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Primary Propulsion of Electrothermal, Ion, and Chemical Systems for Space-Based Radar Orbit Transfer

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PRIMARY PROPULSION OF ELECTROTHERMAL, ION, AND CHEMICAL SYSTEMS

FOR SPACE-BASED RADAR ORBIT TRANSFER

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SUMMARY

An orbit transfer mission concept has been studied for a Space-Based Radar (SBR) where 40 kW required for radar operation is assumed available for orbit transfer propulsion. Arcjet, pulsed electrothermal (PET), ion, and storable chemical systems are considered for the primary propulsion. Transferring two SBR per shuttle flight to 1112 km/60° using electrical propulsion systems offers an increased payload at the expense of increased trip time, up to 2000 kg each, which may be critical for survivability. Trade-offs between payload mass, transfer time, launch site, inclination, and height of parking orbits are presented.

BACKGROUND

A large number of high power and high inclination Low Earth Orbt (LEO) spacecraft have been proposed. The net spacecraft mass which can be delivered is a sensitive function of the characteristics of both the shuttle delivery capability and the upper stage propulsion technology. The Space-Based Radar (SBR) is a concept which may represent a class of the future missions. This paper presents preliminary results of an analysis of the impact of propulsion technology including a typical chemical system and a range of electrical propulsion options.

The baseline concept (ref. 1) is to launch two SBR spacecraft to a shuttle parking orbit with a single flight as sketched in figure 1. Each SBR is then deployed and transferred to a higher mission orbit by a dedicated propulsion system. This study includes trade-offs among different propulsion systems, shuttle parking orbits (278 to 556 km), and inclinations (28.5° to 57°) from Eastern Test Range (ETR) or 70° from Western Test Range (WTR), as shown in figure 2. The 70° inclination is the closest to the 61° destination which can be reached by the shuttle from WTR without a disposal problem for the external tank.

ASSUMPTIONS

The initial mass (M_0) of each SBR delivered by the shuttle is plotted in figure 3 as a function of the parking orbit inclination for both 278 and 556 km altitudes. Two assumptions have been made: (1) the shuttle payload capability is based on the 109 percent SSME, (2) a mass of 3180 kg is used for supporting structure and mission reserve per shuttle flight.

The payload mass of SBR (M_D) in the final orbit is assumed to be:

$$M_D = M_O - M_P - M_T$$

Where M_P is the propellant mass and M_T is the propulsion system dry mass as summarized in table I (to be discussed later). The payload mass includes the 40 kW power supply required for on-orbit radar operation and batteries to allow operation (including thrusting) during occultation.

To calculate propellant usage, different thrusting strategies were assumed for chemical and electrical propulsion systems. A multi-burn Hohmann transfer was used for chemical thruster while the solar arrays of the SBR remained in a stowed position. However, the solar arrays had to be deployed at the parking orbit to supply an assumed 40 kW power for a continuous low thrust from the electrical thrusters. Due to the low thrust, the delta V for the electric propulsion missions was up to 35 percent higher than for the chemical system. We assumed a NASA nominal atmospheric model in the worst case year (1992). The drag was important under an altitude of 600 km and was included in the study.

PROPULSION SYSTEMS

Three electric propulsion systems including a state-of-the-art ion system, a near-term arcjet system, and advanced pulsed electrothermal (PET) system are considered, along with a storable bipropellant chemical system.

Arcjet Thruster

The arcjet is a type of electrothermal thruster which utilizes an electric arc to heat a gaseous propellant and produces the thrust through a nozzle. The schematic of a typical arcjet is shown in figure 4. Most of the performance data for arcjets were obtained during the 1960's and are documented in the literature (refs. 2 to 4). A near-term arcjet system is considered with rather conservative performance, based roughly on the 1960's technology: an Isp of 650 sec with an efficiency of 0.30, using ammonia propellant. Both the Isp and the efficiency are expected to improve as a result of current research. However, the life time is still to be demonstrated beyond the approximate 500 hr attained in early tests.

Since the detail of an arcjet system on a flight prototype level is currently unavailable, the system mass can only be estimated. Component masses that vary with power input include the power processor, the thermal rejection system, and the rest of the thruster module, which includes the arc chamber and nozzle. The power processor consists of a single power supply with the associated control system. The specific mass of such supply has been previously estimated (ref. 5) and assumed here to be about 2.1 kg/kW.

For ammonia at 650 sec of Isp, a major fraction (about 55 percent) of the energy input is lost in frozen flow losses. The thermal loss of about 12 percent was assumed along with a small fraction of 3 percent lost in the divergent nozzle passage. The specific mass of the thermal rejection system is assumed (ref. 6) to be 27 kg/kW of dissipated heat. Without discriminating different parts of heat rejection a conservative specific mass is thus assumed to be

3.2 kg/kW of input power. The rest of the thruster module is estimated to have a specific mass of about 1.9 kg/kW which is based, in part, upon reference 7. Thus, the total specific mass for the arcjet is estimated to be 7.2 kg/kW which should be obtainable in the near term.

