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AN EVALUATION OF ELECTRIC AND HYDRAZINE PROPULSION SYSTEMS FOR ORBIT MAINTENANCE

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The preferred system for either spacecraft configuration is the monopropellant hydrazine system because of its low cost and because such a system is already included on the spacecraft for orbit insertion.

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LIST OF SYMBOLS

F_D	force exerted on satellite from atmospheric drag
F_T	thrust force
I	total impulse
I_{sp}	specific impulse
$I_{sp\ opt}$	optimum specific impulse relative to overall system efficiency, power system mass, propellant tankage fraction and thrust duration
M	system mass
M_f	fixed mass which do not vary greatly with power (valves, structure, etc.)
M_p	mass of the propellant
M_{ps}	mass of the power system
M_t	total power plus propulsion system mass
MW	molecular weight of the propellant
$\dot{M}_p$	propellant mass flow rate
P	electrical power required by electric thruster
$R(\tau)$	probability of successful operation over operating time τ
T	absolute temperature of exhaust gas
t	total thrust duration
ΔV	velocity increment
α_p	electric power supply specific power
δ	propellant tankage mass fraction
η_b	beam power efficiency of an electric engine

List of Symbols (Cont'd)

η_o	overall efficiency of an electric engine
η_{pc}	efficiency of power conditioning equipment
λ	failure rate
τ	duration of operation

SUMMARY

This report presents an evaluation of electric propulsion systems capable of providing thrust to counteract drag action on a certain satellite in a low earth orbit. The orbit maintenance system is required to produce an average thrust of 1.6 mlb_f over a mission duration of up to 2 years. Five candidate systems were studied, they include: 1) a cesium bombardment ionization thruster, 2) a mercury bombardment ionization thruster, 3) a colloid thruster, 4) a hydrazine resistojet, and, as a standard of comparison, 5) a monopropellant hydrazine thruster (not electric). These propulsion devices were evaluated for two spacecraft configurations, one with a solar cell/battery power system, and the other with a reactor/thermoelectric power system.

The preferred system for either spacecraft configuration is the monopropellant hydrazine system because of its low cost and because such a system is already included on the spacecraft for orbit insertion.

1. INTRODUCTION

A spacecraft in low earth orbit encounters drag forces caused by the residual atmosphere at these altitudes. These drag forces will decelerate the spacecraft, which brings about a slowing down of orbital velocity and a gradual loss of altitude. Eventually, the spacecraft will de-orbit unless some propulsion force is used to overcome the drag.

In this report, two types of propulsion systems are evaluated for orbit maintenance on a certain low-earth orbiting satellite: electric propulsion devices and a monopropellant catalytic hydrazine thruster. The electric propulsion systems use electric power to accelerate ions or small charged particles to produce a thrust. Another form of electric thrusters considered are resistojets, in which the propellant is heated electrically to produce thrust. The monopropellant hydrazine system uses a catalyst for a spontaneous decomposition of the hydrazine. The ensuing expansion of the gases produces the thrust.

In its initial, or baseline, design configuration, the spacecraft incorporates an integrated monopropellant hydrazine system for orbit insertion, attitude control, and orbit maintenance[1*]. However, using an electric propulsion system for the orbit maintenance function offers a potential mass reduction due to the increase in thruster performance. The mass, volume, cost, and performance data are presented for each of the candidate electric propulsion systems for the given mission. A comparison of these figures is made to the baseline design of the monopropellant catalytic hydrazine thruster.

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*numbers refer to the list of references.

2. MISSION REQUIREMENTS

The proposed mission requires a satellite in a circular, near-polar orbit at an altitude of 225 nmi. The orbit is sun-synchronous (inclination $\sim 98^{\circ}$) and the angle between the orbit plane and the Earth-sun axis remains throughout the year at 45°. The period of the orbit is approximately 92.8 minutes, with an eclipse duration of 33 minutes, or 35% of the orbit.

All spacecraft subsystems shall be capable of operating for at least one year with a desired goal of two years. The reliability goal for the orbit maintenance system on the two-year mission is 0.95. The two-year goal also implies that enough expendables, e.g. propellants, be on board for the duration of the mission.

The satellite is required to be within 2 nmi of the nominal 225 nmi orbit altitude at all times. Drag calculations on a typical spacecraft, suitable for this mission, indicate that the expected average drag force acting on the satellite is[1]

$$F_D = 1.6 \times 10^{-3}\ lb_f$$

The figure is based on the "1962 U. S. Standard Atmosphere",[2] with diurnal variations in atmospheric densities averaged over the entire orbit. The spacecraft is powered by an orbit-oriented solar array, which is parallel to the orbital velocity vector. This type of solar array presents a minimum projected array area, and hence, a minimum source of drag.

It should be noted that the 1962 U. S. Standard Atmosphere densities are on the high side of the range expected to be encountered in the mid-1970's. However, the 1962 densities are typically a factor of two less than the maximum values, which are expected in the early 1980's.[3] By designing the propulsion system to counteract a "1962 atmosphere", the spacecraft will be capable of at least 2 years operational life if launched anytime in the 1975-1980 period.

The 1.6 mlb_f drag force imparts a total impulse to the spacecraft in 2 years of approximately 101,000 $lb_f \cdot s$. This is the total impulse that the orbit maintenance propulsion system must supply to keep the satellite in the desired orbit.

The spacecraft is to be launched by a Titan III booster. Final orbit insertion is performed by four 50 lb_f monopropellant hydrazine thrusters. Once in orbit, the baseline system will use two of the four 50 lb_f thrusters for orbit maintenance. On the average of once every 11 days the drag make-up maneuver is required. This consists of two burns, lasting

7.6 s each, one made at perigee and the other at apogee (Hohmann-type transfers).

The baseline design of the spacecraft requires three-axis stabilization from the attitude control system. All disturbing torques must be counteracted by momentum wheels and/or attitude control jets. These jets produce 5 lb_f thrust and consume monopropellant hydrazine. Common propellant tanks hold the hydrazine for both the 5 lb_f attitude control thrusters and the 50 lb_f orbit injection/orbit maintenance thrusters.

If there were a failure in the baseline orbit maintenance system (e.g. all four 50 lb_f quit), the 5 lb_f attitude control jets could be used as a backup system. In such a case, the orbit maintenance maneuver would require more time (76 s per burn). Since the monopropellant hydrazine is stored in common tanks, and the performance of the 5 lb_f and 50 lb_f engines are comparable, the mission could continue despite the failure of the larger thrusters.

The electrical power system on the spacecraft is designed to provide 3000 watts 35% of the time and 1100 watts 65% of the time. Either a solar cell-battery or a reactor-thermoelectric power system have been considered in the baseline design to provide these electrical loads.

If a 3 kW_e reactor system were incorporated, approximately 1900 W would be available for electric thrusters 65% of the time. However, with a solar cell-battery power system no "extra" power would be available because all components are sized to meet the power requirements specified above. The requirement for additional power for the electric propulsion system will increase the size of the solar-cell-battery power system. However, increases in the drag force due to the enlarged solar array will not be considered in this study.

3. CANDIDATE PROPULSION SYSTEMS

3.1 Introduction

The applicability of low-thrust propulsion devices is dependent upon the thrust level, mission duration, engine performance, and system mass. The operating regimes of various propulsion systems are summarized in Figure 3-1. Minimum system mass defines the operating regimes, which are shown in terms of thrust level versus total impulse delivered. It should be noted that each regime has very loose boundaries which cannot be properly illustrated. Since clearcut boundaries do not exist, a choice cannot be certain until a system analysis has been performed.

The total impulse required (101,000 $lb_f \cdot s$) can be delivered continuously at the 1.6 mlb_f thrust level or at a higher thrust level over some fraction of the mission duration.

Electric propulsion systems that operate effectively in the thrust range of 1 to 10 mlb_f with a total impulse range of 80,000 to 120,000 $lb_f \cdot s$ shall be considered in this study. The candidate systems that appear feasible

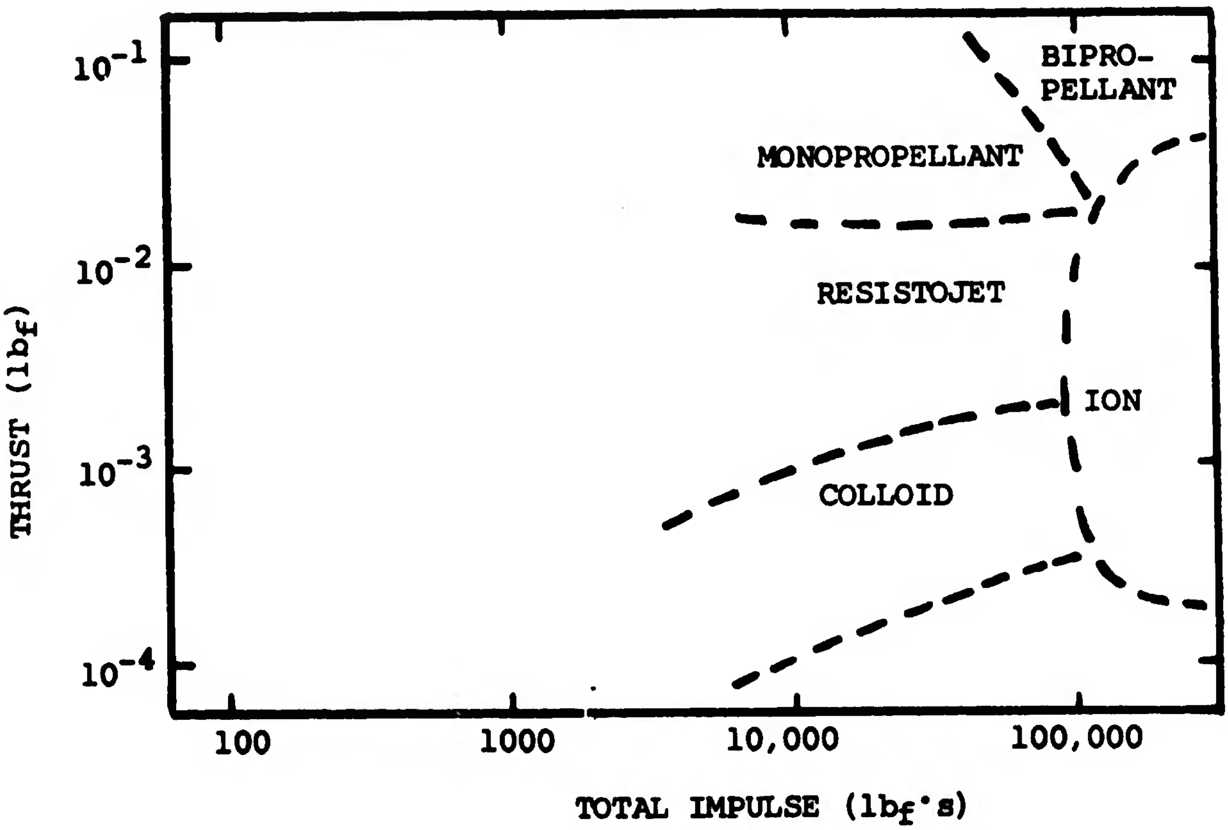

Figure 3-1. Operating Regimes of Low-Thrust Space Propulsion Systems[4]

for this mission include:

1) cesium bombardment ionization thruster
2) mercury bombardment ionization thruster
3) colloid thruster
4) resistojet

   and as standard of comparison:

5) monopropellant hydrazine (not electric).

A description of these thrusters and the status of their development will be given in this section. In addition, the key features, problems and performance characteristics of the system will be identified.

3.2 Mercury and Cesium Bombardment Ionization Thrusters

3.2.1 Description

Electron bombardment ionization thrusters (either mercury or cesium propellant) consist of a propellant tank, feed line, ionization chamber, acceleration electrodes, and a beam neutralizer, as shown in Figure 3-2. The propellant is stored in the liquid state. In operation, the propellant is delivered to a hot porous plug varpoizer located near the end of the feed line. The flow through the vaporizer is regulated by control of its temperature. The vaporizer also separates the liquid and vapor phases of the propellant. If liquid were to migrate down-stream to the ionization chamber, electrical breakdown and shorting could occur.

The vapor is introduced into a cylindrical ionization chamber, where the propellant is ionized by electron bombardment. Electrons are supplied by a cathode which is

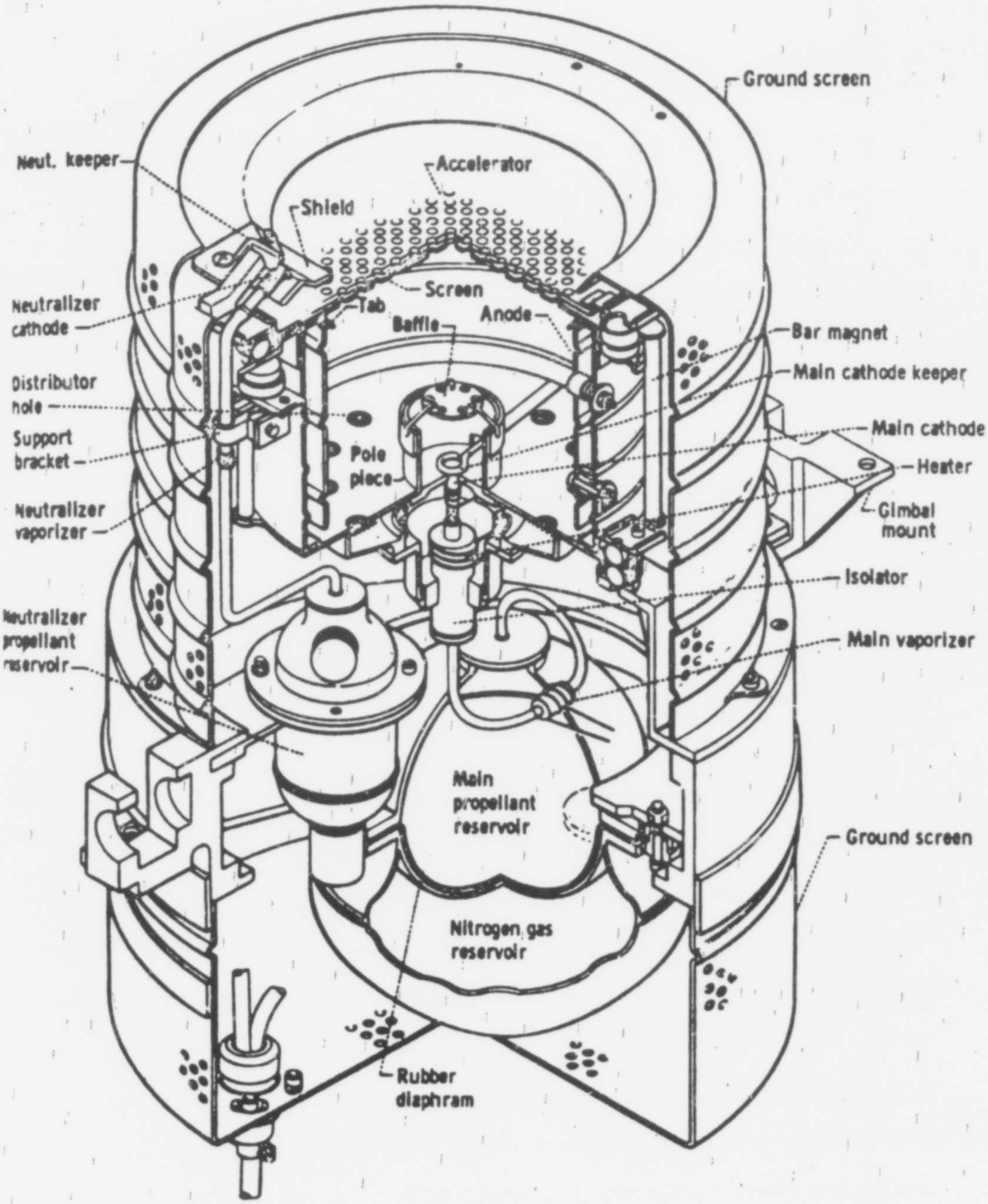

Figure 3-2. Typical Electron Bombardment Engine System (30 cm diameter).

located on the axis of the chamber near the rear of the thruster. A cylindrical anode, which is at a higher voltage potential than the cathode, attracts the electrons.

A divergent magnetic field surrounds the ionization chamber and prevents the direct migration of electrons from the cathode to the anode. Instead, the electrons tend to spiral along the magnetic field lines and thereby increase their residence time in the ionization chamber. Thus the probability of the electrons striking propellant atoms is increased. The magnetic field can be produced by permanent magnets or electromagnets. The electric fields in the ionization chamber tends to direct the newly-formed propellant ions to the accelerating system at the downstream end of the ionization chamber.

The acceleration system consists of two electrodes: one screen electrode attached to the downstr m end of the ionization chamber, and an accelerator elelctrode located 0.025 to 0.25 cm away. Both electrodes have a similar pattern of apertures, and typical accelerating potentials are in the order of 5000 volts (screen electrode, + 500 V up to + 2500 V; accelerating electrode, -250 V down to -2500V). The apertures in the electrodes focus and accelerate the ions, producing many individual positive ion beams downstream.

Neutralization of the exhaust beam is necessary for the successful operation of all electric thrusters. The expulsion of positively charged ions leads to a negative charge buildup on the spacecraft, which could reach negative potentials of a few thousand volts. Neutralization of the positive ion beams is accomplished by injecting low-energy electrons into the exhaust from a neutralizer. A "plasma-bridge" type neutralizer is preferred over thermonic electron emitters for reasons of efficiency and operating life. Plasma-bridge neutralizers incorporate a propellant reservoir, a vaporizer, and a heated cathode, which operates in a manner very similar to those components in the thruster. In operation, a conductive plasma bridges the space between the neutralizer and the exhaust ion beam. The result is that low-voltage electrons are conducted into the beam while the neutralizer may be located far enough away from the beam to prevent degradation from ion bombardment.

A more detailed description of each component in the bombardment ion thrusters is given in reference 5.

3.2.2 Status

The NASA Lewis Research Center has built mercury bombardment engine systems that have flown on the Space

Electric Rocket Test (SERT) Vehicles. SERT I successfully demonstrated the mercury bombardment engine in a ballistic flight to an altitude of 2500 miles on July 20, 1964. The mercury engine produced 6.37 mlb_f for 20 minutes. The engine was then turned off, restarted and allowed to operate for another 10 minutes. The estimated specific impulse was close to 5000 $lb_f \cdot s/lb_m$.[6] The SERT II, launched on February 4, 1970, used a 6.2 mlb_f mercury bombardment engine to raise its orbit. The system has operated successfully.

