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**COST-EFFECTIVE TECHNOLOGY ADVANCEMENT
DIRECTIONS FOR ELECTRIC PROPULSION
TRANSPORTATION SYSTEMS IN EARTH-
ORBITAL MISSIONS**

John D. Regetz, Jr.
Lewis Research Center
Cleveland, Ohio

and

C. H. Terwilliger, Jr.
Boeing Aerospace Company
Seattle, Washington

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COST-EFFECTIVE TECHNOLOGY ADVANCEMENT DIRECTIONS FOR ELECTRIC
PROPULSION TRANSPORTATION SYSTEMS IN EARTH-ORBITAL MISSIONS

by John D. Regetz, Jr.
NASA Lewis Research Center
Cleveland, Ohio 44135

and

C. H. Terwilliger, Jr.
Boeing Aerospace Co.
Seattle, Washington 98124

ABSTRACT

This paper presents the results of a study to determine the directions that electric propulsion technology should take to meet the primary propulsion requirements for earth-orbital missions of the next three decades in the most cost-effective manner. Discussed are the mission set requirements, state-of-the-art electric propulsion technology and the baseline system characterized by it, adequacy of the baseline system to meet the mission set requirements, cost-optimum electric propulsion system characteristics for the mission set, and sensitivities of mission costs and design points to system-level electric propulsion parameters. It is found that the efficiency-specific impulse characteristic generally has a more significant impact on overall costs than specific masses or costs of propulsion and power systems.

INTRODUCTION

Historically, this nation's space program has been the cutting edge for new technology. The goals and objectives of mission planners are sufficiently ambitious as to require continual progress in the development of scientific instruments, spacecraft subsystems, and space transportation vehicles. As a result, the National Aeronautics and Space Administration (NASA) must continually reassess the direction of its research and development efforts to ensure that the requisite technologies will be in-place to support its goals and missions.

It is particularly appropriate that technology needs in the field of electric propulsion be re-examined at this time for at least two important reasons. First, past and current programs have been aimed at the perfection of the 8 and 30-cm mercury ion bombardment thruster systems into useful items of mission hardware. With work on flight test hardware for the 8-cm system now in progress, and with the commitment of the 30-cm system to a major flight program imminent, these goals are nearing fruition. Second, the decade of the seventies has seen the development of a powerful new means of access to near-Earth space, the Shuttle-based space transportation system (STS). With the approach of the STS era, new missions have

been suggested to make use of this versatile new tool and to benefit mankind, missions which are bolder, more aggressive, and more numerous than have heretofore been attempted. In addition to the STS, many of these new missions will require advances in other supporting technologies, such as electric propulsion.

Recognizing these circumstances, NASA's Lewis Research Center in early 1978 contracted for a study, the results of which are given in this paper. (Contract NAS3-21346, Boeing Aerospace Co.) The objective of this study was to identify those areas in the field of electric propulsion technology where advances in the state-of-the-art are required to allow development of propulsion systems which will meet the requirements and constraints of the probable near-Earth space mission set for approximately the next three decades, and to establish the general nature of these advances as guidelines for ensuing technology efforts.

Certain constraints were imposed to guide the conduct of the study. These groundrules helped ensure that the study results would be of maximum usefulness to the NASA, and would be complementary to other current investigations.

(1) This study was restricted to missions in the "near-Earth" region only. Any consideration of deep space exploration missions was avoided, as their requirements are being addressed by others.

(2) This study was restricted to consideration of primary propulsion applications only.

(3) This study was originally restricted to consideration of ion bombardment electric propulsion systems only. As the study progressed, the effort was directed toward a parametric examination of system impacts and sensitivities, and this guideline became of less importance. In the end, the final conclusions are believed to be valid for any type of electric propulsion system.

(4) This study considered that any propulsion-dedicated power sources were photovoltaic only. This constraint forced a consideration of the effects (time and cost) of solar array degradation, and introduced additional complications (trajectory optimization and steering penalties) into the calculations of system performance.

(5) The study was directed to make maximum use of past results and of the data and experience base that exists. In particular, an extensive literature search was performed.

APPROACH

The approach to achieving the study objectives was as follows. First, a set of missions was selected to provide a basis for the assessment of electric propulsion technology. A review of available related literature was conducted in support of this task. Next, comprehensive analyses of each of the selected missions was performed to define the requirements and constraints of each payload and to determine the characteristics of each of the several types of trajectories. This activity established a data base to be used for the remaining study efforts. Next, a simplified analytical model of the payload, propulsion system and mission was developed

to evaluate the cost and performance of electric propulsion systems with particular characteristics across the set of missions. The model was then exercised to determine the benefits of various changes in the model elements that characterize the electric propulsion technology. Studies were also conducted using this model to establish the sensitivity of the outputs (design parameters, costs) to the input assumptions.