Pulsed Electrothermal Thruster

The Pulsed Electrothermal (PET) Thruster, shown schematically in figure 5, is an advanced thruster being investigated by NASA and industry (GT-Devices). As electric arc is pulsed in a small chamber and heats a propellant to produce high pressure and temperature gases. The gases are then accelerated by a nozzle and are exhausted at high velocity. The high pressures attained during the pulse results in recovery of ionizational energy that would normally be lost. Efficiencies of 0.37 to 0.56 at Isp of 1000 to 1750 sec have been measured (ref. 8) in single-shot experiments using polyethylene propellant. A PET thruster system using water as the propellant may ease the feed problem encountered with solid propellant. Table II shows the component masses of a proposed system design for a 40 kW PET thruster (ref. 9). The power dependent components contribute about 8 kg/kW. Thrust efficiency of 0.5 at 1000 sec Isp is expected from calculations for this system. Experimental verification of this advanced thruster performing in a multi-shot mode is to be demonstrated under current research. System performance under long duration needs further verification.

Ion Thruster

The mercury ion thruster, shown schematically in figure 6, is recognized as having a relatively known technology. As shown in table III, the ion thrusters work at higher Isp, but with higher specific masses (ref. 10). Recent improvements, especially in the power processor, have significantly reduced the specific masses. The state-of-the-art mercury ion thruster module of 3000 sec Isp and 0.632 efficiency has the component masses shown in table IV. Not included in this study are other propellant options using inert gases such as xenon, which are of current interest because of their potential for minimizing contamination concerns and certain technical characteristics such as power processor simplification and thruster quick start capability. The performance and system level characteristics are expected to be close to those of the mercury thruster (ref. 6).

Chemical System

A storable bi-propellant MMH/N_2O_4 system at 290 sec of Isp is used as the baseline system. This system is assumed to be the state-of-the-art, based on the proven shuttle reaction control system technology. Performance can be improved to greater than 310 sec of Isp by addition of higher area ratio nozzle, reduction in the fuel film cooling, and/or increase in the combustion chamber length. Continuous improvements of the technology are expected to increase the Isp up to about 340 sec.

A summary of all the propulsion systems considered in this study is listed in table I.

RESULTS

Typical results of SBR payload mass reaching the final orbit of 1112 km at 61° are plotted versus trip (transfer) time as shown in figures 7 to 9 for the case of 278 km parking orbits. Data points are calculated for different types of propulsion systems. For each set of data the inclination of the parking orbit is varied from 28.5° to 57°, launched from ETR, and also 70° from WTR.

Baseline Chemical Propulsion

As shown before in figure 3, the maximum shuttle delivery capability is about 15 000 kg of initial mass for each SBR of the two per shuttle flight to 278 km parking orbit at 28.5° launched from ETR. Using the baseline chemical system the payload mass that can be transferred to the final orbit of 1112 km at 61° in only about 2500 kg (fig. 7), because a large amount of propellant is necessary for changing the inclination. Decreasing the inclination change during orbit transfer will save orbit transfer propellant at the expense of launching to a higher inclination parking orbit with lower delivery capability. However, the net effect is to increase the payload mass up to a total of about 8300 kg from the parking orbit at 57°, launched from ETR. Launched from WTR the payload mass can only reach about 5800 kg after orbit transfer from the parking orbit at 70°, primarily due to lower shuttle lift capability from WTR. The trip time for using this chemical system is normally within 1/2 to 2-1/2 days.

Electrothermal Propulsion Systems

Propulsion at higher Isp and lower thrust (arcjet and PET) generally means less propellant usage with longer trip time. The arcjet system offers the payload mass of about 3000 to 8700 kg from parking orbit at 28.5° to 57°, launched from ETR, with trip time longer than 34 days. Note that the slope in figure 7 is less steep for arcjet relative to chemical systems, as a result of small increase in payload mass, but longer trip time. This trend is continued changing to the next higher Isp thruster-PET, which enables the payload mass in a range of 6000 to 9500 kg from the parking orbits at 28.5° to 57°, launched from ETR. Trip time is in a similar range of arcjet. If launched from WTR to 70° inclination, the payload mass can be transferred is about 6100 kg by arcjet and 7100 kg by PET, with a trip of 51 days.

