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# EVALUATION OF THE USE OF ON–BOARD SPACECRAFT ENERGY STORAGE FOR ELECTRIC PROPULSION MISSIONS

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16. Abstract

On-board spacecraft energy storage represents an under-utilized resource for some types of missions that also benefit from using relatively high-specific-impulse capability of electric propulsion. This resource can provide an appreciable fraction of the power required for operating the electric propulsion subsystem in some missions. The most probable mission requirement for utilization of this energy is that of geostationary satellites which have secondary batteries for operating at high power levels during eclipse. The study summarized in this report selected four examples of missions that could benefit from use of electric propulsion and on-board energy storage. Engineering analyses were performed to evaluate the mass saved and economic benefit expected when electric propulsion and on-board batteries perform some propulsion maneuvers that would conventionally be provided by chemical propulsion. For a given payload mass in geosynchronous orbit, use of electric propulsion in this manner typically provides a 10% reduction in spacecraft mass.

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# FOREWORD

The work described in this report was performed by a study team, with participation from Hughes Research Laboratories and Hughes Space and Communications Group. Program management and definition of electric-propulsion-technology specifications were provided by Hughes Research Laboratories. Identification of specific missions, technology comparisons, and calculation of benefits was performed by the staff of Hughes Space and Communications Group. Emphasis has been placed on establishing the perspective of the spacecraft system designer in selecting missions and investigating the benefits of using electric propulsion and/or on-board energy storage to satisfy spacecraft propulsion requirements. The key contributors are listed below:

| | |
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# INTRODUCTION

The energy storage carried on-board some spacecraft represents an un-tapped power resource that could provide some fraction of the spacecraft propulsion requirements using electric propulsion technology. The study summarized here was an investigation of the benefits that would be obtained by using electric propulsion technology to perform some of the propulsion requirements for certain categories of near-term missions that are either under development or that have already been studied. Attention was focused on missions that would benefit from the use of on-board energy storage to supply the power required for an electric propulsion system. The objective of the study was to quantitatively evaluate this benefit in comparison with chemical propulsion. Our approach toward achieving this objective was to define and analyze the chemical propulsion maneuvers for several representative missions. The scope of the study was limited to those mission concepts for which data were readily available to define the power, mass, and propulsion requirements of the spacecraft.

There was no attempt under this study to formulate new mission concepts that might be performed more favorably by electric propulsion. Consequently, most missions that fall within the scope of the study were formulated originally to exploit chemical-propulsion technology, and do not require an electric propulsion system in order to meet mission objectives. Although some ambitious missions of planetary exploration have previously been described which take full advantage of the high-specific-impulse capability of electric propulsion, these missions were also not included since they cannot be performed using chemical propulsion. By process of elimination, therefore, propulsion maneuvers for earth-orbiting satellites were found to be the only candidate missions suitable for comparison of chemical and electric propulsion systems.

The selection of the power and propulsion technologies that were compared under the study was governed by guidelines from several sources: (1) those provided under the contract, (2) the requirements of candidate propulsion maneuvers, and (3) the availability of data that are representative of the state of the art. The study team adopted a relatively conservative posture that is consistant with Hughes Aircraft Company's approach to manufacturing high-reliability commercial and military spacecraft in identifying acceptable power and propulsion technologies and in specifying their performance characteristics. Consequently, the already appreciable benefits projected through the use of electric propulsion are considered to be conservative estimates, and even greater benefits could be identified by selecting more advanced thruster and/or power technologies.

## ANALYSES AND RESULTS

The discussion of the analyses and results of the study reported here follows the outline shown in Table 1. In identifying the candidate mission set, emphasis was placed first on defining a generic mission that would be representative of a class of missions and then on formulating a spacecraft design that could make use of data from previous studies or ongoing programs for defining the spacecraft characteristics (mass, power, etc.). Four such missions were identified; both chemical and electric-propulsion-system technnologies were then modeled for each mission to be consistent with the study guidelines and the mission requirements. Some technology tradeoffs were analyzed to obtain the results shown in this report. In each case, the baseline spacecraft was designed using chemical propulsion exclusively and the electric propulsion alternatives were obtained by removing only that portion of the chemical propellant that was budgeted for the maneuver being performed by the electric propulsion subsystem. A complete set of chemical-propulsion hardware was retained in all cases for performing attitude-control maneuvers. The major comparison made was in the overall spacecraft mass (on-orbit) relative to the baseline spacecraft. The mass reduction that is obtained using electric propulsion technology represents a benefit that can be realized either in increased payload or reduced launch costs (in comparison with the baseline spacecraft). While the assessment of the

Table 1. Study Elements

- Identification of Electric Propulsion Missions
- Definition of Technologies to be Compared
- Formulation of Point Designs for Comparing Chemical and Electric Propulsion Systems
- Comparison of Benefits of Battery Powered and Solar-Cell Powered Electric Propulsion Systems

monetary value of this mass benefit is admittedly subjective, we have provided our estimate of this benefit for each case studied.

## A. ELECTRIC PROPULSION MISSIONS

Propulsion requirements for any given mission are characterized by the velocity increment required, the payload mass and the thrusting time. The main advantage offered by electric propulsion over chemical propulsion is the capability for providing higher propellant exhaust velocity (or specific impulse, $I_{sp}$), thereby reducing the mass of propellant required to achieve a given velocity increment ($\Delta v$). Figure 1 compares the initial mass of a spacecraft (1000 kg dry mass) that uses a chemical propulsion system ($I_{sp}$ = 400 sec) with one that uses an electric propulsion system ($I_{sp}$ = 3000 sec) as a function of the velocity increment, $\Delta v$, required for certain maneuvers. The mass differential represents the difference in the required propellant mass. At a low value of $\Delta v$, the differential in propellant mass is small, and there is little or no incentive to use an electric propulsion system. To achieve equivalence in spacecraft mass using electric propulsion to replace chemical propulsion for a given maneuver requires that the mass of the chemical propellant that would have been used for that maneuver must equal or exceed the combined mass of the electric propulsion hardware and propellant. In fact, to be economically attractive, the mass of the chemical propellant removed must exceed the mass of the electric propulsion system by an amount that offsets the cost of the electric propulsion hardware. An analysis of the required mass differential is described later in Section B.2.

Some propulsion maneuvers can be shown to be attractive from both a mass and economics viewpoint but are of questionable user acceptability. For instance, maneuvers such as transfer from low earth orbit (LEO) to geosynchronous orbit (GEO) appear extremely attractive from the standpoint of the propellant mass

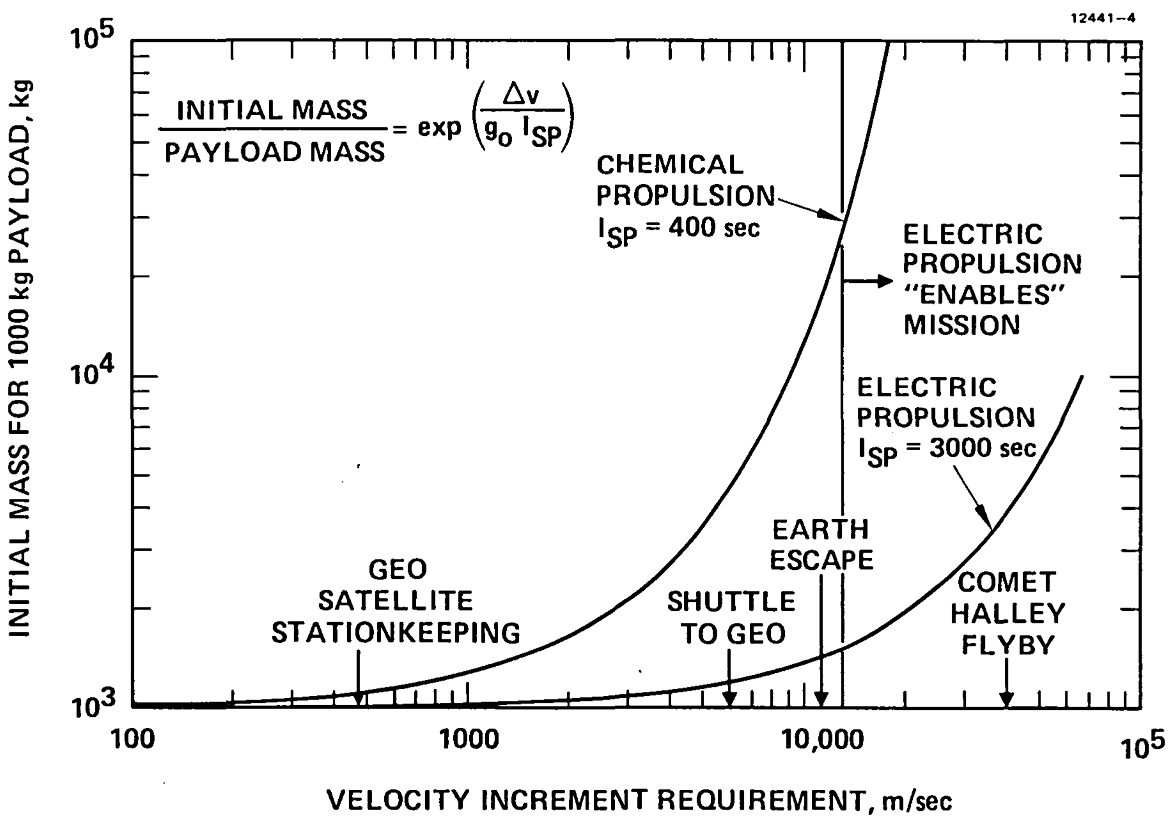


Figure 1. Ion propulsion benefits (vis-à-vis chemical propulsion).

differential. Use of electric propulsion may not be particularly attractive however, because the transfer time may be unacceptably long. Unless the payload is itself a power source, the mass of the power source required to achieve relatively short transfer times (approximately less than 60 days) becomes comparable to the propellant mass differential and electric propulsion loses its advantage. If longer transfer times are compatible with mission objectives, however, electric propulsion can provide a substantial improvement in payload mass-fraction for orbit transfer maneuvers.

