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Electric Propulsion Options for the SP-100 Reference Mission

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SUMMARY

Analyses were performed to characterize and compare electric propulsion systems for use on a space flight demonstration of the SP-100 nuclear power system. The component masses of resistojet, arcjet, and ion thruster systems were calculated using consistent assumptions and the maximum total impulse, velocity increment, and thrusting time were determined, subject to the constraint of the lift capability of a single Space Shuttle launch. From the study it was found that for most systems the propulsion system dry mass was less than 20 percent of the available mass for the propulsion system. The maximum velocity increment was found to be up to 2890 m/sec for resistojet, 3760 m/sec for arcjet, and 23 000 m/sec for ion thruster systems. The maximum thrusting time was found to be 19, 47, and 853 days for resistojet, arcjet, and ion thruster systems, respectively.

INTRODUCTION

Space nuclear power systems are attractive because they offer the potential for power sources that are not limited by a specific orbit or orientation with the sun, as well as allowing for power levels which exceed current solar array technology. Presently, these space nuclear power systems are being developed through the joint DOD/DOE/NASA SP-100 program (ref. 1). The goal of this program is to develop power systems, in the 50 kW to 1 MW power range, that can operate for up to 10 yrs and which can be delivered to orbit from a single Space Shuttle launch. As part of the SP-100 program a flight demonstration has been proposed using an electric propulsion system as an active load for the operating reactor (ref. 2). It has been suggested in reference 1 that the flight represent a realistic mission, such as orbit transfer, and that the electric propulsion system be capable of long-term operation. In addition, the electric propulsion system is required to be a low-risk technology with the potential for growth to power levels higher than the initial 100 to 300 kW demonstration range being considered.

A study was conducted to determine the maximum velocity increment and total impulse of electric propulsion systems using the SP-100 power source. Resistojet, arcjet, and ion thruster systems were considered in this study for use on the flight demonstration. The resistojet, shown in figure 1, and the arcjet, represented in figure 2, are both electrothermal devices, obtaining thrust by heating a gaseous propellant and expanding that propellant through a nozzle. The resistojet uses resistively heated elements, such as rods or coiled tubes, to heat the propellant while the arcjet uses an arc discharge to heat the gas. In the electrostatic ion thruster, shown in figure 3, ions generated in a discharge chamber are accelerated through a set of grids by an electrostatic field to develop thrust.

In this study the dry masses for propulsion systems using each of these devices were calculated and the maximum mission velocity increment (ΔV), total impulse, and thrusting time of each system were determined based on Space Shuttle launch constraints. A consistent set of assumptions was used throughout the study to calculate masses for components such as power processor, thermal control, and structure. Total power levels of 150 and 300 kW were used for the analyses. Previous studies have concentrated on specific missions using electric propulsion (refs. 3 to 7). In this study, however, the analyses concentrated on the maximum mission capability, in terms of velocity increment and operating time, available with each system.

MODEL DESCRIPTION

The model used to perform the analyses contained two parts: the propulsion system mass model and a generic mission model. In the propulsion system model each thruster system was characterized, and the masses of the components were calculated. The propulsion system mass model was based on a previous study (ref. 8). In this model the propulsion system included both a thrust module and an interface module. As shown in table I, the thrust module included the thruster, power processors (PPU), and associated equipment while the interface module contained the propellant storage, reconfiguration units (power distribution cabling), controllers, and converters. The power processor for the resistojet and arcjet systems consisted of a single supply for the resistor or arc, while the ion thruster system had discharge, beam, and low-voltage supplies because the ion thruster requires both low and high voltage supplies for operation. Thermal control for the power processor was included, but no thermal control was included for the thrusters as the thrusters were assumed to be self-radiating. Gimbals for the thrusters were included for all systems. Allowances were made for housing structure in both the thruster and interface modules. Also included was a separate thruster structure to cantilever the thruster/gimbal assembly away from the interface module. The component masses were calculated from expressions given in reference 8 with the exception of the resistojet (ref. 9) and arcjet (ref. 10) thruster masses. No redundancy has been included in the system with the exception of the reconfiguration units, controllers, and converters because loss of the thruster would not negate the primary SP-100 mission objective of operating the reactor system. Also, the mass of a guidance/navigation system was assumed to be a part of the spacecraft and hence was not included in the evaluation of the thruster systems. The sum of the thrust module mass and interface module mass (not including the propellant, tankage, and tankage structure mass) was known as the propulsion system "dry" mass. A summary of the equations used to calculate the component masses is given in table II.

The performance parameters of the three thruster systems are summarized in table III. Each of the electric propulsion systems used in the analyses was assumed to be low risk, with reasonable lifetimes achievable with state-of-the-art technology. The resistojet system used a 65 kW tungsten resistojet, with performance parameters scaled from thrusters in previous studies (ref. 11). This resistojet was assumed to run with either hydrazine (N_2H_4), ammonia (NH_3), or hydrogen (H_2). The arcjet used was a conventional design thruster as described in other studies (refs. 12 to 14). Calculations for both ammonia and hydrogen arcjets were performed at a thruster input power of 26 kW. The resistojet and arcjet power levels were chosen to match the 150 and 300 kW total input power levels to the thruster systems such that an integer number of thrusters was obtained. The 800 sec specific impulse ammonia arcjet was similar to the baseline system in analyses performed by Deininger and Vondra (ref. 15) while the 1000 sec hydrogen arcjet was similar to systems described by Wallner and Czika (ref. 16).

Mercury and xenon ion thruster systems were also considered. The mercury system had a specific impulse range of 1800 to 3810 sec while the xenon system had a range of 2220 to 4710 sec. Both 30 and 50 cm ion thrusters were considered, with performance parameters calculated from previous studies (refs. 17 and 18). I_{sp}-efficiency curves for the ion thruster system are given in figure 4 (ref. 19). For the ion thruster, the thruster input power was a function of specific impulse. For most cases an identical match of the thruster system input power with the full 150 or 300 kW input power could not be achieved. Therefore, with the constraint placed on the study that the total thruster system input power could not exceed the 150 or 300 kW power level, the number of ion thrusters required was an integer number that would allow the total thruster input power to be close to, but not exceed, the 150 or 300 kW limit. The equation for the number of thrusters is given in table II.