Two parameters are useful in determining the practicality of electric thrusters; they are the specific impulse (I_{sp}) and beam power efficiency (η_b). The specific impulse is a measure of the impulse (thrust x time, $lb_f \cdot s$), delivered from a thruster relative to the amount of propellant (lb_m) consumed. The thruster beam power efficiency is defined as the ratio of exhaust beam power to electric power put into the engine. This ratio includes all losses such as particle charge to mass distribution, particle energy loss due to charging and beam divergence losses. Both quantities, specific impulse and efficiency, are dependent upon thrust level, as shown in Section 4.1. In general, electron bombardment ionization engines operate most effectively in the thrust range of 10^{-6} to 10^{-2} lb_f with specific impulses in the range of 2000 to 7000 $lb_f \cdot s/lb_m$[7] and beam power efficiencies of 50 to 80%.

Mercury bombardment engines have not demonstrated high efficiency and long life concurrently. For short periods (less than 1000 hours), efficiencies of 60% to 80% have been achieved with the specific impulse in the range of 2000 to 5000 $lb_f \cdot s/lb_m$. Over a lifetime of 5000 hours, the efficiency is on the order of 65% to 70%.

A cesium electron bombardment ion engine has operated for 8200 hours at a thrust level of 6.7 mlb_f with a specific impulse of 5000 $lb_f \cdot s/lb_m$.[5] The thruster, neutralizer and feed system consumed 1 kW of power for a power efficiency of 73%, not considering the inefficiency of a power conditioning unit. A power conditioner converts the input bus power to the various forms (+ and - dc voltages) required by the engine.

The efficiency of the power conditioning unit (η_{pc}), when packaged for spacecraft, is approximately 85% to 95%. The overall efficiency (η_o), is formed by the products of the beam power efficiency and the power conditioner efficiency (i.e., $\eta_o = \eta_b \times \eta_{pc}$). The overall efficiency of the electron-bombardment engines is around 65%.

The beam power efficiencies of electron-bombardment engines, as a function of specific impulse, are given in Figure 3-3. Mercury engines are represented by a wider band

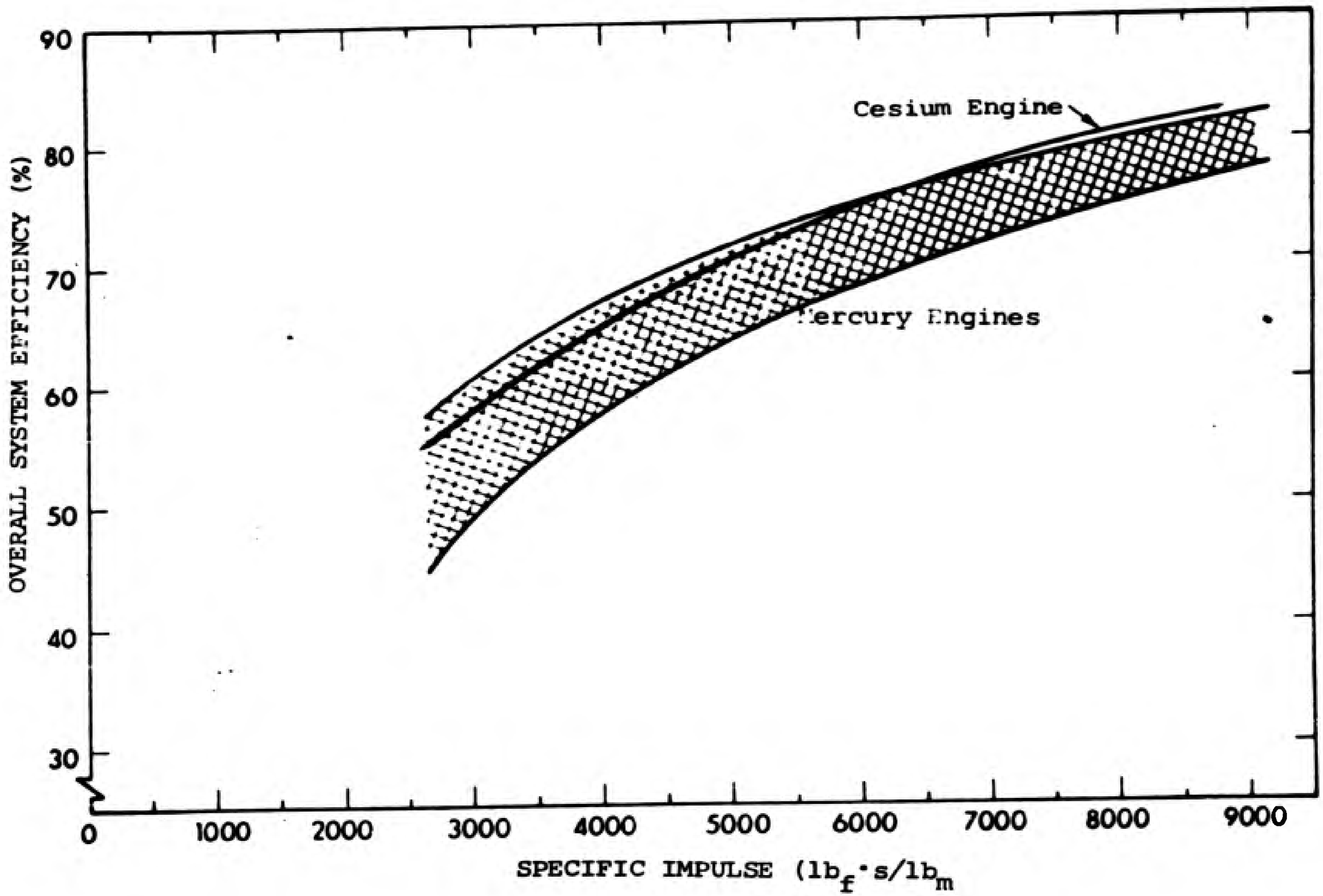

Figure 3-3. Overall Electron Bombardment Engine Efficiency versus Specific Impulse.[5]

in this figure because there have been numerous engines built at different thrust levels, and their performances have varied.

The two propellants used in bombardment engines have different characteristics which influence the performance of the engine. Some of the properties of mercury are: (1) high density, which implies small storage volume; (2) high atomic mass, which makes possible a high thrust per unit area; and (3) ease of handling (cesium requires special handling because it is pyrophoric). However, for long-life engines, mercury ion thrusters have disadvantages which have yet to be overcome. These problems are short cathode life and electrode sputtering. Typical cathode lifetimes are on the order of a few thousand hours, and not many cathodes are designed to last more than 5000 hours. Sputtering damage comes about from ions impinging on the electrodes and freeing atoms of electrode material. Sputtering deteriorates the electrodes, lowers engine performance, and can eventually lead to engine failure. Improved design of the screen and accelerator electrodes, as well as increases in propellant utilization (reduction in un-ionized or neutral mercury atoms) can reduce the sputtering effects. These problems are also present with cesium propellants, but to a lesser degree. Hence, cesium engines have operated up to 8200 hours.[8]

The unique physical properties of cesium make it ideal for use in electron-bombardment ion engines. Cesium has the lowest known ionization potential and a large ionization cross section. This causes a large fraction of the propellant to become ionized, thus mass utilization efficiency is high. The ionization chamber in a cesium engine can use a weaker magnetic field than a mercury engine and this can save some weight.

3.3 Colloid Thruster

3.3.1 Description

The colloid engine, like an ion thruster, produces thrust by accelerating charged particles through an electric field. The colloid thruster concept is schematically shown in Figure 3-4.

A liquid propellant, normally glycerol with 19.3% by weight sodium iodide, is fed into a capillary tube (also called a needle). The emitting rim of the tube is centered within the circular aperture of the extractor electrode. The miniscus at the end of the needle forms microscopic jets which accentuate the electric field (10^7V/cm) causing a continuing emission and acceleration of invisible charged colloid droplets (approximately 100Å in diameter). The capillary tubes are generally held at a positive potental around 5 to 10 kV. The extractor electrode (aperture) is maintained at a negative potential of approximately -500 to -1000 V.

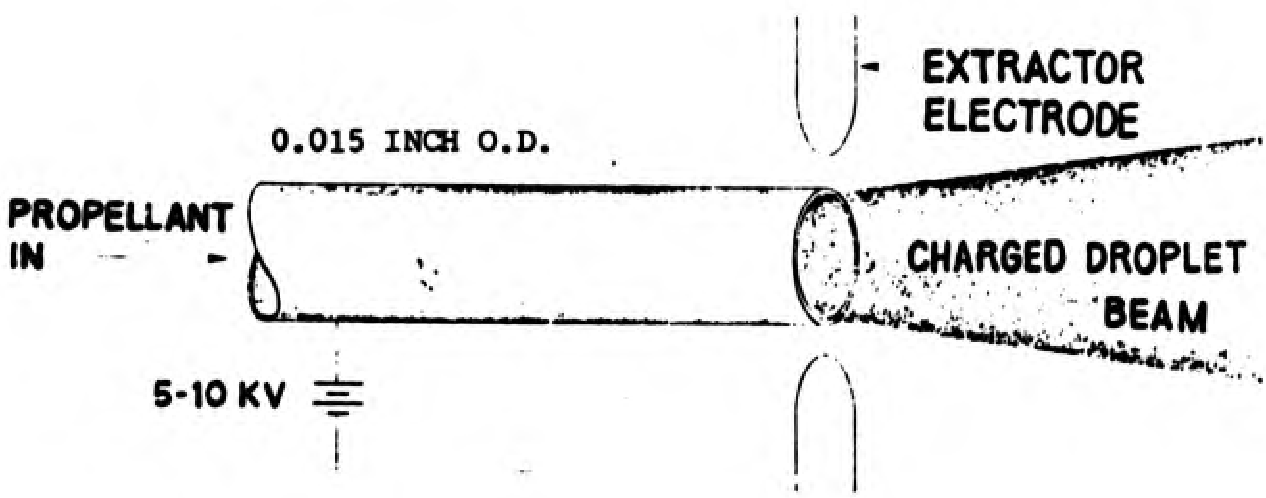

Figure 3-4. Schematic of Colloid Thruster Concept.[9]

A thruster of 5 μlb_f per needle can be obtained, but it may be throttled down to 1 μlb_f by decreasing the propellant flow rate or reducing the capillary potential. Similarly, the specific impulse may be varied between 600 and 1500 $lb_f \cdot s/lb_m$. Higher thrusts are achieved by grouping many needles.

Limited work on a new linear slit geometry (LSG) thruster has demonstrated a thrust density which is higher than that of needle-type thrusters. The linear slit geometry consists of a pair of parallel, closely spaced blades which form an emitting slit, and this replaces the capillary tube as the "nozzle" of the thruster. The slit is then placed within a rectangular extractor aperture in order to achieve the field strength required for charged-particle formation.

3.3.2 Status

Work on colloid thrusters has been sponsored by the Air Force at Electro-Optical Systems and at TRW Systems. Both companies have built and tested needle and linear slit geometry (LSG) thrusters.

The beam power efficiency of the colloid thruster is approximately 70% at specific impulses around 1000 $lb_f \cdot s/lb_m$. It appears feasible to achieve up to 80% efficiency at specific impulses up to 1500 $lb_f \cdot s/lb_m$ because the colloid thruster loses virtually no power for ionization as do the electron-bombardment ion engines. Power conditioning equipment for the colloid thrusters operates near 88% efficiency, for an overall engine efficiency close to 74% at 1500 $lb_f \cdot s/lb_m$.

Some of the problems associated with the colloid thruster that have required special attention include thrust density, beam divergence and lifetime. The linear slit geometry may be the answer to increasing the thrust density, but for the time period required for this mission, only multi-needle systems appear to be available. A nominal 30^o half-cone angle exists in the colloid beam, which is similar to that of the bombardment engines. The lifetime of the colloid thruster is essentially unknown. A 1000 hour life test at TRW on a 36-needle module revealed no appreciable deterioration of the thruster electrodes. A 10,000 hour life test is planned by the Air Force for a flightworthy thruster in 1973.

3.4 Resistojet

3.4.1 Description

A resistojet uses an electrical resistance-heated coil to increase the energy of a gaseous propellant before expansion through an aerodynamic nozzle. The specific impulse of a propellant can be significantly improved with increases in its temperature, or by reduction of its molecular weight through decomposition, and/or by an exothermic reaction. The specific impulse (I_{sp}) is proportional to the square root of the absolute temperature (T) of the gas before ejection divided by its molecular weight (MW) i.e.[6]

$$I_{sp} = \sqrt{\frac{T}{MW}}$$

A typical resistojet is shown in Figure 3-5.

The most promising resistojet propellant to date has been ammonia, although other gases have been used (e.g. nitrogen, hydrogen, argon, freon, and hydrazine). Ammonia (NH_3) decomposes endothermically to nitrogen and hydrogen, reducing the molecular weight of the expelled gases, and thereby raising the specific impulse. The power required to heat the ammonia, and thereby raise its specific impulse, is shown in Figure 3-6 for various thrust levels.

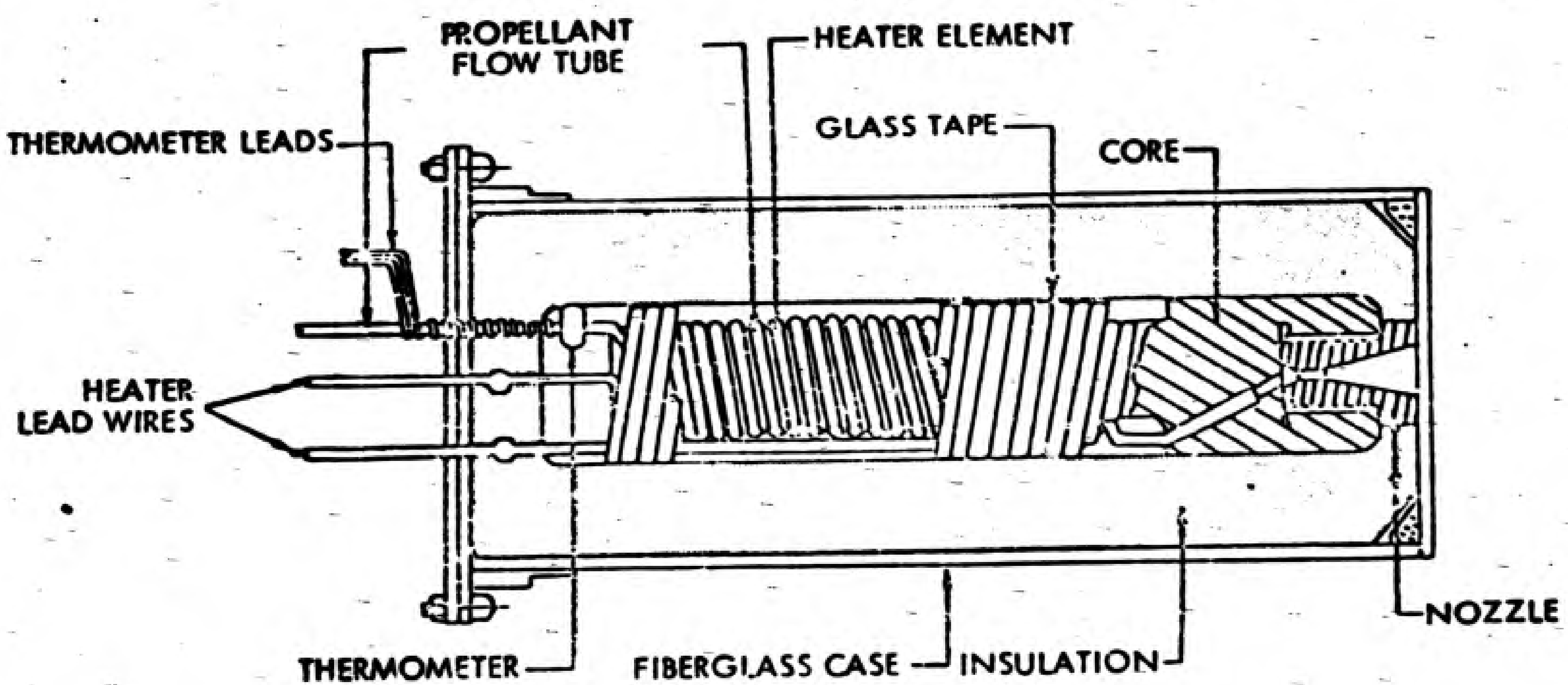

Figure 3-5. Thermal Storage Ammonia Resistojet[9]

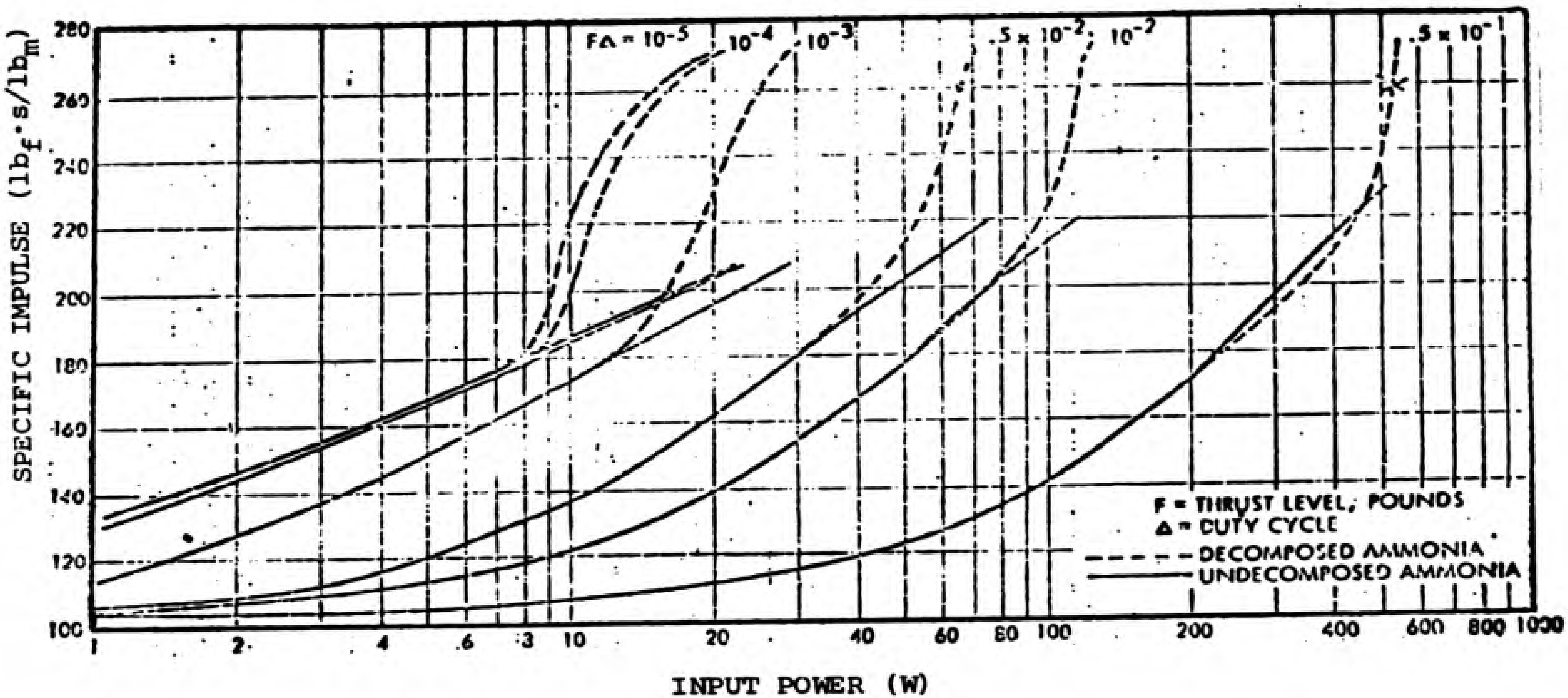

Figure 3-6. Ammonia Resistojet Specific Impulse versus Input Power for Various Thrust Levels.[9]

For the mission considered in this study, an average thrust of 1.6 mlb_f is required. In order for ammonia to exceed the specific impulse of monopropellant hydrazine (I_{sp} = 230 $lb_f \cdot s/lb_m$) at a thrust of 1.6 mlb_f, approximately 30 watts of electrical power are required. A specific impulse of approximately 270 $lb_f \cdot s/lb_m$ could be achieved with an input power around 40 W. Unfortunately, this small improvement in specific impulse (270 - from 230 - $lb_f \cdot s/lb_m$) achieved by substituting an ammonia resistojet for a monopropellant hydrazine system results in only a 69 lb_m propellant mass savings (370 lb_m vs. 439 lb_m). Enlarging the power system to provide the additional 40 W results in a mass increase of 40 lb_m. Because of the increased complexity associated with adding the ammonia system, and the small mass savings (if any when additional tanks, feed lines, and thrusters are considered), the ammonia resistojet will not be considered further.

