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# Advanced Propulsion for LEO and GEO Platforms



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# ADVANCED PROPULSION FOR LEO AND GEO PLATFORMS

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## ABSTRACT

Mission requirements and mass savings applicable to specific low Earth orbit and geostationary Earth orbit platforms using three highly developed propulsion systems are described. Advanced hypergolic bipropellant thrusters and hydrazine arcjets can provide about 11% additional instrument payload to 14,000 kg LEO platforms. By using electric propulsion on a 8,000 kg class GEO platform, mass savings in excess of 15% of the beginning-of-life platform mass are obtained. Effects of large, advanced technology solar arrays and antennas on platform propulsion requirements are also discussed.

## NOMENCLATURE

| | |
|---|---|
| $I_{sp}$ | Specific impulse, s |
| $P_e$ | Input power to thruster's power processor |
| T | Thrust, N |
| $\Delta V$ | Spacecraft velocity change, m/s |

## INTRODUCTION

Over the next two decades unmanned, space platforms will be placed in Low Earth Orbit (LEO) and Geostationary Earth Orbit (GEO) to gain a better understanding of Earth science and also provide a test bed for advanced technology systems (refs. 1-10). Mission models include LEO platforms at inclinations from 0 to 90 degrees and GEO platforms. In this paper LEO refers to any orbit with an altitude less than 1000 km. In the near term, free-flying platforms in the 3,000 to 4,000 kg class will be launched and/or retrieved by the U.S. Space Transportation System (STS) and other launch capabilities. These platforms will be deployed at an altitude of about 300 km. One example is the European Retrievable Carrier (EURECA) platform which will provide about six months mission operation for fifteen experiments (ref. 6). The STS will also retrieve the Japanese Space Flyer Unit which will be a test bed for a number of advanced technology experiments (ref. 5). Another program development for LEO platforms is proceeding under the Earth Observing System (Eos) initiative which involves the launch of polar platforms by NASA, ESA, and Japan (refs. 1-4). The 10,000 to 15,000 kg spacecraft for the Eos missions will be launched by an upgraded Titan IV to an elliptical transfer

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orbit and then will use onboard apogee propulsion to reach a 705 km circular orbit. Other LEO missions will be accomplished using payloads attached to the Space Station Freedom (SSF) and platforms co-orbiting with the SSF at about 500 km altitude with an orbit inclination of 28.5 degrees.

Most platforms and communication satellites use either monopropellant hydrazine, hydrazine resistojet, or hypergolic bipropellant thrusters for the on-orbit propulsion operations (refs. 4, 7). The baseline on-board propellant subsystems for the Eos platform are hypergolic bipropellants for apogee propulsion and end-of-mission disposal into the ocean and monopropellant hydrazine systems for on-orbit operations (ref. 4). Figure 1 shows the on-orbit propellant mass fractions for the baseline Eos platform. More than 1800 kg, or 13% of the beginning-of-life (BOL) platform mass, is propellant (ref 4). In addition, the amount of on-board propellant to be used for orbit acquisition is 935 kg or about 6% of the platform transfer orbit mass. Figure 1 also shows that the propulsion mass fractions for a GEO platform and INTELSAT V are significant. Both systems employ resistojets ($I_{sp} \sim 300$s) for North/South (N/S) stationkeeping and monopropellant hydrazine ($I_{sp} \sim 200$ s) for all the other functions (refs.10-12). Figure 1 indicates that 23% or 1870 kg of the baseline GEO platform is propellant (ref. 8). For comparison purposes, the smaller INTELSAT V has a BOL propellant mass fraction of 18% (refs. 7, 10). The INTELSAT V propellant mass fraction is smaller because the design life is 7 yr versus 10 yr for the GEO platform.

Previous work has generally shown the influence of on-board propulsion on overall mission performance of LEO and GEO satellites and characterized emerging high performance propulsion technologies (refs. 11-13). General applicability of on-orbit electric propulsion to SSF co-orbiting platforms and polar platforms has also been analyzed (ref. 14). Electric propulsion options for over thirty LEO free flying spacecraft have been investigated (ref. 13). The $\Delta V$'s and propellant requirements for LEO transfer, reboost to overcome atmospheric drag, and inclination changes were determined with a sensitivity to life-cycle propellant mass savings. GEO platform servicing and payload delivery missions have been analyzed using chemical and 0.1 to 1 MW electric orbit transfer vehicles (ref. 15).

This paper describes three advanced propulsion system technologies which were applied to large LEO and GEO platforms and were compared to baseline systems which employed monopropellant hydrazine for all on-orbit propulsion and conventional nitrogen tetroxide/monomethyl hydrazine (NTO/MMH) for LEO acquisition and deboost. In all cases, the advanced propulsion system dry mass estimates and thruster life requirements were very conservative and generally were based on state-of-the-art component and subsystem characterizations. The advanced propulsion technologies comprised hypergolic bipropellant devices with high temperature thrust chambers for orbit acquisition and end-of-life (EOL) disposal and hydrazine arcjets or xenon ion thrusters for on-orbit propulsion. Monopropellant hydrazine thrusters provided the remaining orbit maintenance and control functions. The low power arcjets and ion thrusters were not considered for Eos platform orbit acquisition since atmospheric drag exceeds the attainable thrust levels at the perigee altitude (185 km) of the transfer orbit. Platform servicing and refueling were not considered in this analysis since these complex operations have generally not been considered for near-term polar or GEO platforms. High power orbit transfer to half-GEO and GEO using electric propulsion is beyond the scope of this paper and has been reported elsewhere (refs. 16-18).

## ADVANCED PROPULSION SYSTEMS

Advanced hypergolic bipropellant (NTO/MMH) thrusters, low-power arcjets, and ion thrusters are being developed in focused technology programs with a view towards near term applications on communication satellites, as well as planetary and LEO spacecraft (refs. 19-23). The chemical and electric propulsion systems can also accommodate platform requirements for boost/deboost and on-orbit propulsion, respectively. The characteristics and technology status of the advanced propulsion systems are described.

### Bipropellant Thrusters

Conventional NTO/MMH bipropellant thrusters are fabricated from columbium, which is coated with silicides for oxidation protection. These thrusters are used on the Space Shuttle Orbiter and a wide variety of GEO communication satellites (ref. 11). In many spacecraft, both apogee injection and on-orbit propulsion are performed by the NTO/MMH thrusters. Thrust levels range from 22 N to hundreds of Newtons at a specific impulse of about 280 to 310 s. Advanced development bipropellant thrusters have thrust chambers fabricated from rhenium coated with iridium by a chemical vapor deposition process (refs. 19,24,25). The basic fabrication sequence is illustrated in Fig. 2. This fabrication method allows operating temperatures to be increased by about 800 K, thus eliminating the need for film cooling, which was used to prevent failure of silicide coatings. The resulting improvement in propellant mixing provides an increase in specific impulse of 20 to 30 s. Because the rhenium/iridium system offers an operational capability approaching 2300°C, trades involving film cooling, performance, and lifetime can be made including operation at lower temperatures to produce greater than a tenfold increase in operating life (ref. 19). Technology demonstrations have been performed using 22 N and 440 N thrusters. The 22 N thrust chamber is capable of operating at temperatures in excess of 2200°C with storable propellants for durations in excess of 15 hours (ref. 25).