MISSION SET SELECTION

A set of 30 near-Earth missions was selected to provide a framework within which to conduct the cost-effectiveness studies reported herein. This baseline set was selected after an extensive search of current literature sources describing potential space missions over the coming three decades, DOD and NASA mission models, and the like. Figure 1 gives the complete listing of the baseline mission set, a possible time frame, and shows the division into groups as well as the missions felt to be representative of each group. These representative missions were selected for the sensitivity analyses described below. The purpose of the grouping was to combine those missions having similar propulsion system requirements which therefore could potentially be satisfied by a single set of system design parameter values. The primary bases for the grouping were payload mass and density. These determine the structural flexibility characteristics of the spacecraft and through this, the requirements imposed upon the nature of the primary propulsion application. For instance, low-mass, high density payloads can be treated as rigid bodies and the propulsive thrust can be applied at a single central point. At the other end of the scale, the high-mass, low density payloads will exhibit high flexibility, and because of the large dimensions involved, will necessitate distribution of the propulsive thrust over the entire payload structure to avoid any deleterious flexibility effects. Between these extremes lie missions encompassing various combinations of mass and density, in which either the propulsion or power source (or both) cannot be centralized but need not be fully distributed and may be manifested in a number of judiciously placed modules.

The grouping characteristics were as follows:

- Group 1 - Low mass, high density
- Group 2 - Low mass, moderate density
- Group 3 - Moderate mass, low density
- Group 4 - Moderate mass, high density
- Group 5 - High mass, low density

In general, the higher numbered groups tend to contain more advanced missions having more stringent payload requirements, and with launch dates farther in the future.

For each of the 30 missions in the baseline set, mission requirements and characteristics were determined, providing a data base for the economic modeling and analyses to follow. Trajectory requirements were established from trajectory optimization studies (conducted as part of the overall contract effort) which covered all the various types of trajectories and environmental conditions involved in the mission set. Parameters such as mission velocity increment (ΔV), and performance penalties due to radiation degrada-

tion, solar occultation, and delays in restarting thrusters were determined in this phase.

SYSTEM-LEVEL COST/PERFORMANCE MODELING

To provide a mechanism with which to perform economic analyses, an analytical cost/performance model of the mission/spacecraft/propulsion system was developed. The emphasis in this paper being on results, only a functional description of the model will be given here. More detail on the model is provided in the Contract Final Report (Ref. 1). The model is a "simplified" one in the sense that only "system-level" characteristics were included in the modeling. Figure 2 provides a functional block diagram of the model, showing the various "systems" into which it is divided. The "system-level" parameters which characterize each of these blocks are also shown on this figure. The interactions of each of the blocks with the Electric Propulsion System (EPS) were included in the model, but in general not those between individual parameters within a specific block. The EPS "scar" included in the payload parameters refers to any mass or cost increase to the payload as a result of the use of an electric propulsion system with it. Table 1 gives the baseline values of the various system-level parameters and the range of interest over which the parameters were varied in the course of the study. The baseline parameter values correspond to the current state-of-the-art of the various technologies represented, and the parameter sensitivities described later in this paper were determined about these baseline values. Note that overall efficiency of the EPS is modeled as a function of specific impulse. The function shown is that which is characteristic of the current state-of-the-art of Mercury Ion Bombardment thrusters, but provision was made to vary the constants as shown.

Two additional features were included in the model. The first of these is a provision for "trip-time" costs. These costs comprise two factors. One of these is the cost of operating the various tracking and other ground systems during the (often quite lengthy) propulsion time. The second represents the cost of money. This parameter accounts for the fact that the payload sponsor's investment is "frozen" for the transfer period.

The second feature is a provision for variable cost functions associated with certain hardware, such as the solar array, the earth launch system, and EPS production and propellant costs. The philosophy here is that technologies will progress in such a way as to decrease the per unit cost of these items for large spacecraft. Thus, for example, in the case of solar arrays, the baseline cost of \$350/watt was assumed to decrease to the vicinity of \$1/watt for megawatt sizes. For the other parameters, cost coefficients were varied on the basis of system mass.

In the following, "delivery cost" is defined as the total of all costs incurred in the delivery of the payload from the earth to its final orbit (excluding the cost of the payload itself), divided by the payload mass.

SYSTEM DESIGN POINTS

In the course of the study, four different design points were identified, each representing major differences in the philosophical approach to designing the electric propulsion system for a given mission or mission set. The design points are:

(1) The state-of-the-art system - provides an assessment of the capabilities of the current technology, and serves as a point of departure for the remaining studies.

(2) The cost-optimum system - mission cost is judged to be of paramount importance and the size and operating conditions of the system are adjusted to minimize this quantity.

(3) The minimum-power system - minimization of the size/cost of the power source is determined to be more critical than the mission cost here, and the system design is adjusted accordingly - specifically the thrusters are utilized to the limit of their lifetime.

(4) The minimum-time system - in this case, mission time is critical, allowing a sacrifice of cost and power level. (Since true minimum time requires an infinite power source, an approximation to it is shown on the graph.) Such a case might come about through payload reliability considerations, for example.

The relationship between these design points is illustrated in Fig. 3, where the trip time and delivery costs are plotted as functions of source power. The particular plot shown is for the group 2 representative mission, but similar plots were obtained for each mission in the set.

Of these design points, the first two will be of greatest interest to mission planners and the emphasis in the following discussion will be on these.