Once calculations were made of the individual propulsion system dry masses the mission model was used to calculate the mission capabilities of each of the systems. The following assumptions were used for the mission model:

1. The mission must be accomplished with a single Shuttle Orbiter launch from the Eastern Test Range (ETR).

2. The Shuttle Orbiter payload mass capability was 27 270 kg at 296 km (160 nmi).

3. The entire payload volume, including the propulsion system, reactor and orbit transfer vehicle must not exceed the usable Space Shuttle cargo bay volume of 286 m^3.

4. The Airborne Support Equipment (ASE) mass was 15 percent of the Shuttle payload capability.

5. The SP-100 reactor mass was 7200 kg while the spacecraft to support the reactor was 1000 kg. This reactor mass was based on estimates for the 300 kW system. These reactor and spacecraft masses were held constant for both the 300 and 150 kW power levels considered in this study because operation at 150 kW would be achieved with the 300 kW reactor tested at a lower-power throttled condition.

6. A bipropellant chemical propulsion system (I_{sp} = 295 sec) was used to lift the payload from a Shuttle orbit of 296 km (160 nmi) to a nuclear reference orbit of 834 km (450 nmi). This chemical propulsion orbit transfer vehicle (OTV) had a propellant mass fraction of 0.85 of the entire OTV mass. It should be noted that additional analyses were performed using a Shuttle orbit of 377 km. However, at higher initial orbits the available mass for the electric propulsion system was reduced due to a decreased Shuttle payload capability, thus reducing the velocity increment capability of the electric propulsion systems. Only results using the 296 km initial orbit will be included.

7. A contingency of 10 percent of the Shuttle payload capability less the ASE mass was included.

Using these assumptions a calculation was made of the mass available for the entire electric propulsion system, which included the propulsion system dry mass, propellant, tankage, and tankage structure. This available mass was the Shuttle Orbiter payload capability less the ASE, OTV, contingency, reactor, and spacecraft masses. The available mass was 9562 kg, constant for any propulsion system. The electric propulsion dry mass was then subtracted from this available mass to give a propellant plus tankage plus tankage structure mass. The structure to support the propellant and tankage was assumed to be 4 percent of the propellant and tankage masses, and the tankage mass was taken to be a fixed percentage of the propellant mass, given in table IV for each propellant. The tankage fractions for the hydrogen and ammonia systems were based on a study by Palaszewski (ref. 20). These tankage fractions actually include the propellant distribution systems as well as the tankage because the distribution system is not negligible for ammonia or hydrogen. Contingency and structure included in the Palaszewski study were eliminated in the current study as these masses had been included elsewhere. The hydrogen system was a ground-based, cryogenic storage system with a cylindrical tank while the ammonia system was a ground-based system with a spherical tank. The tankage fraction for hydrazine was assumed to be the same as that for ammonia. The tankage fractions for xenon and mercury propellants were based on the model in reference 8, modified so that the tankage fraction was not a function of the amount of propellant (that is, the small amount of additional mass which is required to account for Shuttle Orbiter crash loads was not included). The propellant distribution masses for xenon and mercury systems are small and hence were neglected in this study.

Based on the performance parameters given in table III and reference 19 the mission capabilities of the various propulsion systems could be calculated. The maximum velocity increment and maximum total impulse were calculated from the I_{sp} and the available propellant mass as a function of input power level. The final mass used to calculate the velocity increment included the propulsion system dry mass, tankage, tankage structure, the spacecraft, and the SP-100 masses. The maximum thrusting time was also calculated based on the maximum total impulse capability and thrust produced by each system.

RESULTS AND DISCUSSION

Figures 5 and 6 show the number of thrusters as a function of specific impulse for the resistojet, arcjet, and ion thruster systems at 150 and 300 kW, respectively. For each system the number of thrusters was reduced by approximately a factor of two for the 150 kW case compared to the 300 kW system. For the resistojet and arcjet systems the thruster input power was assumed to be constant for each propellant type, thus holding the number of thrusters constant for a given power level. For the ion thruster system the power input to each thruster increased as the specific impulse increased. Therefore, the number of ion thrusters required decreased as the specific impulse increased. It was recognized that, for most applications, the large number of ion thrusters required in the low specific impulse ion cases was unrealistic. However, it is difficult to make a judgment as to how many thrusters would be allowed for a realistic system. For the sake of completeness, consideration will be given to the medium to high specific impulse range for the ion thruster systems in the remainder of the study.

Figure 7 shows the propulsion system dry mass (which did not include propellant, tankage, or tankage structure) as a function of specific impulse for the three propulsion systems at 150 kW. The dry masses of each system increased by approximately a factor of two when the total input power was increased to 300 kW, corresponding to a twofold increase in the number of thrusters. From the figure it can be seen that the dry masses of the resistojet and arcjet systems were nearly identical at 840 kg. This dry mass was nearly the same for both the ammonia and hydrogen systems. As mentioned above, the number of thrusters did not change with propellant type, thus the propulsion system mass did not change. The dry masses of the 30 cm ion thruster systems ranged from 1.4 to 2.3 times that of the resistojet or arcjet systems, decreasing with increasing specific impulse. This difference in dry mass was due primarily to the larger number of thrusters required for the ion thruster systems. It should be noted, however, that at the higher specific impulses, where the ion thruster would probably be run, the dry masses of the resistojet, arcjet, and ion thruster systems were similar. For a 50 cm ion thruster system the dry mass ranged from 1.1 to 1.5 that of the resistojet or arcjet systems. Again, if the higher specific impulse cases were considered the dry masses of the three systems were found to be similar. Therefore, for the region of specific impulse of interest for near-term application, the dry masses of the resistojet, arcjet, and ion thruster systems were similar and in all cases a small fraction of the total mass available for the total propulsion system.