### Hydrazine Arcjet

A one-kilowatt class hydrazine arcjet system including the arcjet, catalyst bed, power processor and interconnecting power cable has been developed to an engineering model level, and vibration, thermal/vacuum, and electromagnetic compatibility tests have been performed (ref. 26). The arcjet, valve, and gas generator are illustrated in Fig. 3. Mission average specific impulse levels of 450 to over 500 s have been demonstrated at thrust levels of about 0.2N. The low power arcjet offers significant specific impulse improvement compared to 180 to 220 s for the conventional monopropellant hydrazine thrusters, 280 to 300 s for hydrazine resistojets, and about 290 s for a 22 N NTO/MMH bipropellant thruster (ref. 11). Extended tests of laboratory class arcjets have been demonstrated over 1000 hr and 500 cycles (ref. 27), which would satisfy the requirements for about 15 years of on-orbit lifetime for 2000 kg class GEO spacecraft. Post-test component evaluation revealed limited erosion on both the cathode and anode. Thruster and power processor vibration tests and thermal vacuum tests have been successfully undertaken (ref. 26). System level electromagnetic compatibility tests and plume characterization are in progress to assure there will be no impact on spacecraft operations (ref. 28). The General Electric Company has recently baselined the low power arcjet for stationkeeping of American Telephone and Telegraph Company's Telstar IV spacecraft (refs. 21, 29).

Xenon Ion Thruster

Xenon ion thrusters have reached a high level of maturity and have demonstrated specific impulse levels in the 2500 to 5000 s range for input powers from 1 to 5 kW (refs. 21-23, 30-32). The basic ion thruster assembly is shown in Figure 4. Further demonstration of ion thruster performance is not a critical issue, since overall efficiencies of 65 to 80 percent have been readily obtained. Extended tests from 500 to 4000 hours have given confidence that there are no life limiters that would preclude operation of 1 to 5 kW xenon thrusters for periods up to 5000 hours (refs. 23, 30, 33, 34). In addition to 1.4 kW thruster performance and life documentation, a breadboard-model power processor and a flight-prototype pressure regulator have been exercised in subsystem tests (ref. 30). A radio frequency ion propulsion system has been qualified for an experiment on the European Retrievable Carrier (EURECA) which will be launched in 1991 or 1992 by the Space Shuttle (ref. 32). In 1992, the Japanese will use xenon ion propulsion as the prime N/S stationkeeping system for the Engineering Test Satellite-VI (ref. 23).

## EOS CLASS PLATFORM WITH ADVANCED PROPULSION

The Earth Observing System (Eos) is a long term, international program to study land, ocean and atmospheric processes (ref. 2). The Eos program is envisioned to comprise at least four platforms; two provided by NASA, one by the European Space Agency, and one by Japan. The propulsion system trades will use Eos-A as the baseline platform system (Table I). Eos-A is a polar orbiting platform and will be launched from the Western Test Range into a 185 X 705 km elliptical orbit by a Titan IV. The baseline platform mass in the transfer orbit and the final 705 km circular orbit will be about 15,000 kg and 14,000 kg, respectively (ref. 4). Typical instrument payload mass will be about 3500 kg, and the total power available to the platform is 6 kW. The observatory components were designed for 5 yr life, but propellant requirements are set for 7.5 yr. No servicing or resupply is planned for Eos-A. At the end-of-life the platform will be safely propelled into the ocean.

Platform propulsion requirements, shown in Table II, are for orbit acquisition, 7.5 year orbit maintenance, backup attitude and momentum control, and safe ocean disposal (ref. 4). The baseline platform was to be transferred into the 705 km sun-synchronous orbit by a set of three-445 N hypergolic bipropellant apogee thrusters ($I_{sp}$=298 s); another three-thruster set provides redundancy for orbit acquisition. The mission velocity increment ( $\Delta$V) for orbit acquisition from the 185x705 km transfer orbit was 164 m/s. The gravity loss $\Delta$V penalty associated with finite propulsion times and thrust-to-weight ratios of about 0.01 was assessed. Gravity losses for Eos-A orbit acquisition, for example, have been determined to be less than one percent of the impulsive velocity increment. The analysis was performed using the Program to Optimize Simulated Trajectories (POST) code described in reference 35.

The in-plane orbit maintenance included drag makeup and eccentricity control whose $\Delta$V's were 32 m/s and 7 m/s, respectively and the inclination makeup $\Delta$V was 36 m/s (ref. 4). The baseline Eos-A platform used monopropellant hydrazine thrusters ($I_{sp}$=200 s) for all orbit maintenance functions. Monopropellant hydrazine thrusters ($I_{sp}$=180 s) were assigned 50 kg of propellant to cover the backup attitude control requirement (ref. 4). The bipropellant thruster system also provided a deboost trajectory for positive re-entry of the platform with a $\Delta V$ of 232 m/s (ref. 4). The overall propellant requirements for the baseline system, including 3% propellant residuals and 10% margin, were 2105 kg of bipropellants and 675 kg of hydrazine for 7.5 years of operation (Table III).

Next, the propellant budget for the advanced propulsion system was determined using advanced bipropellant thrusters ($I_{sp}$ = 323 s), 700 W hydrazine arcjets for orbit maintenance, and conventional monopropellant hydrazine thrusters for backup attitude control. The advanced bipropellant thrusters primarily involve an improved thrust chamber, so there should be no significant dry mass penalty for replacement of the conventional thrusters. A set of eight 700 W arcjets were dedicated to in-plane orbit maintenance and inclination control. The eight arcjets would perform the same orbit maintenance as the eight 2.2 N monopropellant hydrazine thrusters baselined for Eos-A. Each arcjet burn would involve the operation of two thrusters, configured to minimize unwanted disturbance torques, with a total power of 1400 W to the two power processors. Seven hundred watt arcjets were chosen to minimize power requirements and still perform routine orbit maintenance in a relatively short time period. In the first advanced propulsion case described in Table III, it was assumed arcjet power would be provided by the Eos-A baseline power system. An average arcjet system burn time of about 35 minutes per day would be required. Arcjet burns might be scheduled during housekeeping periods when some of the major power users are not operational. The average energy demand by the arcjet system was 817 W-hr/day. Burn times and energy requirements were based on the arcjet performance defined in Table II (ref. 36).

The arcjet system elements, which included eight thrusters and eight power processing units (PPU), are described in Table IV. Component masses were obtained from references 26, 37, and 38. A power processor efficiency of 0.90 was assumed (ref. 26). Dissipated power from the power processors was assumed to be handled by the thermal control system with a specific mass of 40 kg/kW (ref. 37). The Interface Module contained a housekeeping converter, controller, wire harness and filters (refs. 38, 39). The total mass of the arcjet system was 96 kg which included a margin of 30%.

The propellant budget for the advanced propulsion system using Eos-A power system (Table III) is 459 kg less than the Eos baseline. No dry mass benefit was included when the eight baseline monopropellant hydrazine thrusters were removed and replaced by the arcjet system. After adding the arcjet system (96 kg) and reducing the tankage needed for hydrazine by 35 kg, the overall mass savings is 398 kg (Fig. 5). A 0.064 tankage fraction (the ratio of the mass of propellant and pressurant tanks to the mass of propellant) was assumed based on the Geostationary Operational Environmental Satellite (GOES I) experience (ref. 40). The high performance bipropellant system reduced the baseline propellant budget by 127 kg, while the arcjet system provided a net savings of 271 kg. Only 239 kg of hydrazine was required to perform 7.5 years of orbit maintenance using the arcjet system. If it is assumed that four thrusters consume most of the propellant, on average about 60 kg of hydrazine would be used by each of the four arcjets. This is not a very demanding requirement. For example, the comparable mass throughput of hydrazine required by near term communication satellites, which have two active arcjet systems, is 96 kg per thruster (ref. 26). The overall mass savings using advanced propulsion and the Eos-A baseline power system is about 14% of the baseline propellant budget and is equivalent to an additional payload contingency of 11%.