Baseline System Design Point

The baseline, or state-of-the-art (SOA), electric propulsion system is characterized by the nominal parameter values shown in Table 1. It is representative of a system assembled from eight 30-cm ion bombardment Mercury thrusters, a 25-kW solar array, and associated structure and electronics. The performance of this system for each mission of the mission set was examined. Figure 4 presents the payload specific delivery cost and also the required trip time using the SOA system, for a range of payload masses which encompasses the mission set. In this figure, all model parameters were constrained to their nominal SOA values. Also called out on this figure, as well as on subsequent figures, are the payload masses corresponding to the representative mission for each group. It can be seen that the specific delivery cost decreases, as would be expected, with increasing payload up to about 10 000 kg, because the size and hence the cost of the EPS is fixed. For payloads larger than this, however, the specific cost increases again (dashed curve on Fig. 4). This occurs because for these higher payloads, thrusting times increase greatly, and the "trip-time" cost penalties begin to predominate. These higher thrusting times also exceed the thruster lifetime limit for payloads above 7000 kg. This technical limitation of the

SOA EPS is the primary one which prohibits its application to these heavier missions. One way around this limitation is to increase the system size (add more solar array and larger engine systems) and this approach is equivalent to adopting one of the other design philosophies discussed below. A second way around this lifetime limitation, however, is to postulate a system wherein sufficient spare thruster units are provided so that, with suitable duty cycle management, the utilization time of each individual thruster just equals its expected lifetime. The effect of this "sparing" philosophy was studied by altering the value of EPS specific mass (21 kg/kW in Table 1), effectively increasing the electric propulsion system mass. The result is given by the solid curve in Fig. 4. The increased EPS mass increases the EPS component of delivery cost and also slightly lengthens the required trip time, increasing that component of cost also. It can be seen that although this method permits all missions in the set to be performed by the SOA system, it results in a slight increase in delivery cost. It therefore is obvious that, for the more ambitious missions, systems larger than the 25 kW baseline, or having longer lifetimes (or both) will be required.

Minimum Source Power and Minimum Time Design Points

As previously indicated, these two design philosophies are of less interest to mission planners and will only be briefly discussed here.

Minimum source power design point. - Because required thrusting time is inversely proportional to source power, the minimum power condition is realized when the engine system lifetime is just equal to the required (average) thruster utilization time for that particular mission. For each mission, there is a value of thrust system specific impulse which minimizes the power under this condition. The consideration of the minimum power design condition in this study was directed toward uncovering any shifts in the value of this optimum I_{sp} across the mission set. It was found that, while there is a direct relationship between minimum power level and payload mass, the optimum I_{sp} levels all fell in a narrow band centered about the current technology development point (~3000 sec). Because the minimum power point is influenced by the system lifetime assumption, the minimum power points were determined for lifetimes ranging from 10 000 to 50 000 hours. Within this range, the optimum I_{sp} was found to be unaffected by lifetime.

Minimum time design point. - To obtain an absolute minimum trip time requires the application of infinite power (equivalent to reducing the payload mass to zero). Of greater interest, as well as realism, is the case where the trip time is constrained to be some relatively short, nonminimum preselected value. As before, under this condition there exists for each mission an optimum value of specific impulse which minimizes the required power. Again, as before, while the minimum power level required was a direct function of payload mass, the optimum specific impulse values center in a narrow band around the 3000 second value and are independent of the value of trip time to which the mission is constrained.

Cost-Optimum System Design Point

Design point determination. - For any given mission, if the specific impulse is fixed, there is an optimum size for the power source from the viewpoint of delivery costs. Below that optimum, the system is under-powered and the charges associated with the transportation time duration drive the delivery costs up. For higher powered systems, the point of diminishing returns has been reached regarding decreasing trip time, and the increased costs of larger solar arrays and engine systems cause the delivery costs to increase. Likewise, for a fixed power level, there is an optimum value of specific impulse which minimizes the delivery cost. For lower values of I_{sp} , larger propellant masses are required, which increases the earth-launch costs and decreases the initial acceleration that can be achieved, thus increasing the trip time and associated cost. For I_{sp} 's higher than the optimum, the thrust level that can be achieved from a fixed-power system decreases, which in turn increases the trip time duration and costs.

The minimum cost design point was found for each member of the mission set by performing an optimization on both power and specific impulse simultaneously, holding all other parameters at their nominal values. The sensitivity studies to be described subsequently are all "centered" about these design points. Figures 5(a) and (b) present these design points as functions of payload mass. As was noted for the previous philosophies, the power required is a direct function of the size of the payload, but the cost-optimum I_{sp} is nearly independent of this parameter, falling off slightly at the higher payload masses. This falling-off occurs because the larger payloads increase the trip-time cost penalty, and since earth-launch costs were assumed to decrease with larger masses, the I_{sp} decreases to obtain larger thrust levels at the sacrifice of propellant mass.

Figure 6(a) shows the trend toward decreased specific delivery costs with increasing payload size that occurs for cost-optimized systems. This is in sharp contrast to the cost trends for the baseline (SOA) system shown in Fig. 4(a). For the cost-optimum system, the increased hardware cost resulting from the generally larger systems is more than offset by the reduced penalties resulting from shorter mission times. Figure 6(b) shows that the average thruster operating time also increases as payloads become larger and for a number of missions exceed the lifetime assumed for current (SOA) technology. Therefore the development of longer-functioning components would be beneficial to the implementation of cost-optimum electric propulsion systems for the far-term missions.