Ambitious propulsion maneuvers with $\Delta v$ requirements in excess of about 12 km/sec would require quantities of chemical propellant that exceed shuttle launch capacity (for a 1000-kg spacecraft) and thereby the use of electric propulsion becomes mission enabling. For this study, however, the range of missions was restricted to those in the intermediate range ($400 \text{ m/s} < \Delta v < 12{,}000 \text{ m/s}$) where either chemical or electric propulsion could perform the mission so that mass and economic benefits could be compared. This restriction more or less constrains the missions of interest to be "earth orbital" missions.

A baseline mission set was defined that consists of four generic missions which are representative of the majority of "earth orbital" missions. Table 2 lists these missions with their distinguishing characteristics and representative data base. We studied two large communications satellites so that the differences between spin-stabilization and 3-axis stabilization could be assessed. A large radar satellite was studied as an example of the largest satellite that could be put into orbit with a single shuttle launch using electric propulsion (but requiring several shuttle launches and LEO assembly using chemical propulsion). The military satellite is a totally conceptual mission. For this mission, we used characteristics like those proposed in a composite of unclassified studies to specify the spacecraft mass, and

Table 2. Baseline Mission Selections

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| MISSION | INFORMATION BASE |
|---|---|
| ● COMMUNICATION SATELLITE<br>– STS LAUNCHED<br>– 10 YEAR LIFE<br>– GEOSYNCHRONOUS EARTH ORBIT<br>– SPIN STABILIZED | ● STUDIES OF NEXT GENERATION OF GEOSYNCHRONOUS ORBITING COMMUNCIATIONS SPACECRAFT (INTELSAT VI) |
| ● COMMUNICATION SATELLITE<br>– STS LAUNCHED<br>– 10 YEAR LIFE<br>– GEOSYNCHRONOUS EARTH ORBIT<br>– 3-AXIS STABILIZED | ● STUDIES OF NEXT GENERATION OF GEOSYNCHRONOUS ORBITING COMMUNICATIONS SPACECRAFT (TDRSS, SATCOM) |
| ● RADAR SATELLITE<br>– STS LAUNCHED LARGE SPACECRAFT<br>– 10 YEAR LIFE<br>– GEOSYNCHRONOUS EARTH ORBIT<br>– 3-AXIS STABILIZED | ● RECENT UNCLASSIFIED STUDY EMPHASIZING THE MISSION APPLICATIONS OF HIGH ENERGY DENSITY (HED) BATTERIES |
| ● MILITARY SATELLITE<br>– MOLNIYA (CRITICALLY INCLINED) ORBIT<br>– STS LAUNCHED<br>– 10 YEAR LIFE<br>– 3-AXIS STABILIZED | ● GENERIC UNCLASSIFIED MISSION STUDIES |

propellant and power budgets. All of these missions require sufficiently large maneuvers ($\Delta v$) to benefit from electric propulsion's higher-specific-impulse capability, and all have a requirement for secondary batteries so that the use of on-board energy storage could be evaluated.

## B. PROPULSION SYSTEM TECHNOLOGIES

In comparing the relative performance of chemical and electric-propulsion system capabilities, it is necessary to compare technologies of the same relative maturity. Similarly, power and energy-storage technologies were selected that we considered representative of present or achievable state-of-the-art. Table 3 lists the technologies considered and designates the ones used in the study. Monopropellant-hydrazine thruster systems were not considered because their performance is inferior to bi-propellant thrusters. Nuclear reactors were not included in the study as candidate power sources because their development status is not mature enough to quantitatively assess their capability. Fuel cells were not included because they are considered inferior to solar cells in specific mass and lifetime for the missions selected. Batteries are considered to be the only energy-storage technology with maturity comparable to the other technologies in the study. Nickel-hydrogen battery technology represents the state-of-the-art in secondary battery technology and sodium-sulfur (or other alkali battery technology is a probable advanced battery. Nickel-cadmium-battery technology was not evaluated under the study because we consider the mature nickel-hydrogen battery to be superior in both cycle-life and depth of discharge properties (based on available data).

Of the three generic types of electric thrusters listed in Table 3, only the mercury ion thruster satisfies both the operating-characteristics and maturity guidelines governing our study approach. Teflon pulsed-plasma thrusters have the requisite maturity but for operation at appreciably smaller

Table 3. Propulsion System Technologies

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| TECHNOLOGIES CONSIDERED | USED IN STUDY |
|---|---|
| CHEMICAL PROPULSION SYSTEM | |
| MONOPROPELLANT (HYDRAZINE) | NO |
| BI-PROPELLANT (MONOMETHYL HYDRAZINE, NITROGEN TETROXIDE) | YES |
| ELECTRIC PROPULSION SYSTEM | |
| ● POWER SOURCES | |
| PHOTOVOLTAIC (Si, GaAs) | YES |
| NUCLEAR REACTOR | NO |
| FUEL CELL | NO |
| ● ENERGY STORAGE | |
| BATTERIES ($NiH_2$, NaS) | YES |
| RECHARGEABLE FUEL CELLS | NO |
| ● THRUSTERS | |
| ELECTROSTATIC (ION) | YES |
| ELECTROMAGNETIC (PLASMA) | NO |
| ELECTROTHERMAL | NO |

thrust levels and for total impulse less than the mission requirements we have considered. Similarly, electrothermal thrusters have maturity in operational ranges at lower specific impulse than we studied here ($I_{sp} > 1500$ sec). Pulsed MPD thrusters and other pulsed-plasma thrusters have not reached a state of development that permits a realistic assessment of thrust-subsystem-mass properties and power requirements. Ion-thrust-subsystem mass performance was modeled to fit the characteristics of the flight-ready NASA/Hughes 8-cm-diameter 5-mN thruster and its demonstrated extended performance capabilities and/or the NASA/Hughes 30-cm-diameter 130 mN thruster and its demonstrated operating characteristics at lower power levels.

## C. DESCRIPTION OF THE SPACECRAFT FOR THE BASELINE MISSION SET

In defining the spacecraft for each mission, we postulated a value of mass for the total spacecraft that we perceived to be representative of the generic mission type in future applications. Then, we scaled all of the remaining characteristics and specifications to be consistent with this overall mass for a baseline spacecraft designed to use a totally chemical-propulsion system. We analyzed each baseline spacecraft design and the electric propulsion variations on each spacecraft design in enough detail to estimate the mass and power characteristics; however, the details of packaging for launch and deployment were not studied.

### 1. Mission I - Spin-Stabilized Communcations Satellites

The characteristics for the baseline spin-stabilized satellite (Mission I) are listed in Table 4. North-South station-keeping is the only propulsion maneuver that can be performed to advantage with electric propulsion for this mission. Consequently, the full complement of chemical thrusters is retained and only the chemical propellants required

Table 4. Characteristics for Mission I

| Specification | Value or Comment |
|---|---|
| Mission | 10 Year Geosynchronous Orbit, Spin-Stabilized Communication Satellite |
| BOL Mass | 2,460 kg |
| EOL Mass | 2,000 kg (Approximately) |
| Maximum Eclipse Duration | 1.2 hrs |
| Batteries | 1982 $NiH_2$, Capable of 5,000 Cycles at 0.8 Depth of Discharge, $\alpha_B$ = 63.3 kg/kW-hr |
| Recoverable Energy Storage | 2.0 kW/Hrs |
| Payload Battery Cycles | 1,000 AT 0.7 Depth of Discharge |
| Propulsion System - Solid | Solid Rocket Motor Injection into transfer to GEO |
| - Liquid | $MMH/N_2O_4$ Bipropellant for Augumentation and on-orbit use |
| Proposed Electric Propulsion Maneuvers | N-S Stationkeeping performed 26 times per year Chemically; 464 m/sec |
| Corresponding Chemical Performance | Thrusters operate at an average Isp of 280 to 305 sec. 430 kg propellant is required |
| Solar Panel Technology | 1982 Silicon, Cylindrical Geometry, $P_{BOL}$ = 2.3 kW, $P_{EOL}$ = 2.15 kW, $\alpha_{SP}$ = 67.8 kg/kW |

for North-South stationkeeping are off-loaded. To minimize the electric-propulsion-subsystem hardware mass and cost, only two mercury ion thrusters would be used, mounted as shown in Figure 2 (one to satisfy the propulsion requirements, the other for redundancy to satisfy the "single failure tolerance" requirement). No gimbals are required, and any perturbations from net non-axial thrust components can be readily removed using the chemical thrusters with negligible mass penalty. With the thrusters mounted in this manner the exhaust plume from the thrusters will have minimal impact on the spacecraft.