A distribution of the arcjet propulsion system dry mass is shown in figure 8. This mass distribution applies to the resistojet system as well since the dry masses of the systems are nearly the same. For the resistojet and arcjet systems the majority of the dry mass was thermal control for the PPU, with the power processor being the next largest component. For the 30 cm mercury and xenon ion thruster systems at 150 kW, shown in figures 9 and 10, respectively, the power processor mass formed the largest portion of the system, followed by the thruster/gimbal mass. At 300 kW the masses of the ion thruster systems increase; however, the mass distribution remains the same. The differences in the mass distributions of the three different thruster systems were primarily due to the larger number of thruster subsystems required for the ion thruster propulsion system as discussed above. The 50 cm ion system mass distribution was similar to the 30 cm ion system. The number of ion thrusters can be reduced by an increase in the total accelerating voltage or diameter, as shown in figure 11. However, these technology advances may lead to higher risk options. Details of this higher power operation can be found in reference 19.

From an examination of the distribution of the Shuttle payload mass a comparison could be made of the propulsion system mass and the masses of the other payload components. From figure 12 it can be seen that the mass available for the electric propulsion system was approximately 35 percent of the total Shuttle payload capability. The SP-100 reactor and spacecraft (S/C) made up 30 percent of the payload, while the ASE mass was 15 percent. The remainder of the payload was the chemical orbit transfer vehicle (11 percent) and contingency (8 percent). For the resistojet or arcjet systems at 300 kW total input power the mass available for the propulsion system was divided into dry mass (thruster, PPU, thermal control, etc.) and propellant/tankage/tankage structure mass. As can be seen in the figure, the dry mass was 1671 kg, or approximately 17 percent of the available mass. For the 150 kW power level the dry mass represented a slightly smaller portion of the propulsion system mass (approximately 9 percent) due to a reduction in the number of thruster subsystems required. System masses of the resistojet and arcjet systems are given in tables V and VI, respectively.

A payload distribution for the 30 cm xenon ion thruster system, shown in figure 13, gives a result much the same as that found for the resistojet and arcjet systems. For a 300 kW system at a specific impulse value of 4710 sec the propulsion system dry mass was 2515 kg, or approximately 26 percent of the mass available for the propulsion system. Table VII gives a detailed description of the component masses of the ion thruster systems. As discussed above, at the higher specific impulses the dry masses of the ion thruster systems were similar to those of the resistojet or arcjet systems. Thus, for most systems the dry mass was less than 30 percent of the entire mass available for the propulsion system. This result implies that, regardless of the electric propulsion system selected, the propulsion system dry mass was small compared to the masses of the other components shown in the Shuttle payload distribution.

In the analyses constraints were placed not only on the mass that could be placed in the Space Shuttle cargo bay but also on the volume available for the payload. Taking into account the volumes of the SP-100 payload (ref. 21), the Airborne Support Equipment, and the OTV, the volume available for the propulsion system was approximately 71 m^3. From calculations of tankage and thruster subsystem volumes it was determined that all systems considered fell within the Shuttle volume limits.

In addition to calculations of the propulsion system masses estimates of the maximum velocity increment and total impulse capability of each system were made. Because the system dry masses were found to be similar the mission capabilities of each system were determined primarily by the specific impulse and tankage fraction of each system. Figure 14 shows the maximum velocity increment for each propulsion system as a function of specific impulse given the assumptions stated earlier. It can be seen that the maximum ΔV for the resistojet system at 150 kW was 1550 m/sec with hydrazine, 1780 m/sec with ammonia, and 2890 m/sec with hydrogen. The arcjet offered a maximum velocity increment of 3760 sec with ammonia and 3620 m/sec with hydrogen. For the hydrogen arcjet there was a slight decrease in the velocity increment compared to the ammonia arcjet system. In this case the large percentage of tankage, 53 percent of the propellant mass for hydrogen compared to 24 percent for ammonia, caused a significant decrease in the propellant available and thus a reduction in velocity increment. For the resistojet the large increase in I_{sp}, from 380 sec with ammonia to 800 sec with hydrogen, overwhelmed the increase in tankage fraction and thus led to an increase in ΔV when a hydrogen resistojet was used instead of an ammonia system. At 150 kW the ion thruster system allowed for maximum mission velocity increments ranging from 15 000 to 21 000 m/sec for a 30 cm Hg system and from 17 000 to 23 000 m/sec for a 30 cm xenon system. The maximum ΔV increased modestly as the ion thruster size increased to 50 cm due to a decrease in the system dry mass. Also, the maximum ΔV decreased as the total input power increased to 300 kW due to an increase in the propulsion system dry mass. However, the primary factors affecting the velocity increment were the specific impulse and tankage fraction. In general, increases in specific impulse caused increases in ΔV.

Calculated values of the maximum total impulse are shown for selected cases in table VIII. Because the total impulse is dependent on the velocity increment the trends in this table correspond in general to those discussed above.

Finally, an assessment was made of the maximum thrusting time for each system. The maximum thrusting time was defined as the length of time to use all the available propellant while providing the nominal reactor output power to the electric propulsion system. The maximum thrusting time resulted from the maximum total impulse obtainable for each system. For an increased power level this thrusting time was significantly reduced because the mass flow increased and the mass calculated for the propellant decreased with an increase in power. Due to its relatively low specific impulse, the maximum thrusting time for a resistojet ranged from 3.9 to 19.0 days for a 150 kW system, as shown in figure 15. For an arcjet the maximum thrusting time, given in figure 16, was 46 days for an ammonia system and 47 days for a hydrogen system. An increase in power level to 300 kW reduced the maximum thrusting time by approximately a factor of 2 for each of these systems. For a 30 cm xenon ion system the thrusting times for a 150 kW power level ranged from 501 to 853 days and from 192 to 347 days for a 300 kW system, as indicated by figure 17. For the mercury ion system the thrusting time, given in figure 18, ranged from 357 to 575 days for a 30 cm thruster at 150 kW and from 131 to 237 days for a 300 kW system. In most cases use of the 50 cm thrusters increased the thrusting time slightly. A cross of the 30 and 50 cm curves in figure 17 was a result of the mismatch between the total input power of 150 kW and the total thruster system power. For the 30 cm xenon case at 4710 sec the thrust system input power was less than 150 kW due to the constraint that the number of thrusters be such that the thruster input power not exceed the total SP-100 reactor power output. The 50 cm thruster system, on the other hand, gave a total thruster input power close to 150 kW, thus reducing the thrusting time below the 30 cm case.