In another design scenario, the solar array could be uprated to provide additional power to enable two arcjets to be fired simultaneously and independent of the platform core and payload power demands. In this case, the two arcjet systems would require an additional 1.6 kW including added wire harness losses and 10% margin. The solar array specific mass was assumed to be about 29 kg/kW plus an additional 30% mass margin (ref. 8). About 61 kg of solar array would have to be integrated with the platform, and the overall mass reduction would now be about 337 kg.

## GEO PLATFORMS

This stationkeeping propulsion benefits analysis will examine a single GEO platform generally described in Table I and Figure 6 (refs. 8, 41). The platform would be transferred to GEO by using either perigee/apogee propulsion from a GEO transfer orbit or by direct insertion employing an orbit transfer vehicle. In either case an advanced launch vehicle would be required or platform assembly would be performed in LEO after multiple launches. For this study, the beginning-of-life mass in GEO was 7,000 to 9,000 kg depending on the type of propulsion, and the payload was fixed at 3840 kg. The platform power system was designed to provide 3.2 kW to the power bus at the end of the 10 year design life. No on-orbit servicing was considered.

Relatively large passive microwave radiometers (PMR) are mounted on the main truss of this platform (Fig. 6). Bus subsystems and Earth pointing instruments are housed in two 3-meter modules located on the main truss. Power is supplied by two solar array panels, each of which is about 15 $m^2$ (ref. 5). The design is based on the Advanced Photovoltaic Solar Array (APSA), which employs an accordion folded blanket with a specific mass of 14.3 kg/kW (refs. 8, 42). Nickel-Hydrogen batteries were selected for this study. Sizing routines yield 81 kg battery mass for a 3.2 kW power supply and a 1.2 hour eclipse period (ref. 8). Table V identifies some of the basic analysis assumptions related to the platform propulsion, power and thermal subsystems.

The GEO platform baseline propulsion system has monopropellant hydrazine thrusters performing all on-orbit propulsion operations. The propulsion analysis will show the benefits of using either hydrazine resistojets, hydrazine arcjets, or xenon ion thrusters to perform N/S stationkeeping while $\Delta V$'s associated with E/W stationkeeping, momentum control, some orbit acquisition maneuvers, and EOL boost would be undertaken using either 2.2 or 22 N monopropellant hydrazine thrusters in all situations. Propulsion requirements are shown in Table VI. The platform consisted of about 30 $m^2$ of solar array and two large passive microwave radiometers with diameters of about 7.5 and 15 m, resulting in an effective platform area-to-mass of approximately 0.06 $m^2/kg$ (ref. 8). This area-to-mass is sufficiently small so that just as with typical GEO satellites, the N/S stationkeeping $\Delta V$ requirement dominates E/W stationkeeping.

N/S stationkeeping thruster performance is shown in Table VII. The nominal specific impulse assumed for the monopropellant hydrazine, resistojet, arcjet, and xenon ion thrusters was 200, 290, 450 and 2800 s, respectively. Power to each resistojet subsystem was 400 W (refs. 8, 43), while 1400 W was supplied to each arcjet and ion thruster subsystem. Input power and performance levels are representative of flight-type or engineering model thrusters (refs. 11, 12, 26, 30). Trades could also be made using higher power ($\sim$ 5kW) hydrazine arcjets and xenon ion thrusters since these devices are currently being developed under NASA's focused technology programs but, at present they have not reached the maturity of the 1.4 kW systems (refs. 19,30). The number of 1.4 kW arcjets and ion thrusters to perform stationkeeping is dictated by guidelines associated with maximum burn time per day, as well as thruster lifetime and redundancy. The stationkeeping maneuvers were assumed to be restricted to housekeeping periods of about 90 minutes when some payload instruments would not be operational. This operation mode will result in about 1.6 kW of payload power available for propulsion (ref. 8). Reducing stationkeeping operations to less than 90 minutes per day also minimizes interaction times of propulsion with platform payloads or experiments. The prime lifetime limiters for the 1.4 kW arcjet and ion thruster are anticipated to be the hydrazine gas generator and ion optics charge exchange erosion, respectively. The lifetime target for the arcjet was selected to be 1000 to 1500 hr while the ion thruster target was 10,000 hr based on 1990 NASA technology goals. In order to satisfy the maximum burn time guideline and also reduce thruster lifetime requirements two sets of two arcjets (or two sets of four ion thrusters) were selected to share N/S stationkeeping. Table VIII summarizes the power requirements for the baseline platform and the advanced technology versions using arcjets or ion thrusters (ref. 8). An additional 1.0 kW and 3.8 kW of power above the baseline system was required for the arcjet and ion thruster systems, respectively. The power system mass for the arcjet system and the ion thruster system was in excess of the baseline system by 30 kg and 122 kg, respectively.

Next, the on-orbit propulsion system dry mass was estimated. References are cited in Table IX for the mass estimate of each element of the propulsion systems. The ion thruster mass is 12.7 kg per thruster, and was based on a laboratory model device (ref. 44). To date, no attempt has been made to reduce the mass of the 1.4 kW xenon ion thruster by using high strength, low mass materials. Gimbal mass was taken to be 30% of the thruster mass (ref. 44). Ion thruster gimbaling was assumed to insure there were no roll moments produced by uncertainty or movement of the platform center-of-mass. Thrust modulation can maintain spacecraft attitude when yaw torques are produced (refs. 45). Except for the ion propulsion system, each of the other N/S stationkeeping options has two thrusters on both the North and South faces of the platform, with two redundant thrusters on each face. The ion system has four thrusters on each of the North and South faces and two redundant thrusters on each face. Electric propulsion hardware redundancy and power utilization on communication satellites is discussed in reference 22. Thermal control mass for the propulsion systems was taken to be 40 kg per kilowatt of power dissipated from the power processor (ref. 37). Tankage fractions of 0.064 and 0.15 were assumed for hydrazine and xenon, respectively (refs. 30, 40). Tank structure was taken to be 4% of the sum of the propellant and tankage masses (ref. 39). Including tankage, the arcjet system had the lowest dry mass of 296 kg, followed by resistojet, monopropellant hydrazine, and ion systems with dry masses of 348, 376, and 610 kg, respectively. Further reductions in all system dry masses might be possible by mass optimization during a flight development program.

Table X and Fig. 7 show the BOL GEO platform mass summary. The arcjet and ion system propellant masses are 51% and 25% of the hydrazine required by the baseline system which employed monopropellant hydrazine thrusters for N/S stationkeeping. After factoring in the propulsion dry masses and power system masses, the GEO platforms with the monopropellant hydrazine, resistojet, arcjet, and ion thruster options had total masses of 8734, 7973, 7430, and 7162 kg, respectively. The resulting mass benefit using the arcjet system was about 1304 kg and the ion system produced a mass benefit of 1572 kg. The electric propulsion systems for platform N/S stationkeeping can provide mass savings equivalent to 15% to 18% of the baseline BOL platform mass.