A hydrazine resistojet, which is under development at AVCO, appears to offer a potential specific impulse in the 280 to 325 $lb_f \cdot s/lb_m$ range with moderate power requirements.[10]

Two thruster concepts are being pursued at AVCO Systems Division. One system uses electrical heat to initiate hydrazine disassociation. Once the exothermic reaction starts, it is completely self-sustaining, and the electrical power can be turned off. Thrusters in the range of 1 to 10 mlb_f have achieved a specific impulse of approximately 200 $lb_f \cdot s/lb_m$. The performance of a comparable catalytic hydrazine millipound thruster is around 140 $lb_f \cdot s/lb_m$. The second concept involves a continual input of electrical power to assist in thermal disassociation of the hydrazine. A 4.3 mlb_f thruster, consuming 55 W, has demonstrated a specific impulse of 285 $lb_f \cdot s/lb_m$ during continuous operation (overall efficiency 50%). It is this second concept which has promise for the mission studied in this report.

A 19% propellant mass savings (354 lb_m from 439 lb_m) could be gained by using a hydrazine resistojet (I_{sp} = 285 $lb_f \cdot s/lb_m$). The resistojet system is attractive because the spacecraft has a monopropellant hydrazine system already on board for orbit injection and attitude control. Therefore, the hydrazine resistojet will be evaluated for this mission.

3.4.2 Status

Several low-thrust resistojets have been flown on spacecraft for attitude and orbit control. The VELA satellites use nitrogen resistojets. Ammonia resistojets have been used on the ATS-III, IV and V, and several classified NRL satellites. These types of resistojets have successfully operated for thousands of hours, and virtually unlimited operating time is anticipated. Development of a hydrazine resistojet, which is not significantly different from the other resistojets, is being carried out by AVCO under NASA sponsorship.[11] The hydrazine thruster should be ready for spaceflight in 1973. The principal contractors that have built resistojets include: TRW Systems, Marquardt, AVCO, and General Electric.

3.5 Monopropellant Hydrazine

3.5.1 Description

Monopropellant hydrazine is a liquid compound (N_2H_4) which, when properly activated, can be made to release its chemical energy by disassociating exothermically into large volumes of hot gas. Such systems are inherently simple, reliable and consequently modest in cost.

In 1963, a catalyst (Shell 405) that produces spontaneous ignition of hydrazine was developed. Hydrazine thrusters have since been developed which are capable of numerous restarts without requiring catalyst bed heaters or the addition of an oxidizer injection system for initiation of hydrazine decomposition. Thus, the system consumes only one type of fuel (hence the name monopropellant), and it does not require any electrical power to produce thrust.

Monopropellant hydrazine thrusters in the thrust range of 1 to 75 lb_f have demonstrated a delivered specific impulse of 230 - 235 $lb_f \cdot s/lb_m$ in a continuous operating mode.[4]

A schematic of a high-performance hydrazine system is shown in Figure 3-7. Included in this diagram are the pressurant tank, which holds the nitrogen pressurizing gas, fill and start valves, and the propellant tanks. Figure 3-8 shows a typical section of a monopropellant thruster.

3.5.2 Status

Monopropellant hydrazine propulsion systems are state of the art today, and they will be used extensively in spacecraft for the next 5 to 10 years.[3] The major problem associated with these systems is the high freezing point of the hydrazine propellant, +34°F. This can constrain the spacecraft thermal design, especially during long eclipse periods. One possibility for lowering the freezing point of hydrazine is using a different propellant formulation, called hydrazine azide.[4] This form of the propellant does not cause a loss in performance relative to the standard hydrazine.

Major manufacturers of the monopropellant hydrazine thrusters include: Rocket Research, TRW and Hamilton Standard.

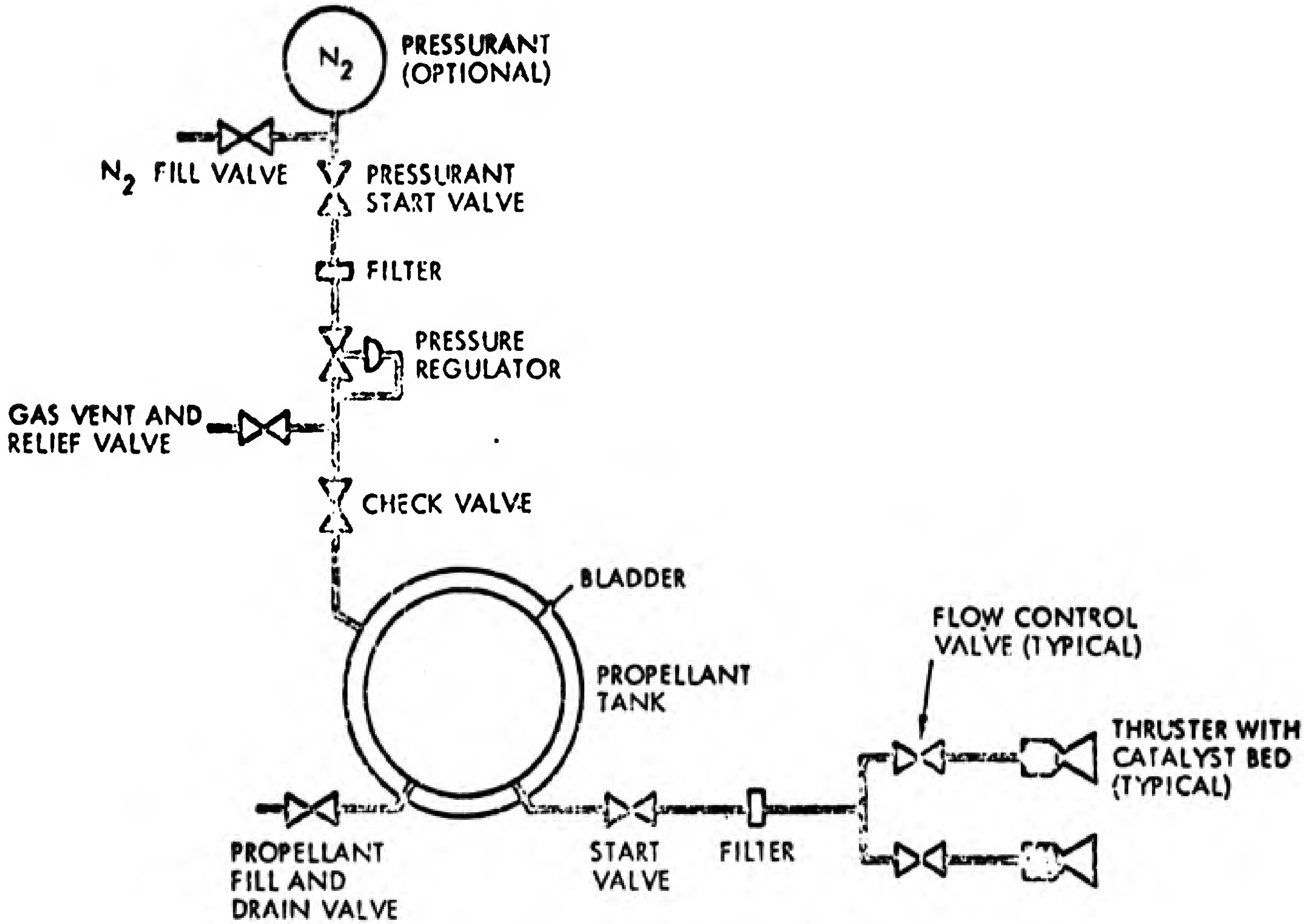

Figure 3-7. Typical High Performance Hydrazine System.[9]

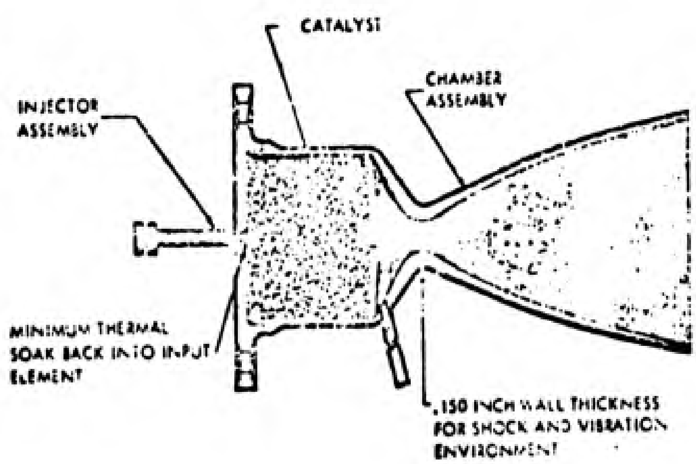

Figure 3-8. Typical Monopropellant Thruster.[9]

3.6 Other Electric Propulsion Devices

Electric thrusters which do not appear feasible for this mission, but are worthy of mention for completness, include:

1) contact ionization thruster
2) pulsed plasma
3) arc jet.

3.6.1 Cesium Contact Ionization Thruster

The cesium contact ionization thruster consists of basically the same components as the electron bombardment ionization thruster except for the ionization device. The contact ionization method passes cesium vapor through a heated tungsten grid or porous slab where ionization occurs upon contact with the tungsten. This type of engine has operated successfully in space on the SNAPSHOT flight in 1965.[6]

The principal disadvantage of the cesium contact ion engine is that large power losses are associated with the hot porous tungsten ionizing surface.[5] The efficiency of the contact engine is lower than the electron bombardment engine in the 1 to 10 mlb_f thrust range. Therefore, the cesium contact engine has not been evaluated for this mission.

3.6.2 Pulsed Plasma

Basically, all pulsed plasma engines operate on the same principle. Electrical energy is discharged across an ablating surface, vaporizing and accelerating the material, or gas, out through a nozzle. The electric energy is stored in capacitors prior to discharge, and the pulse repetition frequency is typically in the range of 1 to 58 Hz. Unfortunately, a high-energy-density capacitor (10 joules per pound) is inefficient and has limited lifetime. Current pulsed plasma designs are approximately 10 to 30% efficient with impulse bits in the range of 1 to 100 $mlb_f \cdot s$ and specific impulses in the range of 200 to 2000 $lb_f \cdot s/lb_m$.[4]

The low efficiency, high power consumption, and low state of development are the reasons that this thruster will not be considered in this study.

3.6.3 Arc Jet Thruster

An arc jet thruster uses an electric arc to heat the propellant gas. A high-intensity arc burns within arc chambers through which the propellant passes. The heated propellant is then exhausted for thrust. Unfortunately, the arc jet is inefficient when compared to a resistojet, which also uses electrical energy to heat gases. In this study arc jets will not be considered as candidate system.

4. EVALUATION OF ORBIT MAINTENANCE

4.1 Optimization of the Specific Impulse

The specific impulse is often used to measure the performance of a thruster. It is determined by the quotient of the thrust of the engine and the propellant mass flow rate required to maintain that thrust, i.e.[12]

$$I_{sp} = \frac{F_T}{\dot{M}_p} \tag{1}$$

where:

I_{sp} = specific impulse, $lb_f \cdot s/lb_m$

F_T = thrust, lb_f

$\dot{M}_p$ = propellant mass flow rate, lb_m/s

The electric power required by the electric engine is dependent upon the specific impulse, thrust level and overall system efficiency. This electrical power is given by[12]

$$P = 21.8 \frac{F_T \, I_{sp}}{\eta_o} \tag{2}$$

where:

P = total electrical power required by thruster, W

η_o = overall system efficiency, which includes beam and power conditioning efficiencies.

Mission constraints usually call for minimum system mass. This includes the power system mass and propellant mass. The mass of the power system, which produces the power for the electric engine, can be determined from the power consumed (P in Equation 1) and the specific power of the power system, α_p, i.e.

$$M_{ps} = \frac{P}{\alpha_p} \tag{3}$$

where:

M_{ps} = mass of the power system, lb_m

α_p = specific power of the electrical power system, W/lb_m.

The propellant mass is determined by the total impulse to be delivered divided by the specific impulse of the propulsion system, i.e.

$$M_p = \frac{F_T \cdot t}{I_{sp}} = \dot{M}_p t \tag{4}$$

where:

M_p = propellant mass, lb_m

t = total duration of thrust, s.

The total system mass is composed of the power system mass (M_{ps}), the propellant mass (M_p), propellant tankage and some fixed masses which do not vary greatly with power.

Combining Equations 2, 3, and 4, and adding terms for propellant tankage and fixed masses, the total system mass is given by[13]

$$M_t = 21.8\,\frac{F_T\, I_{sp}}{\eta_o\, \alpha_p} + \frac{F_T\, t}{I_{sp}}\,(1 + \delta) + M_f \qquad (5)$$

where:

M_t = total system mass, lb_m

δ = propellant tankage mass fraction, dimensionless

M_f = fixed masses which do not vary greatly with power (valves, structure, etc.), lb_m.

As shown above, there is a direct relation between specific impulse and power system mass, and an inverse relationship exists between specific impulse and propellant mass. A graphical representation of Equation 5 is shown in Figure 4-1, with each term of the equation plotted separately versus specific impulse. The values used for the variables in Figure 4-1 are:

F_T = 1.6×10^{-3} lb_f

η_o = 0.50

t = 6.3×10^7 s (i.e. 2 yr)

α_p = 1 W/lb_m

δ = 0.1

M_f = 15 lb_m

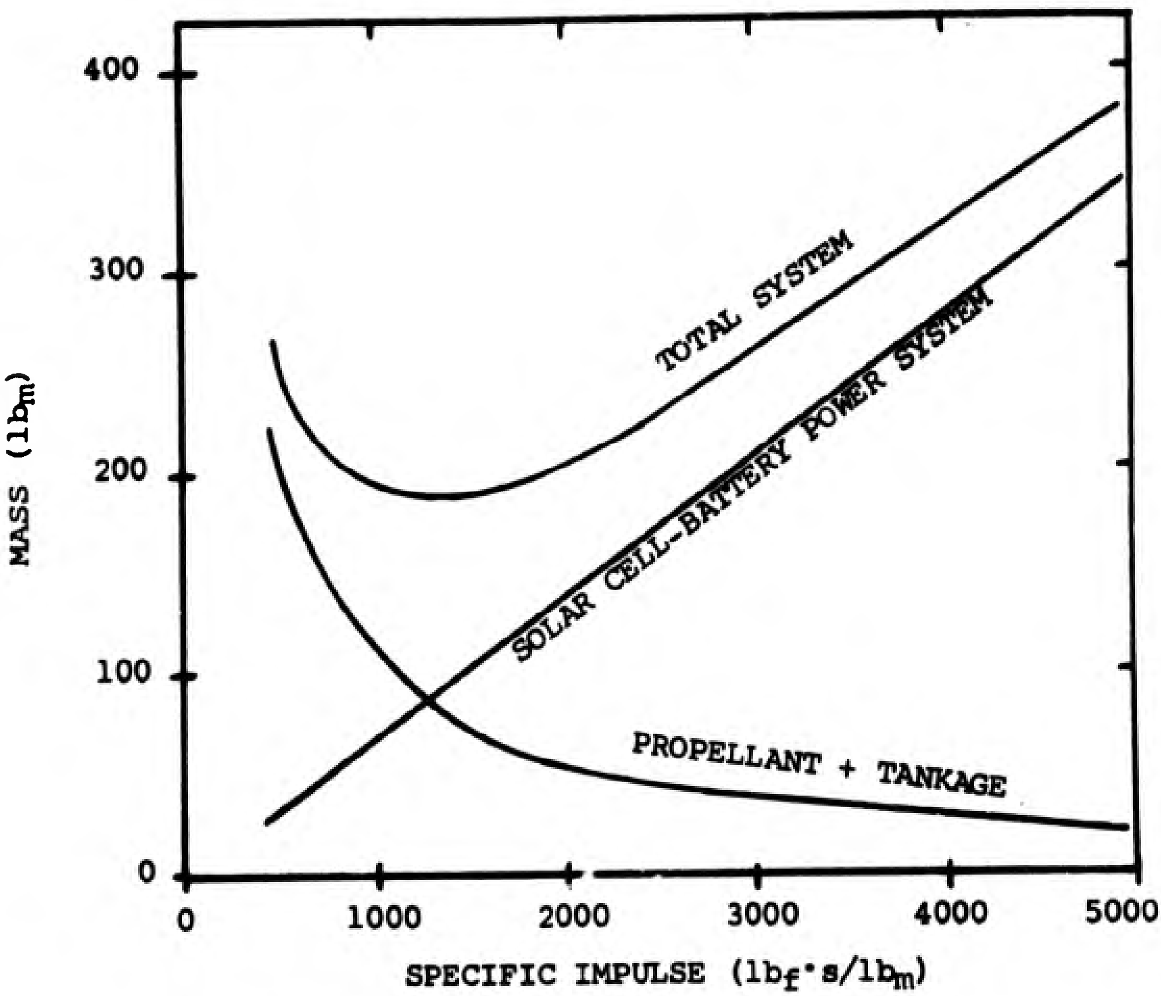

Figure 4-1. Propulsion System, Power System, and Total System Masses versus Specific Impulse.