CONCLUSIONS

A study was conducted to compare resistojet, arcjet, and ion thruster systems for use as an active load on a flight demonstration of the SP-100 nuclear power system. The dry masses of each propulsion system were calculated based on a consistent set of assumptions and assessments were made of the mission capabilities of each system based on the capability of a single Shuttle launch. From the analyses it was found that the mass available for the electric propulsion system was approximately 35 percent of the Shuttle payload capability. A majority of the available mass for the electric propulsion system was propellant and associated tankage and structure. The ion thruster system had a dry mass that ranged from 1.1 to 2.3 times that of the resistojet and arcjet systems. However, for levels of specific impulse expected for near-term operation the dry masses of all the systems were similar and less than 20 percent of the available mass of the propulsion system. The maximum velocity increments of the systems were found to be up to 2890 m/sec, 3760 m/sec, and 23 000 m/sec for resistojet, arcjet, and ion thruster systems, respectively. The maximum thrusting times were up to 19 days for resistojet, 47 days for arcjet, and 853 days for ion thruster systems.

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TABLE I. - PROPULSION SYSTEM
MODEL DESCRIPTION

Thrust module
Thrusters (resistojet, arcjet, ion) Power processor Gimbal (fixed fraction of thruster mass) Propellant distribution Thermal control (power processors only) Structure
Interface module
Propellant Propellant storage Reconfiguration (cabling) Converter Thrust system controller Thermal control Structure

TABLE II. - SUMMARY OF MASSES AND EQUATIONS[a]

Thrust module
Thruster (MTHR):
25 kg resistojet (ref. 9)
3.1 kg arcjet (ref. 10)
11.4 kg 30 cm ion
20.4 kg 50 cm ion
Gimbal (MG):
MG = 0.34 MTHR
Total number of thrusters (N):
N = (NPPU x POWER)/PT
NPPU = Power processor efficiency
POWER = Total input power (150 or 300 kW)
PT = Thruster power = PD + PB + PLO
PD = Discharge supply power (kW)
PB = Beam supply power (kW) (ion only)
PLO = Low voltage supply power (kW) (ion only)
Total thruster/gimbal mass (MTGT):
MTGT = N(MTHR + MG)
Power processor (MPPUT):
Discharge supply:
MD = 2.5 $PD^{3/4}$ + 1.8 $PD^{1/2}$ + 0.1 PD + 3.0
Beam supply:
MB = 2.5 $PB^{3/4}$ + 1.8 $PD^{1/2}$ + 0.1 PB + 7.6 (ion)
Low voltage supply:
MLO = 8.0 (ion only)
Total power processor mass (MPPUT):
MPPUT = N(MD + MB + MLO)
Thermal control (MTC):
MTC = 27 POWER (1.0 - NPPU)
Thruster structure (MSTRT):
MSTRT = 0.31 N (MTHR + MG)
Interface module
Converter (MC):
MC = $PC^{3/4}$ + $PC^{1/2}$ + 0.1 PC + 0.9
PC = 0.08 N (converter power)
Controller (MTSC):
MTSC = 4.0
Reconfiguration unit (MRU):
MRU = 0.15 (POWER + PLD)
PLD = 3/22 POWER (discharge supply dissipated power)
PRU = N(POWER + PLD) (reconfiguration unit power)
Thermal control (MTHIM):
MTHIM = 27 (PLRU + PLC + PLTSC)
PLRU = 0.005 PRU (reconf. unit dissipated power)
PLC = 1/9 PC (converter dissipated power)
PLTSC = 0.015 (controller dissipated power)
Interface module mass (MIMP):
MIMP = 2 MRU + 2 MC + 2 MTSC
Housing structure (MSTRH):
MSTRH = 0.04 (MIMP + MTGT + MPPUT + MSTRT + MTC)

[a]All masses from reference 8 except as noted.

TABLE III. - THRUSTER SYSTEM DESCRIPTION

Thruster	Input power, kW	PPU efficiency	I_{sp}, sec	Thruster efficiency
Resistojet				
Hydrazine	65	[a]0.88	330	0.8
Ammonia	65	[a]0.88	380	0.5
Hydrogen	65	[a]0.88	800	0.8
Arcjet				
Ammonia	26	[a]0.88	800	0.4
Hydrogen	26	[a]0.88	1000	0.5
Ion (xenon, mercury)[b]				
30 cm	(c)	[a]0.88 [f]0.93 [g]0.625	(d)	(e)
50 cm	(c)	[a]0.88 [f]0.93 [g]0.625	(d)	(e)

[a]Discharge supply.
[b]Total PPU efficiency for ion systems is not constant as specific impulse varies as the ratio of beam to discharge power varies. The efficiency range is 0.91 to 0.93.
[c]Function of specific impulse (ref. 19).
[d]Calculated from net ion accelerating voltage (refs. 17 and 18).
[e]Obtained from experimental data (refs. 17 and 18).
[f]Beam supply.
[g]Low voltage supply.

TABLE IV. - TANKAGE FRACTIONS FOR VARIOUS PROPELLANTS

Propellant	Percent
Xenon	14.4
Mercury	2.0
Ammonia	24
Hydrazine	24
Hydrogen	53

TABLE V. - RESISTOJET COMPONENT MASSES

Propellant	NH_3, N_2H_4		H_2	
Total power, kW	150	300	150	300
Thruster/gimbal	67	134	67	134
Thermal control	486	972	486	972
Power processor (PPU)	162	325	162	325
Structure	53	106	53	106
Miscellaneous	76	141	76	141
Total dry mass, kg	844	1678	844	1678
Propellant	6759	6114	5478	4955
Tankage	1624	1467	2905	2626
Tankage structure	335	303	335	303
Total propulsion system mass,[a] kg	9562	9562	9562	9562

[a]Total propulsion system mass = Dry mass + Propellant + Tankage + Tankage structure

TABLE VI. - ARCJET COMPONENT MASSES

Propellant	NH_3		H_2	
Total power, kW	150	300	150	300
Thruster/gimbal	21	42	21	42
Thermal control	486	972	486	972
Power processor (PPU)	218	436	218	436
Structure	39	77	39	77
Miscellaneous	77	144	77	144
Total dry mass, kg	841	1671	841	1671
Propellant	6763	6120	5481	4959
Tankage	1623	1468	2905	2629
Tankage structure	335	303	335	303
Total propulsion system mass,[a] kg	9562	9562	9562	9562