The N/S stationkeeping propellant requirements for the ten year GEO platform imply each of the four active arcjets must handle a hydrazine throughput of about 176 kg, which implies a total operating time of about 1080 hours. Present generation arcjets are required to process about 96 kg of hydrazine over 607 hours for communication spacecraft applications (ref. 20). Thus, the 1.4 kW arcjets and the hydrazine gas generators used for platform stationkeeping will have a longer life requirement than needed on smaller GEO spacecraft, but 1000 hr thruster life has been readily demonstrated in ground tests (ref. 27). On average each of the eight ion thrusters would be required to process about 14 kg of xenon over the ten year mission life. Ion thruster operation time would only be about 1750 hr. Xenon ion thrusters, with 0.023 N thrust levels, will perform N/S stationkeeping on the Japanese Engineering Test Satellite (ETS VI), starting in 1992 (ref. 46). The ETS-VI ion thrusters deliver about one-third the thrust of the 25 cm diameter thruster specified for the platform stationkeeping. An ETS-VI thruster will process about 20 kg of xenon over its 6500 hour lifetime. The GEO platform ion thruster lifetime requirements are less demanding than those of ETS-VI since eight higher power ion thrusters are involved in the stationkeeping process.

Arcjets and ion thrusters are presently being developed for near term applications on 2000 kg class GEO spacecraft. By using electric propulsion on 8000 kg GEO platforms, mass savings equivalent to 15% to 18% of baseline platform mass can be obtained with modest lifetime requirements of 1080 hr and 1750 hr for hydrazine arcjets and xenon ion thrusters, respectively. Major issues concerning implementation of electric propulsion, such as system integration, power utilization, flight qualification, and particle and field interactions are summarized in references 22, 26.

## ADVANCED GEO PLATFORM REQUIREMENTS

Advanced platforms will employ larger antennas and solar arrays, and thus the area-to-mass of platforms may increase dramatically. The large systems will increase the East/West drift in GEO orbit. The drift is caused by solar radiation pressure and disturbances due to the Earth's triaxiality (ref. 8). Figure 8 shows the yearly E/W stationkeeping $\Delta V$ versus area-to-mass. N/S stationkeeping is generally unaffected. Daily corrections of the E/W disturbance have generally not been done because autonomous control systems have not been implemented. Such control systems may produce significant benefits as platform area-to-mass approaches about 0.3 $m^2$/kg. Figure 8 also shows the sensitivity of E/W $\Delta V$ to triaxiality disturbances, which are a function of platform longitudinal position.

The baseline GEO platform had an E/W stationkeeping $\Delta V$ requirement of about 5.3 m/s/yr and a mean area-to-mass ratio of 0.06 $m^2/kg$. If, for example, a platform included a 40 m diameter passive microwave radiometer (PMR), the area-to-mass ratio would increase by a factor of 4.6. The resulting E/W stationkeeping $\Delta V$, 38 m/s/yr, is nearly comparable to the N/S stationkeeping requirement.

Figure 9 shows how platform propellant mass varies as a function of BOL platform mass with area-to-mass as a parameter. Either resistojets ($I_{sp}$ = 300 s) or ion thrusters ($I_{sp}$ = 2500 s) perform the stationkeeping maneuvers which dominate all propulsion requirements. A 7000 kg platform's propellant budget, using resistojet propulsion, would be increased by approximately 700 kg to 1940 kg if the mean platform area-to-mass was increased from 0.06 to 0.28 $m^2/kg$. By using ion propulsion, the total propellant mass would only be 200 to 300 kg. As platform systems advance to larger solar arrays and antennas, propellant mass fractions in excess of 30% of BOL platform mass may be required using conventional propulsion systems. Ion propulsion systems, for example, could reduce the propellant mass fraction of such platforms to less than 5% of BOL platform mass.

## CONCLUSIONS

Mission requirements and mass savings applicable to specific LEO and GEO platforms are described using three highly developed propulsion systems. Advanced Ir/Re bipropellant thrusters performed the apogee motor function, while hydrazine arcjets or xenon ion thrusters were used for N/S stationkeeping. When advanced bipropellants and arcjets were considered for 14,000 kg platforms, similar to Eos-A, mass savings which were equivalent to 14% of the baseline propellant budget or 11% of the baseline payload were obtained. Arcjet propellant throughput and total impulse requirements were less demanding than those pertaining to near term communication satellites. When electric propulsion was considered for N/S stationkeeping of 8 MT class GEO platforms, the arcjet and ion system propellant masses were only 51% and 25% of the hydrazine required by the baseline system which employed monopropellant hydrazine thrusters. The electric propulsion systems provided overall mass savings of 1304 to 1572 kg, which are equivalent to 15% to 18% of the BOL platform mass. Each of the four 1.4 kW arcjet systems used for platform stationkeeping would be required to operate for 1080 hr, which is about 470 hours longer than required for near term communication satellites. However, routine operation of arcjets has been experienced during 1000 hr design verification tests. The lifetime requirement for each of eight ion thrusters was about 1750 hr, which is a factor of three lower than that required by systems to be flown in 1992 by the Japanese ETS VI spacecraft. Results of the GEO platform study should be considered quite conservative because of the use of state-of-the-art propulsion technology. Ongoing programs are developing higher power (3 to 5 kW) arcjet and ion thrusters which would reduce the number of thrusters required for platform stationkeeping.

As platforms advance to larger solar arrays and antennas, the E/W stationkeeping requirement may approach the magnitude of N/S stationkeeping. Propellant mass fractions in excess of 30% of beginning-of-life (BOL) spacecraft mass may be required using conventional propulsion. If ion propulsion systems were used for both maneuvers, for example, the propellant mass fraction could be reduced to less than 5% of BOL platform mass. The leverage electric propulsion exerts on platform systems increases dramatically from Eos class LEO platforms to GEO platforms with relatively large area-to-mass ratios.


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TABLE I - PLATFORM CHARACTERISTICS

| DESIGNATION | EOS-A PLATFORM | GEO PLATFORM |
|---|---|---|
| REFERENCE | 4 | 8,9 |
| DIMENSIONS UNDEPLOYED | | |
| DIA./WIDTH, m | 4.4 | 3 |
| HEIGHT, m | 12.4 | 18 |
| LAUNCH VEHICLE | TITAN IV | - |
| TRANSFER ORBIT ALTITUDE, km | 185 X 705 | (1) |
| INCLINATION, deg. | 98.2 | 0 |
| PLATFORM TRANSFER ORBIT MASS, kg | 15,000 | (1) |
| BOL ON-ORBIT MASS, kg | 14,000 (2) | 8,000 - 9,000 |
| TYPICAL PAYLOAD MASS, kg | 3,500 | 3,800 |
| TOTAL POWER, kW | 6.0 | 3.2 |
| ON-ORBIT SERVICING | NO | NO |
| DESIGN LIFETIME FOR PROPULSION, yr | 7.5 | 10 |

(1) Transfer to GEO by onboard perigee/apogee motors or by an orbit transfer vehicle.

(2) Includes ocean disposal.