The minimum total system mass for the electric propulsion system in the example above occurs near a specific impulse of 1260 $lb_f \cdot s/lb_m$. A more exact value can be derived mathematically by differentiating Equation 5 with respect to I_{sp}, and setting the result equal to zero,

$$\frac{d\,M_t}{d\,I_{sp}} = 21.8\,\frac{F_T}{\eta_o\,\alpha_p} - (1+\delta)\,\frac{F_T\,t}{I_{sp}^2} = 0 \qquad (6)$$

By rearranging and solving for I_{sp}, we find:

$$I_{sp}\ \text{opt} = \sqrt{\frac{\eta_o\,\alpha_p\,(1+\delta)\,t}{21.8}} \qquad (7)$$

Substituting the same values for the variables used in deriving Figure 4-1 into Equation 7, the optimum specific impulse is found to be 1262 $lb_f \cdot s/lb_m$.

One of the most important factors which influences the optimum specific impulse is the power system specific power, α_p. The effect of this parameter on the optimum specific impulse, power requirements, and system masses is seen in Table 4-1. With increased specific power (α_p), more power

α_p (W/lb_m)	I_{sp} opt ($lb_f \cdot s/lb_m$)	P (W)	M_{ps} (lb_m)	M_p (lb_m)	M_t (lb_m)
1	1262	88	88	88	191
2	1784	124	62	62	139
4	2524	176	44	44	103
8	3569	249	31	31	77
16	5047	352	22	22	59

Table 4-1. Optimum Specific Impulse (I_{sp} opt), Power (P), and System Masses versus Power System Specific Power (α_p).

is consumed but power system mass, and propellant masses are decreased.

The effect of power system specific power is extremely important for the mission considered in this study. If the spacecraft is solar cell-powered, the power system must be enlarged to handle the additional power demanced by an electric propulsion system. However, if the spacecraft is reactor-powered, approximately 1900 W would be available for electric propulsion up to 65% of the time. The implications that these two different power systems have on the optimum specific impulse are discussed on the following pages.

4.1.1 Optimum Specific Impulse on a Solar Cell-Battery-Powered Spacecraft

The solar cell-battery power system consists of an orbit-oriented solar cell array, battery, and the associated power conditioning equipment. Enlargement of this system to handle the electric propulsion requirements will increase the size of all the system components. For this low orbit mission, the specific power of the power system is approximately 1 W/lb_m[14]

One potential scheme for reducing the power system mass is offered by electric thrusting during the sunlight portions of the orbit only. This type of operation eliminates the need for additional batteries, which would store energy for night time thrusting. Unfortunately, this mission requires a sun-synchronous orbit with the angle between the orbit plane and Earth-Sun axis remaining at 45° throughout the year. Thus, that portion of the orbit nearest the sun, will always be in the sunlight throughout the mission. Conversely, the eclipsed portion of the orbit will never see sunlight.

The mass saving scheme presented above results in electric thrusting over only one portion of the orbit, namely the sunlit side. Thus a positive velocity increment ($+\Delta V$) will be added to one side of the orbit, but no corresponding ΔV will be added on the dark-side of the orbit. In fact, the atmosphere will produce a negative ΔV by the drag acting on the spacecraft in the shadow. The net effect of adding a ΔV on the sun side of the orbit, and subtracting a ΔV on the eclipse side of the orbit is an eccentric orbit with the altitude continually lowering on the sunlit side. This is a direct consequence of Hohmann-type transfers, where a velocity change at one point in an orbit affects the altitude of a point on the opposite side of the orbit. Because a negative ΔV is always present on the dark side of the orbit, the altitude on the sunlit side will lower approximately 42 ft each orbit, or 1 nmi in less than 10 days. Thus, electric thrusting only during the sunlight portions of the orbit, in an effort to reduce power system mass, is impractical for this mission.

The determination of the optimum specific impulse for solar cell-powered spacecraft will be based on a power system specific power of 1 W/lb. The other factors which influence the optimum specific impulse (Equation 7) are overall system efficiency (η_o) and total integrated duration of the thrust (t). A graphical representation of the relationship between these three variables is given in Figure 4-2 (note: the units of t are presented in years).

A higher optimum specific impulse results from longer total thrust durations at any given overall system efficiency. At t=2 years, the electric thruster must operate continuously, this is referred to as a 100% thrust duty cycle. Lower thrust duty cycles require higher thrust levels from the propulsion system in order to provide the same total impulse with shorter total thrust times.

Increasing the thrust increases the power demanded by the propulsion system, which in turn reduces the optimum

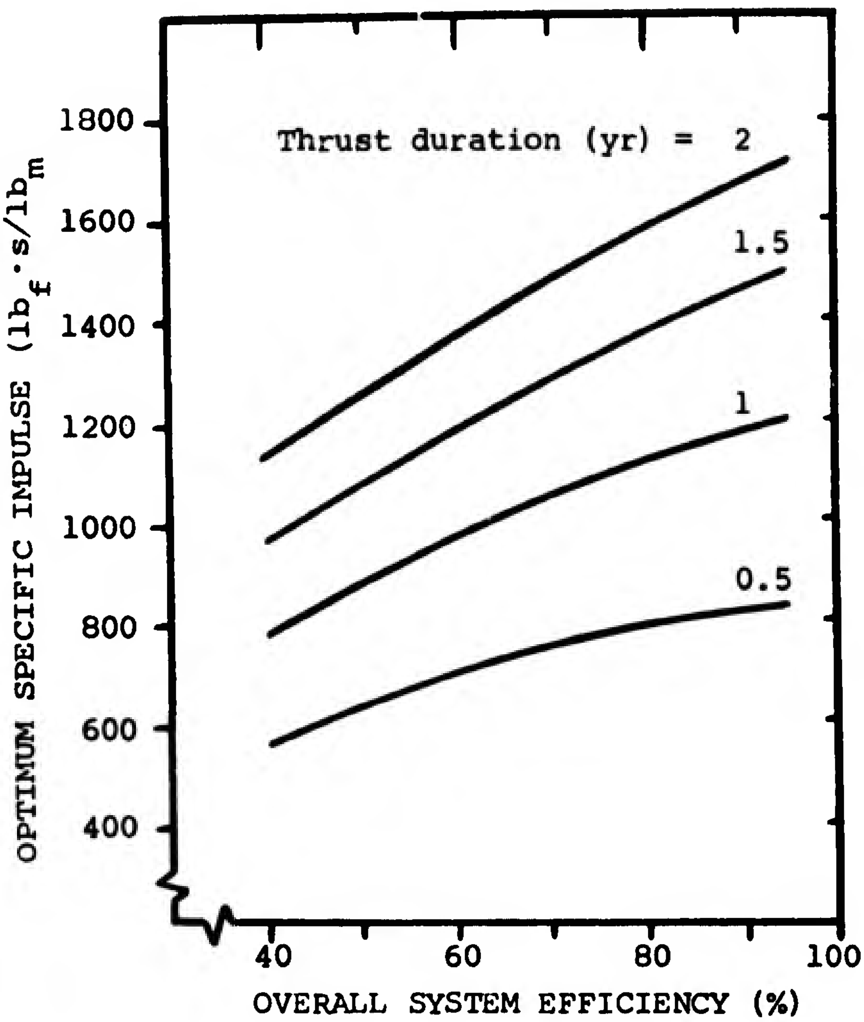

Figure 4-2. Optimum Specific Impulse versus Overall System Efficiency for Various Total Thrust Durations.

specific impulse such that the total system mass is minimized. In Figure 4-3, the total system mass is plotted against system efficiency for two thrust duty cycles. The 100% thrust duty cycle case requires one-half the thrust of the 50% case;

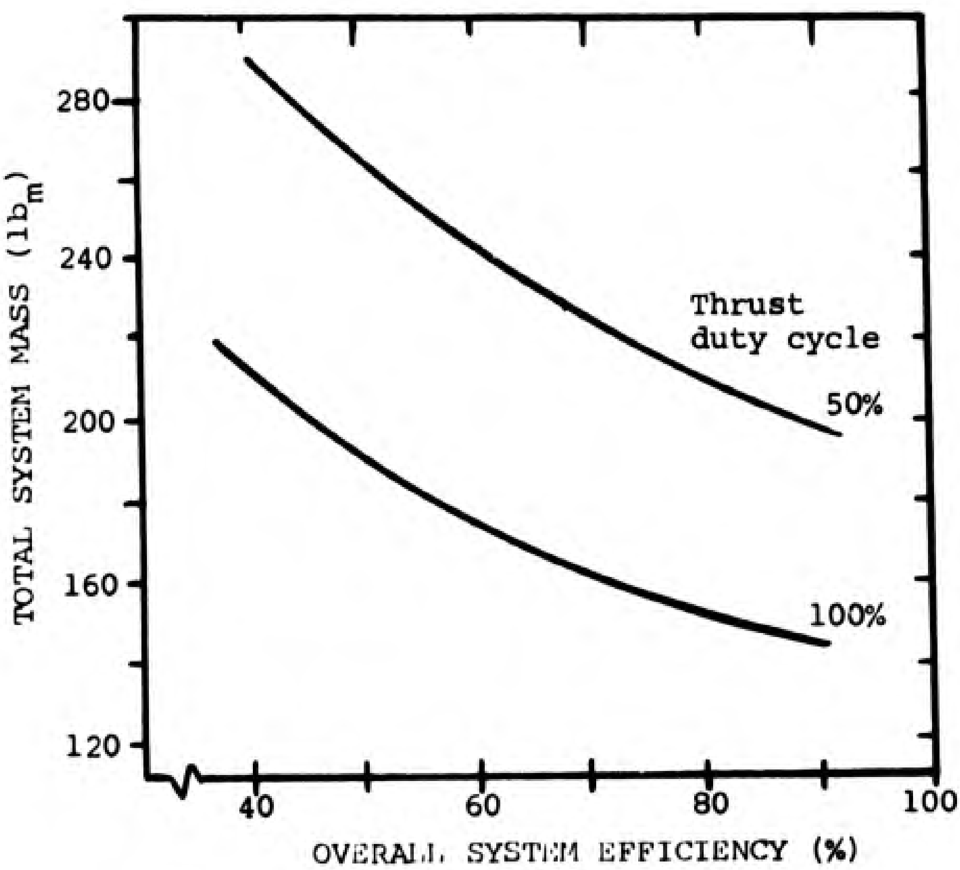

Figure 4-3. Total System Mass versus Overall System Efficiency for Two Thrust Duty Cycles.

hence the 100% case consumes less power, has a higher optimum specific impulse and has a lower total system mass than the 50% case. It can be concluded that, from a mass standpoint on a solar-powered spacecraft, it is best to operate an electric propulsion engine at the lowest thrust possible and over the longest thrust duty cycle.

In terms of the candidate propulsion systems for this mission, only the colloid engine is capable of both efficient operation ($\eta_o \simeq 0.50$) and specific impulses in the range of 600 to 1500 $lb_f \cdot s/lb_m$. The bombardment engines cannot operate effectively below 2000 $lb_f \cdot s/lb_m$ The maximum I_{sp} possible with the hydrazine resistojet is 285 $lb_f \cdot s/lb_m$. The implications of these constraints on the solar-cell-powered spacecraft are discussed later.

4.1.2 Optimum Specific Impulse on a Reactor Thermoelectric-Powered Spacecraft

The reactor-thermoelectric power system produces 3 kW_e of power continuously, of which approximately 1900 W are available for electric propulsion some 65% of the time. This implies that unlike the solar power system, the reactor system does not need to be enlarged to handle the additional electrical power demands from the propulsion system. Therefore, minimum total system mass is achieved by operating the electric propulsion engine at its maximum specific impulse. This minimizes propellant mass and does not influence the size of the power system, which is fixed on the reactor-powered spacecraft.

With 1900 W of "free" power available on the spacecraft, a trade off must be made to determine the thrust level and thrust duty cycle. The variation in thrust level and power required versus thrust duty cycle is shown in Figure 4-4.

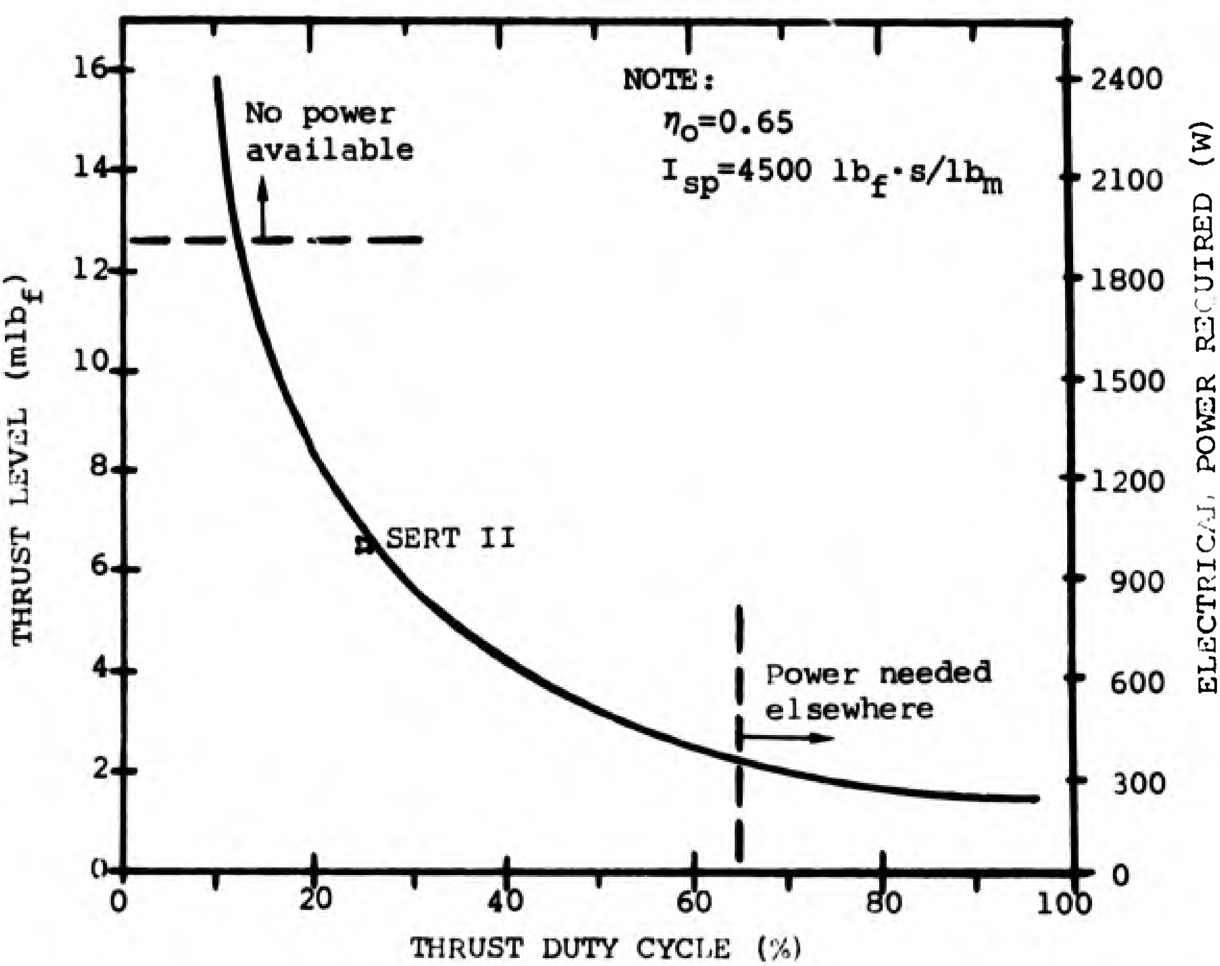

Figure 4-4. Thrust Level and Power Required versus Thrust Duty Cycle.

The dashed vertical line at the 65% thrust duty cycle indicates the point at which the 35% spacecraft power duty cycle requires the entire 3 kW_e. The horizontal broken line indicates the maximum power available (1900 W) from the reactor during the spacecraft 1100 W-duty cycle periods. The thrust duty cycle range of interest is roughly 12% to 65%, with thrust levels of 12 to 2.5 mlb_f, respectively. The electric power required varies from 1900 W to 375 W.

In terms of the candidate propulsion systems for this mission, both the mercury bombardment ($I_{sp} = 4500\ lb_f \cdot s/lb_m$)[15] and the cesium bombardment ($I_{sp} = 5000\ lb_f \cdot s/lb_m$)[16] engines are quite adaptable to the power and thrust range available, as indicated in Figure 4-4. The SERT II mercury bombardment engine, which is indicated in Figure 4-4, produces 6.2 mlb_f and consumes nearly 1 kW. The SERT II system operated successfully in space for its planned 6 months. If a similar system were employed on a reactor-powered spacecraft, it could operate at a thrust level of 6.4 mlb_f and a thrust duty cycle of 25%. The total thrust duration of such a system would be 6 months.

Cesium bombardment engines, suitable for this application, have been built and ground tested. Two engines operated at 6.7 mlb_f and an I_{sp} of 5000 $lb_f \cdot s/lb_m$ for 3700 and 8200 hours. The tests were terminated when the cesium propellant supplies were depleted. These propulsion systems were composed of flight-type hardware, but they were never flown in space. For this mission, a 6.4 mlb_f cesium bombardment engine could be used on the reactor-powered spacecraft. The thrust duty cycle would be 25% with an I_{sp} of 5000 $lb_f \cdot s/lb_m$.

The colloid engine, in its present configuration, is limited to a specific impulse of 1500 $lb_f \cdot s/lb_m$[17] and to a thrust level of 1 mlb_f. The overall efficiency of the system is near 50%, and the power demanded is 65 W. With the limited I_{sp}, the only possible way for a colloid system to use some of the "free" power available from the reactor system is through use of multiple 1 mlb_f engines. Therefore, the thrust and power demanded increases linearly with each engine added.