Ion Propulsion System

Launching from ETR to a lower inclination parking orbit within 28.5° to 57°, the advantage of larger shuttle lift capability is offset by the increased amount of propellant for the larger plane change to 61°, while using either the arcjet, PET, or chemical propulsion system as shown above. However, for an ion system of high enough Isp, the amount of propellant is small enough that advantage can be taken of the larger shuttle lift capability to lower inclination orbits, as shown in figure 8. Also note that the initial altitude has to be raised a little to lower the drag such that the ion thruster can produce a positive thrust. Here the payload mass is between 9800 and 10 600 kg; the mass margin is up to above 2000 kg over chemical system at the expense of long trip

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time, 110 to 620 days. Launching from WTR to 70° inclination parking orbit (closest to 61° without tank disposal problem) offers about 8000 kg of payload mass in 160 days.

Alternate

The effect of a higher (556 km) parking orbit was also analyzed (fig. 9). In general, this resulted in much lower payload mass as a direct consequence of the largely reduced shuttle capability (fig. 3). The trends of payload mass as a function of transfer time and technology were, however, similar to those presented for the lower parking orbit.

Finally, the effect of increased final orbit and higher performance chemical propulsion was briefly evaluated (table V). The payload mass of SBR transferred to 1112 km at 61° from 278 km at 57° would increase 380 kg for chemical propulsion using advanced 340 sec Isp bipropellant. The advantage of baseline arcjet thruster would then be marginal. However, if the final orbit altitude is higher (1853 to 4077 km), the payload mass margin of arcjet is between 1040 to 1750 kg over bipropellant chemical engine at 290 sec Isp and between 590 to 1150 kg over the advanced chemical engine at 340 sec Isp, as shown in table V. Although a more complete parametric study is not within the scope of this paper, this table does indicate the increased benefits of using electrical thrusters for transferring SBR lie on these missions to higher final orbits.

CONCLUSION

A baseline chemical propulsion system is shown capable of transferring two SBR (8300 kg each) to an orbit of 1112 km at 61° from one shuttle flight. The parking orbit is 278 km at 57° launched from ETR. Alternatively, the electric propulsion systems with higher Isp offer the potential for heavier SBR (8700 to 10 600 kg) and the choice of the parking orbit inclination (28.5° to 57°) launched from ETR at the expense of longer trip time (34 to 620 days). If launched from WTR to the closest parking orbit inclination (70°) to 61°, the electric propulsion systems also offer the option of heavier SBR (6100 to 8000 kg) at longer trip time (51 to 160 days) to the baseline chemical system (5800 kg). Also discussed are the possible benefits of using electrical thrusters for transferring SBR to higher final orbits and the payload range of using the advanced chemical system - bipropellant at 340 sec of Isp.

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TABLE I. – SUMMARY OF THE PROPULSION SYSTEMS

[Where G is the system mass per unit power,
and A is the tankage mass fraction. An
the propulsion system dry mass in kilogram
is $M_T = G \times P + A \times M_p$. Where P is
the input power in kW, and Mp is the
propellant mass in kilogram.]

System	Arcjet	PET	Ion	Chemical
Propellant	NH_3	H_2O	Hg	MMH/N_2O_4
Isp, sec	650	1000	3000	290
Thrust, N	[a]3.77	[a]4.08	[a]1.72	4000
System efficiency	0.30	0.50	0.632	--------
G, kg/kW	7.2	8.0	14.3	--------
A	0.20	0.16	0.04	0.084

[a]For 40 kW.

TABLE II. - COMPONENT MASSES OF A 40 kW PET SYSTEM

```
Propellant handling, kg . . . . . . . . . . . . . . . . 131
    Water tank and holdup, 4 atm . . . . . . . . . . . 71
    He tankage, 230 atm . . . . . . . . . . . . . . . . 24
    Main valve, regulator, and filter . . . . . . . . . . 6
    Thruster filters (20), valves (20), and lines . . . 30
Power conditioner, kg . . . . . . . . . . . . . . . . . 181
    dc-dc chargers, 5 kV (2) . . . . . . . . . . . . . . 100
    Capacitors, 10 J (20) . . . . . . . . . . . . . . . 16
    Inductor, wiring, insulation . . . . . . . . . . . . 10
    Thruster connect latching relays (20) . . . . . . . 30
    PFN switches, DPPT (20) . . . . . . . . . . . . . . 25
Thruster and thermal control, kg . . . . . . . . . . . 45
Other structures, kg . . . . . . . . . . . . . . . . . 40
Total, kg . . . . . . . . . . . . . . . . . . . . . . . 397
```

TABLE III. - ION THRUST SYSTEM CHARACTERISTICS

System parameter	Flight system			Thrust module	
	SERT II	IAPS	SEPS 1979	1985 projections	
Total efficiency	0.59	0.36	0.62	0.54 to 0.75	0.56 to 0.75
Specific impulse, sec	4200	2550	3020	2000 to 6500	2500 to 8000
Thrust/thruster, mN	28	5	130	194 to 618	194 to 618
Power/thrust unit,[a] kWe	0.98	0.175	3.1	3.5 to 26	4.2 to 32
Demonstrated thruster life, hr	3800	15 000	5000	- ----------	-----------
Total impulse, 10^6 Ns	0.6	0.13	7	10 to 33	10 to 33
Specific mass,[b] kg/kWe	~35	103	22	12.5 to 5.6	11.3 to 5.3
Thruster dia., cm	15	8	30	30	30
Propellant	Hg	Hg	Hg	Hg	Xe

[a]Power to power processor.
[b]Includes thruster, power processor, thermal control, gimbals, structure; excludes power source, tankage, transmission line.