TABLE II - EOS CLASS PLATFORM PROPULSION SYSTEMS

| MANEUVER | DELTA-V REQUIREMENT, m/s, (ref. 4) | EOS-A BASELINE, (ref. 4) | ADVANCED PROPULSION |
|---|---|---|---|
| ORBIT ACQUISITION | 164 | NTO/MMH<br>SILICIDE-COATED NB CHAMBER<br>Isp = 298 s<br>T = 445 N | NTO/MMH<br>IR-LINED RE CHAMBER<br>Isp = 323 s<br>T = 445 N |
| ORBIT MAINTENANCE | | N2H4<br>Isp = 200 s<br>T = 2.2 N | N2H4 ARCJET<br>Pe = 700 W<br>Isp = 450 s<br>T = 0.091 N |
| -DRAG MAKEUP | 32 | | |
| -ECCENTRICITY CONTROL | 7 | | |
| -INCLINATION CONTROL | 36 | | |
| ATTITUDE CONTROL | | N2H4<br>Isp = 180 s<br>T = 22 N | N2H4<br>Isp = 180 s<br>T = 22 N |
| DISPOSAL OF PLATFORM | 232 | .........SAME AS ORBIT ACQUISITION | .................. |

TABLE III - PROPELLANT BUDGETS FOR EOS CLASS PLATFORMS

| | EOS BASELINE (1) kg | | ADVANCED PROPULSION USING BASELINE POWER SYSTEM (6 kW), kg | | ADVANCED PROPULSION WITH UPGRADED POWER SYSTEM (7.6 kW), kg | |
|---|---|---|---|---|---|---|
| | BIPROP | HYDRAZINE | BIPROP | HYDRAZINE | BIPROP | HYDRAZINE |
| ORBIT ACQUISITION | 819 | 6 | 758 | 6 | 758 | 6 |
| ORBIT MAINTENANCE | | 532 | | 239 | | 239 |
| ATTITUDE CONTROL | | 50 | | 50 | | 50 |
| DEBOOST/DISPOSAL | 1039 | 8 | 988 | 8 | 988 | 8 |
| TOTAL PROPELLANT WITH 3% RESIDUALS AND 10% MARGIN | 2105 | 675 | 1978 | 343 | 1978 | 343 |
| TOTAL PROPELLANT | 2780 | | 2321 | | 2321 | |
| ARCJET SYSTEM PLUS ADDED POWER (2) | | | 96 | | 157 | |
| TANKAGE MASS REDUCTION (3) | | | 35 | | 35 | |
| MASS SAVINGS OVER BASELINE | | | 127 | 271 | 127 | 210 |
| | | | 398 | | 337 | |

(1) Transfer orbit mass 15,000 kg, ref. 4.
(2) Includes 4% harness loss, 10% power margin and 30% mass margin for added solar array. Specific mass of 29.4 kg/kW, ref. 8.
(3) 0.064 tankage fraction, ref. 40; tankage structure is 4% of the sum of propellant plus tankage masses, ref. 39.

TABLE IV - ARCJET PROPULSION SYSTEM MASS

| | |
|---|---|
| 8 THRUSTERS (1) | 9 kg |
| 8 POWER PROCESSORS (700 W each) (1) | 36 |
| THERMAL CONTROL (2) | 6 |
| THRUSTER STRUCTURE (3) | 3 |
| INTERFACE MODULE (3) | 20 |
| 30% MARGIN | 22 |
| TOTAL MASS | 96 kg |

(1) Scaled from ref. 26. Power to PPU/thruster: 700 W.
(2) Based on 40 kg/kW of dissipated power, ref. 37.
(3) Estimated from 5 to 10 kW systems, ref. 38.

TABLE V - GEO PLATFORM SUBSYSTEM DEFINITION AND ANALYSIS ASSUMPTIONS

- N/S STATIONKEEPING PERFORMED BY ONE OF FOUR PROPULSION OPTIONS
- E/W STATIONKEEPING, MOMENTUM CONTROL, ORBIT TRIMMING, AND END-OF-LIFE MANEUVER PERFORMED BY 2.2 N AND 22 N MONOPROPELLANT HYDRAZINE THRUSTERS
- 30% MARGINS INCLUDED IN POWER AND PROPULSION SYSTEM DRY MASSES
- PROPELLANT MASS INCLUDES 3% RESIDUALS AND 10% MARGIN
- 50 MICROMETER SILICON SOLAR CELLS (14.3 kg/kW), 50 VDC POWER SYSTEM, ref. 41
- MAGNETIC BEARING REACTION WHEEL ATTITUDE CONTROL SYSTEM
- PASSIVE THERMAL CONTROL SYSTEM (40 kg/kW), ref. 37

TABLE VI - PROPULSION SYSTEM REQUIREMENTS FOR A 10 YEAR PLATFORM

| PROPULSION FUNCTION | DELTA-V REQUIREMENT, (m/s), ref. 8 | THRUSTER SYSTEM |
|---|---|---|
| N/S STATIONKEEPING | 440 | FOUR OPTIONS |
| E/W STATIONKEEPING | 53 | MONOPROPELLANT HYDRAZINE |
| MOMENTUM CONTROL | | |
| ROLL | 1.4 | |
| PITCH | 13.1 | |
| YAW | 0.1 | |
| ORBIT ACQUISITION | 45 | |
| EOL ORBIT 300 km BOOST | 13 | |

## TABLE VII - TYPICAL THRUSTER CHARACTERISTICS

| PROPULSION SYSTEM | THRUST, N | SPECIFIC IMPULSE, s | POWER, W | REFERENCE |
|---|---|---|---|---|
| MONOPROPELLANT HYDRAZINE | 2.2, 22 | 200 | - | 11 |
| HYDRAZINE RESISTOJET | 0.23 | 290 | 400 | 11,12 |
| HYDRAZINE ARCJET | 0.20 | 450 | 1400 | 26 |
| XENON ION THRUSTER | 0.061 | 2800 | 1400 | 30 |

## TABLE VIII - POWER SYSTEM MASS SUMMARY

| SYSTEM ELEMENT | 3.2 kW SYSTEM kg | 4.2 kW SYSTEM [Two sets of two arcjets share operation] kg | 7.0 kW SYSTEM [Two sets of four ion thrusters share operation] kg |
|---|---|---|---|
| SOLAR ARRAY | 54 | 71 | 119 |
| BATTERY | 81 | 81 | 81 |
| POWER MANAGEMENT AND DISTRIBUTION | 31 | 38 | 61 |
| SUBTOTAL | 167 | 190 | 261 |
| 30% MARGIN | 50 | 57 | 78 |
| TOTAL | 217 | 247 | 339 |

* Power generation: 50 micrometer Si solar cells
  50 micrometer ceria-doped coverglass
  Carbon-loaded Kapton substrate
* Power storage: Two NiH2 batteries
  45 A-hr capacity
  48 cells in series
  70% depth of discharge