Clustering of colloid engines presents some problems which require further consideration. These include the requirement for the thrust duty cycle to be less than 65%, so that the propulsion system uses only the "free" available power; the area availability on the spacecraft for multiple engines; propellant storage and plumbing requirements; and reliability and redundancy needs. These problems are discussed in later sections.

The specific impulse of hydrazine resistojets is limited to 285 $lb_f \cdot s/lb_m$.[18] For this mission, a 6.4 mlb_f thruster will be evaluated because it permits direct comparison to the ion engines and it needs to operate only 25% of the time. Higher thrusts, and shorter thrust duty cycles do not offer any savings in mass because of the limit on I_{sp}.

4.1.3 Summary of the Specific Impulse Optimization

The optimum specific impulse for an electric propulsion system on a solar cell-battery-powered spacecraft is 1262 $lb_f \cdot s/lb_m$ (overall efficiency assumed to be 50%). The "free" available power on the reactor-thermoelectric-powered spacecraft results in an infinitely large optimum I_{sp}. Only the limitations of each candidate propulsion system restrict the achievement of these optimum I_{sp}'s. The range of specific impulses achievable, for each candidate propulsion system, are given in Table 4-2.

CANDIDATE PROPULSION SYSTEMS	I_{sp} RANGE ($lb_f \cdot s/lb_m$)	REFERENCES
Cesium Bombardment	2500 - 5000	4,5,16
Mercury Bombardment	2020 - 4500	4,5,15
Colloid	600 - 1500	4,9,17,19
Hydrazine Resistojet	180 - 285	4,18
Monopropellant Hydrazine	130 - 230	4,20,21

Table 4-2. Range of Specific Impulses Achievable for the Candidate Propulsion Systems.

Only the colloid system is capable of operating at the optimum I_{sp} on the solar-powered spacecraft. Neither bombardment engine can operate below 2000 $lb_f \cdot s/lb_m$. On the reactor-powered spacecraft, high I_{sp}'s are desirable. Here the bombardment engines are best with 5000- and 4500 - $lb_f \cdot s/lb_m$ I_{sp}'s for the cesium and mercury engines, respectively.

4.2 Reliability

An important consideration in the design of the orbit maintenance system is its reliability. Determination of the number of engines needed to insure mission success depends upon the reliability of each engine and the redundancy scheme of the back up systems. For the mission considered in this report, a reliability of 0.95 is required for the two year mission.

One way of expressing the reliability of a system is in terms of its operating time (τ) and failure rate (λ), i.e.

$$R(\tau) = e^{-\lambda\tau} \tag{8}$$

where:

$R(\tau)$ = probability of successful operation over an operating time τ

λ = failure rate (h^{-1})

τ = duration of operation (h)

The failure rate represents the reciprocal of the mean time to failure.

With the requirement for R = 0.95 at the end of two-years of operation, the minimum failure rate acceptable is $2.93/10^6$h. If the thrust duty cycle were reduced to 50%, the duration of operation (τ) is halved, and R = 0.975 for the minimum failure rate given above. Conversely, the minimum failure rate acceptable could be relaxed to $\lambda = 5.96/10^6h$ in order to achieve $R = 0.95$ over the operating lifetime (1 year in this particular case). The effect of shorter thrust duty cycles is obvious - shorter operating times permit higher reliabilities or less stringent failure rates for the system. For purposes of this study, the reliability of the on-off switching is not assumed to be a function of the thrust duty cycle.

Another important aspect of shortened operating times (reduced thrust duty cycles), appears in the testing and qualifying of the propulsion system. Testing electric propulsion engines on the ground requires a special vacuum chamber (with charge isolating screens). Running ground tests can be costly and time consuming, it is therefore desirable to run the shortest tests possible. If a thrust duty cycle of 25% were selected, the total operating duration required of the system would be 6 months. Without question, it is far easier to demonstrate a 6 month life capability rather than a 2 year capability. Failures during long duration test of a system which is to operate continuously (100% thrust duty cycle) for 2 years could cause serious time delays for the entire program.

The reliability of the orbit maintenance system is a function of the number of engines required to operate, the number of back up, or redundant, engines, and the thrust duty cycle which determines the total duration of the thruster operation. In order to achieve the required 0.95 reliability, the modes of operation of each candidate propulsion system must be selected, and then the number of back-up systems must be determined.

4.2.1 Modes of Operation

The manner in which the electric propulsion systems are incorporated and operated on the spacecraft is dependent upon the engine thrust capabilities and the desired thrust duty cycles. These factors determine the modes of operation of each system.

Variation of the system total mass with thrust duty cycle is given in Table 4-3 for the following input values:

$$I = 101000 \ lb_f \cdot s$$

$$t = 2 \ yr$$

$$\alpha_p = 1 \ W/lb_m$$

$$\delta = 0.1$$

$$\eta_o = 0.50$$

$$M_f = 15 \ lb_m$$

DUTY CYCLE (%)	THRUST (MLBF)	ISP,OPT (LBF-S/LBM)	M,PRO (LBM)	M,PS (LBM)	M,TOT (LBM)
10.00	15.99	399.10	278.32	27.83	321.15
20.00	7.99	564.41	196.80	39.36	251.16
30.00	5.33	691.26	160.69	48.20	223.89
40.00	3.99	798.20	139.16	55.66	209.82
50.00	3.19	892.41	124.47	62.23	201.70
60.00	2.66	977.59	113.62	68.17	196.80
70.00	2.28	1055.92	105.19	73.63	193.83
80.00	1.99	1128.83	98.40	78.72	192.12
90.00	1.77	1197.30	92.77	83.49	191.27
100.00	1.59	1262.07	88.01	88.01	191.02

Table 4-3. Variation in Total System Mass as a Function of Thrust Duty Cycle.

The lowest system mass occurs at the 100% thrust duty cycle and thrust level of 1.6 mlb_f. However, there is very little variation in total system mass over a duty cycle range from 80% to 100% (192 lb_m to 191 lb_m, respectively). In terms of the mission considered in this study, a 2 mlb_f thrust level operating on an 80% thrust duty cycle offers two important advantages over a 1.6 mlb_f (100% duty cycle) system. First, the colloid engine, the cesium bombardment engine, and the hydrazine resistojet are each being developed to operate at 1 mlb_f. This thrust level would permit two such engines to be combined to meet the 2 mlb_f thrust level required at an 80% duty cycle. Secondly, an 80% duty cycle permits for some margin of error once the system is in operation. If the drag force on the spacecraft ever were to exceed 1.6 mlb_f, the duty cycle on the 2 mlb_f configuration could be increased to avoid a loss in altitude. Thus the 80% thrust duty cycle system offers greater flexibility in operation, if the need should ever arise. Therefore, for this mission, the 2 mlb_f thrust, operating on a 80% thrust duty cycle, will be the mode of operation considered on the solar cell-battery-powered spacecraft.

It should be noted that a mercury bombardment engine does not exist at a 1 or 2 mlb_f thrust level with a specific impulse near 2000 $lb_f \cdot s/lb_m$. However, preliminary designs have been made for a 1 mlb_f, 2020 $lb_f \cdot s/lb_m$ engine.[15] To maintain consistency in comparing the candidate systems, this new 1 mlb_f mercury engine will be considered for the solar cell-powered spacecraft. Therefore, the mode of operation of all candidate propulsion systems on the solar-powered spacecraft is 80% thrust duty cycle at 2 mlb_f thrust.

On the nuclear-powered spacecraft, a thrust level of 6.4 mlb_f and thrust duty cycle of 25% are particularly well suited for the electron bombardment engines. Both cesium and mercury engines have been built and tested near this thrust level. In addition, the 6 month operating duration has been demonstrated by both ion engines. The hydrazine resistojet is also capable of thrust levels up to 7 mlb_f

and it can be operated in the same mode as the bombardment engines.

A colloid thruster system is currently being developed at the 1 mlb_f thrust level. Higher thrust applications will require a multiple combination of these 1 mlb_f engines. Since the specific impulse is limited to a maximum 1500 $lb_f \cdot s/lb_m$ on the colloid system, mass savings will not be gained by operating at thrust levels near 6 or 7 mlb_f. By using three 1 mlb_f colloid engines, the total 3 mlb_f thrust is only required 54% of the time. This configuration permits the colloid engines to use the "free" power which is available from the reactor-thermoelectric power system.

The mode of operation for each candidate propulsion system, in terms of thrust level and duty cycle, is summarized in Table 4-4. Here the applicability of each propulsion system is indicated for the different spacecraft configurations. The hydrazine systems are relatively insensitive to thrust level or duty cycle variations, therefore they are evaluated for all three thrust configurations.

SPACECRAFT POWER SYSTEM:	SOLAR CELL-BATTERY	REACTOR-THERMOELECTRIC	
THRUST LEVEL, mlb_f	2.0	3.0	6.4
THRUST DUTY CYCLE	80%	54%	25%
PROPULSION SYSTEM:			
Cesium Bombardment	x		x
Mercury Bombardment	x		x
Colloid	x	x	
Hydrazine Resistojet	x	x	x
Monopropellant Hydrazine	x	x	x

Table 4-4. Mode of Operation for Each Propulsion System. (Note: the x indicates that the system is applicable to that configuration).

4.2.2 Propulsion System Redundancy

To insure high reliability in the orbit maintenance system, standby redundancy is incorporated in the propulsion system design. By this technique, if one thruster should fail, another thruster can be switched in to complete the mission. The primary reason for employing redundancy in this study stems from the uncertainties in the failure rates of electric propulsion components. Not enough test data exists to make accurate reliability estimates with high confidence.

Redundancy of electric thrusters presents a complicated voltage isolation problem if more than one thruster receives propellant from one tank. If a short circuit should develop in one thruster, the conducting propellants, such as cesium, mercury, or glycerol, could carry the high voltages back to the propellant tanks and on to other thrusters. Thus, a failure in one thruster might lead to failures in all engines fed by a common tank unless a high voltage isolator is incorporated in the feed lines. Work has been done in this area for each of the bombardment engines and the colloid engine. For the purposes of this study, isolators are assumed to be available for each of the three candidate propulsion systems considered for this mission.[15,16,19] If the high voltage isolators were not available, a separate propellant tank would be needed for each electric engine, including the standby redundant engines. However, with the isolators in the system, all engines can be fed from one set of propellant storage tanks, which need only carry a 2 year supply of fuel. High voltage propellant isolation problems do not occur with the hydrazine systems.

A common power conditioning unit (PCU) could be connected to more than one thruster in an effort to reduce total system mass. When a failure occurs in one thruster, the PCU could be switched to the standby thruster to complete the mission. However, the potential mass savings is less than 10 lb_m per PCU, and the additional complexity of the high voltage power switching is not justifiable for such small savings.

The electric propulsion engines considered in this study will consist of (1) one thruster, and (2) one power conditioning unit. All engines will be fed from a common propellant tank. The combination of all engines and their propellant tank comprises the complete propulsion system.

The determination of the number of engines to be incorporated into the spacecraft design must consider the operating mode of the complete propulsion system (i.e. system thrust level, duty cycle and total thrust duration) and engine reliability. The justification of the redundancy scheme used in this study is presented for the colloid system on the solar power-spacecraft. A similar justification exists for the other candidate systems.

The development plan for the colloid engine specifies a reliability goal of 0.95 for a seven years of operation on a 33% duty cycle (i.e. R(2.3yr)=0.95)[22] This implies a failure rate of $\lambda = 2.51/10^6$h, and this figure includes the unreliability of the on-off cycling. Assuming that this failure rate is achieved, the reliability for one engine operating on a 80% duty cycle for 2 years is:

$$R_{1:1}(1.6\ \text{yr}) = 0.965.$$

Here the subscript indicates the number of engines operating versus the number of engines available.

On the solar-powered spacecraft, two engines must operate simultaneously to provide the 2 mlb_f thrust level required. The combined reliability of the two engines is

$$R_{2:2}(1.6\text{yr}) = R_{1:1}(1.6\text{yr}) \times R_{1:1}(1.6\text{yr}) = 0.932$$

This figure is below the reliability goal of 0.95 for the orbit maintenance system specified in the mission requirements section on page 3 of this report. Increasing the system reliability can be achieved by adding a standby redundant engine, such that only two engines need to operate out of the three engines on board. Based on a binomial failure rate distribution, the probability that any two out of three engines will operate successfully is

$$R_{2:3}(1.6\text{yr}) = 0.996$$

Therefore, the redundancy scheme presented provides a comfortable margin of safety above the specified reliability requirement of 0.95.

In order to achieve the minimum acceptable system reliability with this redundancy scheme, i.e. $R_{2:3}(1.6yr)=0.95$, the failure rate can be as high as $\lambda = 10.37/10^6h$. This is approximately four times as many failures per million hours as the colloid system will be designed to meet. It is therefore concluded, that the redundancy scheme of two engines operating, with one standby redundant engine, will be sufficient to meet the mission reliability requirements.

The redundancy scheme incorporated for the colloid system can also be used for the electron bombardment engines and hydrazine resistojet on the solar-powered spacecraft. With such a scheme, the maximum failure tolerable is $10.37/10^6h$, and this rate should be achievable for any of the engines considered.[15,16,18]

On the reactor-powered spacecraft, two modes of operation are considered. First, at least three 1 mlb_f colloid engines are needed to thrust 54% of the time for mission success. The system reliability of three thrusters operating simultaneously with the design-specified failure rate of $\lambda = 2.51/10^6h$ is

$$R_{3:3}(1.08yr) = 0.931$$

Addition of one standby redundant colloid engine, and requiring any three of the total of four engines to operate for mission success, results in a system reliability of

$$R_{3:4}(1.08\text{yr}) = 0.997.$$

The maximum failure rate permissible, and that would still enable an overall system reliability of

$$R_{3:4}(1.08\text{yr}) = 0.95$$

is $\lambda = 15.37/10^6\text{h}$. This is approximately six times as many failures per million hours as the colloid system will be designed to meet. Therefore, on the reactor-powered spacecraft, four 1 mlb_f colloid engines will be included, of which only three need to operate to meet mission requirements.

The second mode of operation on the reactor-powered spacecraft is used for the electron bombardment and hydrazine resistojet systems. Since these three systems can produce the required 6.4 mlb_f individually over the 25% duty cycle, a grouping of multiple engines is not necessary as it was for the 1 mlb_f-limited colloid engine. In order for the orbit maintenance system to meet the mission reliability requirement

$$R_{1:1}(0.5\text{yr}) = 0.95$$

the failure rate must be less than $\lambda = 23.42/10^6\text{h}$. If a standby redundant engine were incorporated into the design, only one of two engines would be required to work, i.e.

$$R_{1:2}(0.5yr) = 0.95$$

and the maximum failure rate permissible becomes $\lambda = 57.78/10^6 h$.

To insure a comfortable margin of safety, and to maintain conformity between operating modes and redundancy schemes, two electron bombardment engines, or two hydrazine resistojets, will be incorporated into the reactor-powered spacecraft design. Only one of the two engines needs to operate for the mission to be successful.

In summary, the redundancy schemes to be used by the various candidate propulsion systems on the different spacecraft configurations permits one complete engine failure without hampering the success of the mission. In other words, there is always one standby redundant system available if the need should arise.

4.3 Orbit Maintenance System Mass

The mass of the electric propulsion system is composed of (1) the mass of the electric engines, which include the thruster, structure, and power conditioning unit (PCU), and (2) the mass of the propellant and associated tankage. The number of engines required on each spacecraft configuration has been determined in the previous sections dealing with modes of operation and redundancy schemes. In addition to the electric propulsion system mass, the increase in demand for electric power necessitates a growth in capacity of the solar cell-battery power system. The additional mass of the solar cell-battery system is 1 lb_m for each watt needed by the electric propulsion engine. The combined masses of the propulsion system and the additional power system comprise the total orbit maintenance system mass.

4.3.1 Cesium Bombardment System

The performance specifications of the bombardment engines required for this study are dependent upon the spacecraft configuration:

(1) The solar cell-battery-powered spacecraft shall incorporate three 1 mlb_f engines, of which any two must thrust 80% of the time. The mass of one thruster and support structure is approximately 9 lb_m, and the power conditioning unit weighs 13 lb_m.[16] The optimum specific impulse for this application is 1129 $lb_f \cdot s/lb_m$, but the minimum I_{sp} achievable by the cesium engine is 2500 $lb_f \cdot s/lbm$.

Propellant and tankage weigh 40.4 lb_m and 4 lb_m, respectively. The overall efficiency of the system is 0.39, which results in a power demand of 140 W for each 1 mlb_f engine. During thrusting operation, 280 W are consumed by the two engines. Since this power is needed only 80% of the time, the power required from the power system can be averaged over the orbit to result in 225 W. The mass of the additional solar cell-battery system equipment is 225 lb_m.

(2) The reactor-thermoelectric-powered spacecraft employs two 6.4 mlb_f engines, of which only one must operate for 25% of the 2 year mission. The 6.4 mlb_f thruster and support structure weigh 20 lb_m, and the mass of the power conditioning unit is 18 lb_m[16] The maximum I_{sp} achievable by the system is used in this application where the electrical power is "free", i. e. I_{sp} = 5000 $lb_f \cdot s/lb_m$. Propellant and tankage masses are 20.2 lb_m and 2 lb_m respectively. The overall efficiency of this system is 0.77.[16]

The engine parameters, power requirements, and cesium engine component masses for the two spacecraft configurations are given in Table 4-5. On the solar-powered spacecraft, the orbit maintenance system weighs 335 lb_m; on the reactor-powered spacecraft, the system weighs 98 lb_m. The primary difference in the masses comes about from the additional power system mass of 225 lb_m on the solar-powered spacecraft.

PARAMETER	SPACECRAFT CONFIGURATION			
POWER SYSTEM:	SOLAR CELL-BATTERY		REACTOR-THERMOELECTRIC	
Thrust Required, mlb_f	2		6.4	
Thrust Duty Cycle	80%		25%	
Individual Engine Thrust, mlb_f	1		6.4	
Number of Engines Operating	2		1	
Total Number of Engines	3		2	
Specific Impulse, $lb_f \cdot s/lb_m$	2500		5000	
Overall System Efficiency	0.39		0.77	
Power Required per Engine, W	140		906	
SYSTEM MASSES, lb_m				
Thruster & Structure	9		20	
Power Conditioning Unit	13		18	
One Engine	22		38	
All Engines		66		76
Propellant		40		20
Propellant Tankage		4		2
Complete Propulsion System		110		98
Additional Power System		225		0
Orbit Maintenance System		335		98

Table 4-5. Cesium Bombardment System Parameters and Component Masses.

