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UPPER STAGES UTILIZING ELECTRIC PROPULSION

David C. Byers
Lewis Research Center
Cleveland, Ohio

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Cleveland, Ohio

ABSTRACT

Electric propulsion offers many benefits for upper stages for space propulsion. Masses of electric propulsion systems required for Low Earth Orbit (LEO) to Geosynchronous Orbit (GEO) payload transfer can be small compared to chemical propulsion systems. This implies great reductions in the number of Earth launches, or the payload requirements on the shuttle. The low acceleration characteristic of electric propulsion is consistent with the low force loading requirements for lightweight large space systems and electric propulsion systems have demonstrated long term compatibility with the geocentric space environment. This paper presents the payload capabilities of upper stages using electric propulsion for a LEO to GEO orbit transfer mission. To determine payload mass it is necessary to define the electric thrust system and its power requirements in detail. This was done using an established methodology. Electric propulsion technology was assumed which is within, or felt to be readily attainable from, demonstrated state-of-art. The impact on payloads of total mass in LEO, thrusting (trip) time, propellant type, specific impulse, and power source characteristics was evaluated and is presented. Dependent upon detailed assumptions, electric stages were found capable of delivering payloads in thrusting time less than 50 days with payloads always initially increasing rapidly with increasing thrusting times. For the shorter thrusting (trip) times the payloads increased with increasing propellant mass and decreasing specific impulse. At very long trip times, however, the payload increased with decreasing propellant mass and increasing specific impulse. Variation of the specific mass of the power source between 5 and 30 kg-kW^{-1} caused the minimum trip times to vary about a factor of three and at short trip times strongly affected the electric stage payload capabilities.

INTRODUCTION

The advent of the Space Shuttle has stimulated interest in new or greatly expanded missions in geocentric space. Such missions include very large communication systems,[1] space manufacturing facilities,[2] and other Large Space Systems for many purposes.[3] The primary space propulsion systems will have a great impact on the approaches, capabilities, and costs of advanced missions. Key issues such as the numer of Earth launches, deployment requirements and options, spacecraft structural requirements, reuse/refurbishment options, and the impacts of the volume and mass constraints of the Shuttle are all dependent to first order on the characteristics of the primary space propulsion system.

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For many of the proposed missions the use of electric propulsion appears to result in enabled or greatly enhanced mission performance along with reduced mission costs. For example, to achieve a transfer from Low Earth Orbit (LEO) to Geosynchronous Orbit (GEO) with chemical propulsion requires a propellant mass several times the payload mass. With electric propulsion the sum of the propellant and thrust system masses can be a small fraction of the payload mass. This results in a dramatic reduction in the total mass required in LEO, and/or payload requirements of Shuttle launches.

To minimize structural mass, fragile Large Space Systems (LSS) require low force loading during orbit transfer. The low thrust characteristics of electric propulsion is consistent with that requirement. Furthermore, electric propulsion has demonstrated long term (10 year) compatibility with the geocentric space environment.[4] This may, in part, enable new return/refurbishment strategies for geocentric spacecraft.

Electron bombarbment ion thruster systems are approaching operational application. Germany,[5] Japan,[6] and the United States[7] all firmly plan demonstration space tests of small mercury ion thrusters developed for on-orbit propulsion. These thruster systems were designed for spacecraft with masses up to about 2500 kg. In addition, the 30-cm mercury ion thruster system is nearing technology readiness[8] and the NASA has announced plans[9] to initiate detailed contracted design studies of a Solar Electric Propulsion System (SEPS).

The designs of the elements of the baseline 30-cm mercury thruster system were predominately influenced by the requirements of planetary missions. These strongly drove the baseline system design in areas such as thruster size, propellant type, and power processor requirements. For several years a number of advanced technologies for electric propulsion systems have also been under active investigation. These technologies include inert gas propellants,[10,11,12] larger diameter thrusters,[13] operation at increased thrust and power per thruster,[14] and improved Power Management and Control (PMAC) concepts.[15] It is certain that future high energy geocentric missions would benefit greatly from the use of advanced electric propulsion technologies. The present state-of-art in electric propulsion allows the characteristics of such advanced electric propulsion systems tc be predicted with a high degree of confidence.

This paper presents the payload characteristics of geocentric missions which utilize electron-bombardment ion thruster systems. A baseline LEO to GEO mission was selected. The impacts on payloads of both mission parameters (such as trip time and mass in LEO) and electric propulsion technology options (such as specific impulse and propellant type) were evaluated. To predict payload mass it is necessary to specify the characteristics of the electric propulsion thrust system and the power requirements. This was done by utilizing a previously developed methodology [16,17] which provides a detailed thrust system description after the final mass on orbit, the thrusting time, and the specific impulse are specified. The characteristics of the thrust system elements were based on experimental data and/or detailed point system de-

signs.[18] Element masses and efficiencies were selected which were consistent with data already obtained (such as thruster efficiency) or are felt to represent straightforward extrapolations from present technology (such as power processor mass as a function of input power).

This paper describes the payload capabilities of electric propulsion for LEO to GEO orbit transfer missions. These data will be useful for mission planners in assessing the capabilities of electric propulsion systems which are basically state-of-art. In addition, the data can also allow evaluation by technologists of the impact of various electric propulsion technology advances.