TABLE IX - ON-ORBIT PROPULSION SYSTEM DRY MASS FOR A GEO PLATFORM

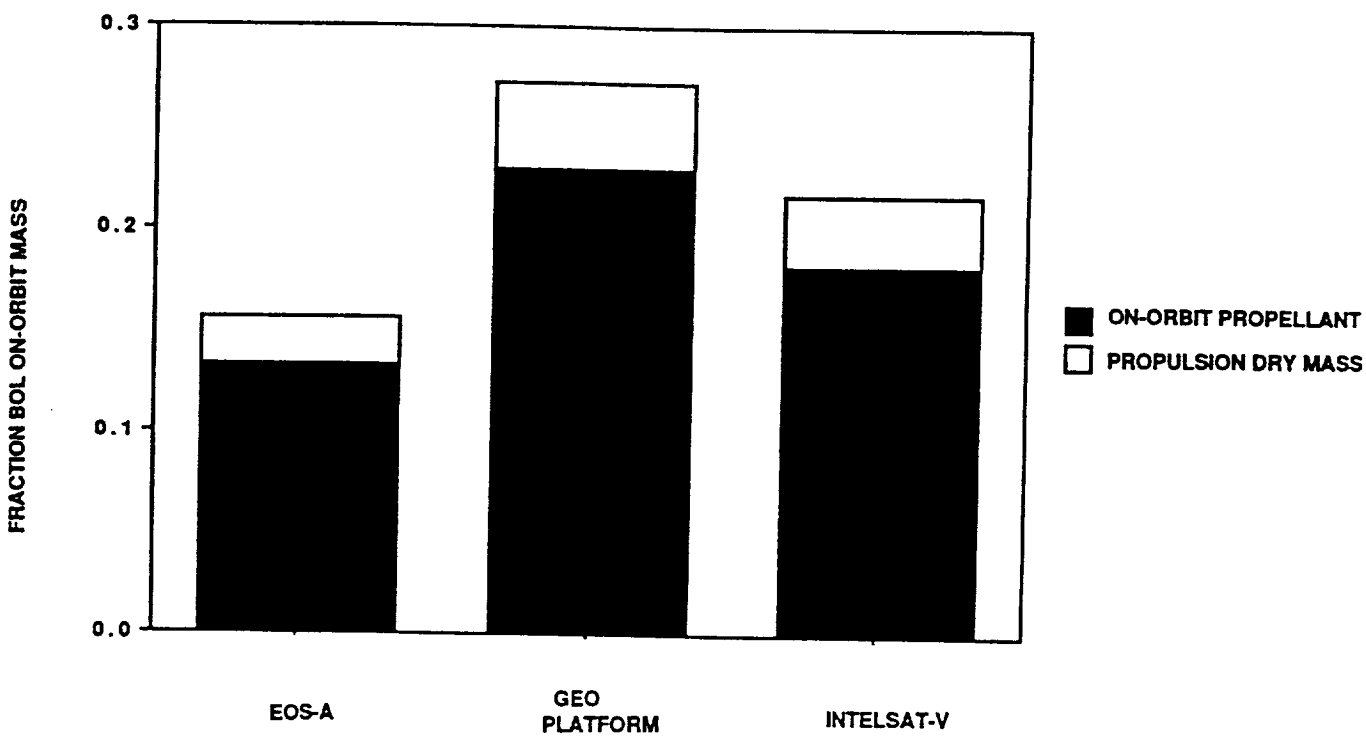
| N/S STATIONKEEPING OPTION ⟶ | MONOPROPELLANT HYDRAZINE | | HYDRAZINE RESISTOJET | | HYDRAZINE ARCJET | | XENON ION THRUSTER | |
|---|---|---|---|---|---|---|---|---|
| SUBASSEMBLY | QUANTITY | MASS, kg | QTY | MASS, kg | QTY | MASS, kg | QTY | MASS, kg |
| N/S STATIONKEEPING THRUSTERS/GIMBALS | 8 | 3 (1,8) | 8 | 7 (1,2) | 8 | 12 (1,3) | 12 | 198 (5) |
| POWER PROCESSORS | | | 8 | 29 (4) | 8 | 37 (3) | 12 | 120 (5) |
| THERMAL CONTROL (6) | | | | 13 | | 11 | | 23 |
| THRUSTER STRUCTURE (7) | | 1 | | 2 | | 3 | | 13 |
| MONOPROPELLANT N2H4 THRUSTERS (8) | 16 | 5 | 16 | 5 | 16 | 5 | 16 | 5 |
| INTERFACE MODULE AND PROPELLANT FEED SYSTEM (7) | | 5 | | 16 | | 20 | | 25 |
| SUBTOTAL | | 14 | | 72 | | 88 | | 384 |
| TANKAGE INCLUDING STRUCTURE (9) | | 275 | | 196 | | 140 | N2H4<br>XE | 54<br>31 |
| MARGIN (30%) | | 87 | | 80 | | 68 | | 141 |
| TOTAL DRY MASS | | 376 | | 348 | | 296 | | 610 |

(1) No gimbal
(2) ref. 12
(3) ref. 26
(4) refs. 8, 43
(5) refs. 30, 44. Gimbal mass = 30% of thruster mass.
(6) Assume 40 kg/kW of dissipated power, ref. 37
(7) Estimates from 5 to 10 kW systems, ref. 38
(8) ref. 8
(9) Hydrazine tankage fraction = 0.064, ref. 40
Xenon tankage fraction = 0.15, ref. 30
Tankage structure fraction = 0.04 of the sum of propellant plus tankage masses, ref. 39

TABLE X - GEO PLATFORM MASS SUMMARY (KG)

| N/S STATIONKEEPING SYSTEM → SUBSYSTEM | BASELINE MONOPROPELLANT HYDRAZINE (1) | HYDRAZINE RESISTOJET | HYDRAZINE ARCJET | XENON ION THRUSTER |
|---|---|---|---|---|
| GN&C, THERMAL(NON-PROPULSION), DMS/TTC, CE, STRUCTURE (2) | 1746 | 1746 | 1746 | 1746 |
| PAYLOAD | 3836 | 3836 | 3836 | 3836 |
| POWER | 217 | 217 | 247 | 339 |
| PROPULSION •DRY MASS | 376 | 348 | 296 | 610 |
| •PROPELLANT (3) -HYDRAZINE | 2559 | 1826 | 1305 | 505 |
| -XENON | | | | 126 |
| BOL PLATFORM MASS IN GEO, kg | 8734 | 7973 | 7430 | 7162 |
| MASS SAVINGS OVER BASELINE, kg | - | 761 | 1304 | 1572 |

(1) ref. 8
(2) GN&C: Guidance, Navigation and Control
DMS/TTC: Data Management System/Telemetry, Tracking and Command
CE: Control Electronics
(3) Includes 3% residuals and 10% margin.




| SPACECRAFT | EOS-A | GEO PLATFORM | INTELSAT-V |
|---|---|---|---|
| ALTITUDE,km | 705 | 35,800 | 35,800 |
| LIFE, yr | 5 * | 10 | 7 |
| BOL SPACECRAFT MASS, kg | 14,000 | 7970 | 980 |
| TYPE OF ON-ORBIT PROPULSION | MONOPROPELLANT HYDRAZINE | HYDRAZINE RESISTOJET | HYDRAZINE RESISTOJET |
| REFERENCE | 4 | 8, THIS PAPER | 7, 10 |

* 7.5 year propulsion design life

FIGURE 1. - BEGINNING OF LIFE (BOL) ON-ORBIT PROPULSION MASS FRACTIONS FOR LEO AND GEO SPACECRAFT

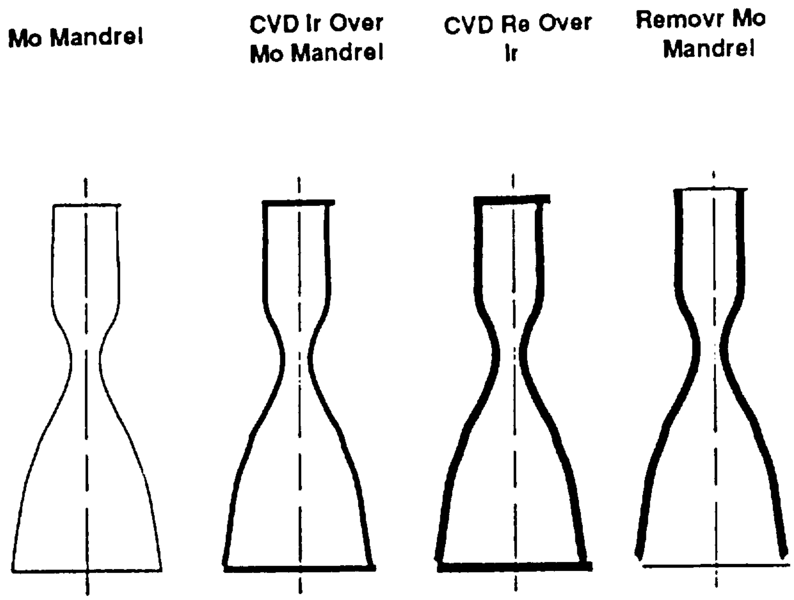



FIGURE 2. - BASIC INSIDE-OUT FABRICATION SEQUENCE FOR BIPROPELLANT THRUST CHAMBERS (ref. 25)