4.3.2 Mercury Bombardment System

The performance specifications of the mercury bombardment system are similar to those of the cesium system, i.e. dependent upon the spacecraft configuration:

(1) The solar cell-battery-powered spacecraft incorporates three 1 mlb_f engines. Currently, no 1 mlb_f mercury engines exist which operate at specific impulses near the optimum of 1129 $lb_f \cdot s/lb_m$ for this mission. Therefore, this study shall consider a new mercury engine undergoing development by NASA-Lewis Research Center. The projected performamce of this new low thrust, low I_{sp} mercury technology is a 1 mlb_f thrust, I_{sp} = 2020 $lb_f \cdot s/lb_m$, and an overall efficiency of 0.49.[15] The thruster and structure weigh approximately 5 lb_m, and the mass of the PCU is 7 lb_m. Approximately 90 W of power are required by this projected engine.

(2) The reactor-thermoelectric-powered spacecraft shall incorporate two SERT-II type thrusters, each capable of producing the required 6.4 mlb_f. Improvements in the SERT-II technology and which can be used for a 1975 flight system are: (a) a specific impulse increase to 4500 $lb_f \cdot s/lb_m$ from 4450 $lb_f \cdot s/lb_m$, and (b) an overall system

efficiency increase to 0.81 from a previous figure near 0.60.[15] The thruster and structure weigh 11 lb_m, and the PCU weighs 10 lb_m. The power required by the engine while operating is 775 W.

The engine parameters, power requirements, and mercury bombardment engine component masses for the two spacecraft configurations are given in Table 4-6. The masses of the orbit maintenance systems on the solar cell- and reactor - powered spacecrafts are 235 lb_m and 66 lb_m, respectively.

4.3.3 Colloid System

A colloid propulsion system is currently being developed by TRW Systems under a U. S. Air Force Contract.[23] The system is designed to produce 1 mlb_f, with a specific impulse capability ranging from 600 to 1500 $lb_f \cdot s/lb_m$, and with an overall system efficiency of 0.50.

The two spacecraft configurations studied in this report require multiple engines with one standby redundant engine. The specifications for the two configurations are:

(1) The solar cell-battery-powered spacecraft incorporates three colloid engines, of which any two must operate 80% of the time. The optimum specific impulse for

PARAMETER	SPACECRAFT CONFIGURATION			
POWER SYSTEM:	SOLAR CELL BATTERY		REACTOR-THERMOELECTRIC	
Thrust Required, mlb_f	2		6.4	
Thrust Duty Cycle	80%		25%	
Individual Engine Thrust, mlb_f	1		6.4	
Number of Engines Operating	2		1	
Total Number of Engines	3		2	
Specific Impulse, $lb_f \cdot s/lb_m$	2020		4500	
Overall System Efficiency	0.49		0.81	
Power Required per Engine, W	90		775	
SYSTEM MASSES, lb_m				
Thruster & Structure	5		11	
Power Conditioning Unit	7		10	
One Engine	12		21	
All Engines		36		42
Propellant		50		22
Propellant Tankage		5		2
Complete Propulsion System		91		66
Additional Power System		144		0
Orbit Maintenance System		235		66

Table 4-6. Mercury Bombardment System Parameters and Component Masses.

this application is 1129 $lb_f \cdot s/lb_m$ (given in Table 4-3), a value which the colloid system can easily achieve. The 1 mlb_f thruster, with structure, weighs 5 lb_m, and the associated power conditioning unit weighs 10 lb_m.[17] The mass of propellant required is 89 lb_m and the associated tankage weighs 18 lb_m. The colloid engine consumes 49 W while producing the 1 mlb_f thrust.

(2) The reactor-thermoelectric powered spacecraft incorporates four colloid engines, of which any three must operate on a 54% duty cycle. The maximum specific impulse of 1500 $lb_f \cdot s/lb_m$ is used with the unlimited power available from the reactor. Propellant and tankage masses are 67 lb_m and 13 lb_m, respectively. Power consumed per engine is 65 W.

The engine parameters and component masses for the colloid system are given in Table 4-7. The orbit maintenance system weighs 229 lb_m on the solar-powered spacecraft, and 140 lb_m on the reactor-powered spacecraft.

PARAMETER	SPACECRAFT CONFIGURATION			
POWER SYSTEM:	SOLAR CELL-BATTERY		REACTOR-THERMOELECTRIC	
Thrust Required, mlb_f	2		3	
Thrust Duty Cycle	80%		54%	
Individual Engine Thrust, mlb_f	1		1	
Number of Engines Operating	2		3	
Total Number of Engines	3		4	
Specific Impulse, $lb_f \cdot s/lb_m$	1129		1500	
Overall System Efficiency	0.50		0.50	
Power Required per Engine, W	49		65	
SYSTEM MASSES, lb_m				
Thruster & Structure	5		5	
Power Conditioning Unit	10		10	
One Engine	15		15	
All Engines		45		60
Propellant		89		67
Propellant Tankage		18		13
Complete Propulsion System		152		140
Additional Power System		77		0
Orbit Maintenance System		229		140

Table 4-7. Colloid System Parameters and Component Masses.

4.3.4 Hydrazine Resistojet System

Development of hydrazine resistojets has evolved to a stage where a specific impulse of 285 $lb_f \cdot s/lb_m$ and an overall efficiency of 0.50 has been demonstrated. This maximum I_{sp} is well below the optimum I_{sp} of 1129 $lb_f \cdot s/lb_m$ for the solar-powered spacecraft, or the unlimited I_{sp} for the nuclear-powered spacecraft. Therefore, the resistojets limited I_{sp} implies a propellant mass of 354 lb_m for either spacecraft configuration. The tankage requirements are 10% of the mass of the fuel, or 35 lb_m in this case.

The power conditioning required by the hydrazine resistojet is minimal because the thruster can be designed to operate at the same voltage as the spacecraft bus line voltage. The mass of the PCU is 1 lb_m for each of the applications studied in this report.[18]

The resistojet technology being developed at AVCO Systems for NASA covers a thrust range of 1 to 100 mlb_f. These resistojets are typically small. An 8 mlb_f thruster is less than 2 inches long and weighs 0.3 lb_m.[18] Because of the relative simplicity of these devices, a resistojet could be made for any one of the three modes of operation considered in this study (solar cell-spacecraft: 2 mlb_f @

80% duty cycle; reactor-spacecraft; 3 mlb_f @ 54% duty cycle, or 6.4 mlb_f @ 25% duty cycle). For this reason all three configurations are evaluated to aid in comparisons between propulsion systems. The redundancy scheme incorporated for the hydrazine resistojet is to have one engine thrust at the required thrust level, and to have one standby redundant engine. The system parameters and component masses are given in Table 4-8.

The mass of the orbit maintenance system is 412 lb_m on the solar-powered spacecraft, and 392 lb_m on the reactor-powered spacecraft. The thrust level and duty cycle have no effect on the total system in the reactor case, because each thruster weighs so little and the propellant mass is determined by the limited I_{sp}.

4.3.5 Monopropellant Hydrazine System

The monopropellant hydrazine propulsion system incorporated in the spacecraft baseline design performs orbit insertion and orbital maintenance with the same set of thrusters. The set is composed of four 50 lb_f engines, which all burn during orbit insertion. Two of the four are

PARAMETER	SPACECRAFT CONFIGURATION		
POWER SYSTEM:	SOLAR CELL-BATTERY	REACTOR-THERMOELECTRIC	
Thrust Required, mlb_f Thrust Duty Cycle	2 80%	3 54%	6.4 25%
Individual Engine Thrust, mlb_f	2	3	6.4
Number of Engines Operating	1	1	1
Total Number of Engines	2	2	2
Specific Impulse, $lb_f \cdot s/lb_m$	285	285	285
Overall System Efficiency	0.50	0.50	0.50
Power Required per Engine, W	25	37	80
SYSTEM MASSES, lb_m			
Thruster & Structure	1	1	1
Power Conditioning Unit	1	1	1
One Engine	2	2	2
All Engines	3	3	3
Propellant	354	354	354
Propellant Tankage	35	35	35
Complete Propulsion System	392	392	392
Additional Power System	20	0	0
Orbit Maintenance System	412	392	392

Table 4-8. Hydrazine Resistojet System Parameters and Component Masses

required for drag make up maneuvers, which occur on the average of once every 11 days. The maneuver consists of two burns on opposite sides of one orbit, and they last 7.6 seconds each. The specific impulse of these engines, when used in the manner described above, is approximately 230 $lb_f \cdot s/lb_m$. For the 2 year mission, 439 lb_m of hydrazine is required, and the propellant tanks will weigh 44 lb_m.

Since the four 50 lb_f engines are required to be on the spacecraft for orbit insertion, no mass penalty need be assessed to the orbit maintenance system. Similarly, no additional mass is added for the power system because the monopropellant system does not consume electricity. The only power required is for valve operations, and this amounts to 20 to 30 W.[21] However, valve opening and closing occurs infrequently, so the average power used over an 11 day cycle is negligible.

In summary, the mass of the orbit maintenance system amounts to 483 lb_m, all of which is hydrazine and propellant tankage.

4.4 Orbit Maintenance System Volume

The most voluminous portion of most propulsion systems studied in this report is the propellant and its associated tankage. Estimations of this volume are made by dividing the propellant mass by its density. Listed below, are the densities of the propellants considered:

PROPELLANT	DENSITY	
	(g/cm^3)	lb_m/ft^3
Cesium	1.87	117
Mercury	13.55	845
Glycerol-sodium iodide	1.46	91
Hydrazine	1.00	62.5

The volume of the orbit maintenance system also includes the additional power system volume on the solar cell-battery-powered spacecraft. The specific volume of the power system is 0.01 ft^3/W.[14] The volumes for each candidate propulsion system are given on the following pages, as they were for the system masses.

4.4.1 Cesium Bombardment System

The 1 mlb_f cesium engines to be used on the solar-powered spacecraft has a diameter of approximately 2.2 inches, and a length of 3.5 inches.[16] The cylinderically-shaped engine occupies approximately 13.3 in^3, or 0.008 ft^3. The PCU can be located in a cube-shaped box, which has a side length of 10 inches.[16] The volume of the PCU is 0.58 ft^3.

The reactor-powered spacecraft incorporates the 6.4 mlb_f engine. This thruster is approximately 6 inches in diameter and 9 inches long.[16] The volume of the thruster is 254 in^3, or 0.15 ft^3. The size of the PCU of the 6.4 mlb_f system is the same as that of the 1 mlb_f system, i.e. 0.58 ft^3.

The component and system volumes for the cesium bombardment system are given in Table 4-9.

4.4.2 Mercury Bombardment System

The 1 mlb_f and 6.4 mlb_f mercury engines are the same size as the comparable cesium bombardment engines.[15] Likewise, the power conditioning units of the bombardment engines are similar, i. e. 0.58 ft^3. The differences in bombardment system volumes occurs in the propellant and power system areas, as indicated in the summary in Table 4-10.

P A R A M E T E R	SPACECRAFT CONFIGURATION			
POWER SYSTEM:	SOLAR CELL-BATTERY		REACTOR-THERMOELECTRIC	
Thrust Required, mlbf	2		6.4	
Thrust Duty Cycle	80%		25%	
Individual Engine Thrust, mlbf	1		6.4	
Number of Engines Operating	2		1	
Total Number of Engines	3		2	
SYSTEM VOLUMES, ft^3				
Thruster & Structure	0.01		0.15	
Power Conditioning Unit	0.58		0.58	
One Engine	0.59		0.73	
All Engines		1.77		1.46
Propellant & Tankage		0.34		0.17
Complete Propulsion System		2.11		1.63
Additional Power System		2.25		0
Orbit Maintenance System		4.36		1.63

Table 4-9. Cesium Bombardment System Volumes.

PARAMETER	SPACECRAFT CONFIGURATION			
POWER SYSTEM:	SOLAR CELL-BATTERY		REACTOR-THERMOELECTRIC	
Thrust Required, mlb_f	2		6.4	
Thrust Duty Cycle	80%		25%	
Individual Engine Thrust, mlb_f	1		6.4	
Number of Engines Operating	2		1	
Total Number of Engines	3		2	
SYSTEM VOLUMES, ft^3				
Thruster & Structure	0.01		0.15	
Power Conditioning Unit	0.58		0.58	
One Engine	0.59		0.73	
All Engines		1.77		1.46
Propellant & Tankage		0.06		0.03
Complete Propulsion System		1.83		1.49
Additional Power System		1.44		0
Orbit Maintenance System		3.27		1.49

Table 4-10. Mercury Bombardment System Volumes.

4.4.3 Colloid System

The 1 mlb_f colloid thruster assembly can fit in a cylinder approximately 13 inches in diameter and 5 inches long.[19] This results in a volume of 663 in^3, or 0.38 ft^3. The PCU fits in a 6 in x 8 in x 7 in envelope,[19] which is 0.23 ft^3. The volumes for the colloid orbit maintenance system are given in Table 4-11 for the two spacecraft configurations.

4.4.4 Hydrazine Resistojet System

The hydrazine resistojet consists of a small cylinderical thrust chamber and nozzel. A typical 6.4 mlb_f thruster is approximately one-half inch in diameter and two inches long.[18] This amounts to a volume of less than 0.4 in^3, or $2.3 \times 10^{-4} ft^3$. When compared to the 5.6 ft^3 of hydrazine propellant, the volume of the individual thrusters can be neglected. The PCU needed for the hydrazine resistojet is approximately 0.02 ft^3 in volume. The volumes for the orbit maintenance system are given in Table 4-12 for all three spacecraft configurations.

PARAMETER	SPACECRAFT CONFIGURATION	
POWER SYSTEM:	SOLAR CELL-BATTERY	REACTOR-THERMOELECTRIC
Thrust Required, mlb_f	2	3
Thrust Duty Cycle	80%	54%
Individual Engine Thrust, mlb_f	1	1
Number of Engines Operating	2	3
Total Number of Engines	3	4
SYSTEM VOLUMES, ft^3		
Thruster & Structure	0.38	0.38
Power Conditioning Unit	0.23	0.23
One Engine	0.61	0.61
All Engines	1.83	2.44
Propellant & Tankage	1.00	0.74
Complete Propulsion System	2.83	3.18
Additional Power System	0.77	0
Orbit Maintenance System	3.60	3.18

Table 4-11. Colloid System Volumes.

PARAMETER	SPACECRAFT CONFIGURATION		
POWER SYSTEM:	SOLAR CELL-BATTERY	REACTOR-THERMOELECTRIC	
Thrust Required, mlb_f	2	3	6.4
Thrust Duty Cycle	80%	54%	25%
Individual Engine Thrust, mlb_f	2	3	6.4
Number of Engines Operating	1	1	1
Total Number of Engines	2	2	2
SYSTEM VOLUMES, ft^3			
Thruster & Structure	$>10^{-3}$	$>10^{-3}$	$>10^{-3}$
Power Conditioning Unit	0.02	0.02	0.02
One Engine	0.02	0.02	0.02
All Engines	0.04	0.04	0.04
Propellant & Tankage	5.66	5.66	5.66
Complete Propulsion System	5.70	5.70	5.70
Additional Power System	0.20	0	0
Orbit Maintenance System	5.90	5.70	5.70

Table 4-12. Hydrazine Resistojet System Volumes.

4.4.5 Monopropellant Hydrazine System

The volume associated with the monopropellant hydrazine orbit maintenance system comes only from the propellant volume. The 439 lb_m of hydrazine requires 7.02 ft^3 of storage space. No additional power system or thruster volumes are assessed to the orbit maintenance system because these systems are incorporated in the baseline design regardless of the propulsion system selected for drag make up.

4.5 Orbit Maintenance System Cost

The costs of developing and procuring the electric propulsion systems considered for this mission cannot be firmly stated. Some systems require further technology development, as well as prototype fabrication and qualification to the mission specifications. Programs have been funded by NASA to develop both types of bombardment engines and resistojets. Similarly, the Air Force is sponsoring the development of the colloid system. So in most cases, the electric propulsion will be developed, tested and ready for use by the Navy for a 1975 mission. Only the 1 mlb_f

mercury bombardment engine needs special development funds because it uses new technology in the low 2000 $lb_f \cdot s/lb_m$ I_{sp} range of mercury engines. However, in the projection of all system costs, great uncertainties arise, and only estimates can be made on the costs of these future flight systems.

The costs of the additional power system needed on the solar cell-battery-powered spacecraft are based on a specific cost of $2140/W.[14] Tankage, plumbing and support structures are assumed to cost $50,000, which when added to the engine costs forms the complete propulsion system.

4.5.1 Cesium Bombardment System

Electro-Optical Systems is currently under contract to the NASA Goddard Space Flight Center to develop and deliver a cesium bombardment engine system for flight on the ATS-F satellite in 1973. System development work has been complete as of July 1969, and current work is on the flight system, which consists of engine, power conditioner, and propellant tank. NASA-Goddard will be spending a total of approximately $1.8 M for this work, spread over the next few years. The ATS-F thruster is designed to provide 1 mlb_f for 6 months in orbit.

Modifications of the ATS-F system, to make a 1 mlb_f thrust applicable for the solar-powered spacecraft studied in this report, would include (1) increasing the propellant tank capacity to give 1.6 years operating life, (2) some circuitry changes to match new spacecraft power system characteristics, and (3) some requalification of the modified thruster. These modifications should not cost more than $300,000.[16]

The NASA contract calls for a delivery of three flight-worthy cesium engines. These engines will be completely fabricated and will have passed acceptance tests. A cost of $450,000 is associated with the delivery of the flight units. A recurring cost per flight unit of $150,000 is assumed by EOS for the 1 mlb_f system.[16]

A 6.4 mlb_f cesium engine, suitable for use on the reactor-powered spacecraft, has been built and tested by EOS. Modification of the system design and qualification to mission requirements will cost approximately $500,000 to $800,000. The recurring cost for one engine is estimated by EOS to be $200,000.[16]

A summary of the cesium orbit maintenance system costs are presented in Table 4-13.

PARAMETER	SPACECRAFT CONFIGURATION	
POWER SYSTEM:	SOLAR CELL-BATTERY	REACTOR-THERMOELECTRIC
Thrust Required, mlb$_f$	2	6.4
Thrust Duty Cycle	80%	25%
Individual Engine Thrust, mlb$_f$	1	6.4
Number of Engines Operating	2	1
Total Number of Engines	3	2
SYSTEM COSTS ($ in 1000's):		
NON RECURRING, DEVELOPMENT	300	800
RECURRING, FLIGHT SYSTEM:		
Electric Engine	150	200
Complete Propulsion System	500	450
Additional Power System	482	0
Orbit Maintenance System	982	450

Table 4-13. Cesium Bombardment System Costs.

