightly more than 5000 hours of thrust, and Saturn captures and all planetary flybys with 10,000 hours. Reactors lifetime requirements for first generation thermionic nuclear electric propulsion should be consistent with these results.

Initial development of a nuclear thermionic auxiliary power plant will be congruous with the demonstrated requirements for electric propulsion, adding to the versatility of this particular space power source.

This paper, in the main, has dealt with nuclear thermionics for prime propulsion. Application of the power for experimental purposes would necessitate a redefinition of the vehicle's payload to include a percentage of the systems power plant. Similarly, the additional reactor operating time would have to be superimposed on the propulsion requirements. The operational mode of the reactor during coast periods could also contribute to increased lifetime requirements.

Fig. 1. Thermionic nuclear electric propulsion
 system design points

Fig. 2. 100 kWe system--optimum performance

Fig. 3. 50 kWe system--optimum performance

Fig. 4. Typical unlimited thrust times

Fig. 5. 100 kWe system--10,000 hr thrust

Fig. 6. 50 kWe system--10,000 hr thrust

Fig. 7. Jupiter flyby.
100 kWe system--limited thrust times

Fig. 8. Jupiter capture.
100 kWe system--limited thrust times

Fig. 9. Jupiter capture.
50 kWe system--limited thrust times

Fig. 10. Saturn flyby.
100 kWe system--limited thrust times

Fig. 11. Saturn capture.
100 kWe system--limited thrust times

Fig. 12. 800-day Jupiter capture.
100 kWe system. Launch window

Fig. 13. 1600-day Uranus flyby.
100 kWe system. Launch window

Fig. 14. 600-day Jupiter capture.
Optimum power selection

Fig. 15. 800-day Jupiter capture.
Optimum power selection

Fig. 16. 1200-day Saturn capture.
Optimum power selection

Fig. 17. 1600-day Saturn capture.
Optimum power selection

Fig. 18. 1600-day Uranus flyby.
Optimum power selection

Fig. 19. 10,000-hr thrust time performance.
Optimum power selection

Fig. 20. Launch vehicles

Fig. 21. 1600-day Uranus flyby. Optimum power selection for different launch vehicles

Fig. 22. Nominal thruster efficiencies

Fig. 23. 800-day Jupiter capture. Final mass increase for potential thruster improvement

Fig. 24. 800-day Jupiter capture. Exhaust velocity increase for potential thruster improvement

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