An important consideration in implementing electric propulsion on spacecraft of this configuration is that the addition of a solar-panel area will increase the spacecraft length and subsequent launch costs, even though the mass added is not appreciable (i.e., spin-stabilized spacecraft tend to be power-limited by limits on spacecraft length dictated by efficient use of cargo volume).

## 2. Mission II - Three-Axis Stabilized Communications Satellites

The characteristics for the three-axis-stabilized communications satellite mission are listed in Table 5. For satellites of this type, several propulsion maneuvers are tractable and the mass benefit achieved is proportional to the maneuvers performed. As in Mission I, the largest potential benefit is obtained by using electric propulsion for North-South stationkeeping. The electric propulsion subsystem can also be used to advantage for final orbit acquisition (circularization) if a 60-day time period (arbitrarily) selected for achieving final orbit can be tolerated. Although there is presently no indication that a 60-day orbit circularization would be accepted by any potential communications-satellite customer, we have evaluated the benefit as an option. East-West stationkeeping

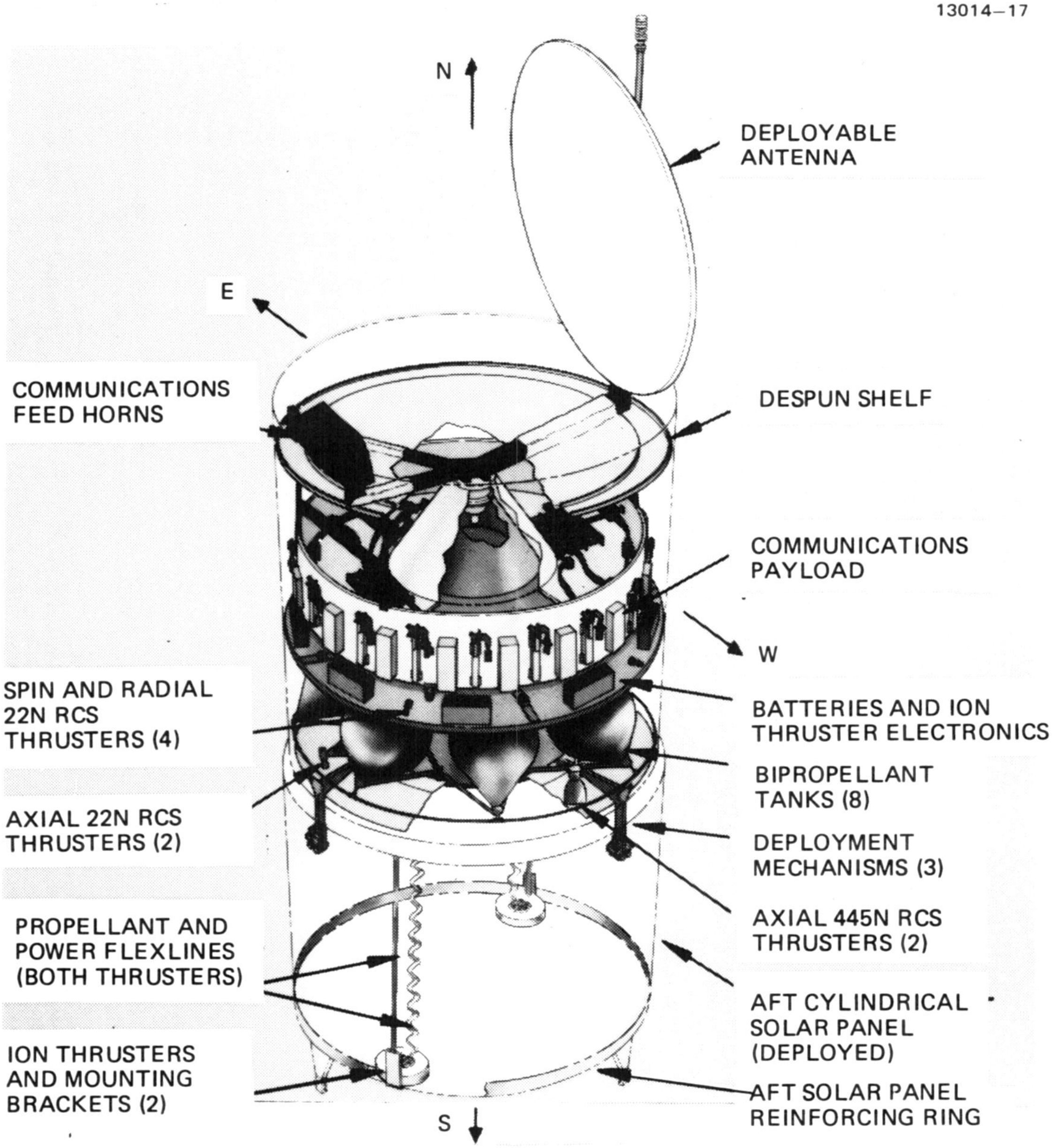


Figure 2. Spin-stabilized communications satellite showing detail of ion thruster integration.

Table 5. Characteristics for Mission II

| Specification | Value or Comment |
|---|---|
| Mission | 10 Year Geosynchronous Orbit, 3-Axis Stabilized Communication Satellite |
| BOL Mass | 2,460 kg |
| EOL Mass | 2,000 kg (Approximately) |
| Maximum Eclipse Duration | 1.2 hrs. |
| Batteries | 1982 $NiH_2$, Capable of 5,000 Cycles at 0.8 Depth of Discharge, $\alpha_B$ = 63.3 kg/kW-hr. |
| Recoverable Energy, Storage | 2.0 kW-hr |
| Payload Battery Cycles | 1,000 at 0.7 Depth of Discharge |
| Propulsion System - Solid | Solid Rocket Motor Injection into transfer to GEO |
| - Liquid | MMH/$N_2O_4$ Bipropellant for augumentation and on-orbit use |
| Proposed Electric Propulsion Maneuvers | Final circularization (Optional), N-S SKPG, and E-W SKPG; $\leq$ 696m/sec |
| Corresponding Chemical Performance | Thrusters operate at an average Isp of 280 to 305 sec. 630 kg of propellant is required |
| Solar Panel Technology | 1982 Silicon, Flat Geometry, $P_{BOL}$ = 2.3 kW, $P_{EOL}$ = 2.15 kW,<br>$\alpha_{SP}$ = 21.6 kg/kW |

can be performed simultaneously with North-South stationkeeping by providing a velocity component along the orbit trajectory, either by a fixed angle of the thrusters or by rotating the satellite body slightly during stationkeeping maneuvers. As in the case of the spinning spacecraft, the ion thrusters must be mounted for thrusting predominantly in a North or South direction and a full complement of chemical thrusters must be retained for attitude control. Traditionally, 3-axis stabilized communications satellites are designed to provide power with flat solar panels extended in both directions along the North-South axis of the satellite. This practice creates a problem for optimal location of the ion thrusters. To avoid interaction between the exhaust plume of the ion thruster and this solar panel configuration, the thrusters have to be mounted at a relatively large cant angle with respect to the North-South axis (30° to 45°).

Mounting the thrusters with large cant angles requires that a higher total impulse be provided and also produces an undesirable, effective East-West drift of the satellite orbit unless the thrust component in the plane of the orbit is cancelled by operating thrusters in pairs. This mode of operation requires a minimum of four thrust subsystems. Therefore, ion propulsion could be implemented most effectively on a 3-axis stabilized satellite if the satellite were configured to provide an unobstructed view along the North or South axis of the satellite like the configuration shown Figure 3. In this assymetric orientation of the solar panel, the solar pressure would be balanced by a deployable cylindrical sail that has negligible mass and cost (in comparison to the solar panel) but which balances the torque resulting from solar pressure on the solar array. Any interaction of the ion-thruster exhaust with the cylindrical sail would have minimal impact on satellite operation, and could, in fact, shield the rest of the satellite from any ion thruster efflux. External reflectivity changes may occur, however these are not critical to the present analysis.