TABLE IV. – MERCURY ION THRUSTER MODULE SYSTEM

Power, kW	6.0
System mass, kg	85.7
PPU	28.1
Thermal control . . .	16.2
Thruster	11.0
Gimbal	3.4
Miscellaneous	7.0
Interface	20

TABLE V. – FINAL PAYLOAD MASS OF SBR AT HIGHER ALTITUDE

Initial orbit/ inclination	Final orbit/ inclination	Arcjet	Chemical, 290 sec	Chemical, 340 sec
278 km/57°	1853 km/61°	8570	7530	7980
	2594 km/61°	8220	6860	7370
	4077 km/61°	7550	5800	6400
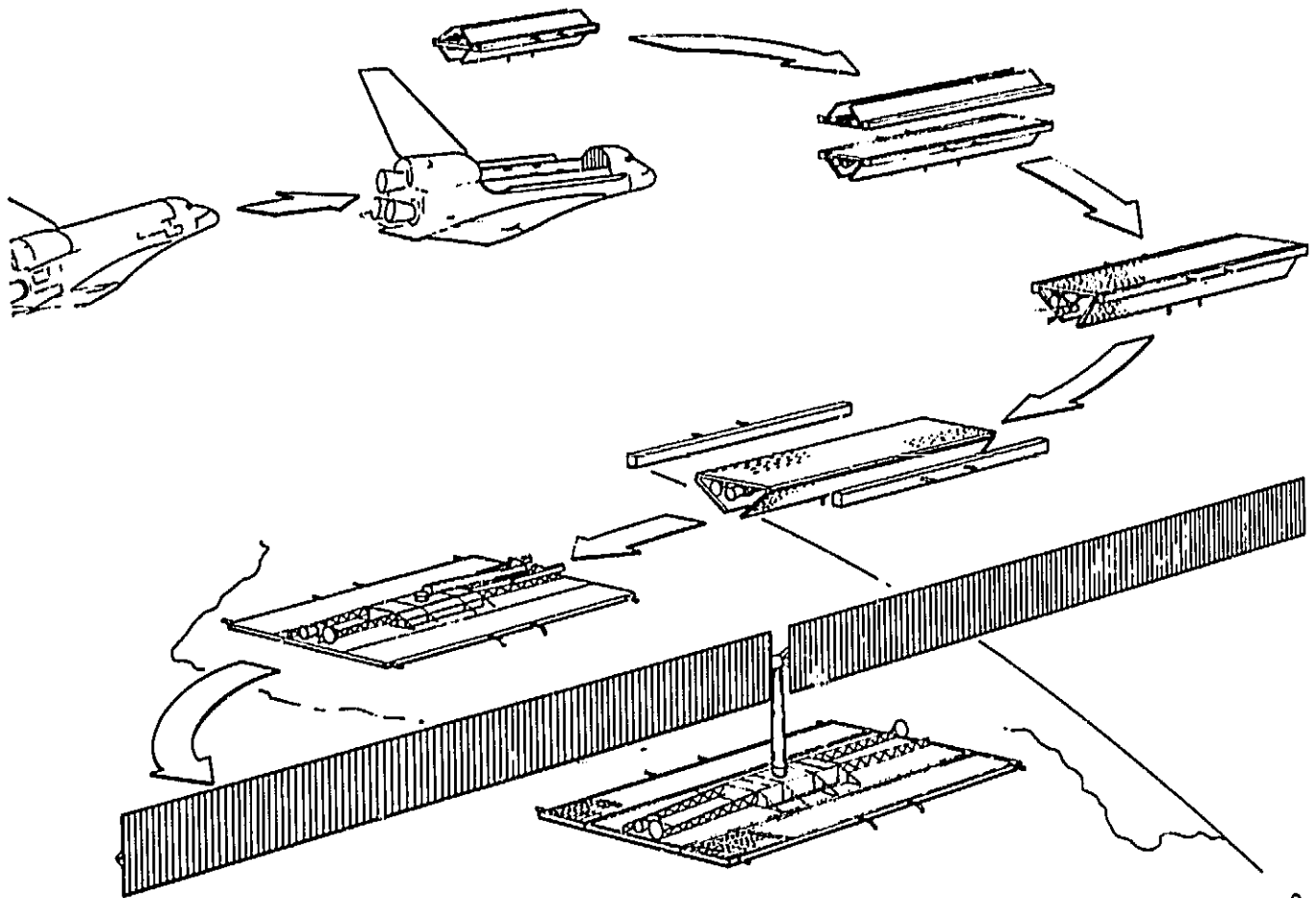
Fig. 1. – SBR separation and deployment. Shuttle parking orbit, 278 km/556 km; spacecraft final orbit, 1112 km, 61°.
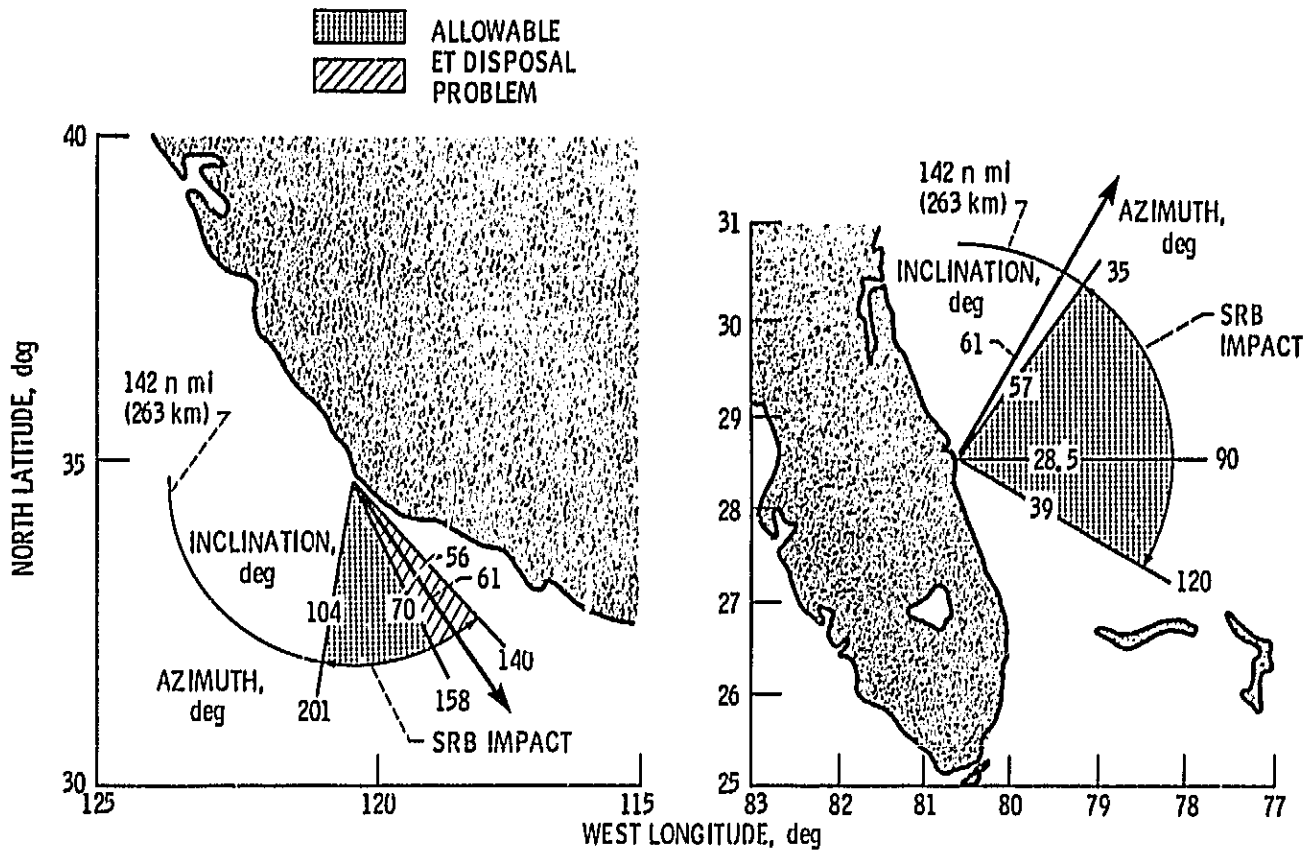
(a) Vandenberg Air Force Base (b) Kennedy Space Center