Design point sensitivity. - Having established a cost-optimum design point for each mission using nominal values of the modeling parameters, it is of interest to define the changes in these design points that result from perturbing various of these parameters. These studies were performed only on the representative mission for each group (see Fig. 1). The parameters perturbed were EPS specific mass components, specific cost, operations cost and efficiency function, the solar array specific mass and specific cost, the launch system delivery cost to LEO, the cost of money, and the mission velocity increment. The range of variation of these parameters is given in Table 1. The results were presented as the locus of points in

power-specific impulse space as the selected parameter was varied over its specified range, and all others were held at their nominal values. Figure 7 shows the extremes of these loci for each parameter varied, for each of the representative missions. The design point extremes are seen to cluster in slightly different regions for each group. For example, group 4 is characterized by slightly higher specific impulses, whereas group 5 requires power levels an order of magnitude higher than the other groups. However, the significance of this figure lies in the fact that none of the parameters examined caused any large changes in the set of cost-optimized design points. The parameters producing the largest changes were the mission velocity increment and the efficiency function. The first of these is predominantly a function of the mission characteristics. The impact of the second is discussed in a separate section below.

Mission cost sensitivities. - The sensitivity of the total mission cost, as well as each of its components (excluding payload cost), to perturbations in the modeling parameters was calculated for each member of the mission set. This data allows an assessment of the potential benefit to be gained from any contemplated technology improvement undertaken. For this analysis, nominal values were used for all parameters except the one being examined. Cost-optimum values were used for system power levels and specific impulses. The values of the sensitivity coefficients thus obtained are presented in Table 2 for the representative mission for each group. It can be seen that the largest effect across the mission set is produced by changes in system efficiency and in mission velocity increment. Because the latter is predominantly a function of mission characteristics, system efficiency appears to be the most fruitful area for technology advancement.

Effect of power system sizing. - The previous results were all obtained under the assumption that sufficient propulsive capacity is installed to utilize all of the power coming from the energy source at the start of the vehicle lifetime (the so-called "beginning of life," or BOL, sizing). As the solar array, for earth-orbit missions, will quickly be degraded by radiation damage, an excess propulsive capability will be carried (as dead weight) for a significant portion of the time. The effect of installing only enough propulsive capability to utilize the solar array output expected at the end of the mission was examined. Figure 8 shows that this "End-of-Life" (EOL) strategy does decrease costs slightly across the mission set, but Fig. 9, which shows the change in design points caused by the EOL sizing as opposed to the BOL sizing, indicates that the strategy selected has little impact on the cost-optimum design points and therefore on the direction in which the technology should be advanced.

IMPACT OF EFFICIENCY FUNCTION VARIATIONS

Throughout the study, it was noted that the optimum specific impulse for the electric propulsion system never varied significantly from the vicinity of 3000 seconds - the nominal, state-of-the-art value. This was true for all design philosophies and despite the fact that system parameters were varied over fairly broad ranges. In the course of determining the sensitivity of the cost-optimum design points to variations in the input param-

ters, the efficiency - I_{sp} function was varied by both scaling (i.e., by multiplying by a constant), and by translating (adding a constant across the range). Figure 10 shows the effect of these two variations on the design point loci. It is seen here that scaling, which has a proportionately larger effect at higher I_{sp} 's, has little impact on the design points whereas translation, which alters efficiencies across the entire I_{sp} range, produced larger changes in the design points than any other parameter variation. It was concluded that the drop-off of the efficiency function in the 3000 sec region and below tends to hold the optimum I_{sp} up, and that, if efficiencies were increased for the lower I_{sp} 's, the optimum I_{sp} would decrease. This would occur because thrust levels and accelerations could then be increased, thereby decreasing trip times and their associated costs. To test this conclusion an efficiency function which was independent of specific impulse was assumed as shown in Fig. 11(a). Figure 11(b) shows that the cost-optimum specific impulses have decreased from about 3000 sec to the 500 to 1000 second range. The various plots of Fig. 11(b) represent the loci of design points as the (constant) efficiency level is varied from 0.25 to 1.0. Figure 12 illustrates the effect of an increase in efficiency at lower specific impulses on delivery cost. Presented here are the efficiency functions required to produce delivery costs which are independent of specific impulse. The functions shown correspond to the nominal cost and to 10 percent above and below nominal. Also shown for comparison is the SOA efficiency function. It is seen that an increase in efficiency at 2000 seconds from 45 to 65 percent will result in a 10 percent decrease in delivery cost. Although this figure is given for the group 1 representative mission, the results are similar for all members of the mission set, with potentially larger savings for the heavier missions (groups 4 and 5).

Finally, it must be emphasized that all design point and mission cost sensitivities as discussed above were obtained while using the SOA efficiency-specific impulse characteristic. Therefore any conclusions drawn concerning the relative impact of a given parameter on technology advancement directions cannot be considered necessarily valid in the event that this efficiency characteristic changes drastically as discussed earlier in this section.