OVERALL APPROACH

The various subsystems of a space system employing electric propulsion are shown on Fig. 1. In order to predict the payloads of missions it is necessary to define the characteristics of the required thrust systems. The use of Eq. (1) is the first step in that process.

$$T_T = 1.16 \times 10^{-5} \frac{g M_F I_{sp}}{\Delta T} \left(e^{\frac{V_m}{I_{sp} \, g}} - 1 \right) \tag{1}$$

In Eq. (1) M_F is the final delivered mass which is comprised of the payload and the thrust system dry masses. The payload (Fig. 1) includes the power used for the orbit transfer and the nonpower payload, which contains any propellant or separate hardware required for on-orbit propulsion functions. The parameter ΔT in Eq. (1) is the thrusting time (in days) where it is assumed that the thrust level of the propulsion system, T_T, is constant when operating. The total trip time is ΔT plus any thrust system off time - such as might occur due to solar occultation of a photovoltaic power source. The system off time is extremely sensitive to the specific technology and mission approaches assumed.[19] Issues such as thruster startup time, initial and transfer orbit inclination, and power source type (e.g., photovoltaic or nuclear) are all important in precisely defining the off times. The off times are expected to be a small (less than 20 percent) fraction of the thrusting time but because of its sensitivity to specific mission approaches its specification is beyond the scope of this paper.

The mission velocity increment, V_m, is that appropriate for low acceleration. In this paper V_m was fixed at 6000 m-sec^{-1} which is representative of a low acceleraion orbit transfer between LEO and GEO with an inclination change of about 28 degrees.

If the specific impulse, I_{sp}, is specified in addition to M_F and V_m the required propellant mass is given by:

$$M_p = M_F \left(e^{\frac{V_m}{g \, I_{sp}}} - 1 \right) \tag{2}$$

The total mass required in LEO, MLEO, is the sum of the final mass, M_F, and the propellant mass, M_p. For later reference, the ratios of the propellant and final masses to the total mass in LEO are given by Eqs. (3) and (4), respectively, as:

$$\frac{M_p}{M_{LEO}} = 1 - \frac{1}{e^{\frac{V_m}{gI_{sp}}}} \tag{3}$$

$$\frac{M_F}{M_{LEO}} = \frac{1}{e^{\frac{V_m}{gI_{sp}}}} \tag{4}$$

As described in Refs. 16 and 17 the detailed characteristics of the thrust system and the power required may be predicted after the system thrust, specific impulse, and approaches for thrust system elements have been specified. The reader is referred to Refs. 16 and 17 for details on the methodology utilized to describe the thrust system. For completeness, however, the major assumptions used in describing the thrust system will be given in the next section. Previous data[17] indicated that the specific impulse and the propellant type very strongly impact the thrust system and power requirements and these two parameters were varied, within allowable limits, to assess the impacts on mission payloads.

After the thrust system and power requirements are defined, the fraction of the final mass on-orbit, M_F, available for power and non-power payloads can be obtained when a specific mass for the power source is assumed. The payloads on orbit were calculated over a wide range of assumed thrusting times, total masses in LEO, and specific masses of the power source.

THRUSTER SYSTEMS

Fig. 2 shows the various elements of a thrust system. In Refs. 16 and 17 all elements of the thrust system are described parametrically. As examples, the output thrust and input power of a thruster were presented as functions of specific impulse and propellant type. The parametric descriptions used herein were obtained or derived from available experimental data or detailed system designs and are felt to be within, or relatively straightforward extrapolations from, state-of-art. The following sections will briefly discuss the major assumptions used in defining the thrust system and the rationales for their selection.

THRUST MODULE

Thruster. Table I shows the characteristics assumed for the thrusters which were assumed to be 50-cm in diameter for all data pre-

sented herein. It was shown in Ref. 17 that the thrust system mass remained nearby constant for thrusters 50-cm in diameter and larger. In addition, thrusters 50-cm[20] and 150-cm[13] in diameter have been built and tested. Those thrusters were operated at higher specific impulses and lower current densities than are considered herein. Technology advances using close spaced dished ion optics have enabled a new generation of low specific impulse, high current density, ion thrusters. Based on present knowledge, however, it is not felt that a 50-cm diameter thruster (or a non-circular thruster with an equivalent active ion acceleration area) with close-spaced ion optics represents a significant extrapolation of state-of-art technology. The individual thruster mass was specified as 20.4 kg, based on the data of Refs. 12 and 17.

Operation with mercury, xenon, and argon propellants was evaluated. An extensive data base[10,11,12] exists for each of these propellants and the values of propellant utilization efficiency, energy required to ionize the propellant, and thrust losses shown on Table I have been previously achieved. The maximum thruster beam current is essentially specified[16,17] for a given propellant by the total voltage which may be applied between the ion accelerator grids. A total voltage of 2000 V was assumed herein which is about two-thirds of maximum values demonstrated to date.[14] The ion beam current densities assumed were 90 percent of limit values, at the total voltage of 2000 V, obtained in experiments.[11] Three grid ion optics were assumed which allow a thruster to be operated at high values of thrust over a range of specific impulse of about 2.1 to 1.[14] The absolute values for the range of specific impulse shown on Table I for each propellant are specified by the total voltage, propellant utilization efficiencies, and thrust losses assumed. Thrusters may be operated at both higher nd lower values of specific impulse than shown on Table I. Such operation, with the assumptions given, will result in a strong decrease in output thrust per thruster[16] and cause thrust system masses to increase due to a requirement for additional thrusters and associated power processing.