[a]Total propulsion system mass = Dry mass + Propellant + Tankage + Tankage structure

TABLE VII. - ION SYSTEM COMPONENT MASSES

(a) 30 cm Xenon

Specific impulse, sec	3684		4710	
Total power, kW	150	300	150	300
Thruster/gimbal	321	657	199	412
Thermal control	315	645	289	600
Power processor (PPU)	760	1557	544	1129
Structure	163	332	108	225
Miscellaneous	83	158	77	149
Total dry mass, kg	1642	3349	1217	2515
Propellant	6657	5222	7014	5923
Tankage	959	752	1010	853
Tankage structure	304	239	321	271
Total propulsion system mass, kg	9562	9562	9562	9562

(b) 30 cm Mercury

Specific impulse, sec	2984		3810	
Total power, kW	150	300	150	300
Thruster/gimbal	397	810	260	519
Thermal control	320	652	309	618
Power processor (PPU)	881	1795	658	1316
Structure	195	398	136	272
Miscellaneous	85	162	83	154
Total dry mass, kg	1878	3817	1446	2879
Propellant	7243	5416	7651	6300
Tankage	145	108	153	126
Tankage structure	296	221	312	257
Total propulsion system mass, kg	9562	9562	9562	9562

TABLE VII. - Concluded.

(c) 50 cm Xenon

Specific impulse, sec	3684		4710	
Total power, kW	150	300	150	300
Thruster/gimbal	191	410	137	273
Thermal control	281	603	301	603
Power processor (PPU)	378	810	328	657
Structure	99	211	78	156
Miscellaneous	72	143	79	144
Total dry mass, kg	1021	2177	923	1833
Propellant	7179	6207	7261	6496
Tankage	1034	894	1046	936
Tankage structure	328	284	332	297
Total propulsion system mass, kg	9562	9562	9562	9562

(d) 50 cm Mercury

Specific impulse, sec	2984		3810	
Total power, kW	150	300	150	300
Thruster/gimbal	246	519	164	328
Thermal control	294	622	294	588
Power processor (PPU)	443	935	356	712
Structure	121	256	88	176
Miscellaneous	76	147	77	142
Total dry mass, kg	1180	2479	979	1946
Propellant	7902	6677	8091	7179
Tankage	158	134	162	144
Tankage structure	322	272	330	293
Total propulsion system mass, kg	9562	9562	9562	9562

TABLE VIII. - TOTAL IMPULSE FOR VARIOUS THRUSTER SYSTEMS

Propellant	I_{sp}, sec	Total input power, kW	Total impulse, N-sec
Resistojet			
N_2H_4	330	150	21.9×10^6
	330	300	19.8
NH_3	380	150	25.2
	380	300	22.8
H_2	800	150	42.9
	800	300	38.8
Arcjet			
NH_3	800	150	53.0×10^6
	800	300	48.0
H_2	1000	150	53.7
	1000	300	48.6
Ion			
Hg	2984	150	210×10^6
	2984	300	158
	3810	150	290
	3810	300	235
X_e	3684	150	240
	3684	300	189
	4710	150	324
	4710	300	273

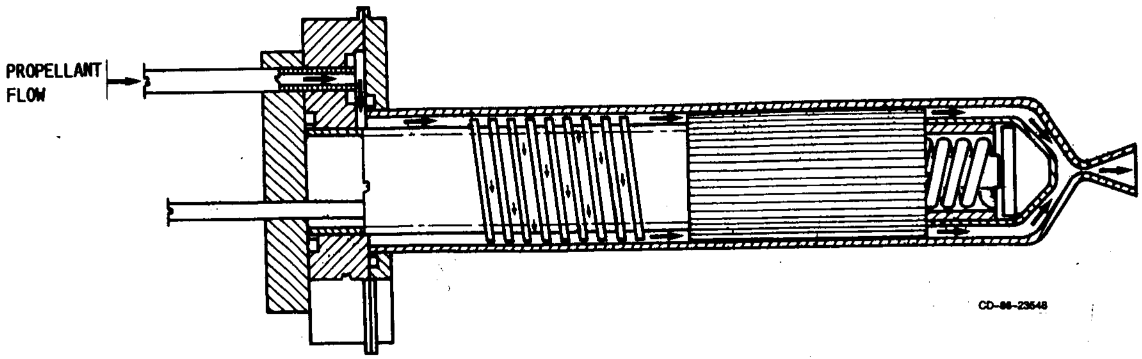

FIGURE 1. - RESISTOJET THRUSTER (FROM REF. 22).

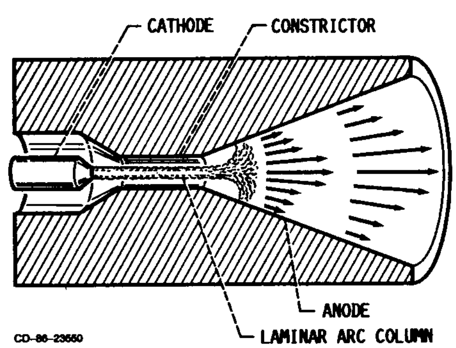

FIGURE 2. - CONVENTIONAL ARCJET THRUSTER.

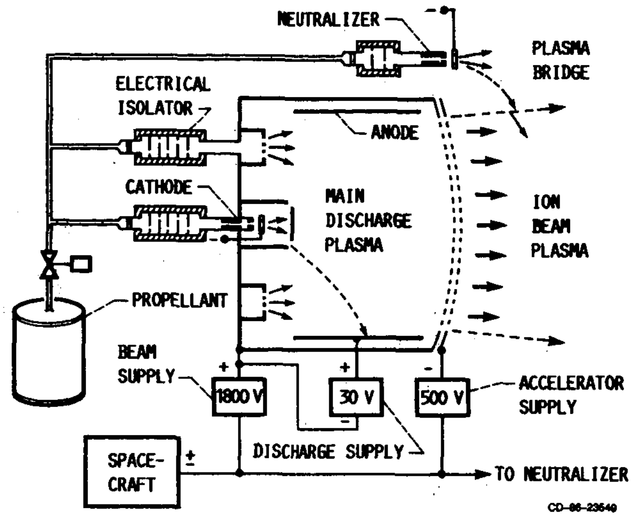

FIGURE 3. - ION THRUSTER.