CONCLUSIONS

From the foregoing results, the following can be concluded. The current state of the art in electric propulsion technology, as manifested in the baseline system, can be used to perform the group 1 and group 2 missions. The cost penalties associated with the use of SOA technology for these missions, although not shown in this report, are small and for some missions the baseline system is very close to being cost-optimum. For higher-mass payloads, however (such as the group 4 and group 5 missions), the baseline system (without redundancy) is inadequate because thruster lifetime is too short to accommodate the longer thrusting times required. Because the largest contributors to the delivery cost of these missions comprises those costs which are dependent upon trip time, advanced thruster systems which can reduce trip times will be required to minimize the costs. This implies that thruster systems larger than the 25 kW baseline, having higher thrust levels

and increased lifetimes, should be developed to meet the needs of these missions.

The 3000 second specific impulse range appears to be optimum for those thruster systems (such as the Mercury bombardment ion thrusters) which have an efficiency characteristic representative of the SOA characteristic used in this study. If, however, technology advancements in the direction of producing higher system efficiencies at lower specific impulses can be achieved, then a decrease in the operating value of specific impulse to the vicinity of 1000 seconds (depending upon the efficiency characteristic obtained) will be accompanied by a significant decrease in delivery costs. Improvement of efficiency appears to be potentially more fruitful than working to reduce specific masses or costs of propulsion and power systems.

REFERENCES

1. Terwilliger, C. H., Jr., et. al., "Study of Electric Propulsion for Near-Earth Space Missions," 1979, Contract NAS3-21346, Final Report.

TABLE I. - SYSTEM-LEVEL MODEL CHARACTERISTIC VALUES AND RANGES

System-level characteristic	Baseline value	Range of interest
E.P. system		
Specific impulse	3000 sec	1000 - 10 000
End-to-end efficiency	1.094 \pm	1.0 - 1.2
	$\left(1 + \frac{6.99 \cdot 10^6}{I_{SP}^2}\right)$	6.5 - 7.5 ($\cdot 10^6$)
	& $\eta_{MAX} = 82\%$	75 - 100
Mass	21 kg/kW +	2 - 30
	0.002 kg/PLkg +	0.0001 - 0.1
	200 kg	0 - 500
Power level	25 kW	10 - 10^6
Purchase cost	\$13 500/kg	150 - 100 000
Start-up delay	30 minutes	0 - 60
Propellant costs	\$15/kg	0 - 25
Operation costs	\$5M/year	1 - 10
Thrust level	~ 1 N.	dependent variable
Lifetime	15 000 hours	5000 - 100 000
Power system		
Power level	25 kW	10 - 10^6
Mass	15 kg/kW	1 - 20
Radiation sensitivity	P/P ₀ = 50% at	no degradation to 50% at
	$3 \cdot 10^{16}$ particles/cm ²	0.75 - 50 ($\cdot 10^{16}$)
Purchase cost	\$350/w	0.5 - 500
Earth launch system		
Delivery costs	\$700/kg	25 - 1000
Payload characteristics		
Mass		
Envelope/volume }	payload peculiar	-----
Cost of money	7%/year	0 - 20
EPS "Scar" - mass	+0.0075 kg/kg	-0.06 to +0.03
- cost	+0.0065 \$/\$	-0.06 to +0.03
Trajectory requirements		
Velocity increment (ΔV)	mission dependent	3000 - 9000 m/sec

TABLE 2. - MISSION COST SENSITIVITY COEFFICIENTS

[Representative partial derivatives for the given parameters in $\Delta\% \text{ transportation}/\Delta\% \text{ parameter.}]$

Group Parameter	Efficiency	EPS specific mass	EPS payload-dep. mass	EPS constant mass	Solar array specific mass	EPS specific cost
1	-0.41	0.23	0.03	0.18	0.05	0.30
2	-.40	.23	.02	.10	.05	.25
3	-.40	.22	.04	.07	.04	.22
4	-.36	.18	.04	.02	.04	.18
5	-.38	.24	.05	.0026	.05	.25
Group Parameter	Solar array specific cost	Operations cost	Cost of money	Earth- launch cost	Propellant cost	Mission velocity increment
1	0.19	0.34	0.10	0.09	0.0003	0.47
2	.23	.25	.20	.12	.0005	.48
3	.23	.19	.25	.14	.0003	.46
4	.25	.13	.22	.20	.0006	.44
5	.23	.01	.32	.35	.0009	.30