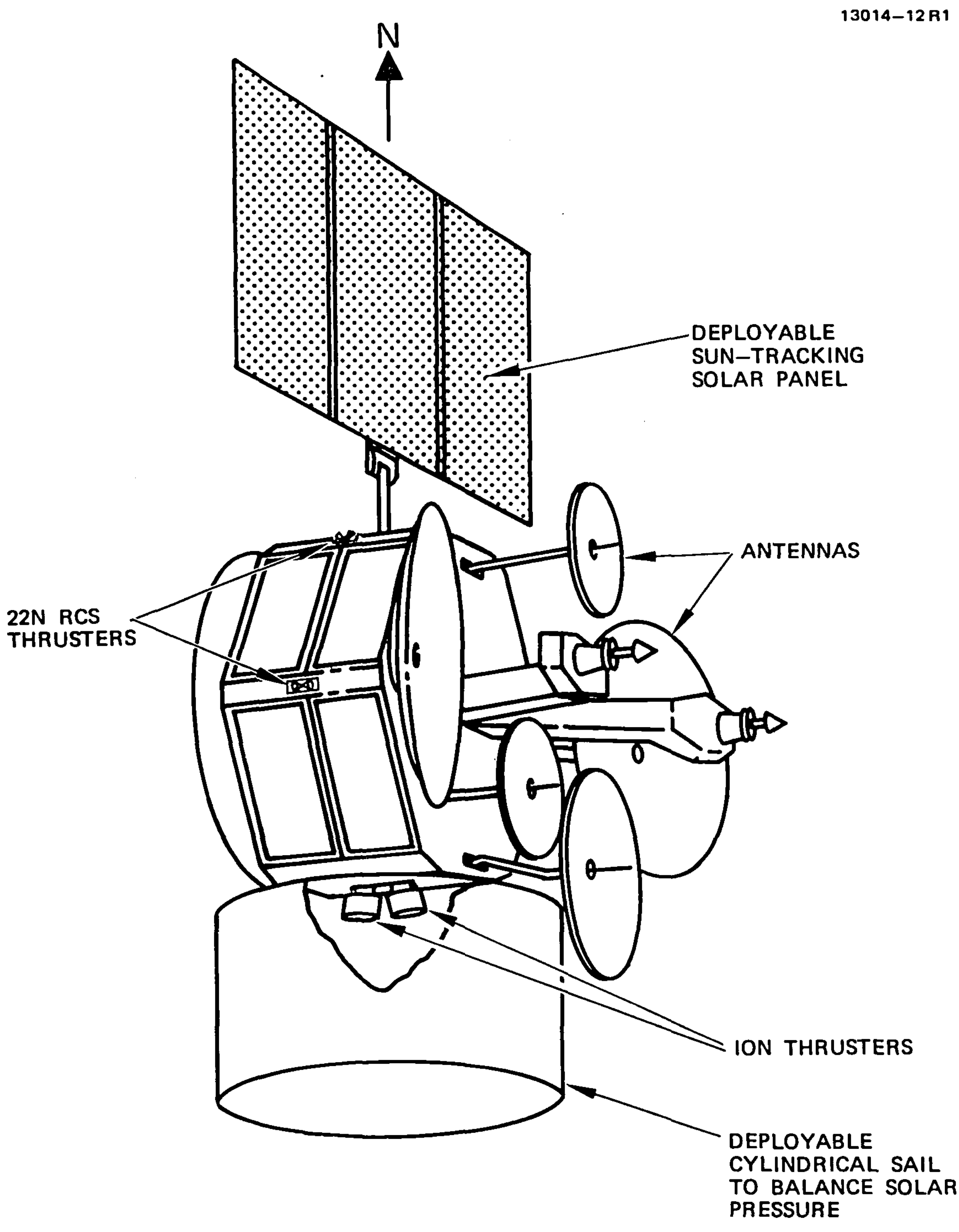


Figure 3. Three-axis stabilized spacecraft configuration for optimum utilization of ion propulsion with minimum hardware.

Other non-conventional assymmetric satellites can also be conceived to provide the ion thruster a relatively unobstructed beam path along the North-South axis. In this study, we did not consider configurations that require mounting the thrusters at large angles to avoid interaction with a North-South oriented solar panel.

### 3. Mission III - Large Antenna, 3-Axis, Stabilized Satellite

The third mission analyzed was designed to propel a high-power large-antenna radar satellite into geosynchronous orbit with the largest possible mass that could be achieved with a single shuttle launch and using electric propulsion for orbit transfer. The characteristics for the mission are shown in Table 6. The radar antenna was designated to be approximately 60 m in diameter; however, deployment and storage considerations were not addressed (analysis was based on mass only). Figure 4 shows the satellite in the orbit-transfer configuration. The orbit-transfer propulsion is supplied by 15 30-cm-diameter mercury ion thrusters operated at extended performance capability for approximately 6700 hours using the 125 kW of solar-panel power which is postulated to be available during orbit transfer. Note that all technologies have been advanced to relatively optimistic projections. Figure 5 shows the satellite configuration in geosynchronous orbit. For North-South stationkeeping, 30-cm mercury ion thrusters were mounted in pairs on gimbals on opposite sides of the antenna. The thrusters would be operated in pairs and the effective torques would be balanced by adjustment of the thruster gimbal angles. East-West stationkeeping was also provided by minor gimbal-angle adjustment. The mass benefit for this mission is (by design) large enough to require additional shuttle launches for the chemical propulsion baseline.

Table 6. Characteristics for Mission III

| Specification | Value or Comment |
|---|---|
| Mission | 10 Year Geosynchronous Orbit, 3-Axis Stabilized Radar Satellite (RADSAT) |
| BOL Mass | *46,489 kg Deployed in LEO; 18,285 delivered to GEO chemically |
| EOL Mass | 14,880 kg (Approximate) |
| Maximum Eclipse Duration | 1.2 hrs |
| Batteries | 1982 $NiH_2$, Capable of 5,000 Cycles at 0.8 Depth of Discharge, $\alpha_B$ = 63.3 kg/kW-hr |
| Recoverable Energy | 187 kW-hr; with 20% distribution loss, 150 kW-hr seen at the load |
| Payload Battery Cycles | 1,000 |
| Propulsion System - Solid | (Not Employed) |
| - Liquid | $H_2/O_2$ Cryogenic OTV; $MMH/N_2O_4$ RCS for on-orbit use |
| Proposed Electric Propulsion Maneuvers | All except uncontrolled attitude recovery below 500 km in LEO and on-orbit altitude control (RCS) |
| Corresponding Chemical Performance | OTV Isp = 470 sec; Isp = 300 sec |
| Solar Panel Technology | Advanced flat panel gallium arsenide $P_{BOL}$ = 167 kW, $P_{EOL}$ = 125 kW, $\alpha_{SP}$ = 8.4 kg/kW |

* The chemical baseline spacecraft mass deployed in LEO requires 2 shuttle launches.