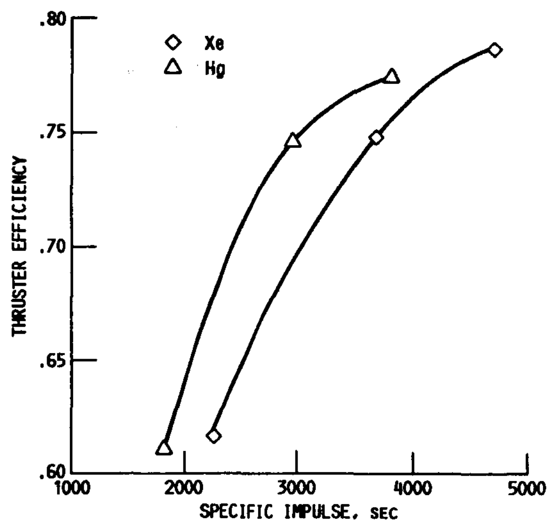

FIGURE 4. - EFFICIENCY VERSUS SPECIFIC IMPULSE FOR MERCURY AND XENON ION THRUSTERS.

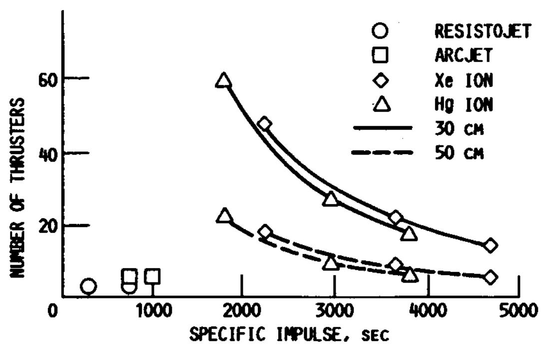

FIGURE 5. - NUMBER OF THRUSTERS VERSUS SPECIFIC IMPULSE AT 150 kW TOTAL INPUT POWER.

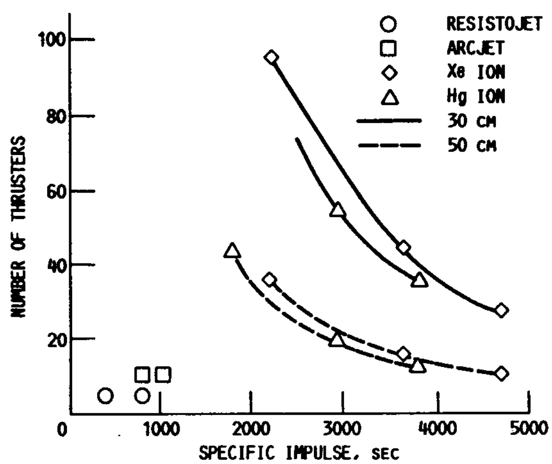

FIGURE 6. - NUMBER OF THRUSTERS VERSUS SPECIFIC IMPULSE AT 300 kW TOTAL INPUT POWER.

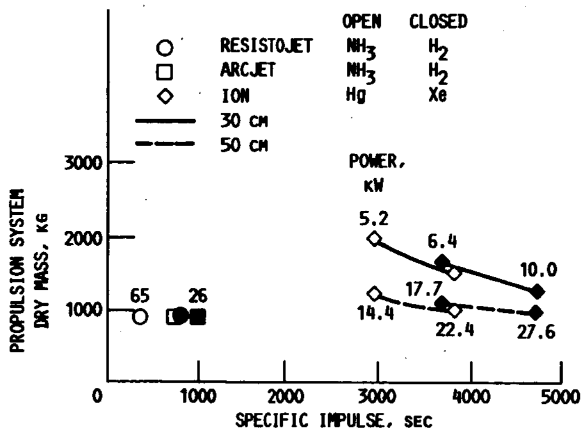

FIGURE 7. - PROPULSION SYSTEM DRY MASS VERSUS SPECIFIC IMPULSE AT 150 kW TOTAL INPUT POWER. DRY MASS = THRUSTER/GIMBAL + PPU + RECONFIGURATION UNITS + CONTROLLERS + CONVERTERS + THERMAL CONTROL + STRUCTURE.

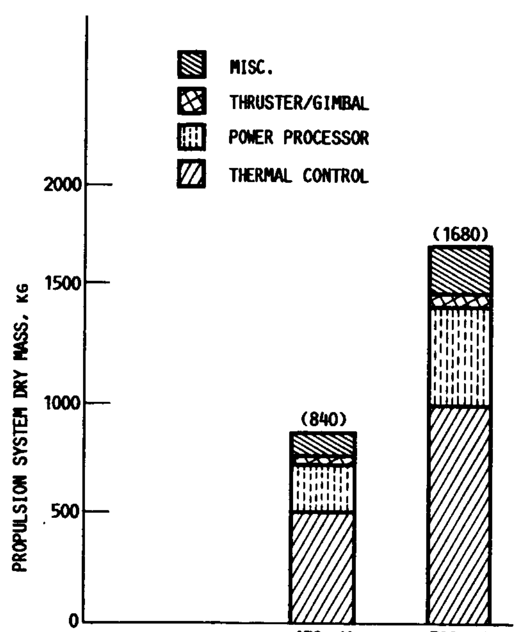

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FIGURE 8. - DISTRIBUTION OF THE ARCJET PROPULSION SYSTEM DRY MASS.