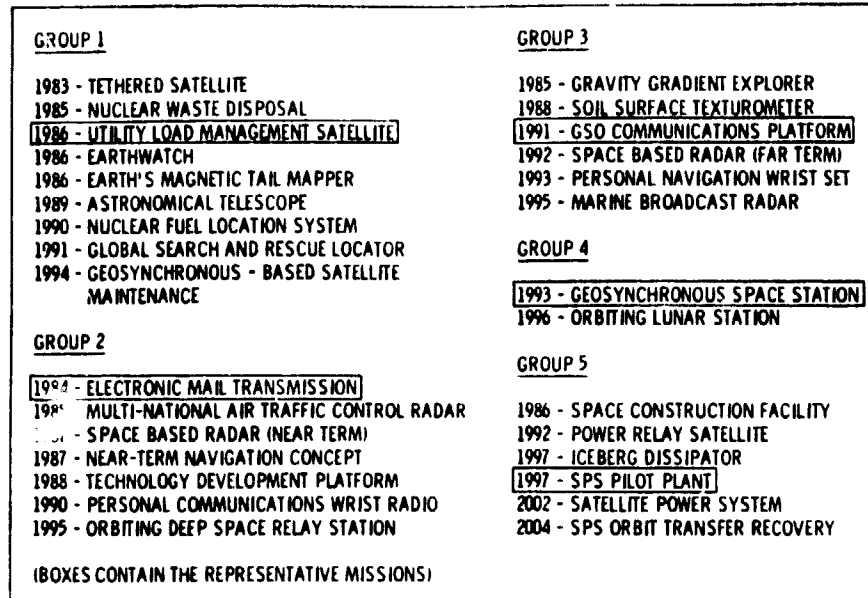


Figure 1. - Mission set, mission groups, and representative missions.

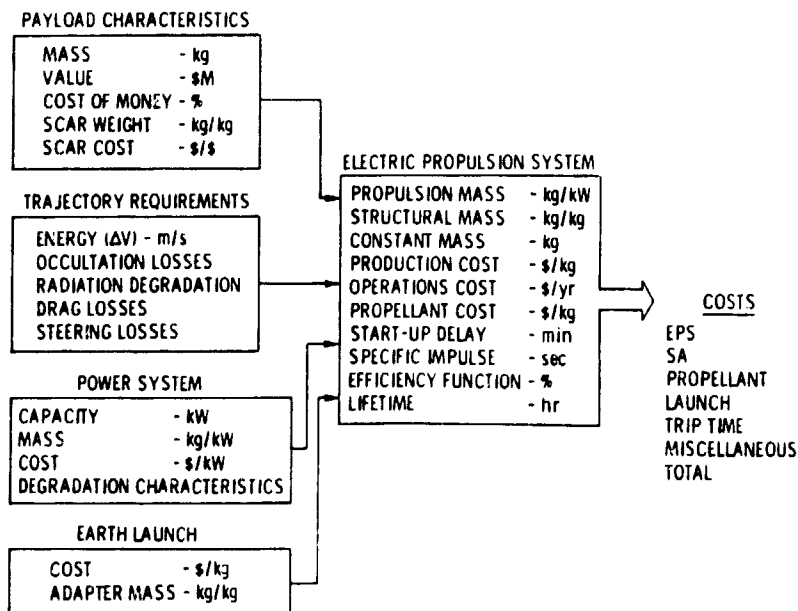


Figure 2. - Cost/performance model elements and parameters.

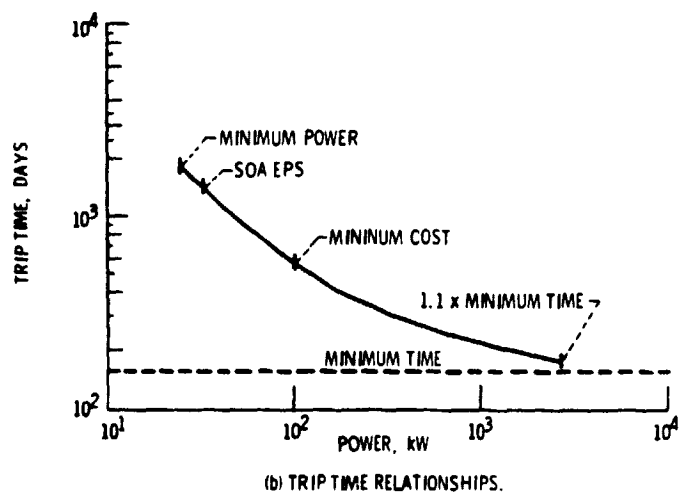
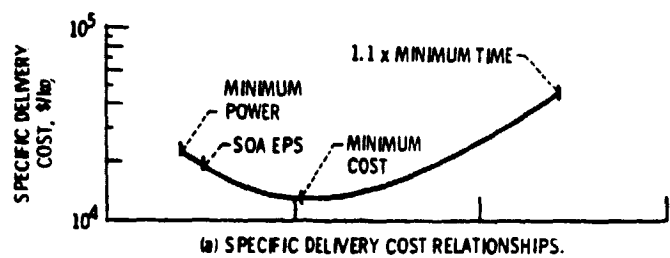
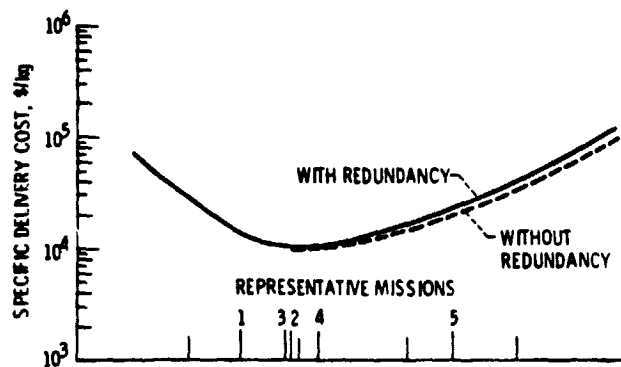
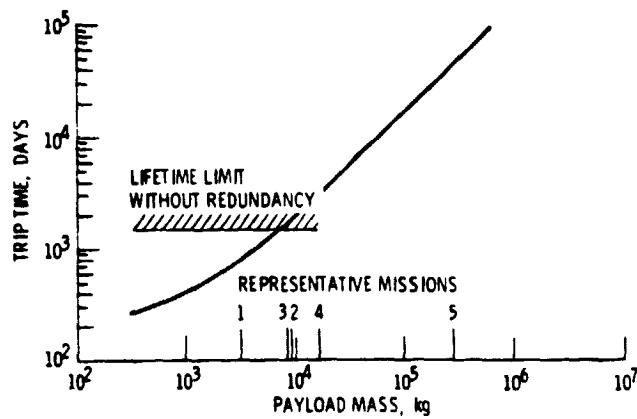


Figure 3. - Typical relationships of system design points.