Fig. 2. - Launch azimuth and inclination limits from VAFB (WTR) and KSC (ETR).
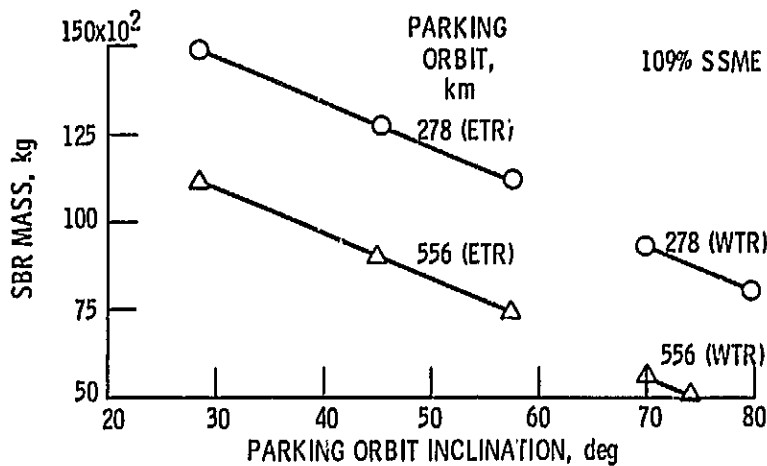
Fig. 3. - Initial mass of SBR spacecraft that can be delivered to parking orbits of two altitudes and various inclinations launched from ETR or WTR.
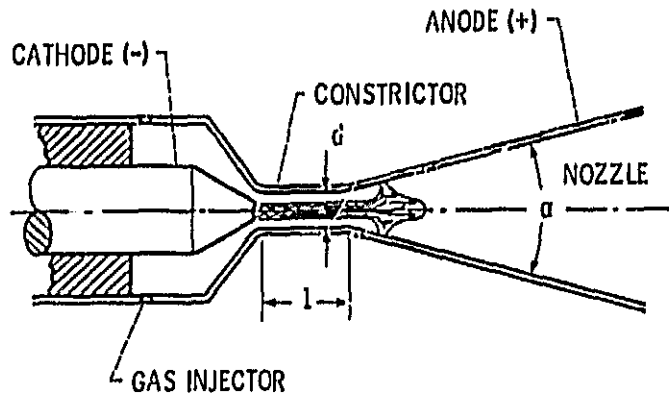
CATHODE (-)

ANODE (+)

CONSTRICTOR

NOZZLE

GAS INJECTOR

Fig. 4. - Schematic of a typical arcjet.

CONTAINMENT TUBE

12° NOZZLE

CATHODE (-)

CONSUMABLE LINER

POLYETHYLENE LINER

ANODE (+)

S

HV POWER SUPPLY

Fig. 5. - Schematic of PET thruster.

BACKGROUND PRESSURE, $\leq 1 \times 10^{-5}$ torr

NEUTRALIZER

PLASMA BRIDGE

ANODE

KEEPER, 14 V

ISOLATOR

Hg^0

VAPORIZER (300 °C AND 250 torr)

CATHODE (1 torr)

MAIN DISCHARGE PLASMA (10^{-4} torr)

$1e^{-1}$

Hg^+ ION BEAM PLASMA

Hg^{+1}

$1e^{-1}$

$5e^{-1}$

LIQUID MERCURY

1500 torr

NITROGEN

$5e^{-1}$

$6e^{-1}$

1100 V

37 V

500 V

0.005 e^{-1}

SPACE CRAFT

± VOLTAGE BIAS

$1e^{-1}$

$1e^{-1}$

TO NEUTRALIZER

$1e^{-1}$

Fig. 6. - Schematic of mercury ion thruster.

Fig. 4. - Schematic of a typical arcjet.