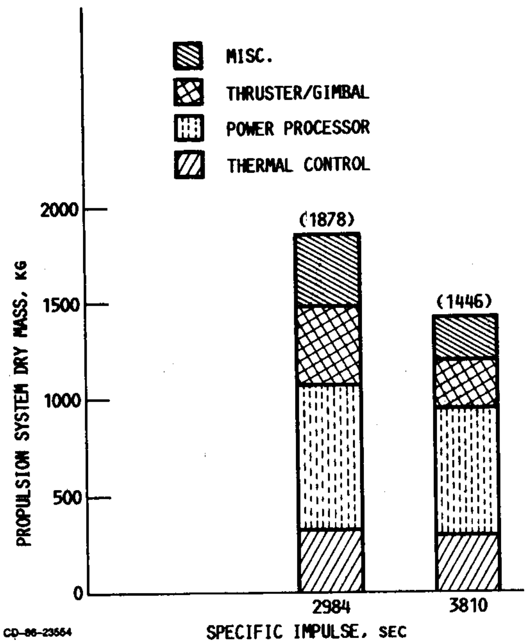

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FIGURE 9. - DISTRIBUTION OF THE 30 CM MERCURY ION PROPULSION SYSTEM DRY MASS AT 150 KW.

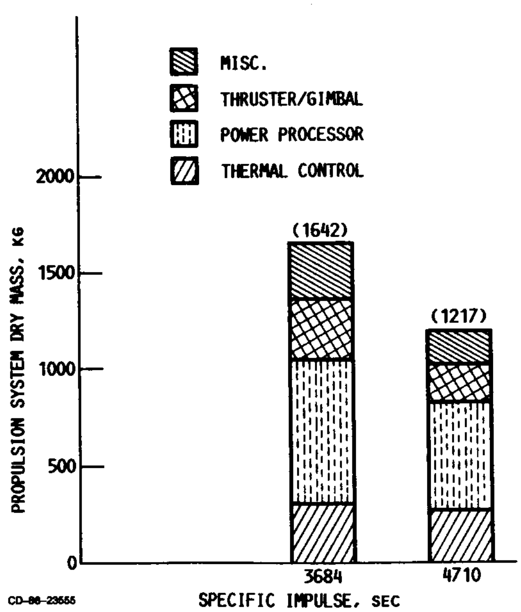

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FIGURE 10. - DISTRIBUTION OF THE 30 CM XENON ION PROPULSION SYSTEM DRY MASS AT 150 KW.

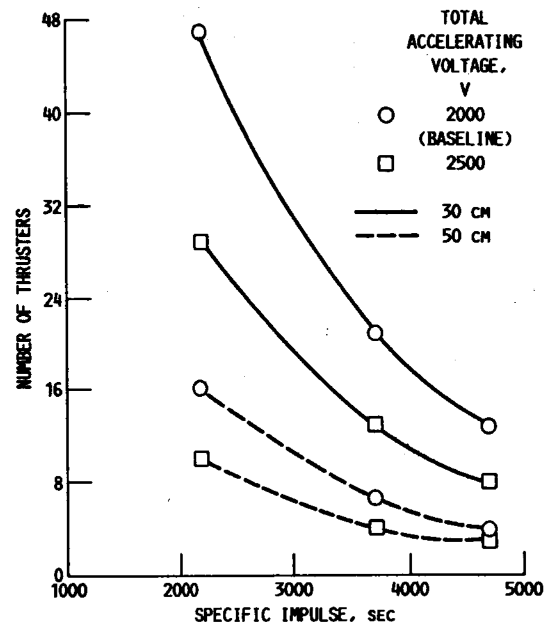

FIGURE 11. - NUMBER OF THRUSTERS VERSUS SPECIFIC IMPULSE FOR A XENON ION THRUSTER SYSTEM AT VARIOUS TOTAL ACCELERATING VOLTAGES. TOTAL INPUT POWER = 150 KW.

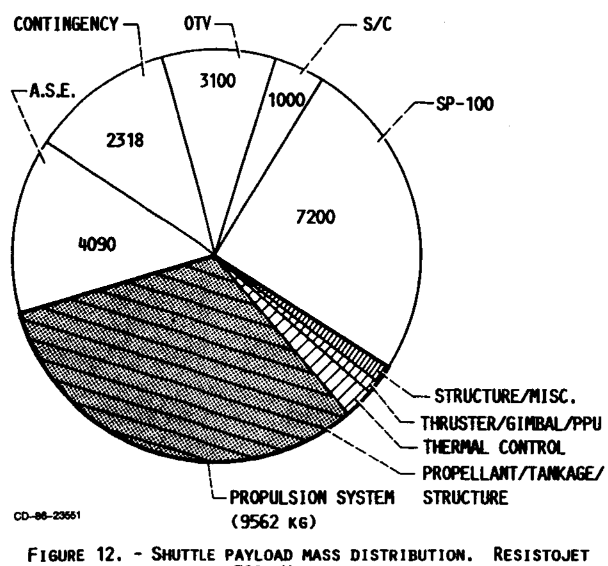

FIGURE 12. - SHUTTLE PAYLOAD MASS DISTRIBUTION. RESISTOJET AND ARCJET SYSTEMS, 300 KW TOTAL INPUT POWER, TOTAL MASS = 27270 KG AT 296 KM.

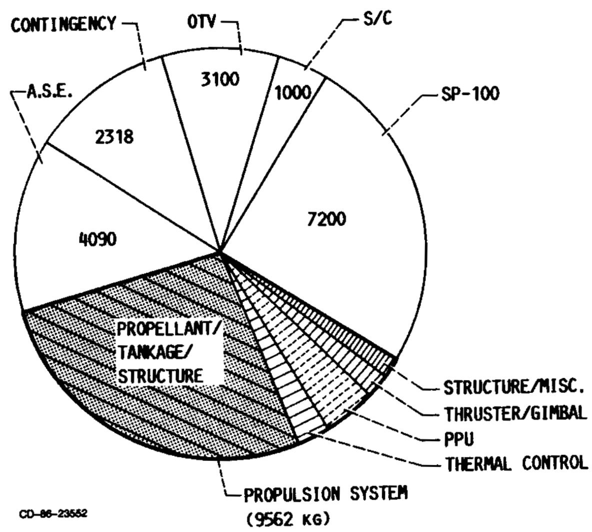

FIGURE 13. - SHUTTLE PAYLOAD MASS DISTRIBUTION. 30 CM XENON ION, ISP = 4710 SEC, 300 KW TOTAL INPUT POWER, TOTAL MASS = 27270 KG AT 296 KM.