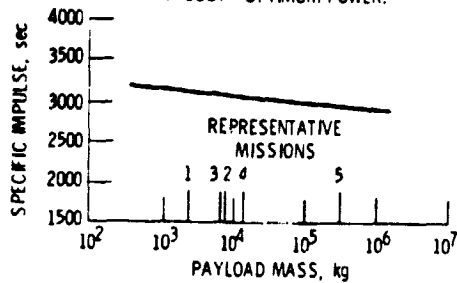
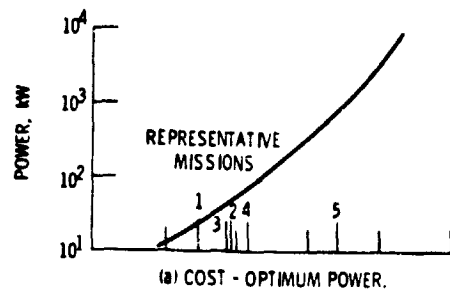


(a) SPECIFIC DELIVERY COST WITH AND WITHOUT REDUNDANCY.

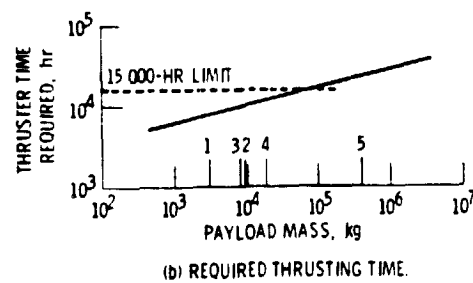
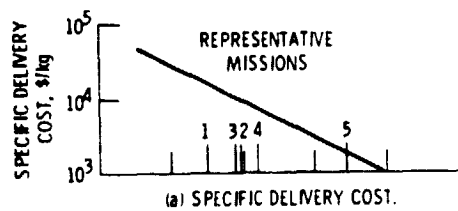


(b) REQUIRED TRIP TIMES.

Figure 4. - Performance of SOA system over the mission set.



(a) COST - OPTIMUM POWER.
(b) COST - OPTIMUM SPECIFIC IMPULSE.
Figure 5. - Variation of cost-optimum design point over mission set.



(a) SPECIFIC DELIVERY COST.
(b) REQUIRED THRUSTING TIME.
Figure 6. - Cost-optimum design point performance.

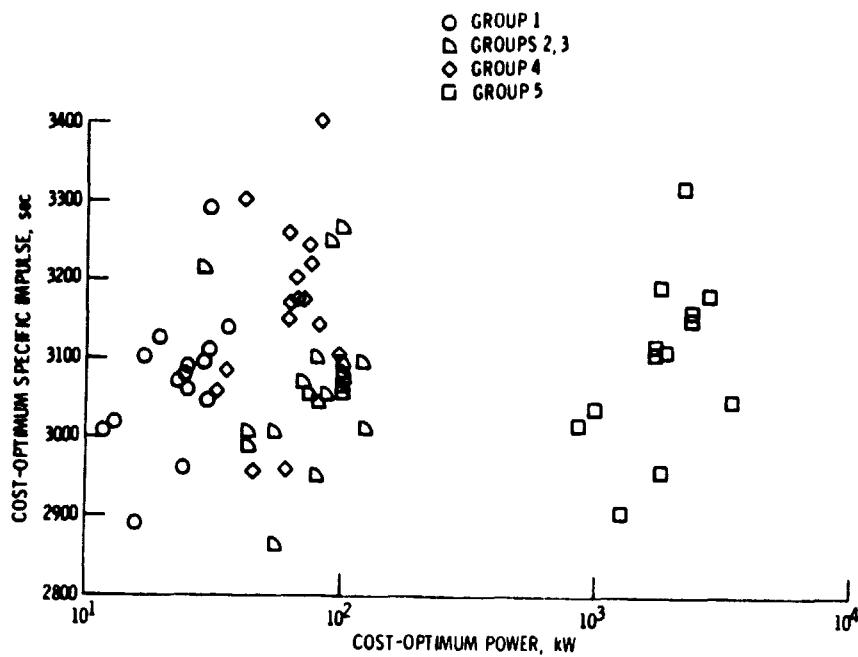


Figure 7. - Cost-optimum design point variation extremes.