It is not felt that thruster lifetime represents a major concern for the missions presented herein. Thrusting times of eight months or less are of interest for geocentric orbit raising missions. Such duration are much less than the two year operating design lifetimes[8] of the baseline, 2 A beam current, 30-cm mercury thruster. The thrusters proposed herein operate at higher ion current densities than the baseline thruster. It has been found,[14] however, that the discharge voltage at which bombardment thrusters operate decreases with increasing ion current density. This has resulted[14] in decreased values of discharge chamber erosion (the major life limiting phenomena) with increased ion current density.

The assumptions used for the thrusters are well within demonstrated envelopes of operation. It is expected that the ongoing technology programs will result in improved and extended performance capabilities with consequent decreases in thrust system masses and power requirements and, therefore, increases in the payload capabilities presented herein.

The total number of operating thrusters in the thrust system was obtained by first dividing the system thrust (Eq. (1)) by the thrust of an individual thruster and then rounding off to the next highest integral number. The total number of thrusters was then obtained by arbitrarily adding two spare thrusters (and their associated power supplies, gimbals, and propellant distribution) to the number of operating thrusters.

Gimbal. The gimbal mass was based on a detailed point system design[18] and was assumed to be 0.34 times the thruster mass.

Propellant Distribution. From Ref. 18 a mass of one kilogram per thruster was charged for propellant distribution.

Power Management and Control. The Power Management and Control (PMAC) subsystem is comprised of the beam, discharge, and low power supplies in the thrust module and the beam and discharge reconfiguration units, the distribution inverter, the converter, and the thrust system controller in the interface module. The numbers, dissipated powers, and masses of all elements in the PMAC subsystem were all taken directly from Ref. 17.

Thermal Control. The thermal control subsystems for both the thrust and interface modules were assumed to use heat pipes and radiators which rejected heat from one side only. The radiators were sized to maintain critical baseplate temperatures at 323° K and to accommodate a 15 percent view factor to a solar array at one astronomical unit. Based on the detailed optimization of Ref. 18, the mass of the thermal control subsystems (in kilograms) was taken to be 31 times the dissipated power in kilowatts.

Structure. The thrust module structure serves to cantilever the thrusters and gimbals away from the interface module. The thrust module structure was assumed to be 0.31 times the sum of the thruster and gimbal masses as was derived in the detailed point design[18] of a thrust system.

INTERFACE MODULE

The PMAC and thermal control subsystems of the interface module were described in the previous sections.

Propellant Storage. Mercury and xenon were assumed to be stored noncryogenically. Mercury is a liquid at normal temperatures. Xenon required a pressure of 1×10^7 Pa to store it supercritically. Cryogenic storage was assumed for argon and the masses for all of the propellant tanks were taken directly from Ref. 17.

Structure. From Ref. 17 the interface structural mass was taken as four percent of the sum of the thrust system mass, the propellant mass, and the other dry mass of the interface module. This value of interface module structural mass is about three times that calculated for a mercury thrust system in a very detailed point system design.[18] The increase over the reference design value was assumed to provide design

margin for thrust systems which may be less compact than the design of Ref. 18.

TRANSMISSION SUBSYSTEM

The power transmission subsystem was assumed to carry direct current (dc) power from the power source and was assumed to utilize copper wires and be 100 meters in length. The masses and dissipated powers of the transmission subsystems were calculated using the techniques described in Ref. 17 and as a result the dissipated power in the transmission subsystem remained nearly constant at about 5.5 percent of the total system power for all the data presented herein.

POWER SOURCE

A dc power source was assumed which was characterized by a specific mass (in $kg-kW^{-1}$). To simplify mission calculations, the power available to the thrust system was assumed to be constant during the mission. This would be typical of nuclear power sources and, to first order, hardened solar arrays, some backlight array systems, and solar thermal power systems. For planar solar arrays of present design, significant degradation will occur during the orbit transfer and, as discussed later, such degradation can be easily factored into the results presented. The payloads presented in this paper can be conservatively estimated if degradation is assumed to occur before the start of the mission and the initial power required is adjusted appropriately.

Power source specific masses between 5 and 30 $kg-kW^{-1}$ are assumed in this paper. The lower value has been described by many authors[21,22,23] as representative of thin cell solar arrays at Beginning of Life (BOL). The baseline SEPS solar has a specific mass of 15 $kg-kW^{-1}$ which is also representative[23] of a number of nuclear power sources above about 100 kW.

MISSION PAYLOADS

In the following, payloads will be presented for an orbit transfer mission with a mission velocity increment of 6000 $m-sec^{-1}$. This is representative of a low thrust LEO to GEO mission which includes a 28.5 degree inclination change. The effects of total mass in LEO and thrusting time were investigated. Operation on mercury, xenon, and argon propellants was evaluated over the range of specific impulse appropriate for each propellant (Table I). Finally, the effect of variation of the specific mass of the power source is presented. As stated previously, attempts were made throughout to use assumptions within, or felt straightforwardly attainable from, state-of-art in order to provide an assessment of present electric propulsion technology for orbit raising missions.