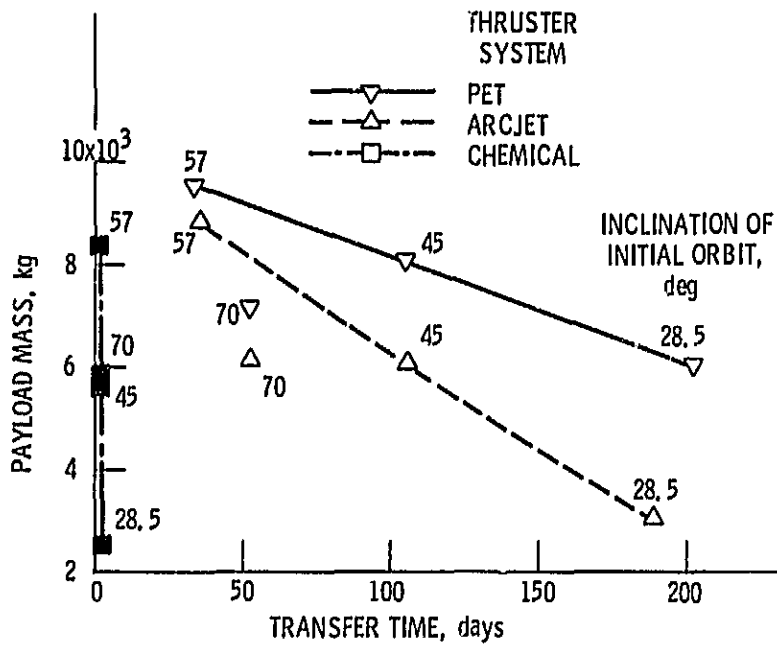Fig. 5. - Schematic of PET thruster.
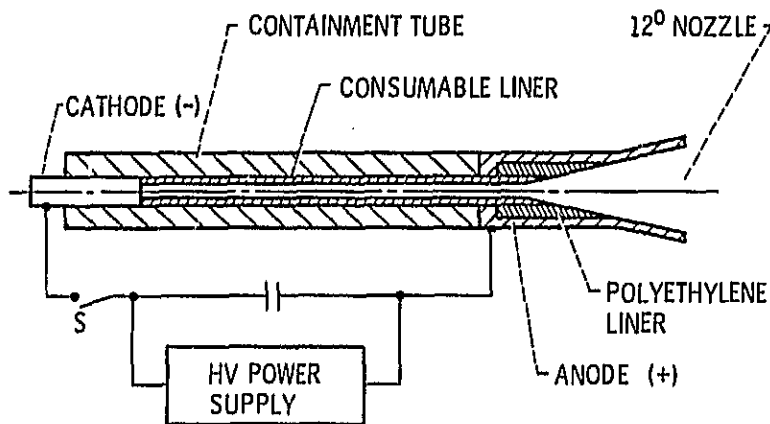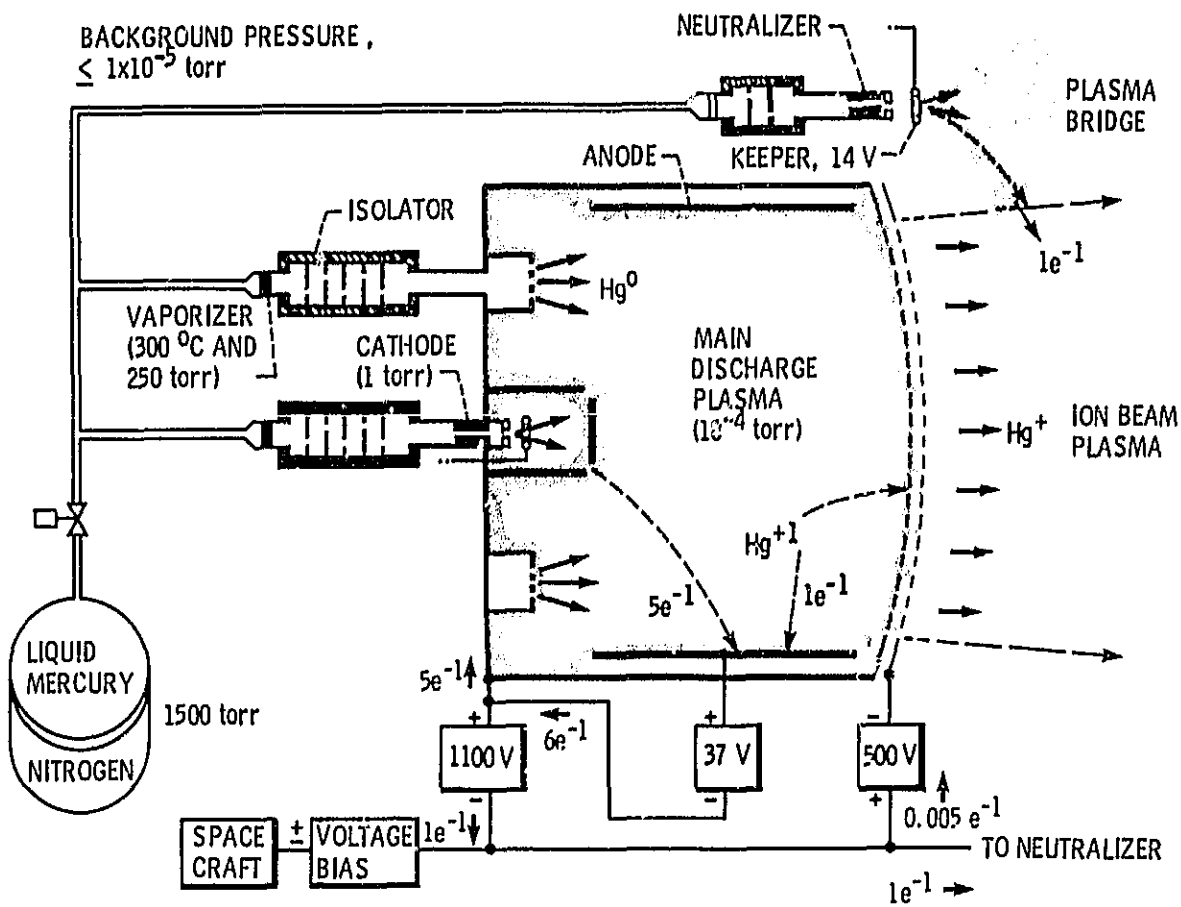
Fig. 6. – Schematic of mercury ion thruster.