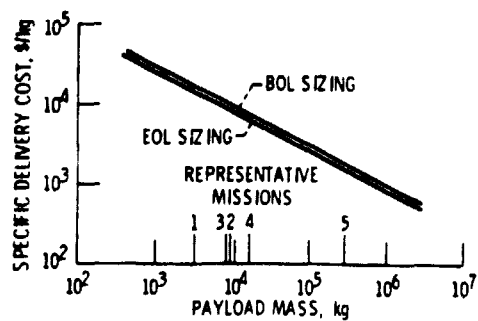


Figure 8. - Comparison of BOL and EOL sizing.

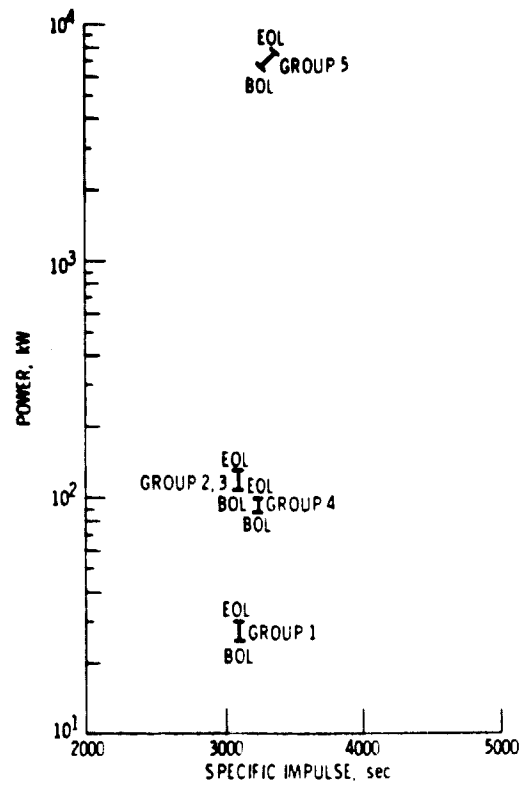


Figure 9. - EOL sizing impact on design points.

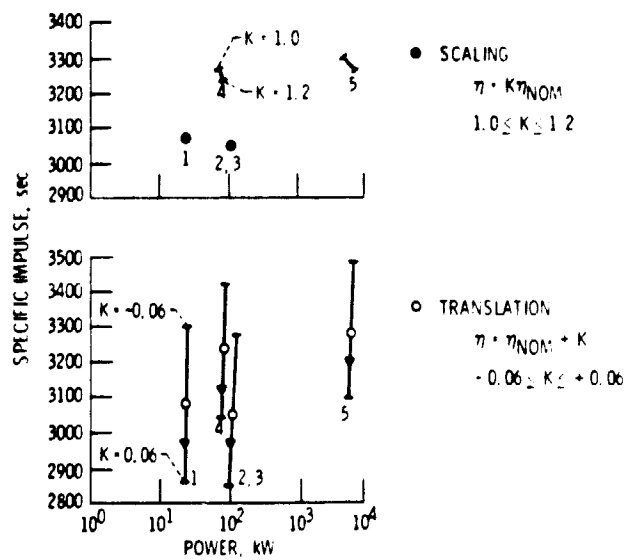
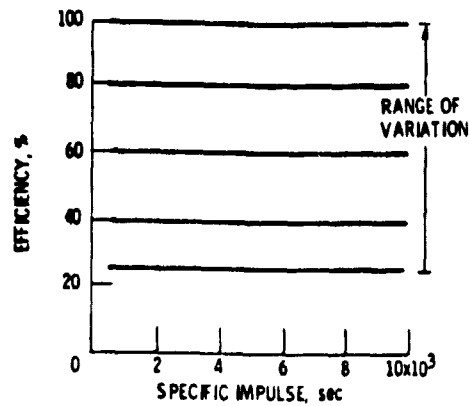
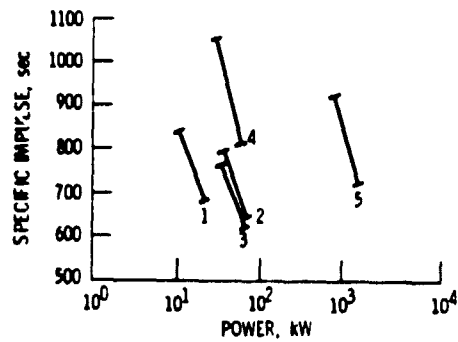


Figure 10. - Cost-optimum design point sensitivity to efficiency function variations.



(a) CONSTANT EFFICIENCY FUNCTIONS.



(b) IMPACT ON DESIGN POINTS.

Figure 11. - Effect of constant efficiency on cost-optimum design points.

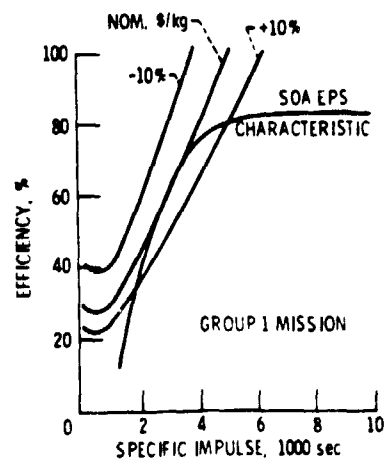


Figure 12. - Comparison of SOA efficiency function with that required for (various) constant delivery costs.

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