MASS IN LEO

The mass in LEO, MLEO, is comprised of the propellant, thrust system, the power payload, and the nonpower payload (Fig. 1). No mass pen-

alty was assumed herein for a structural interface between the space system and the Shuttle due to uncertainty in overall requirements.

Fig. 3 shows the ratio of the total payload mass to MLEO and the ratio of power (in kW) to MLEO plotted as a function of MLEO over a range of thrusting times for a mercury thrust system operating at a specific impulse of 1800 seconds. It is seen that for a given specific impulce and thrusting time both the total payload mass and power ratios are basically insensitive to MLEO for values of MLEO greater than about 10^4 kg. Although not shown for clarity, the ratios of thrust system mass to MLEO and propellant mass to MLEO were equally insensitive to MLEO. This behavior was found true for all combinations of specific impulse, propellant type, and thrusting times presented in this paper.

The various paylaod mass and power ratios presented subsequently in this paper were all obtained for an MLEO of 10^4 kg. From the remarks above, hower, the mass and power ratios are insensitive to MLEO for values of MLEO greater than about 10^4 kg and are accurate to within a few percent for values of MLEO down to about 5×10^3 kg. This feature allows the payloads of a very broad set of potential missions to be plotted in a highly condensed manner.

GENERALIZED PAYLOAD CHARACTERISTICS

Fig. 4 shows the total payload and power ratios of Fig. 3 as a function of thrusting time. for mercury, xenon, and argon thrust systems operating at the upper and lower values of specific impulse selected for each propellant. Again, the data of Fig. 4 are independent of MLEO.

The data of Fig. 4 are very general and can be used to define all the major subsystems of Fig. 1: as will be illustrated by the following example.

An arbitrary value of MLEO of 2×10^4 kg will be assumed along with a mercury thrust system operating at a specific impulse of 1800 seconds for a thrusting time of 100 days. From Fig. 4(a) the ratio of total payload mass to MLEO is obtained as 0.54 which implies a total payload mass of 1.08×10^4 kg. From Eq. (3), the ratio of propellant mass to MLEO is obtained, at a ⌐pecific impulse of 1800 seconds, as 0.29 which implies a propellant mass of 5.8×10^3 kg. The thrust system mass is MLEO minus the sum of the total payload and propellant masses and is, therefore, equal to 3.4×10^3 kg.

The ratio of power to MLEO at 100 days thrusting time is shown on Fig. 4(a) as 10×10^{-3} kW-kg^{-1} which implies a power of 200 kW. As stated previously, this power was assumed to be available throughout the mission. An arbitrary power degradation of 25 percent of initial power will be assumed to occur during the mission which implies a beginning of mission power of 267 kW.

In order to obtain the nonpower payload it is necessary to select a specific mass of the power system. From the above, the specific mass should be that representative of Beginning-of-Life (BOL) for the particular power source selected. For a BOL specific mass of 5 kg-kW^{-1} the

power payload mass is 1.34×10^3 kg. The nonpower payload mass is 9.47×10^3 kg and was obtained by subtracting the power payload from the total payload.

For the example given, the mass of the power payload became equal to the total payload mass at a BOL power source specific mass of about 40 kg-kW^{-1}. At that point the nonpower payload vanished and the selected thrusting time is at its minimum value for that specific impulse and propellant unless lighter power sources and/or thrust systems are assumed.

The data of Fig. 4 can be conveniently used to specify the total payloads and power payloads for a very large range of LEO to GEO missions which utilize electric propulsion thrust systems within, or readily attainable from state-of-art technology. As discussed previously, the nonpower payloads may also be obtained from Fig. 4. The nonpower payload cannot, however, as generally plotted because the specific mass and degradation of the power source must be specified. To provide clearer insights, the following sections will present nonpower payload masses to determine the impact of various parameters.

NONPOWER PAYLOADS

Fig. 5 shows the ratio of nonpower payload mass to MLEO as a function of thrusting time for mercury, xenon, and argon propellants at the upper and lower values of specific impulse specified for each propellant. A power source specific mass of 15 kg-kW^{-1} was assumed and for convenience no degradation of the power was assumed to occur during the mission.

For all propellants the nonpower payloads were maximum for the lowest specific impulse near the minimum thrusting times. With increasing thrusting times the nonpower payload becomes less sensitive to specific impulse and then becomes maximum at the largest value of specific impulse assumed for each propellant. This behavior is best explained by reference to Fig. 6 which shows the four elements of MLEO (Fig. 1) as a function of specific impulse for a mercury thrust system at thrusting times of 50 and 250 days. It is first seen that the propellant mass decreases by about a factor of two over the assumed range of specific impulse but, as shown by Eq. (2), is not a function of thrusting time. The ratio of thrust system mass to MLEO drops strongly with increasing thrusting time but, on the other hand, is insensitive to specific impulse for a given thrusting time. This occurs due to a tradeoff[17] between the decreases (due primarily to the decreased number of thrusters and power processors) and the increases (due primarily to increased power processing and thermal control system masses) in thrust system mass with increasing specific impulse (and power).