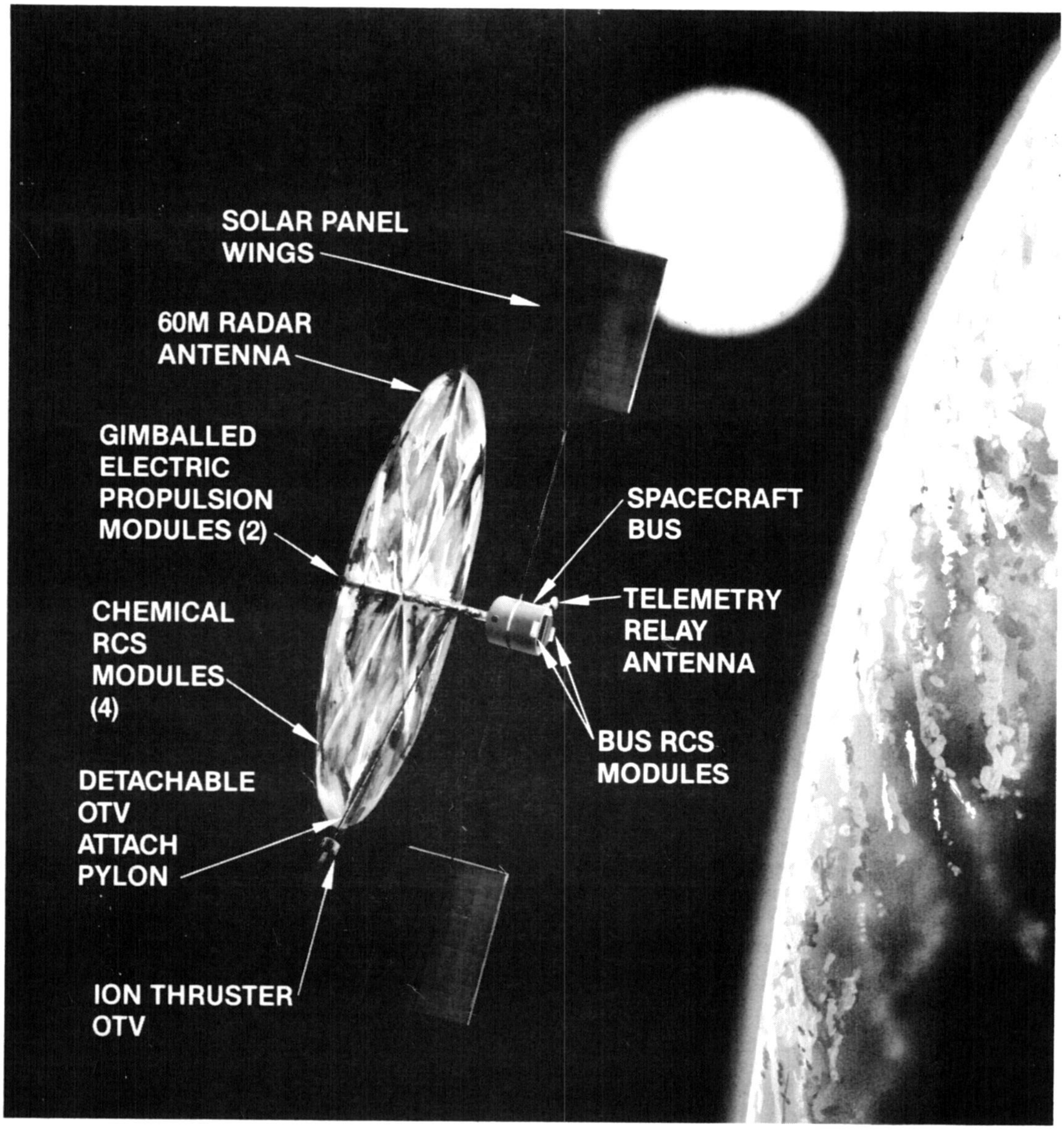


Figure 4. Spacecraft configured for Mission III showing in low drag orientation for orbit raising from LEO to GEO.

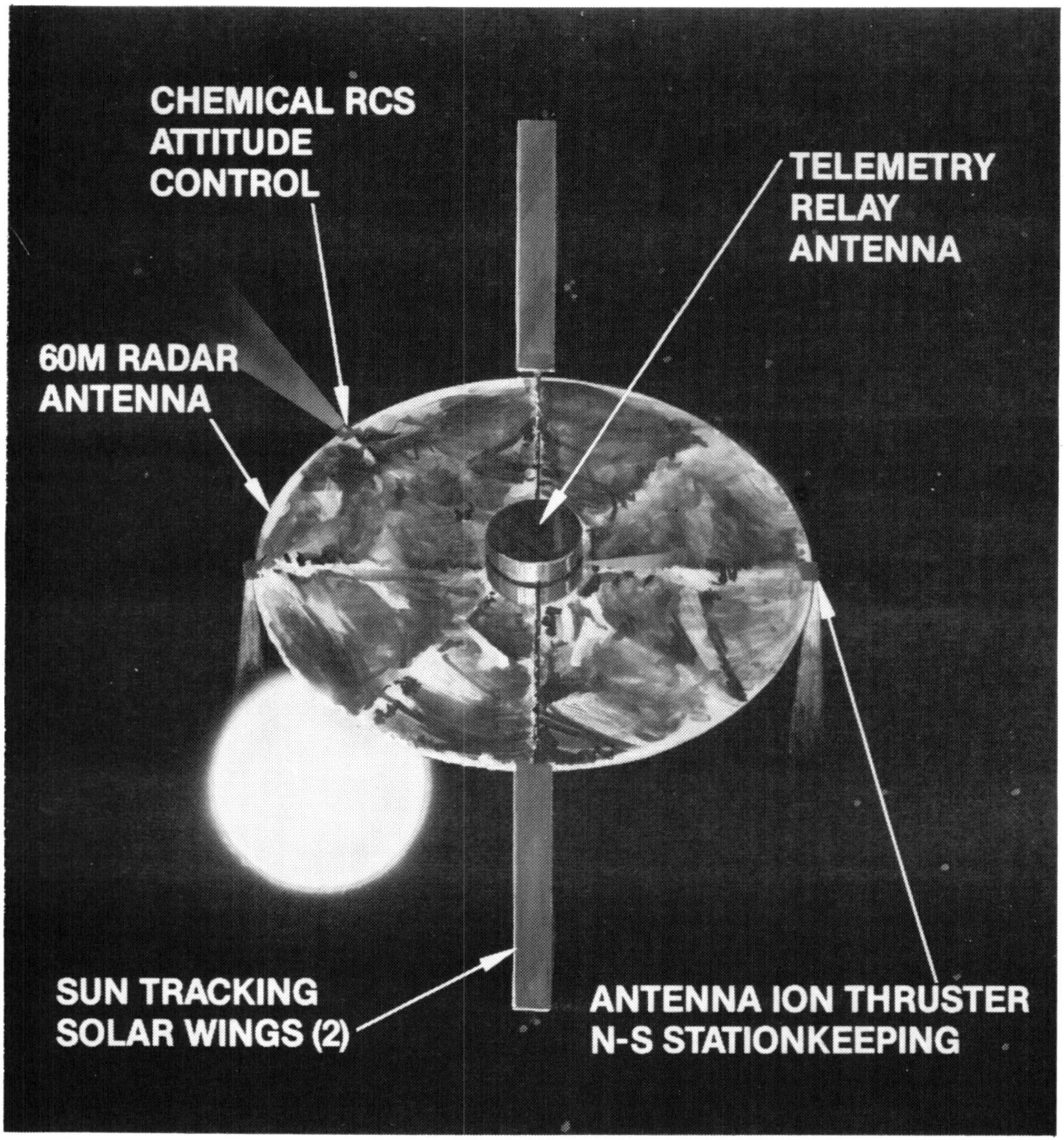


Figure 5. Spacecraft configuration for Mission III showing on-orbit orientation and stationkeeping thruster locations.

### 4. Mission IV - A Military Type Satellite

The characteristics for Mission IV, a military type satellite, are shown in Table 7. The mission scenario calls for a 3-year storage period at low earth orbit (LEO) and some orbit maintenance in the final critically inclined elliptic orbit (CIEO). A possible spacecraft configuration is shown as Figure 6. The mission scenario calls for using electric propulsion to provide the apogee raising portion of orbit transfer from LEO to CIEO during the 3-year storage period, and cyclic drag make-up. Power available on such a satellite could vary from 2 to 20 kW; consequently, the net mass benefit was evaluated at both extremes.

## D. RESULTS OF THE POINT DESIGN ANALYSES

Using analytic models to determine the thruster-subsystem mass and power requirements (see Appendix A) as functions of thrust and specific impulse, we performed trade-off studies to determine the effect of operating for longer times at lower thrust, or vice-versa. We also examined the relative merit of augmenting the power subsystem with either batteries or solar panels. Our conservative performance models for the ion-thruster-subsystem required the addition of some power in all cases. The results of each design variation were quantified as a net mass benefit in comparison to the baseline spacecraft that is designed to perform the mission with chemical thrusters. We begin discussion of these results by comparing Missions I and II.

Table 7. Characteristics for Mission IV

| Specification | Value or Comment |
|---|---|
| Mission | 3 Year LEO storage and 7 year active mission in critically-inclined elliptic orbit (CIEO) |
| BOL Mass | 17,200 kg deployed from shuttle (middle of STS capability) |
| EOL Mass | 6,320 kg (Approximate) |
| Maximum Eclipse Duration | 2/3 hr in LEO; 1 hr in CIEO |
| Batteries | 1982 $NiH_2$, Capable of 5,000 Cycles at 0.8 Depth of Discharge, $\alpha_B$ = 63.3 kg/kW-hr |
| Recoverable Energy | 2.0 kW-hr and 20.0 kW-hr variants studied |
| Payload Battery Cycles | 2,8000 in CIEO |
| Propulsion System - Solid | (Non Assumed) |
| - Liquid | $MMH/N_2O_4$ Bipropellant for LEO and CIEO use |
| Proposed Electric Propulsion Maneuvers | 144 m/sec of LEO apogee raising; 160 m/sec on-orbit |
| Corresponding Chemical Performance | 1,190 kg at an average Isp of 300 sec |
| Solar Panel Technology | Advanced flat panel gallium arsenide $P_{BOL}$ = 2.7 and 27 kW variants, $P_{EOL}$ = 2.0 and 20 kW (respectively), $\alpha SP$ = 8.4 kg/kW |