4.5.2 Mercury Bombardment System

Most of the development work for mercury bombardment systems has been done by NASA at their Lewis Research Center. This work has been primarily at technology development and is not system oriented. However, NASA-LRC has built the SERT-II system which has been successfully demonstrated in orbit. Estimations of development and recurring costs of NASA mercury engines is difficult because so much of the work was done internally at NASA. Procurement of engines for many spacecraft would require a private contractor for production, because NASA is essentially a research and development organization. Two contractors that have had some experience with mercury systems are EOS and TRW Systems.

The development and testing of a new 1 mlb_f mercury engine, which operates at an I_{sp} near 2000 $lb_f \cdot s/lb_m$ could cost up to two million dollars.[15] Since most of the work at NASA is done on high performance engines, the low I_{sp} technology is not as far developed, and much work is needed if the system is to be built. A great portion of the non-recurring costs associated with the 1 mlb_f mercury engine development is in the qualification testing. The system is expected to operate for 2 years on an 80% duty cycle, which implies an integrated operating duration of 1.6 years. Accelerated testing is not possible for the actual life tests, because so much time and money is required for the demonstration of this system. Recurring costs for the 1 mlb_f mercury engine will be comparable to those of the similar cesium engine, i. e. $150,000 per flight unit.[15]

The reactor-powered spacecraft incorporates a 6.4 mlb_f thruster which would be of similar design to the SERT-II system. The 6.4 mlb_f engine used for this mission would incorporate the advancements in mercury bombardment technology, that were not available for the finalized SERT II design in 1966. These improvements imply that additional testing and qualification expenses are to be expected. The development of the 6.4 mlb_f mercury system may cost up to $1,000,000. Once qualified, the recurring costs of subsequent engines will be near $200,000.[15]

The mercury bombardment orbit maintenance system costs are summarized in Table 4-14.

PARAMETER	SPACECRAFT CONFIGURATION	
POWER SYSTEM:	SOLAR CELL-BATTERY	REACTOR-THERMOELECTRIC
Thrust Required, mlb_f	2	6.4
Thrust Duty Cycle	80%	25%
Individual Engine Thrust, mlb_f	1	6.4
Number of Engines Operating	2	1
Total Number of Engines	3	2
SYSTEM COSTS ($ in 1000's):		
NON RECURRING, DEVELOPMENT	2000	1000
RECURRING, FLIGHT SYSTEM:		
Electric Engine	150	200
Complete Propulsion System	500	450
Additional Power System	308	0
Orbit Maintenance System	808	450

Table 4-14. Mercury Bombardment System Costs.

4.5.3 Colloid System

The Air Force is currently sponsoring the development of a 1 mlb_f colloid thruster system at TRW Systems. The overall $4.6 M program includes the building and testing of numerous prototype, bread-board, and life-test systems.[9] Two thrusters are planned to be tested for 10,000 hours each, concurrently in a vacuum chamber at Wright Patterson Air Force Base. With the successful completion of this development program in 1973, the colloid system will be space qualified for missions requiring 1 mlb_f thrusters.

Nonrecurring cost to users of the colloid system should only include integration expenses, which are typically less than $250,000.[17] Recurring costs for the 1 mlb_f thruster is estimated to be $150,000 each.[17] Both the solar- and nuclear-powered spacecraft incorporate this 1 mlb_f thruster. A summary of the orbit maintenance system costs are given in Table 4-15.

4.5.4 Hydrazine Resistojet

The AVCO Corporation has been funded by the NASA-Goddard Space Flight Center for the technical development

PARAMETER	SPACECRAFT CONFIGURATION	
POWER SYSTEM:	SOLAR CELL-BATTERY	REACTOR-THERMOELECTRIC
Thrust Required, mlb_f	2	3
Thrust Duty Cycle	80%	54%
Individual Engine Thrust, mlb_f	1	1
Number of Engines Operating	2	3
Total Number of Engines	3	4
SYSTEM COSTS ($ in 1000's):		
NON RECURRING, DEVELOPMENT	250	250
<u>RECURRING, FLIGHT SYSTEM</u>:		
Electric Engine	150	150
Complete Propulsion System	500	650
Additional Power System	165	0
Orbit Maintenance System	665	650

Table 4-15. Colloid System Costs.

of hydrazine resistojets. To develop a thruster for a particular application, in which the thrust and configuration are specified, will cost up to $250,000. The building and qualifying of such a thruster requires an additional $200,000. Therefore, the total nonrecurring cost expected for a particular thruster is approximately $450,000.[18]

Once development and qualification are complete, the thrusters can be procurred directly. The expected cost of an individual thruster in the 2 to 7 mlb_f range is approximately $50,000 each.[18] This figure includes the thruster, flow controller, housekeeping instrumentation, and control equipment. These costs are summarized in Table 4-16.

4.5.5 Monopropellant Hydrazine System

The monopropellant hydrazine thrusters, that are incorporated into the baseline spacecraft designs, represent current technology and need no special development. No costs are associated with the four 50 lb_f thrusters used for orbit maintenance because these engines are part of the spacecraft. The only costs associated with the baseline orbit maintenance system are the tankage and hydrazine. If the

PARAMETER	SPACECRAFT CONFIGURATION		
POWER SYSTEM:	SOLAR CELL-BATTERY	REACTOR-THERMOELECTRIC	
Thrust Required, mlb_f	2	3	6.4
Thrust Duty Cycle	80%	54%	25%
Individual Engine Thrust, mlb_f	2	3	6.4
Number of Engines Operating	1	1	1
Total Number of Engines	2	2	2
SYSTEM COSTS ($ in 1000's)			
NON RECURRING, DEVELOPMENT	450	450	450
<u>RECURRING, FLIGHT SYSTEM:</u>			
Electric Engine	50	50	50
Complete Propulsion System	150	150	150
Additional Power System	43	0	0
Orbit Maintenance System	193	150	150

Table 4-16. Hydrazine Resistojet System Costs.

hydrazine is stored in two, 22 inch-diameter "standard" tanks, a cost of less than $35,000 will be incurred for this system.[24]

5. COMPARISON OF ORBIT MAINTENANCE SYSTEMS

5.1 Orbit Maintenance System Mass

The mass of the orbit maintenance system is composed of the propulsion system and additional power system. These masses have been determined in Section 4.3, and the totals are presented in Table 5-1.

On the solar-powered spacecraft, the colloid system is the lightest at 229 lb_m. This is due to the fact that the colloid engine can operate at the optimum specific impulse for this application. The mercury bombardment system represents the next lightest system at 235 lb_m.

5.2 Orbit Maintenance System Volume

The volume of the orbit maintenance systems was determined in Section 4.4, and the totals for each candidate propulsion system are presented in Table 5-2.

The mercury bombardment system requires the least volume for either spacecraft configuration, 3.27 ft^3 and 1.49 ft^3 on the solar cell- and reactor-powered designs,

P A R A M E T E R	SPACECRAFT CONFIGURATION		
POWER SYSTEM	SOLAR CELL-BATTERY	REACTOR-THERMOELECTRIC	
Thrust Level, mlb_f	2	3	6.4
Thrust Duty Cycle	80%	54%	25%
PROPULSION SYSTEM			
Cesium Bombardment	335		98
Mercury Bombardment	235		66
Colloid	229	140	
Hydrazine Resistojet	412	392	392
Monopropellant Hydrazine	483	483	483

Table 5-1. Orbit Maintenance System Mass (lb_m) for the Candidate Propulsion Systems.

PARAMETER	SPACECRAFT CONFIGURATION		
POWER SYSTEM	SOLAR CELL-BATTERY	REACTOR-THERMOELECTRIC	
Thrust Level, mlb_f	2	3	6.4
Thrust Duty Cycle	80%	54%	25%
<u>PROPULSION SYSTEM</u>			
Cesium Bombardment	4.36		1.63
Mercury Bombardment	3.27		1.49
Colloid	3.60	3.18	
Hydrazine Resistojet	5.90	5.70	5.70
Monopropellant Hydrazine	7.02	7.02	7.02

Table 5-2. Orbit Maintenance System Volume (ft^3) for the Candidate Propulsion Systems.

respectively. The baseline monopropellanc hydrazine system occupies 7.02 ft^3, hence the orbit maintenance system volume could be reduced significantly by incorporating an electric propulsion engine for the drag make up function.

5.3 Orbit Maintenance System Cost

The cost of developing and using a propulsion system consist of a nonrecurring and recurring cost, respectively. Many of the propulsion systems considered are being developed by Government agencies. These include: (1) development of a 1 mlb_f cesium bombardment system by NASA-Goddard Space Flight Center for a flight on the ATS-F satellite in 1973, (2) Air Force development and qualification of a 1 mlb_f colloid system, and (3) NASA-Goddard development of hydrazine resistojets. The technology is well in hand for the building of 6.4 mlb_f cesium or mercury bombardment engines suitable for the reactor-powered spacecraft considered for this mission. Similarly, the technology required to build a 1 mlb_f mercury engine with an I_{sp} near 2000 $lb_f \cdot s/lb_m$ is well understood.

The only pertinent costs for the planners of the mission being studied are: (1) the nonrecurring cost of modifying, or developing the system and then qualifying it for the mission requirements, and (2) the recurring costs of buying the orbit maintenance system flight units for future spacecraft. These costs are summarized in Table 5-3. The top figure indicates the nonrecurring costs that the using agency must pay, and the "+ /OMS" indicates the recurring cost for each orbit maintenance system flight unit.

Clearly, the monopropellant hydrazine system represents the least costly system because there are no costs for development or thrusters, only the cost for additional tankage.

PARAMETER	SPACECRAFT CONFIGURATION		
POWER SYSTEM	SOLAR CELL-BATTERY	REACTOR-THERMOELECTRIC	
Thrust Level, mlb_f	2	3	6.4
Thrust Duty Cycle	80%	54%	25%
PROPULSION SYSTEM			
Cesium Bombardment	300 +982/OMS		800 +450/OMS
Mercury Bombardment	2000 +808/OMS		1000 +450/OMS
Colloid	250 +665/OMS		250 +650/OMS
Hydrazine Resistojet	450 +193/OMS	450 +150/OMS	450 +150/OMS
Monopropellant Hydrazine	0 +35/OMS	0 +35/OMS	0 +35/OMS

Table 5-3. Orbit Maintenance System Costs for the Candidate Propulsion System (note: the "/OMS" refers to the Cost per Orbit Maintenance System Flight Units).

5.4 Spacecraft Integration

The integration of the orbit maintenance system to the spacecraft must consider the possible interactions that the thruster may have on the entire spacecraft. In most cases the location of the thrusters will be on the rearward-most surface of the spacecraft, such that the exhaust does not impinge upon spacecraft surfaces. Serious problems could arise with electromagnetic inferference (EMI) or surface coatings if the charged particle thrusters or hydrazine systems contaminated these sensitive surfaces.

The exhaust dispersion of the hydrazine thrusters is on the order of 15^{o} in half-cone angle.[20] The colloid and bombardment engines both have the major portion of the beam (over 95%) within a 30^{o} half-cone angle. However, both bombardment engines need a 2 steradian field of view because of electrode sputtering in all directions downstream from the thruster. It has been calculated by TRW that if a solar array were located in this 2 steradian field of view of a mercury bombardment engine, it would become coated with sputtered molybdenum (the accelerating electrode material) within 30 days. Exposure of the solar array to an ion thruster should not be permitted, because it could jeopardize the performance of the power system and consequently the success of the mission.

Electromagnetic interference may also be created by charged particles interacting with spacecraft electronic devices, primarily the antennae and solar cells. Another form of EMI problems could arise from an electric potential being built up on the spacecraft due to the expulsion of electrically charged particles. Both the bombardment and colloid engines incorporate beam neutralizers to counteract this charge build up problem by injecting electrons into the exhaust beam. However, despite the neutralizers, spacecraft floating potentials of up to 50 volts may be encountered.[25] This electrical potential may create some EMI problems, and must not be over looked if spacecraft integration detailed studies are performed. The 50 V potential is not expected to create major problems, but more study is needed.

Location of the propulsion system on the spacecraft can also be a source of integration problems because of the area required by some of the electric thrusters. One 6.4 mlb_f bombardment thruster occupies 28 in^2, so two will require 56 in^2 of area on the trailing end of the spacecraft. The colloid engine requires the most area, 133 in^2, or nearly 1 ft^2. On the solar-powered spacecraft 3 ft^2 must be allocated to the colloid thrusters, and almost 4 ft^2 are needed on the reactor-powered spacecraft. In terms of this area requirement, the colloid system is the most difficult system to intergrate into the spacecraft. Both hydrazine systems have very small thruster areas, typically these areas are less than 1 in^2 and 10 in^2 for the resistojet and monopropellant hydrazine thrusters, respectively.

Location of electronic power conditioning units, propellant feed lines, and propellant tankage influences the ease of spacecraft integration. Here again, the monopropellant hydrazine system allows the greatest flexibility in terms of the location of these accessory components. Separation of thruster and hydrazine tanks does not usually create any problems. On the contrary, electric propulsion systems usually require some check out prior to mating with the spacecraft. Hence, it becomes highly desirable to have the propellant tankage and feed lines connected to the thruster. In addition, the PCU must be checked and it facilitates testing if the PCU is connected to the engine as it would be during flight. Therefore, all electric propulsion components should be placed near the thruster and not require disassembly before being mated to the spacecraft. In terms of check out and mating of the propulsion systems, the more flexible and less complex monopropellant system presents the least problems.

In summary, the baseline design monopropellant system presents no major spacecraft integration problems. The large volume of the propellant tanks (7.02 ft^3) creates the greatest concern with this system, but the tanks can be placed virtually anywhere inside the spacecraft. Conversely, the electric engines, principally the bombardment and colloid systems, present numerous problems whose solutions require careful considerations. Some of the problems that must be considered include: beam divergence which may lead to spacecraft surfaces being coated with charged particles, EMI problems could be caused by floating spacecraft potentials, and mating of the complete propulsion system to the spacecraft may require delicate handling and check out problems.

5.5 Orbit Altitude Tolerances

The mission requirements state that the spacecraft must be at an altitude of 225 $\pm$ 2 nmi throughout its two year mission. The baseline monopropellant hydrazine system is expected to meet these requirements by thrusting once every 11 days. The electric propulsion systems are scheduled to operate on higher duty cycles, up to 80% on the solar-powered spacecraft. It can be expected that the orbit can be maintained more closely to the specified 225 nmi, with thrusting on practically every orbit. If the effectiveness of the mission could be significantly improved by maintaining the altitude more cloely to the 225 nmi, rather than in the band of 223 to 227 nmi, electric propulsion can perform this task better than the high thrust (100 mlb_f) monopropellant hydrazine system. This is due to a loss in specific impulse performance of high thrust engines when operating on short burn times (e.g. less than 5 s). Since the mission requirements are specified at $\pm$ 2 nmi, the baseline hydrazine system proves to be quite adequate for the mission.

5.6 Thrust Misalignment and Thrust Vectoring

An important aspect that all propulsion systems must consider, is disturbing torques caused by thrust misalignment. Thrust misalignment can come about in two ways: (1) thrusters are not mounted on the line of the orbital velocity vector which passes through the spacecraft's center of mass, or (2) the thrusters are not aimed properly on their mounts. Both cases have the same effect, that is imparting an undesired torque on the spacecraft.

In the baseline design of the spacecraft, momentum wheels are designed to absorb disturbing torques induced by gravity gradients, solar pressure, aerodynamic pressure, and thrust misalignments.[1] Hence, the 3-axis stabilization system can be expected to handle any angular momentum created by thrust misalignment.

The ability of the stabilization system to absorb these disturbing torques alleviates the need for any re-alignments of the propulsion system's thrust, or commonly referred to as thrust vectoring. Some of the electric propulsion systems have demonstrated electrostatic and electromechanical technique for deflecting the beam or chaged particles, but these sophisticated techniques are not needed for this mission.

5.7 Nuclear Survivability

All Department of Defense space missions being planned must consider the nuclear survivability of the system. A nuclear environment has been specified, and all future spacecraft are expected to meet these criteria.[26]

In terms of the propulsion systems considered in this study, only the colloid and hydrazine systems will be capable of surviving the nuclear environment specified in reference 27. The Air Force development program for the colloid system includes numerous tests of parts, components, and subsystems to verify that the system will survive.

The most vulnerable components on the hydrazine systems are the valves. Radiation exposure can cause some valves to open, and thereby expell propellant. This will lead to spacecraft instability, loss of propellant, and eventual curtailment of the mission. Hardening of the valves is being sponsored by the Air Force, and survivable hydrazine systems will be available for this mission.

On the other hand, the cesium and mercury bombardment engines are being designed and developed by NASA, where survivability is not stressed as strongly as it is in the DoD. The most vulnerable component on these engines is the power conditioning unit. By using careful part selection, component tests, and shielding, a hardened PCU could be built for the bombardment systems, as it will be done for the colloid PCU.[9] However, this development is not being pursued at NASA, and for the near future, the bombardment engines cannot be assumed to be survivable. If the bombardment engines are needed for DoD missions, a hardened program for the system would be initiated. Such a program would probably incorporate many of the components developed for the colloid PCU. In any event, the development of a hardened bombardment system will require additional expenditures and extensive testing.

It can be concluded that only the hydrazine and colloid systems are survivable, because there are current DoD programs underway to insure this. Use of the hardened colloid PCU components could enable the bombardment systems to meet the nuclear criteria, but no programs exist for this development. Hence, the bombardment systems are assumed to be not survivable at this time.

5.8 Orbit Maintenance System Effectiveness

The effectiveness of the orbit maintenance system represents the capability of the system to perform its drag makeup functions in accordance with the mission requirements. A measure of how well the system performs, under a variety of criteria, is provided in the following weighted rating scheme (ref 14).

The effectiveness value for the power system is defined by:

$$E = \sum_i W_i R_i$$

where:

E = Effectiveness of the orbit maintenance system

W_i = Weighting factor of the i-th criterion

R_i = Rating value of the orbit maintenance system for the i-th criterion

i = Index number of the criteria.

The weighting values, W_i, used in the effectiveness model are designed to take into consideration the importance of the criteria in successfully fulfilling the mission. Each criterion is given a weighting value consistent with its definition and reflecting its importance to the mission.