The variation of the power payload mass is, the, the major reason for the different sensitivity of nonpower payload to specific impulse at different thrusting times. At low values of thrusting time the power payload is a major fraction of MLEO and the variation of the power payload mass with specific impulse strongly affects the nonpower payload. At large values of thrusting time the power mass becomes a small frac-

tion of MLEO and the variation of specific impulse has a smaller fractional effect on the nonpower payload.

In the limit of very long thrusting times the nonpower payload mass ratio approaches the final mass ratio given in Eq. (4). This is because the thrust, and power, decrease with increasing thrusting time and, therefore, the masses of the thrust system and power payload become negligible.

Fig. 5 shows that the nonpower payload is quite sensitive to propellant type at the shorter thrusting times. The reason for this behavior is illustrated on Fig. 7 which shows various mass ratios for all three propellants at their lowest values of specific impulse for a thrusting time of 100 days.

The propellant mass decreased with increasing specific impulse (or decreasing atomic mass) as predicted by Eq. (3). The thrust system mass increased with decreasing atomic mass such that the sum of the thrust system and propellant masses remained nearly constant. This resulted in the near independence (Fig. 7) of the total payload mass on propellant type. The variation in nonpo er payload with propellant type at the shorter thrusting time is essentially explained by the differences in power required for the orbit transfer. The major reason for the differences in power for the various propellants is the variation of thrust to power ratio for the thrust systems. The thrust to power ratio varies because of the differences in specific impulse and the system efficiencies assumed for each propellant (Table I).

The sensitivity of nonpower payload to propellant type decreases with increasing thrusting time (Fig. 5). Although not shown, this arises primarily because the thrust and power system masses decrease with increasing thrusting time and their variation with propellant type becomes fractionally smaller with respect to MLEO.

The specific mass of the power source can have a dramatic effect on the nonpower payload. Fig. 8 shows the nonpower payload ratic as a function of thrusting time for three values of specific mass for a mercury thrust system. Variation of the specific mass from 5 to 30 kg-kW^{-1} causes about a factor of three increase in the minimum thrusting times. At the shorter thrusting times the nonpower payload at a fixed th usting time is very sensitive to specific mass but becomes less sensitive at longer thrusting times.

The effect of variation of the power source specific mass is shown on Fig. 9 for mercury and argon propellant thrust systems operated at specific impulses of 1800 and 3400 seconds, respectively. For any thrusting time the fractional effect of the power source specific mass variation from 5 to 30 kg-kW^{-1} is much larger for argon than mercury thrust systems. As an example, for a thrusting time of 100 days the nonpower payload varies by a factor of four with argon and a factor of only two with mercury. This difference in sensitivity to specific mass occurs because the power system is a much larger fractional mass of MLEO with argon (Fig. 7). For both propellants, and for xenon which is not shown for clarity, the six-to-one variation in power system specific

mass caused about a factor of 2.5 to three variation in thrusting times. This implies that the mass of power systems plays an important role in determining the payload capabilities of electric propulsion systems. This is especially true for short thrusting times and for lightweight propellants, which require the largest amounts of power.

CONCLUDING REMARKS

The payload capabilities of electric propulsion upper stages were presented for a baseline orbit transfer from LEO to GEO. To calculate payloads it was necessary to define the detailed characteristics of the electric propulsion thrust systems. This was done using an established methodology which employed assumptions regarding electric propulsion technology which were within, or felt to be directly attainable trow, state-of-art.

The effects on payloads was investigated for variations of total mass in LEO (MLEO), thrusting (trip) times, propellant type, specific impulse, and power source specific mass. It was determined that the ratios of payload masses to total mass in LEO were basically insensitive to MLEO. This fact allowed the overall payload capability of electric propulsion upper stages to be presented in a highly condensed fashion. Dependent upon the detailed assumptions, electric stages were found capable of delivering payloads in thrusting times less than 50 days with the payloads always increasing rapidly with increasing thrusting times. Payload capabilities far in excess of that attainable with chemical propulsion were possible using state-of-art electric propulsion technology.

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TABLE I. - THRUSTER CHARACTERISTICS

	Propellant		
	Mercury	Xenon	Argon
Thruster diameter, cm	50	50	50
Thruster mass, kg	20.4	20.4	20.4
Propellant utilization efficiency	0.95	0.95	0.8
Ionization power per beam ampere, W/A	150	183	250
Thrust losses,[a] percent	5.0	5.0	5.0
Range of specific impulse, sec	1800 to 3800	2200 to 4700	3400 to 7200
Thruster overall efficiency	0.62 to 0.79	0.58 to 0.78	0.44 to 0.63

[a]Thrust losses due to beam divergence and multiply charged ions.