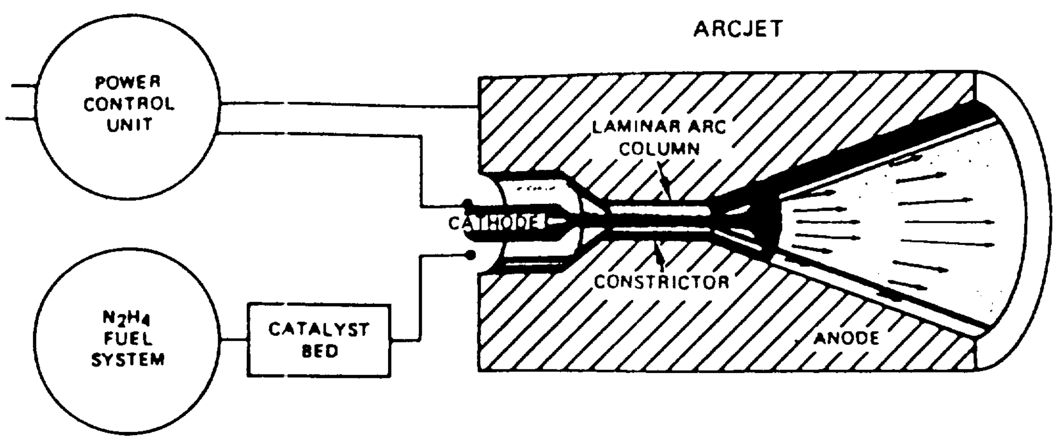



FIGURE 3. - 1.4 kW HYDRAZINE ARCJET (ref. 26)

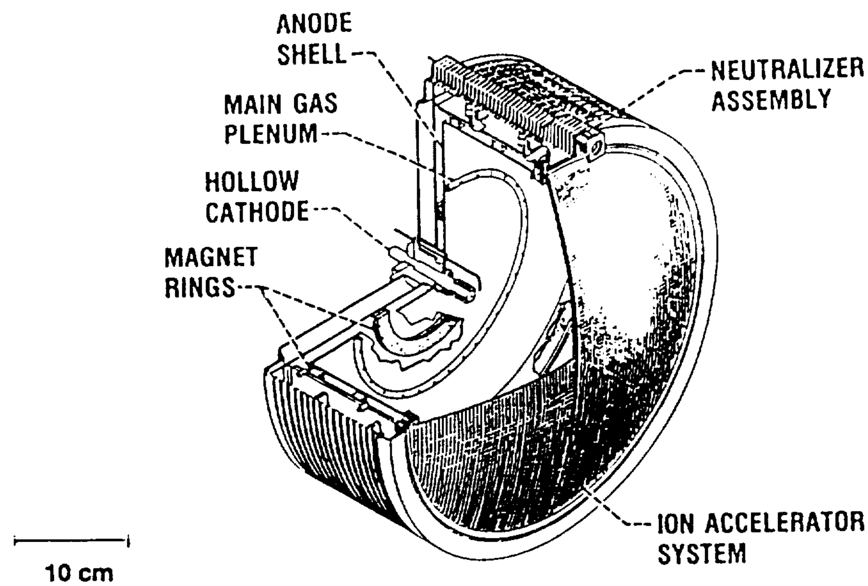
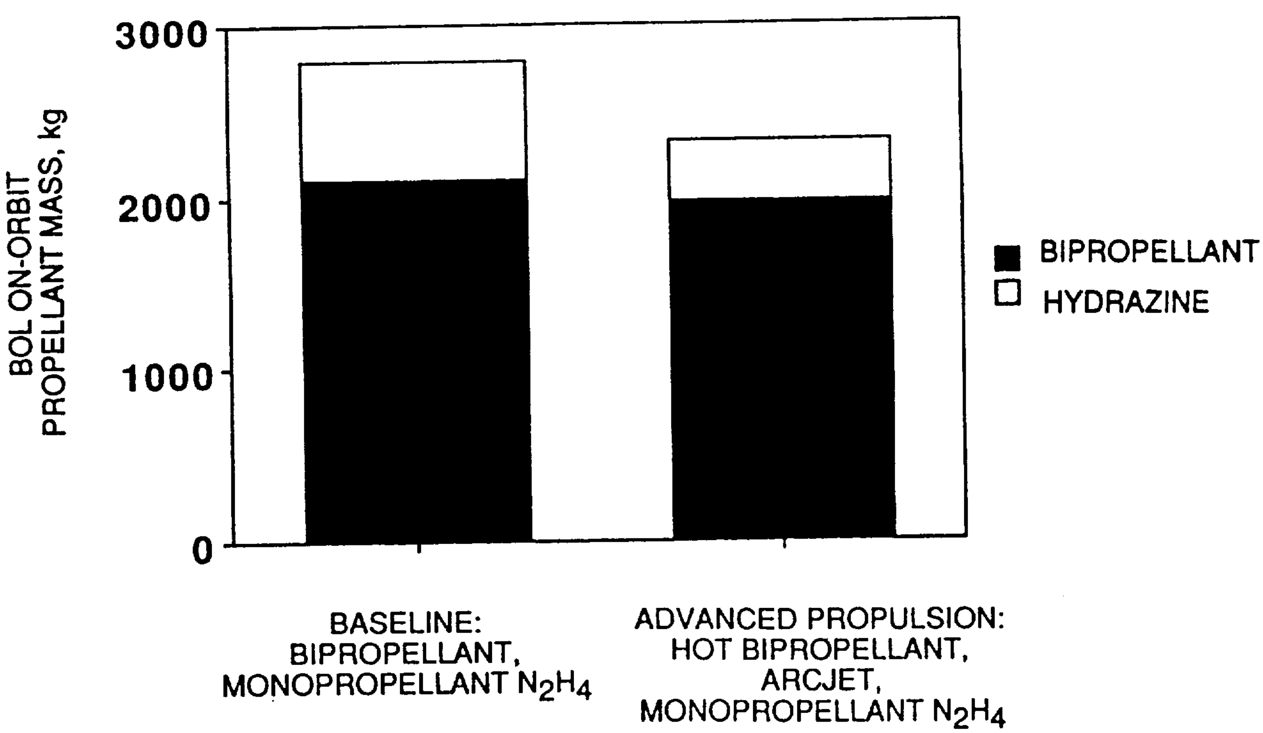