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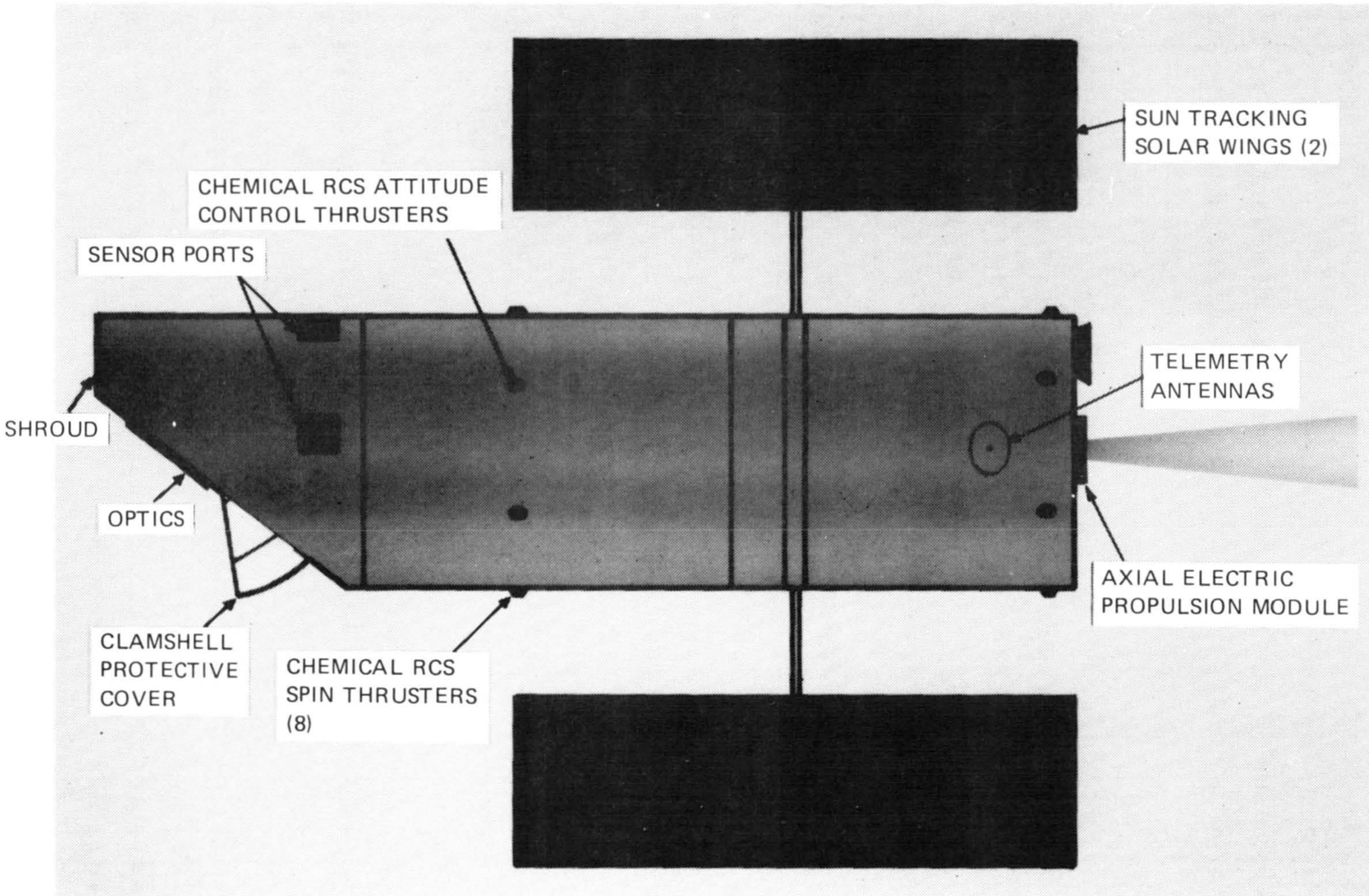


Figure 6. Spacecraft configuration for Mission IV.

## 2. Communications Satellite Missions (I and II)

The results of our point-design analysis produced summary tables such as those shown as Tables 8 and 9. The spacecraft mass and propulsion requirements are relatively constant for both missions and point designs. For this discussion, we have selected a point design for the spin-stabilized satellite that uses a 0.035-N thruster (see Table 8). The operating time required in correspondence to this thrust level is 3.8 hours on each of the 265 days per year that stationkeeping is performed (no stationkeeping is performed on days involving eclipse). Note that this relatively long thrusting period increases the effective velocity increment ($\Delta v$) that must be supplied for North-South stationkeeping.

For the 3-axis-stabilized satellite, we show the results obtained by using a larger (30cm dia.) thruster at higher thrust and power. Table 9 also shows the benefits of using these larger thrusters (in comparison with the smaller thrusters shown in Table 8 for Mission I) for final-orbit circularization (which requires about 60 days). The mass benefit attributable to this maneuver is about 170 kg, and appears as an increase in the BOL mass (assuming the benefit has been implemented as increased payload). In this high-thrust example, the operating time is less than 1 hour per day, and no increase in the velocity increment is required. The power augmentation required is considerable, however, and for the solar-cell-electric-propulsion (SCPEP) example, the power added for EP exceeds the power provided for the payload. By utilizing the battery system, however, the power augmentation becomes more tractable. Providing a higher thrust level reduces the total operating time required for the thrust system summarized in Table 9, and permits qualification of flight hardware in a reasonable time period (e.g. less than one year).

Table 8. Summary of Mission I (Spin-Stabilized Satellite)

| Parameter | Baseline | SCPEP | BPEP |
|---|---|---|---|
| Spacecraft Mass (BOL), kg | 2,460 | 2,460 | 2,460 |
| Propulsion Requirements, m/sec | | | |
| GEO Acquisition | 4,300 (c) | 4,300 (c) | 4,300 (c) |
| N-S Stationkeeping | 464 (c) | 484 (e) | 484 (e) |
| E-W Stationkeeping | 19 (c) | 19 (c) | 19 (c) |
| Disturbance Nulling | ~0 | ~0 | ~0 |
| Power Required for EP, kW | 0 | 0.95 | 0.95 |
| Solar Power added for EP, kW | 0 | 0.95 | 0.42 |
| Propulsion Subsystem- for NSSK | | | |
| Total Thrusters | 8 Chemical | 2 Electric | 2 Electric |
| Operating Thrusters | 1 or 2 (c) | 1 (e) | 1 (e) |
| Unit Thrust, N | 445;22 | 0.035 | 0.035 |
| BPEP Cycles | 0 | 0 | 0 |
| Total Operating Time, hrs | <10 (each) | $10^4$ | $10^4$ |
| Propellant Mass, kg | 430 | 42 | 42 |
| Net Spacecraft Mass Benefit, kg | - | 260 | 300 |
| Economic Benefit (at $50,000/kg) | - | $13M | $15M |
| User Benefit ($10^7/40 kg XPDR) | - | $60M | $70M |

(c) Chemcial Propulsion
(e) Electric Propulsion

Table 9. Summary of Mission II (3-Axis-Stabilized Satellite)

| Parameter | Baseline | SCPEP | BPEP |
|---|---|---|---|
| Spacecraft Mass (BOL), kg | 2,460 | 2460 2630 | 2460 2630 |
| Propulsion Requirements, m/sec- | | | |
| GEO Acquisition (c) | 4,301 | 4300 4070 | 4300 4070 |
| GEO Acquisition (e) | 0 | 0 230 | 0 230 |
| N-S Stationkeeping | 464 (c) | 464 (e) | 464 (e) |
| E-W Stationkeeping (c) | 19 | 19 | 19 |
| Disturbance Nulling | ~0 | ~0 | ~0 |
| Power Required for EP, kW | 0 | 2.9 | 2.9 |
| Solar Power Added for EP, kW | 0 | 2.9 | .9 |
| Propulsion Subsystem- (for Maneuvers Compared) | | | |
| Total Thrusters | 14 Chemical | 2 Electric | 2 Electric |
| Operating Thrusters | 1 or 2 | 1 | 1 |
| Unit Thrust, N | 445;22 | 0.130 | 0.130 |
| BPEP Cycles | 0 | 0 | 2,650 |
| Total Operating Time, hrs | <10 (EACH) | 3000 5000 | 3000 5000 |
| Propellant Mass | 430 630 | 42 66 | 42 66 |
| Net Spacecraft Mass Benefit, kg | - | 220 390 | 260 430 |
| Economic Benefit (at $50,000/kg) | - | $11M $19.5M | $13M $21.5M |
| User Benefit ($10^7/40 kg XPDR) | - | $50M $90M | $60M $100M |

(c) Chemical Propulsion
(e) Electric Propulsion

Comparing the net mass benefits for the two satellites, exclusive of the case involving ion propulsion for final orbit acquisition, both benefits are of the same magnitude, but the lower thrust level of the spin-stabilized satellite shows a somewhat larger mass benefit. Using the higher thrust system for orbit circularization increases the mass benefit by 50% more without increasing the operating time significantly.