SPACE SYSTEM		
THRUST SYSTEM • THRUST MODULE • INTERFACE MODULE • TRANSMISSION SUB- SYSTEM	PROPELLANT	PAYLOAD • POWER • NONPOWER
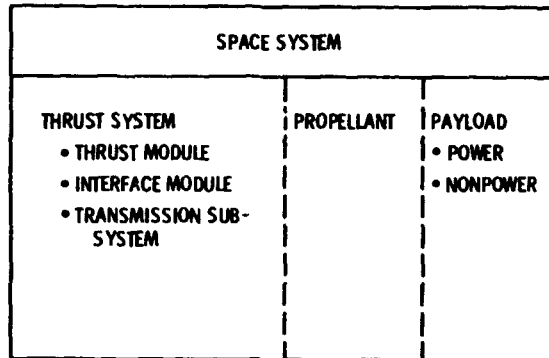
Figure 1. - Elements of an electric propulsion space system.
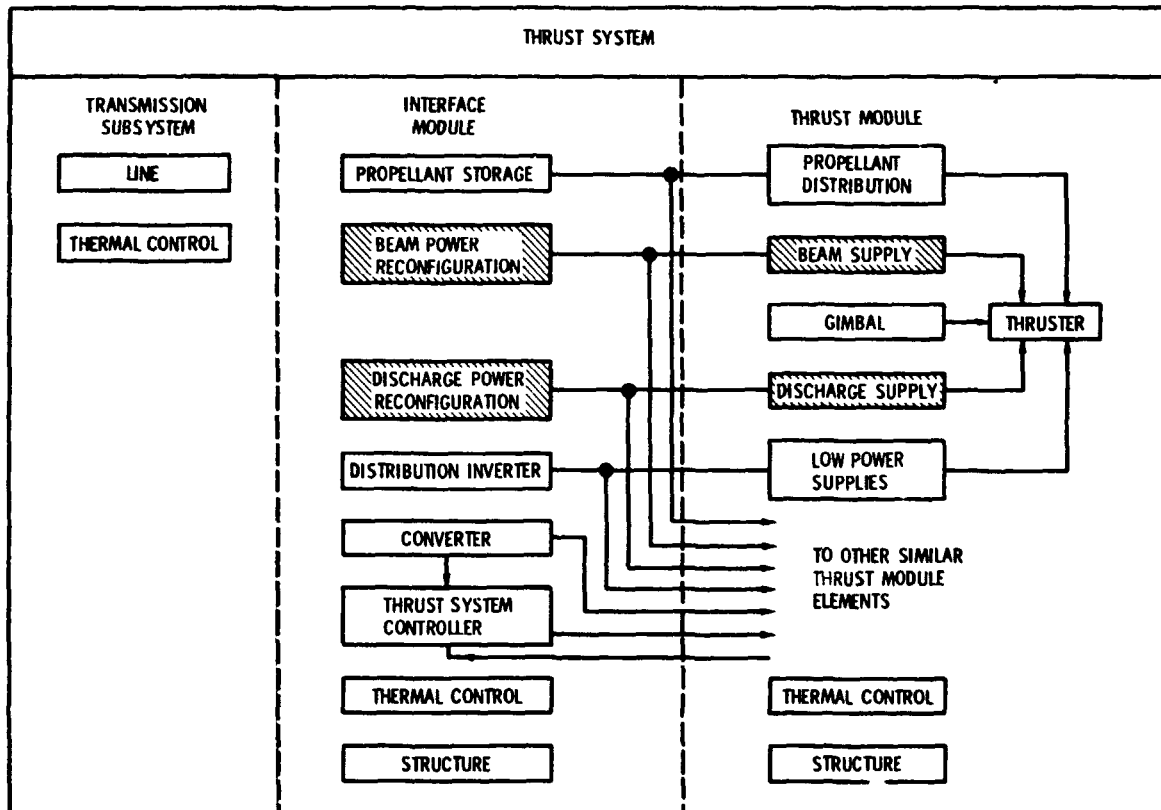
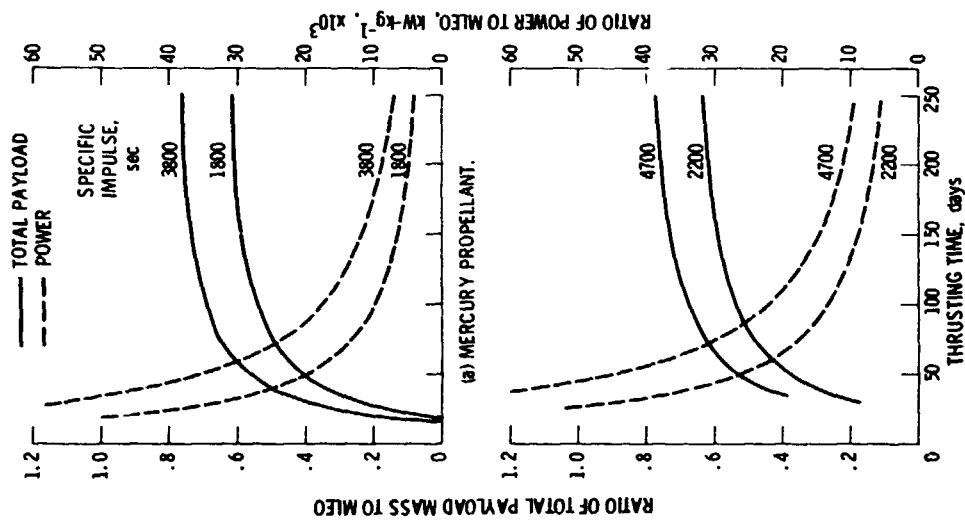