Fig. 7. - Payload mass versus transfer time. Initial orbit, 278 km; inclination, 28.5^0 and 70^0; final orbit, 1112 km; inclination, 61^0.
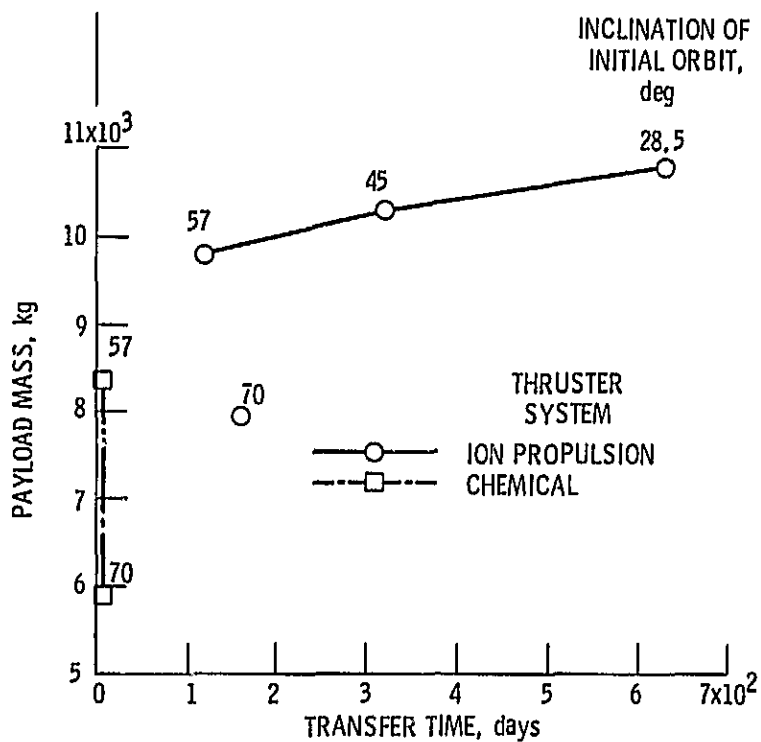

Fig. 8. - Payload mass versus transfer time. Initial orbit, 298 km; final orbit, 1112 km; inclination, 61^0.

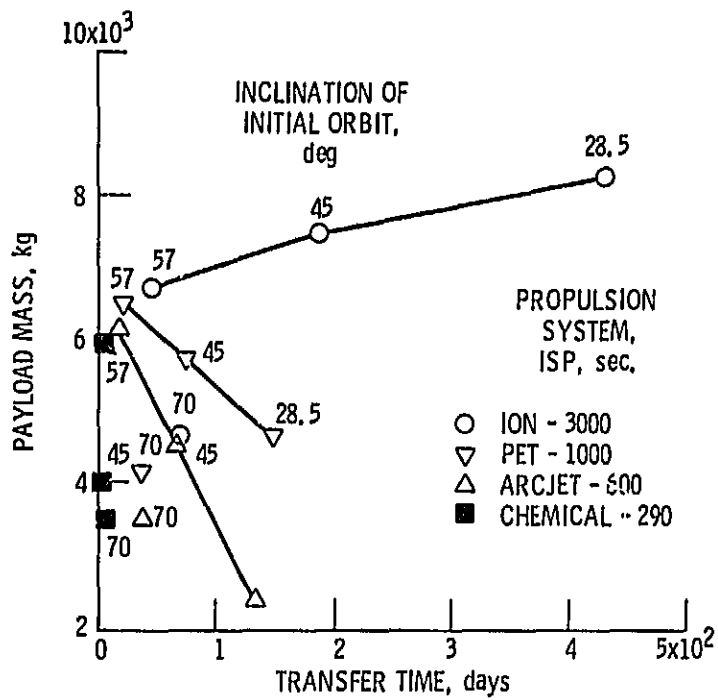Fig. 9. - Payload mass versus transfer time. Initial orbit, 556 km; final orbit, 1112 km; inclination, 61°.

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15. Supplementary Notes

16. Abstract

An orbit transfer mission concept has been studied for a Space-Based Radar (SBR) where 40 kW required for radar operation is assumed available for orbit transfer propulsion. Arcjet, pulsed electrothermal (PET), ion, and storable chemical systems are considered for the primary propulsion. Transferring two SBR per shuttle flight to 1112 km/60° using electrical propulsion systems offers an increased payload at the expense of increased trip time, up to 2000 kg each, which may be critical for survivability. Trade-offs between payload mass, transfer time, launch site, inclination, and height of parking orbits are presented.

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