Assessing a monetary value for this mass benefit is highly subjective. We have set two values on the mass benefit. If one chose to pursue an engineering approach to reduce satellite structural mass, or conversely, to pay increased launch costs for inserting an added kilogram of mass into geosynchronous orbit, a value of $50,000/kg could be considered as a logical "wholesale" value of the mass benefit. If one converts the mass benefit into communications transponders at 40 kg per transponder (including ancillary equipment), the increased revenue per transponder is estimated to be $10M for a ten-year satellite life. One might consider this to be the "retail" value of the mass benefit, or the user benefit. At this point, we have not deducted the implementation costs, which also require discussion.

To determine the net economic benefit of the SCPEP and BPEP spacecraft listed in Tables 8 and 9, we must formulate a hardware pricing schedule. This assessment was based on using or adapting flight-ready ion propulsion technologies and is equally subjective. We will postulate the thrust-subsystem hardware costs be $3M, solar panel costs at $2M/kW, and qualification and integration costs at $10M per year (5000-hour beam operation requires 1 year; 10,000-hour operation requires a 2-year qualification period, spread over the cost of 10 satellites). Table 10 lists the implementation costs and the net benefit for each of the SCPEP and BPEP examples shown in Tables 8 and 9. While the "gross" economic benefits shows relatively insignificant variations between the point designs,

Table 10. Hardware Implementation Costs and Net Economic Benefits for Point Designs Shown in Tables 8 and 9

| Thruster Size, mN | 0.035 | | 0.130 | | | |
|---|---|---|---|---|---|---|
| | Mission I | | Mission II | | | |
| | NSSK M$ | | NSSK Only | | NSSK Plus OA | |
| | SCEP | BPEP | SCPEP | BPEP | SCEP | BPEP |
| Gross Economic Benefit | | | | | | |
| Wholesale | 13 | 15 | 11 | 13 | 19.5 | 21.5 |
| Retail | 60 | 70 | 50 | 60 | 90 | 100 |
| Implementation Costs | | | | | | |
| Thrust System | 3 | 3 | 3 | 3 | 3 | 3 |
| Solar Panel | 1.9 | 0.8 | 5.8 | 1.8 | 5.8 | 1.8 |
| Non-Recurring (per Satellite) | 2.0 | 2.0 | 1.0 | 1.0 | 1.0 | 1.0 |
| Total | 6.9 | 5.8 | 9.8 | 5.8 | 9.8 | 5.8 |
| Net Economic Benefit | | | | | | |
| Wholesale | 6.1 | 9.2 | 1.2 | 7.2 | 9.7 | 15.7 |
| Retail | 53.1 | 64.2 | 40.2 | 54.2 | 81.2 | 94.2 |

the variations in the net economic benefit is appreciable, especially for the higher level of thrust that requires appreciable power augmentation. The examples shown illustrate the sensitivity of the net economic benefit to the selection of thrust level, specific impulse, and thrust system performance (power to thrust ratio).

## 2. Mission III - Large Radar Satellite

The large satellite considered for Mission III was designed to make use of a full shuttle load with a spacecraft using an electric-propulsion orbit transfer vehicle (OTV). This objective resulted in the selection of a spacecraft mass of 14,880 kg at end-of-life and a relatively large spacecraft that would be categorized as a large space structure (LSS). Table 11 summarizes the characteristics of the chemical baseline spacecraft and the solar-cell and battery-powered electric propulsion spacecraft concepts. The spacecraft mass that must be delivered to LEO with the chemical baseline concept is about twice that of the spacecraft that use electric propulsion. Just in terms of launch costs, this would provide a user benefit of about $70M.

For this spacecraft, estimation of a net user cost-benefit would be pure speculation. Both the chemical and the electric propulsion OTV require design and development. Unless the chemical propulsion OTV operates at relatively low thrust, the spacecraft as conceived here would have to be deployed at GEO. Similarly, the use of a single shuttle launch is based on mass only and LEO deployment may require some form of on-orbit construction technology. For the baseline spacecraft, the propellant would constitute the second shuttle load and on-orbit-fuel-loading technology would be required. The relative difficulty and cost of these unknown technologies is considered

Table 11. Summary of Mission III

| Parameter | Baseline | SCPEP | BPEP |
|---|---|---|---|
| Spacecraft Mass (EOL), kg | 14880 | 14880 | 14880 |
| Propulsion Requirements, m/sec– | | | |
| GEO Acquisition | 4,301 (c) | 5,881 (e) | 5,881 (e) |
| N-S Stationkeeping | 464 (c) | 464 (e) | 464 (e) |
| E-W Stationkeeping | 19 (c) | 19 (e) | 19 (e) |
| Disturbance Nulling | 123 (c) | 123 (e) | 123 (e) |
| Power Required for EP, kW | 0 | 17 | 17 |
| Solar Power Added for EP, kW | 0 | 17 | 0 |
| Solar Power Available for OT, kW | 125 | 125 | 125 |
| Propulsion Subsystem– | | | |
| Total Thrusters on OTV | 2 Cryogenic | 20 Electric | 20 Electric |
| Total Thrusters on Spacecraft | 26 Chemical | 4 Electric | 4 Electric |
| Operating Thrusters on OTV | 2 | 15 | 15 |
| Operating Thrusters on Spacecraft | 1 Or 2 (c) | 2 (e) | 2 (e) |
| Unit Thrust, N | 445;22 | 0.392 | 0.392 |
| BPEP Cycles | 0 | 0 | 2,650 |
| Total Operating Time, hrs | <10 (each) | <5000 | <5000 |
| Propellant Mass, kg | 31,600 | 5,400 | 5,400 |
| Spacecraft Mass Delivered to GEO | 18,500 | 16,690 | 16,540 |
| OTV MASS, kg | 31,100 | 7,400 | 7,370 |
| Spacecraft Mass Delivered to LEO, kg | 49,600 | 24,090 | 23,910 |
| Net Mass Benefit, kg | 0 | 25,510 | 25,690 |
| User Benefit (at $70M/Launch) | 0 | $70M | $70M |

(c) Chemical Propulsion
(e) Electric Propulsion

to be equivalent for electric propulsion and chemical propulsion OTVs, to first order. Consequently, the net user benefit is the cost benefit and convenience associated with a single shuttle launch.

### 3. Mission IV - Military Class Satellites

In Table 12, the spacecraft propulsion requirements for a mission scenario exemplifying military satellites are summarized and the electric propulsion spacecraft characteristics are compared with the baseline spacecraft. Although a modest mass benefit can be realized, using either a SCPEP or BPEP approach, we were unable to assign a user-benefit dollar value. Military satellites are usually not mass limited and one of this size would probably have a dedicated shuttle launch (though not necessarily). If the objective for keeping the satellite in a storage orbit is to have it ready for deployment with minimal time delay, the apogee raising that is proposed as an electric propulsion maneuver would not be acceptable, and the mass benefit would be smaller. In our estimation, motivation for using electric propulsion in this application would depend on the logistics of the specific mission (possibility for launch sharing, details of mission operations, etc), and is not clearly definable.

### 5. Impact of Battery Technology Improvement

The results cited above are achievable with state-of-the-art nickel-hydrogen batteries at 0.7 depth of discharge for 5000 cycles (specific mass of available energy storage is 63.3 kg/kW-hr). Improvement in battery technology by either increasing the permissable depth of discharge or by otherwise

Table 12. Summary of Mission IV

| Parameter | Baseline | SCPEP | BPEP |
|---|---|---|---|
| Spacecraft Mass (BOL), kg | 17,200 | 17,200 | 17,200 |
| Propulsion Requirements, m/sec- | | | |
| Momentum Wheel Spin-up (c) | 4.1 | 4.1 | 4.1 |
| LEO Apogee Raising | 145 (c) | 161 (e) | 161 (e) |
| EO Injection (c) | 2,141 | 2,141 | 2,141 |
| Inclination Change to $63.4^0$ (c) | 488 | 488 | 488 |
| Orbit Ttim | 6.1 (c) | 6.9 (e) | 6.9 (e) |
| On-Orbit Maneuvers | 160 (c) | 178 (e) | 178 (e) |
| Power Required for EP, kW | 0 | 17 | 17 |
| Solar Power Added for EP, kW | 0 | 17 | 8.5 |
| Propulsion Subsystem- | | | |
| Total Thrusters | 8 Chemical | 2 Electric | 2 Electric |
| Operating Thrusters | 1 or 2 (c) | 1 (e) | 1 (e) |
| Unit Thrust, N | 445;22 | 0.392 | 0.392 |
| BPEP Cycles | 0 | 0 | 4,908 |
| Total Operating Time, hrs | <10; <10 | <1,900 | <1,900 |
| Net Spacecraft Mass Benefit, kg | 0 | 137 | 208 |
| Economic Benefit (at $50,000/kg) | 0 | $6.9M | $10.4M |
| User Benefit | Not Able to Assess | | |

(c) Chemical Propulsion
(e) Electric Propulsion

decreasing battery specific mass reduces the spacecraft mass for all propulsion options equally (assuming the electric power subsystem is designed to satisfy the requirements of the payload). Consequently, the magnitude of spacecraft-mass reduction is directly proportional to the energy storage required for the baseline spacecraft. Projecting improvements in battery specific mass from 63.3 kg/kW-hr to less than 20 kg/kW-hr decreases the overall spacecraft mass of the Mission I and Mission II communications satellites by about 100kg. The maximum net mass benefit is achieved for both of these systems by providing the additional power requirement with solar panels, even with advanced battery technology. In the case of the Mission I spin-stabilized satellite, the additional solar panel required presents a formidable integration task that may be reduced somewhat by increasing battery capacity. We did not explore this option in any detail. When flat solar panels are used, we expect the addition of solar panel area to be preferred to addition of batteries.