Figure 2. - Elements of an electric thrust system.
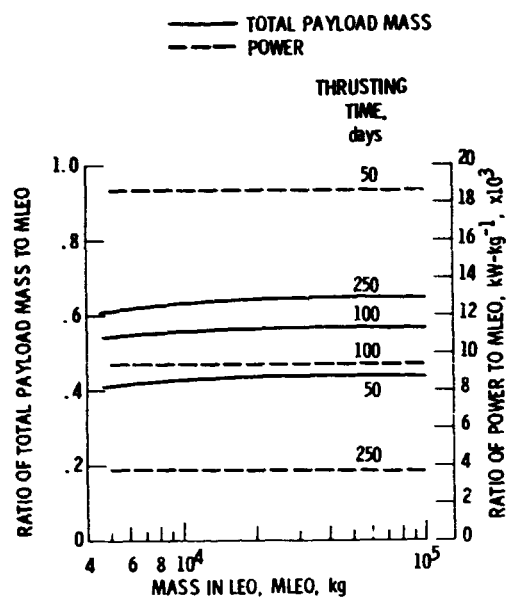
Figure 3. - Ratios of total payload mass and power to mass in LEO. Propellant, mercury; specific impulse, 1800 seconds.
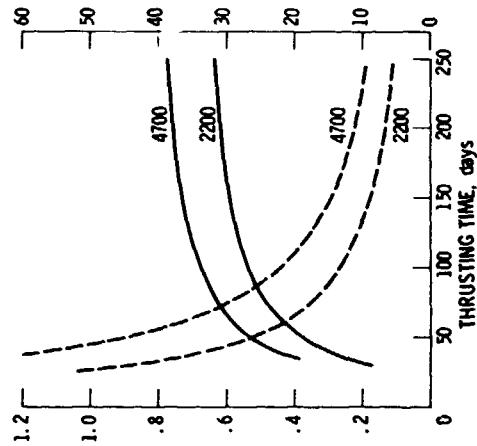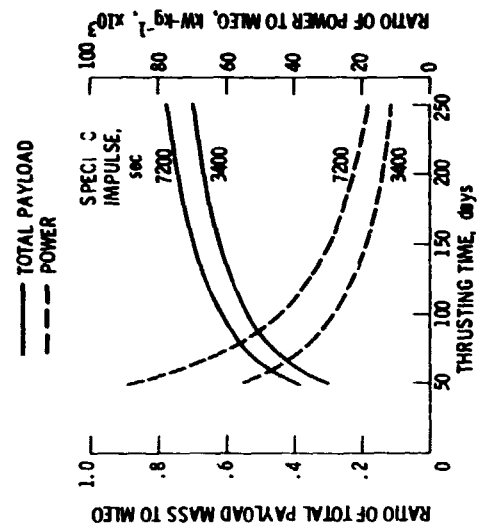
(c) ARGON PROPELLANT.

Figure 4. – Concluded.

(a) MERCURY PROPELLANT.

(b) XENON PROPELLANT.

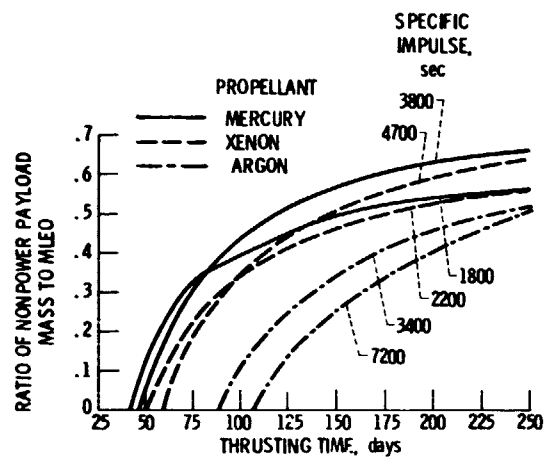Figure 4. – Ratios of total payload mass and power to MLEO as a function of thrusting time.

Figure 5. - Ratio of nonpower payload to MLEO as a function of thrusting time. Power source specific mass, 15 kg-kW^{-1}.

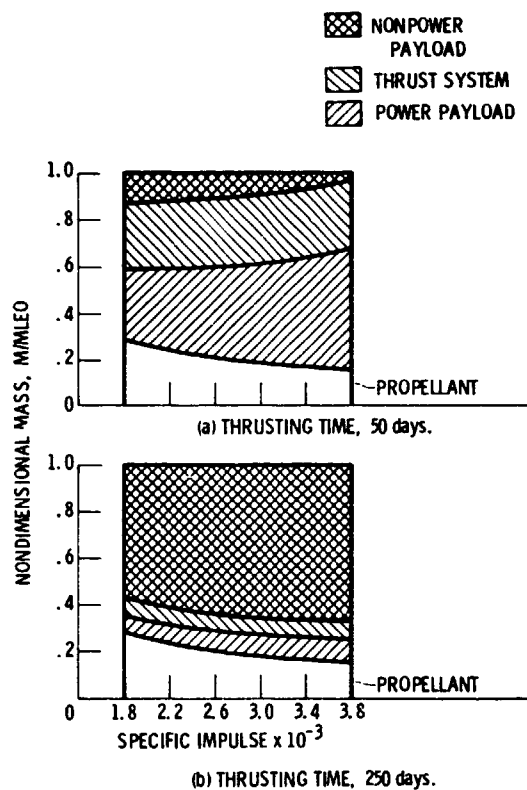(a) THRUSTING TIME, 50 days.