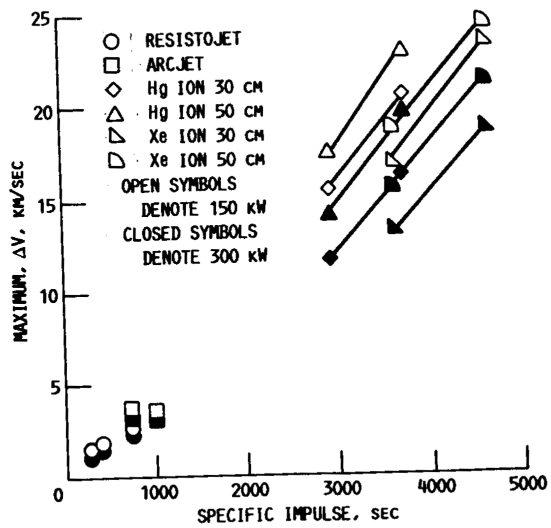

FIGURE 14. - MAXIMUM VELOCITY INCREMENT VERSUS SPECIFIC IMPULSE FOR ARCJET, RESISTOJET, AND ION THRUSTER SYSTEMS.

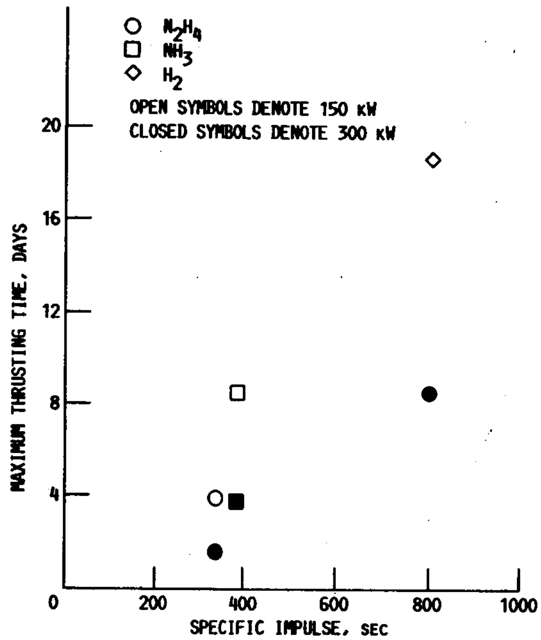

FIGURE 15. - MAXIMUM THRUSTING TIME VERSUS SPECIFIC IMPULSE FOR RESISTOJET SYSTEMS.

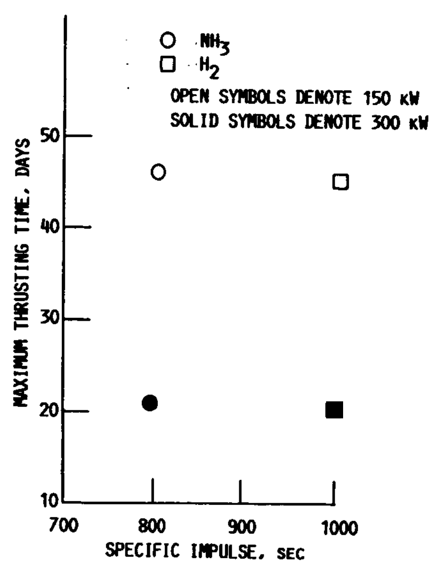

FIGURE 16. - MAXIMUM THRUSTING TIME VERSUS SPECIFIC IMPULSE FOR ARCJET SYSTEMS.

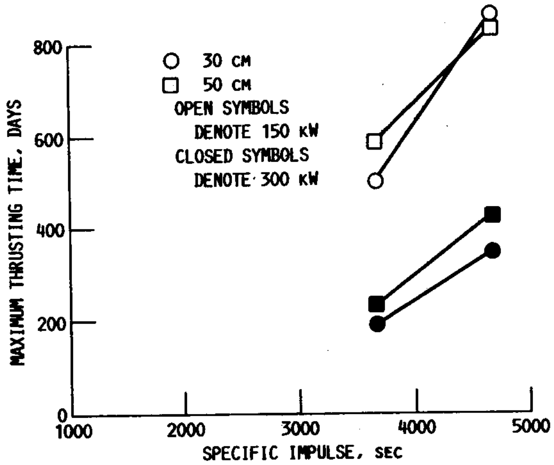

FIGURE 17. - MAXIMUM THRUSTING TIME VERSUS SPECIFIC IMPULSE FOR XENON ION THRUSTER SYSTEMS.

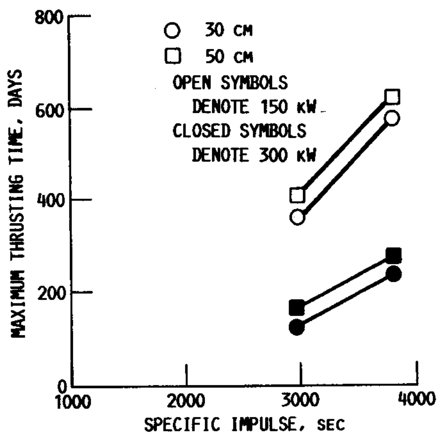

FIGURE 18. - MAXIMUM THRUSTING TIME VERSUS SPECIFIC IMPULSE FOR MERCURY ION THRUSTER SYSTEMS.

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15. Supplementary Notes

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16. Abstract

Analyses were performed to characterize and compare electric propulsion systems for use on a space flight demonstration of the SP-100 nuclear power system. The component masses of resistojet, arcjet, and ion thruster systems were calculated using consistent assumptions and the maximum total impulse, velocity increment, and thrusting time were determined, subject to the constraint of the lift capability of a single Space Shuttle launch. From the study it was found that for most systems the propulsion system dry mass was less than 20 percent of the available mass for the propulsion system. The maximum velocity increment was found to be up to 2890 m/sec for resistojet, 3760 m/sec for arcjet, and 23 000 m/sec for ion thruster systems. The maximum thrusting time was found to be 19, 47, and 853 days for resistojet, arcjet, and ion thruster systems, respectively.

17. Key Words (Suggested by Author(s))	18. Distribution Statement
SP-100; Electric propulsion; Resistojet; Arcjet; Ion thrusters; Propulsion	Unclassified - unlimited STAR Category 20

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