For the large radar satellite of Mission III, the net mass reduction available from improved battery technology is comparable to the on-orbit benefit obtained by using ion propulsion for NSSK. If we assume only the advancement projected for nickel hydrogen batteries (40 kg/kW-hr), the net mass reduction of the spacecraft is about 3800 kg. The mass of the military satellite of Mission IV can be reduced from 50 to 500 kg, depending on the stored energy assumed, by projected advancement in battery technology.

Based on our assessment that nickel-hydrogen battery technology can provide 5000 charge-discharge cycles at 0.7 depth of discharge, the benefits of ion propulsion for reducing spacecraft mass are independent of the benefits projected for improvement in battery technology, and vice-versa.

## CONCLUSIONS

On-board energy storage in the form of secondary batteries was found to provide appreciable benefits when used for North-South stationkeeping of geostationary satellites. The magnitude of the benefit is a complex function of the electric-propulsion subsystem design and operating characteristics. The benefit is realized as a 10- to 15-percent decrease in the BOL mass of the satellite (for fixed payload mass). This mass benefit is independent of the status of battery technology (assuming the availability of batteries with a capability for 5000 charge/discharge cycles at 0.7 depth-of-discharge). Further mass benefits would be realized through improved battery technology, by either increasing the allowable depth-of-discharge or by otherwise reducing specific mass. The benefits of improved battery technology would accrue independent of the use of electric propulsion. The economic value of this mass benefit depends upon how it is used. For commercial communication satellites, for instance, a satellite could produce an additional $50M in revenue over a ten-year period if the mass benefit is used to provide additional transponders. Operation of the stationkeeping ion thrusters at relatively low thrust for long time periods produces the highest mass benefits when power is obtained primarily from the solar panels, but places more stringent qualification requirements on thrust subsystem hardware. Judicious use of battery power alters this conclusion radically, and if the stored energy available is adequate to supply the entire daily stationkeeping requirements, the mass benefit becomes less dependent on the thrust subsystem operating conditions. In this situation, the thrust level can be chosen at a value consistent with a mission operating time of less than 5,000 hours (a value consistent with a one-year qualification period). The NASA/Hughes 30-cm mercury ion thruster technology can provide benefits that are near-optimal.

For geostationary satellites, the largest mass benefit is derived by using electric propulsion for North-South stationkeeping (NSSK). Since the size of the mass benefit is directly proportional to the satellite mass, the monetary value of the mass benefit must exceed the cost of the ion-propulsion hardware itself (and any power augmentation) if a net benefit is to be realized. Based on the costs projected for ion propulsion hardware (including power electronics) the minimum size satellite that can realize a net economic benefit from using ion propulsion to provide NSSK was estimated to be in the 1000 kg to 1500 kg range. This range could be raised or lowered depending on the cost of electric propulsion hardware, the absolute economic value of mass in orbit, and the mass of the propulsion system hardware.

Orbit-transfer maneuvers can also be performed to advantage (for reducing propellant mass) with ion propulsion but the time required for orbit transfer mitigates against use of the ion propulsion system for this maneuver. Batteries are of little or no utility for orbit transfer maneuvers. Other reaction-control maneuvers require impulsive thrusting in too many different directions to be provided by ion propulsion in a cost-effective manner.

To enhance the economic benefits of ion propulsion for NSSK, the thrust-subsystem technology will benefit from improved efficiency and/or increased thrust-to-power ratio up to the point at which no augmentation of the power system is required for implementation of ion propulsion. Furthermore, the size of the satellite that can realize an economic benefit from ion propulsion will be decreased (satellite mass-threshold lowered) if the cost of the ion propulsion hardware can be reduced.

To obtain maximum benefits, satellite configurations have to be designed to provide an unobstructed North- or South-facing field of view so that stationkeeping can be provided with a minimum of two thrusters to minimize hardware costs (provided such a satellite design does not substantially increase the satellite cost). Significant mass benefits can be nullified if too much hardware (thruster or power) has to be used.

For very large satellites, ion propulsion may be enabling if the increased payload-delivery capability of ion propulsion (as compared to chemical propulsion) is exploited to allow the entire structure to be delivered to orbit from a single shuttle load. The logistics of multiple-shuttle launches for construction of a spacecraft appear to be far more demanding technologically and administratively than space assembly and subsequent launch of a single shuttle load. However, the development of an ion propulsion OTV must solve the problem of long residence in the radiation belts to be viable. Development of the LSS technology and a 125-kW power source will delay initiation of a mission like the radar satellite well into the future (beyond 2000 A.D.) Thrust modules that operate at higher thrust levels are required for satellites with mass on the order of 15,000 to 20,000 kg.

Advanced battery technology will provide benefits for any satellite, regardless of its propulsion system. For battery support of electric propulsion systems used in cyclic operation, demonstration of battery technology that is capable of reliably sustaining a large number of charge/discharge cycles (5000 to 10,000) at 80% depth of discharge would enhance acceptability of using the otherwise under-utilized battery resource for propulsion purposes. Similarly, testing batteries at variable depths of discharge is required to verify that variable discharge does not constitute a more severe use of the batteries than the conventional charge-discharge cycling used for proving battery technology.

# APPENDIX A

The ion propulsion system was modeled for this study in accordance with the equations listed in Table A-1. On the basis of mission analyses, the trust, $F$, the specific impulse, $I_{sp}$, and the total thrusting time, $\Delta t$, are determined (for providing a certain velocity increment, $\Delta v$). Using these specified values, the modeling equations determine the mass (including propellant) of the electric propulsion subsystem, $m_{EP}$, and the power input, $P_{in}$, required (including inefficiency in the power supplies).

Table A-1. Mercury Ion Thruster Subsystem Technology Models

| Parameter | 8-CM | 30-CM |
|---|---|---|
| Thrust, mN | F | F |
| Specific Impulse, sec | I | I |
| Total Mission Operating Time, Hrs | $\Delta t$ | $\Delta t$ |
| Thruster Beam Voltage, V | $V = (I_{sp}/73)^2$ | $V = (I_{sp}/90)^2$ |
| Thruster Beam Current, A | $J = F/(2\sqrt{V})$ | $J = F/(1.96\sqrt{V})$ |
| Thruster Qual. Life, Hrs | L = 20,000 | L = 20,000 |
| Design Mission Life, Hrs | $L_{DM} = 10,000$ | $L_{DM} < 10,000$ |
| Ratio of Qual.-To Design Mission Life | 2:1 | 2:1 |
| Propellant | Mercury | Mercury |
| Redundancy<br>- Commercial<br>- Military | <br>R = 2<br>(Not Applicable) | <br>R = 2<br>R = 3/2, 4/3, or 2/1 (Mission & Mission Phase Dependent) |
| Propellant Mass, kg | $m_p = 3.6 \frac{F \Delta t}{I_{sp}\, g_o}$ | $m_p = 3.6 \frac{F \Delta t}{I_{sp}\, g_o}$ |
| Propellant Tankage, Mass, kg | $m_T = 0.04\, m_p$ | $m_T = 0.04\, m_p$ |
| Total Number of Systems | $N = RN_{op}$ | $N = RN_{op}$ |
| Electric Propulsion Implementation Mass, kg | $m_{EP} = N\,[8.1 + 42J] + 5.4\, N_{op} + m_p + m_T$ | $m_{EP} = N\,[22 + 15J] + 5.4\, N_{op} + m_p + m_T$ |
| Total Input Power to Thruster Subsystem, $P_{in}$ | $P_{in} = 1.1\,(VJ + 242J + 59)$ | $P_{in} = 1.05\,(VJ + 211J + 123)$ |

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