(b) THRUSTING TIME, 250 days.

Figure 6. - Nondimensional masses as a function of specific impulse. Mercury propellant. Power source specific mass, 15 kg-kW^{-1}.
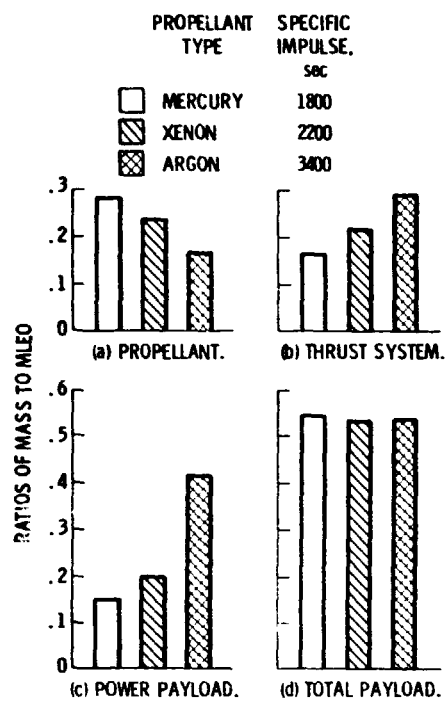
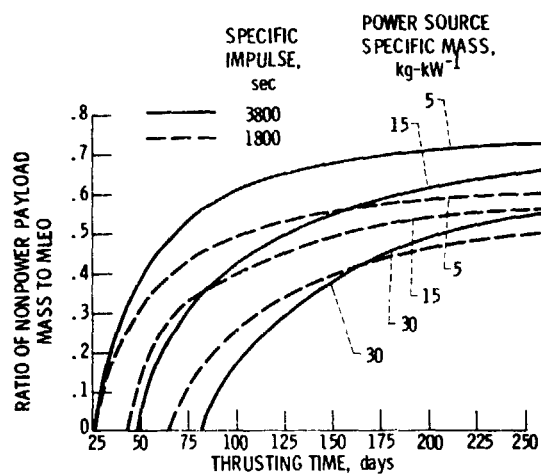Figure 7. - Mass ratios for various propellants.
Thrusting time, 100 days.

Figure 8. - Nonpower payload mass ratio as a function
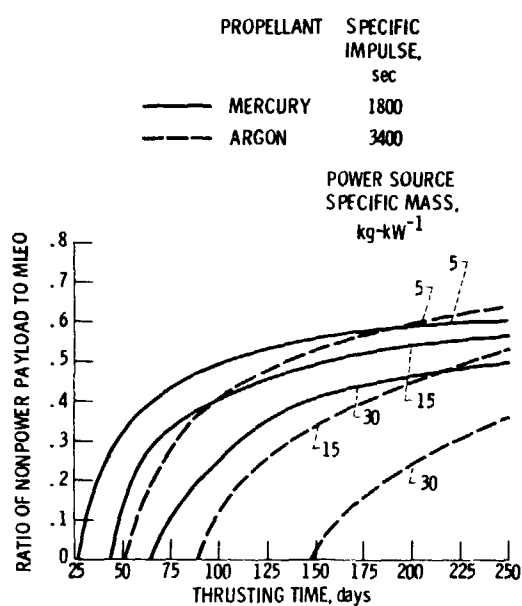of thrusting time. Mercury propellant.

PROPELLANT SPECIFIC
IMPULSE,
sec

———— MERCURY 1800
———— ARGON 3400

POWER SOURCE
SPECIFIC MASS,
$kg\text{-}kW^{-1}$

Figure 9. - Nonpower payload as a function of thrusting time.

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15 Supplementary Notes

16 Abstract

Electric propulsion offers many benefits for upper stages for space propulsion. Masses of electric propulsion systems required for Low Earth Orbit (LEO) to Geosynchronous Orbit (GEO) payload transfer can be small compared to chemical propulsion systems. This implies great reductions in the number of Earth launches, or the payload requirements on the Shuttle. The low acceleration characteristic of electric propulsion is consistent with the low force loading requirements for lightweight large space systems and electric propulsion systems have demonstrated long term compatibility with the geocentric space environment. This paper presents the payload capabilities of upper stages using electric propulsion for a LEO to GEO orbit transfer mission. To determine payload mass it is necessary to define the electric thrust system and its power requirements in detail. This was done using an established methodology. Electric propulsion technology was assumed which is within, or felt to be readily attainable from, demonstrated state-of-art. The impact on payloads of total mass in LEO, thrusting (trip) time, propellant type, specific impulse, and power source characteristics was evaluated and is presented. Dependent upon detailed assumptions, electric stages were found capable of delivering payloads in thrusting times less than 50 days with payloads always initially increasing rapidly with increasing thrusting times. For the shorter thrusting (trip) times the payloads increased with increasing propellant mass and decreasing specific impulse. At very long trip times, however, the payload increased with decreasing propellant mass and increasing specific impulse. Variation of the specific mass of the power source between 5 and 30 $kg-kW^{-1}$ caused the minimum trip times to vary by about a factor of three and at short trip times strongly affected the electric stage payload capabilities.

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