The selection of the weighting values is based upon judgment. This judgment is partly based on the knowledge of the mission requirements with respect to the defined criteria, and partly on the importance of the orbit maintenance system to the mission. Some of the major criteria, such as reliability, are divided into subcriteria to permit a more accurate evaluation.

Each of the criteria was considered separately and each orbit maintenance system was given a rating number from 1 to 10 for each criterion, with 10 being the best rating possible. The rating is intended to indicate how well a given system meets the requirements of the mission for that criterion. The more desirable a feature is, the higher is the rating.

The definition of each evaluation criterion, and the corresponding selection of the weighting factor and rating values are discussed in the following paragraphs. Since two spacecraft configurations were considered in this study, two effectiveness evaluations are required. The results of the weighting-rating procedure are shown in Tables 5-4, and 5-5 for the solar cell- and reactor-powered spacecraft, respectively.

EVALUATION CRITERIA	WEIGHTING FACTOR	ORBIT MAINTENANCE SYSTEM									
		CESIUM BOMBARDMENT		MERCURY BOMBARDMENT		COLLOID		HYDRAZINE RESISTOJET		MONOPRO. HYDRAZINE	
	W	R	W•R	R	W•R	R	W•R	R	W•R	R	W•R
TECHNICAL FEASIBILITY (30)											
Mass	12	4	48	6	72	6	72	2	24	0	0
Volume	6	5	30	7	42	6	36	2	12	0	0
Spacecraft Integration	12	4	48	5	60	3	36	9	108	10	120
DEVELOPMENT (20)											
Development Risk	10	8	80	1	10	8	80	7	70	10	100
Flight Unit Costs	10	1	10	3	30	4	40	8	80	10	100
RELIABILITY (40)											
Operation	35	6	210	5	175	6	210	8	280	9	315
Partial Failures	5	7	35	7	35	7	35	9	45	10	50
NUCLEAR SURVIVABILITY	10	4	40	4	40	9	90	9	90	9	90
EFFECTIVENESS, E	100		501		464		599		709		775
RELATIVE EFFECTIVENESS		0.646		0.599		0.773		0.915		1.000	

Table 5-4. **Effectiveness Evaluation for the Solar Cell-Battery-Powered Spacecraft**

EVALUATION CRITERIA	WEIGHT-ING FACTOR	ORBIT MAINTENANCE SYSTEM									
		CESIUM BOMBARDMENT		MERCURY BOMBARDMENT		COLLOID		HYDRAZINE RESISTOJET		MONOPRO. HYDRAZINE	
	W	R	W·R	R	W·R	R	W·R	R	W·R	R	W·R
TECHNICAL FEASIBILITY (30)											
Mass	12	9	108	10	120	8	96	2	24	0	0
Volume	6	10	60	10	60	7	42	3	18	0	0
Spacecraft Integration	12	4	48	5	60	3	36	9	108	10	120
DEVELOPMENT (20)											
Development Risk	10	6	60	5	50	8	80	7	70	10	100
Flight Unit Costs	10	6	60	6	60	4	40	9	90	10	100
RELIABILITY (40)											
Operation	35	8	280	8	280	7	245	8	280	9	315
Partial Failures	5	6	30	6	30	7	35	9	45	10	50
NUCLEAR SURVIVABILITY	10	4	40	4	40	9	90	9	90	9	90
EFFECTIVENESS, E	100		686		700		664		725		775
RELATIVE EFFECTIVENESS		0.885		0.903		0.857		0.935		1.000	

Table 5-5. Effectiveness Evaluation for the Reactor-Thermoelectric-Powered Spacecraft.

1. <u>Technical Feasibility</u>: In general, technical feasibility refers to the probability that the orbit maintenance system will be able to meet the design specifications. An arbitrary weighting of 30 was given to this criterion for this mission. The weighting and power system rating of each technical feasibility sub-criteria are discussed below:

1.1. <u>Mass</u>:

This subcriterion is defined as the mass of the orbit maintenance system, which includes the masses of the propulsion and additional power system resulting from the use of the particular orbit maintenance system.

a. Weighting: The mass subcriterion was given a weighting factor of 12 because mass savings represents one of the prime reasons for using electric propulsion for drag-makeup.

b. Rating: The masses of the orbit maintenance systems ranged from 66 lb_m (mercury engine/ reactor-powered spacecraft) to 483 lb_m (monopropellant hydrazine). Assigning a rating of 10 to the lightest system, and 0 to the heaviest system, a linear interpolation between these extremes permit a rating to be given to the other systems in between. The ratings are shown in Tables 5-4 and 5-5.

1.2 <u>Volume</u>: The subcriterion volume is defined as the volume of the orbit maintenance system, including propulsion and additional power system volume.

a. Weighting: The volume is important to the mission because the space required for the orbit maintenance system may restrict the volume needed for other spacecraft components. This criterion was given a weighting of 6.

b. Rating: The volumes of the orbit maintenance systems ranged from 1.49 ft^3 (mercury engine/reactor-spacecraft) to 7.02 ft^3 (monopropellant hydrazine). Assigning a rating of 10 to the smallest system, and 0 to the largest system, interpolation of ratings for the systems within the range was possible.

1.3 Spacecraft Integration: This criterion is defined as the degree of complexity of the problems created by integrating the orbit maintenance system to the spacecraft.

a. Weighting: Complex interface restraints will affect the development time and cost of the entire spacecraft system. Integration must consider area, volume, testing, spacecraft mating, and potential interactions that the orbit maintenance system will have on the overall spacecraft. This criterion is considered to be of equal importance to the mass criterion, and therefore carries the same weighting factor (12).

b. Ratings: The ratings are based primarily on the size of the orbit maintenance system, and its effects on the spacecraft once in operation. The electron bombardment engines and the colloid system are rated the lowest because of the charged particles existing in the exhaust beam can interact with any surface within a 2π steradian field of view. The hydrazine systems do not have this problem. Electromagnetic interference problems are more likely to be encountered with these high voltage electric engines and floating spacecraft potentials are also expected. System testing prior to spacecraft mating is more difficult with the charged particle systems. The colloid system was rated lowest at 3 because there is also an area constraint imposed by the use of multiple 1 mlb_f systems. The bombardment systems were rated at 4 because of the problems mentioned above, excluding the area problem present in the colloid case. The hydrazine resistojet imposes very few constraints on the spacecraft, but the additional thrusters and power conditioning units must be located somewhere, hence a rating of 9. The monopropellant system requires no additional integration problems, and the volume of the tanks were already considered in the previous subcriterion. Thus, a rating of 10 was given.

2. Development: The development required to prepare an orbit maintenance system for an operational mission can be related to a development risk and a subsequent flight unit cost. A well-planned development program will optimize non-recurring and recurring costs for a specified application. A weighting of 20 was assigned to development, and the weighting and rating of the subcriteria discussed below.

2.1 Development Risk: This criterion is defined as the probability that the orbit maintenance system will be fully developed and ready for use at the time it is required (approximately 1 year before launch).

a. Weighting: Because of the proximity of the operational use of the system in the later-half of the 1970's, the development risk is considered to be relatively important. A weighting factor 10 was given.

b. Rating: The bulk of the system development for the election bombardment and colloid systems has been sponsored by NASA and the Air Force, respectively. The greatest nonrecurring cost expected is \$ 2 M for the low I_{sp}, 1 mlb_f mercury engine, and the least costly is the monopropellant hydrazine system. Linearly interpolations between these two extremes (\$0 = 10, and \$ 2M = 1) yields the ratings for the other systems.

2.2 <u>Flight Unit Costs</u>: This subcriterion is defined as the recurring cost associated with the operational use of the orbit maintenance system.

a. Weighting: The recurring cost of a system can influence the selection of the preferred orbit maintenance system for an operational spacecraft. A weighting of 10 was assigned to this factor.

b. Rating: The most costly system considered in this study was the cesium bombardment system on the solar cell-battery spacecraft (\$982,000/orbit maintenance system). The least costly is the monopropellant hydrazine system at \$35,000 per spacecraft. Linear interpolation between these extremes, with a rating of 1 for the \$982,000/oms and a rating of 10 for \$35,000, yields the rating values for the other systems.

3. <u>Reliability</u>: In general, reliability is the probability that the system will meet the performance specifications. For this mission, it is considered the most important evaluation criteria, and hence a weighting factor of 40 was assigned to it.

3.1 Operation: The operation reliability criterion is defined as the probability that the orbit maintenance system will meet the specified performance level and operate successfully for the specified life of the satellite.

a. Weighting: The operation of the orbit maintenance system is the most important parameter affecting the performance of the mission. A weighting of 30 was assigned to operation for the mission.

b. Rating: The operation of the electric engines on the solar cell-powered spacecraft requires longer periods of thrusting over higher duty cycles relative to the nuclear-powered spacecraft. The differences in total thrust time, 1/2 year to 1.6 years, should result in a higher reliability to the system which is on less time. Thus, the reactor-powered spacecraft systems were rated slightly higher.

Differences between propulsion systems can also be found in test results. Cesium engines have demonstrated greater life times than mercury or colloid systems so far. However, colloid systems are scheduled to operate for 10,000 hours in tests for the Air Force. It was concluded that the colloid systems should be rated slightly higher than the cesium engines for both spacecraft configurations, assuming that the 10,000 hour test will be successful. The cesium engines are rated slightly higher than the corresponding mercury systems.

The hydrazine resistojet system was rated at 8 for both spacecraft configurations because of the lack of complexity in the system. The monopropellant hydrazine system was given a rating of 9. This system was judged to be less complex than the electrically heated resistojet.

3.2 <u>Partial Failures:</u> This subcriterion is defined as the probability that the orbit maintenance system will be able to perform in the event that some component of the system fails.

a. Weighting: It is important to mission success that the spacecraft can still perform its functions even if a portion of the orbit maintenance system fails.

A system which tends to have only catastrophic failure would thus have a low rating under this criterion. A weighting factor of 5 was given because of the importance assigned to redundancy in the consideration of operating modes (Section 4.2.2).

b. Rating: The prime factor in rating the orbit maintenance systems under this criterion is the over all performance of drag makeup once one or more components fail in the propulsion system. The monopropellant hydrazine system is the least complex system with the greatest redundancy. Drag makeup can be successfully performed with only one out of the four engines operating. In fact, if all four 50 lb_f hydrazine engines failed, the mission could be extended by using the smaller 5 lb_f attitude control jets for some drag makeup. Therefore, a rating of 10 was given to the baseline system.

Partial failures to the hydrazine resistojet could also be offset by increased flow rates through the remaining thruster, or use of the small attitude control jets. In either case, the hydrazine is available to continue the mission. A rating of 9 was given to the resistojet system.

On the solar-powered spacecraft, the bombardment and cesium engines all incorporate the same mode of operation, i.e. 2 engines must operate out of the 3 that are available. A partial failure of one engine, which leads to its shut down and the startup of the standby redundant engine, brings the probability for mission success down because the margin for failures is reduced. If two engines should fail, one engine could possibly be operated at higher thrusts and over longer duty cycles in an effort to maintain the orbit. Thus, with two failed engines, partial mission success could be achieved. A rating of 7 was given to these electric systems.

On the nuclear-powered spacecraft, there is only one standby redundant bombardment thruster, one failure means there is no more margin of safety. Thus the bombardment systems are rated slightly lower (6) on the reactor configuration. The colloid system, with four engines has more flexibility in its mode of operation, hence a rating of 7 was given to it.

4. <u>Nuclear Survivability</u>: This criterion is defined as the capability of the orbit maintenance system to resist damage or destruction due to an exoatmospheric nuclear explosion.

a. Weighting: The importance of nuclear survivability of all systems being considered for space use can not be overlooked. Therefore, a rating of 10 has been assigned to this area.

b. Rating: The colloid, hydrazine resistojet, and monopropellant hydrazine systems are all being designed to be survivable, hence a rating of 9. On the other hand, the NASA-developed bombardment engines are not incorporating hardened components. Therefore, the ion engines are more vulnerable and have been rated at 4.

<u>Results of System Effectiveness Analysis:</u>

The relative effectiveness ratings, normalized with respect to the highest effectiveness (E), are presented below for the two spacecraft configurations:

Solar Cell-Batter-Powered Spacecraft:

Momopropellant Hydrazine	1.000
Hydrazine Resistojet	0.915
Colloid	0.773
Cesium Bombardment	0.646
Mercury Bombardment	0.599

This comparison indicates that the monopropellant hydrazine system is the best system for orbit maintenance on this spacecraft. The hydrazine resistojet is a close second with the colloid and bombardment systems being must less effective.

Reactor-Thermoelectric-Powered Spacecraft:

Monopropellant Hydrazine	1.000
Hydrazine Resistojet	0.935
Mercury Bombardment	0.903
Cesium Bombardment	0.885
Colloid	0.857

The monopropellant hydrazine system is the preferred system for orbit maintenance on the reactor-powered spacecraft based on this comparison. The overall comparison indicates that all candidate systems show a high relative effectiveness (all 5 systems above 0.85). In contrast, the range of relative effectiveness was lower than 0.78 for three systems on the solar-powered spacecraft.

6. CONCLUSIONS

Based on the over-all study evaluation and comparisons, conclusions were reached regarding the possible use of the various candidate electric propulsion systems for the orbit maintenance system. These conclusions are presented for the two spacecraft configurations considered for this mission.

6.1 Solar Cell-Battery-Powered Spacecraft

6.1.1 Cesium Bombardment System

The cesium bombardment engine represents a proven design that is capable of long duration operation. NASA-Goddard is sponsoring the development of a 1 mlb_f engine for flight in 1973. With minor modifications this system could be used on the solar-powered spacecraft. However, such a system is limited to a minimum I_{sp} of 2500 $lb_f \cdot s/lbm$ and this results in a large mass penalty for additional power system. The effectiveness of this system for this mission is low (0.646) because of spacecraft integration problems and recurring costs ($982,000 per flight unit).

6.1.2 Mercury Bombardment System

A 1 mlb_f bombardment system will require extensive development ($2 M) if such a system should ever fly on this mission. In addition, the low thrust, low I_{sp} (2000 $lb_f \cdot s/lb_m$) system will be plagued by the same problems as the corresponding cesium system. The effectiveness of the system resulted in a value of 0.599, the lowest effectiveness of all systems considered for either configuration.

6.1.3 Colloid System

The Air Force is sponsoring the development of a colloid system which will produce 1 mlb_f in the specific impulse range of 1000 to 1500 $lb_f \cdot s/lb_m$. Such a system is ideal for solar powered spacecraft because it can match the optimum I_{sp} required (1129 $lb_f \cdot s/lb_m$). In addition, the colloid system will have its reliability proven and it will be survivable. Spacecraft integration presents the greatest problem for the colloid system. High flight unit costs ($665,000/OMS) help contribute to the moderate effectiveness rating of 0.773.

6.1.4 Hydrazine Resistojet

Development of the hydrazine resistojet by NASA will result in a specific impulse improvement to 285 $lb_f \cdot s/lb_m$ from 230 $lb_f \cdot s/lb_m$. Such resistojets offer mass and volume savings over monopropellant hydrazine systems. No major spacecraft integration problems are expected and the system should be survivable. An overall relative effectiveness rating of 0.915 was given to the system.

6.1.5 Monopropellant Hydrazine

The monopropellant hydrazine system is a flight-proven system capable of performing all propulsion functions required by this mission. Orbit insertion and drag make-up can be performed adequately by the same set of 50 lb_f-thrusters. In addition, attitude control thrusters are monopropellant hydrazine, and all engines can be fed from common propellant tanks.

The monopropellant system is the most massive of all orbit maintenance systems considered for this mission (483 lb_m). However, this system is the least expensive orbit maintenance system because the thrusters are part of the spacecraft already. There are no major spacecraft integration problems with the hydrazine system. Survivability does not present a major problem for hydrazine systems. The monopropellant hydrazine system had the highest effectiveness rating of all systems considered for this mission.

6.2 Reactor-Thermoelectric-Powered Spacecraft

6.2.1 Cesium Bombardment System

A cesium engine capable of producing 6.4 mlb_f has been built and tested by Electro-Optical Systems for 8200 hours. On the reactor-powered spacecraft, such a system would operate at an I_{sp} of 5000 $lb_f \cdot s/lb_m$ for a total duration of 6 months. Low system mass (98 lb_m) and high reliability results in a relative effectiveness rating of 0.885. Spacecraft integration problems and questionable survivability are the major shortcomings of the system.

6.2.2 Mercury Bombardment System

The flight of the SERT II indicates that mercury bombardment systems can be built and reliably operated in space at thrust levels near 6.4 mlb_f, and with an I_{sp} near 4500 $lb_f \cdot s/lb_m$. Use of a similar type system on the reactor-powered spacecraft will result in the lowest orbit maintenance system mass (66 lb_m) and volume (1.49 ft^3). As in the cesium engines, problems are expected with integration and survivability. The relative effectiveness of the system was assessed to be 0.903.

6.2.3 Colloid System

Multiple use of the 1 mlb_f colloid system, being developed by the Air Force, would be incorporated on a reactor powered spacecraft. Three out of four engines would thrust 54% of the time at an I_{sp} of 1500 $lb_f \cdot s/lb_m$. The system encounters the same problems mentioned for it on the solar-powered spacecraft, where integration and high flight unit costs ($650,000/OMS) being the biggest drawbacks. A relative effectiveness rating of 0.857 was the result of the analysis. The colloid system was judged to be the least effective on the reactor powered spacecraft.

6.2.4 Hydrazine Resistojet

As on the solar cell-powered spacecraft, the hydrazine resistojet offers a mass and volume savings over the monopropellant hydrazine system. No problems are anticipated with the resistojet, and therefore a relative effectiveness rating of 0.935 was given to it.

6.2.5 Monopropellant Hydrazine

The same conclusions discussed for the monopropellant hydrazine system under the solar-powered spacecraft (Section 6.1.5) hold here. The system is the most effective for either spacecraft configuration.

7. RECOMMENDATIONS

The preferred propulsion system for the orbit maintenance function on this mission is the baseline monopropellant hydrazine system. This is true for either a solar cell- or reactor-powered spacecraft. This recommendation is based on the fact that a monopropellant thruster will already be on board the spacecraft for orbital insertion, and these same thrusters can perform drag make up maneuvers quite adequately. The complications involved with incorporating an electric propulsion system into the spacecraft are not justified for the savings in mass and volume. Besides, the electric propulsion systems are much more costly than the baseline system.

Only in the event that spacecraft mass becomes a critical issue should electric propulsion be considered again. In such a case, a careful trade off must be made between pounds saved and dollars required to be spent.

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