FIGURE 4. - XENON ION THRUSTER




**MASS SAVINGS OVER BASELINE EOS USING HOT BIPROPELLANTS AND ARCJETS: 398 kg ***

* ADDED ARCJET SYSTEM: +96 kg; REDUCED $N_2H_4$ TANKAGE: -35 kg

FIGURE 5. - PROPELLANT BUDGETS FOR EOS-CLASS PLATFORMS

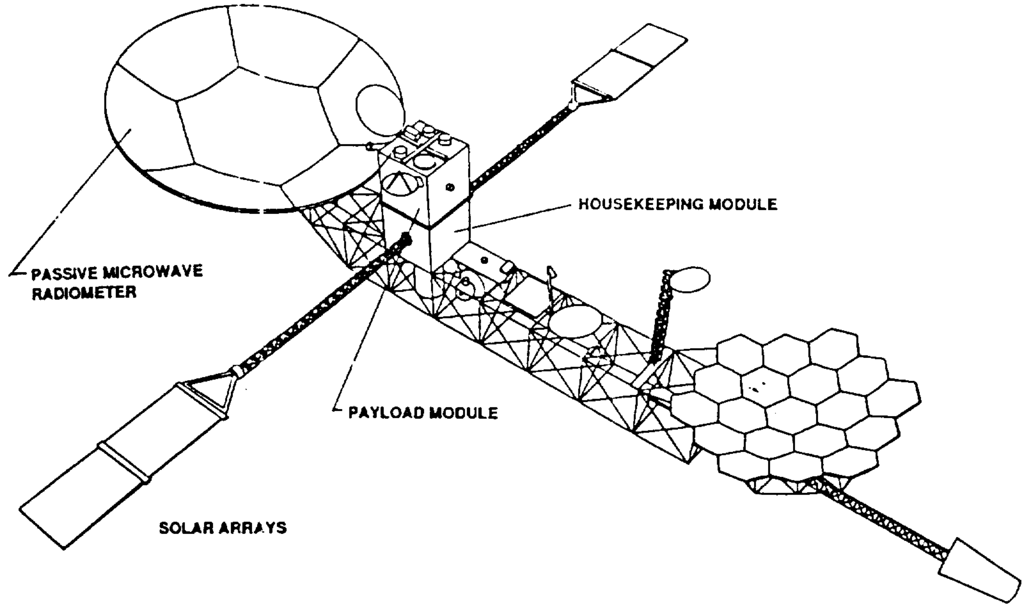
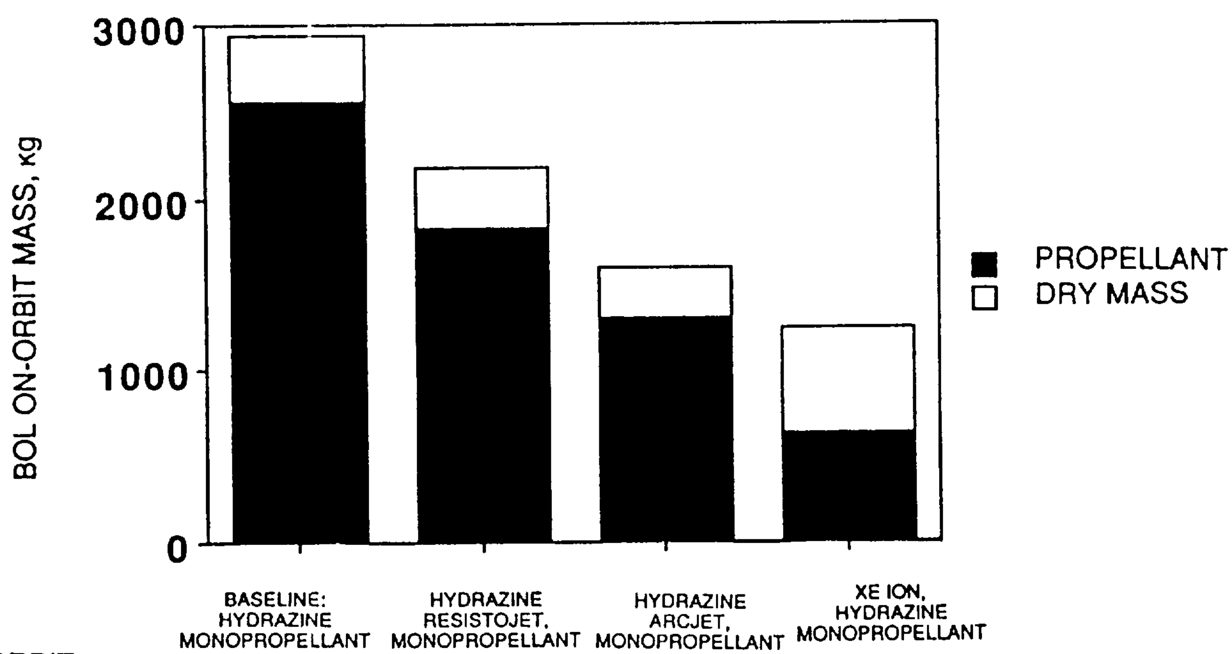



FIGURE 6. - GEOSTATIONARY PLATFORM CONCEPT




| | | | | |
|---|---|---|---|---|
| BOL ON-ORBIT PLATFORM MASS, kg: | 8734 | 7973 | 7430 | 7162 |

**ARCJET AND ION SYSTEMS PROVIDE GEO PLATFORM MASS SAVINGS OF 1304 AND 1572 kg, RESPECTIVELY**

FIGURE 7. - GEO PLATFORM MASS SUMMARY

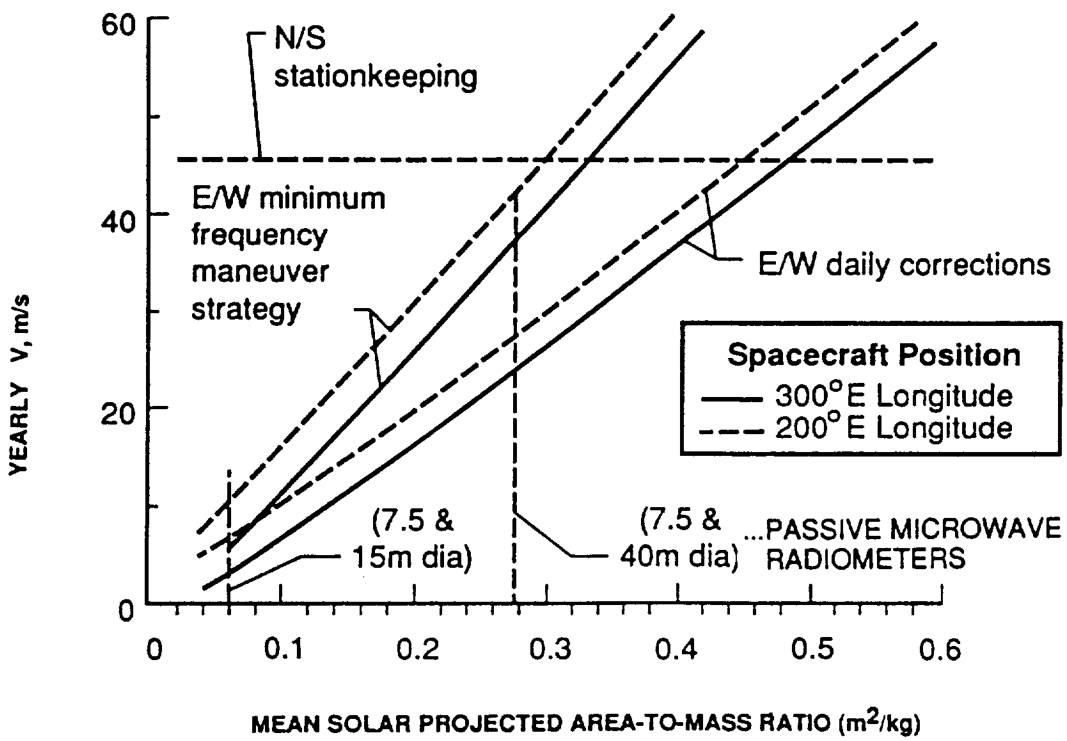



FIGURE 8. - STATIONKEEPING TRENDS: LONGITUDE VARIATION AND CORRECTION FREQUENCY

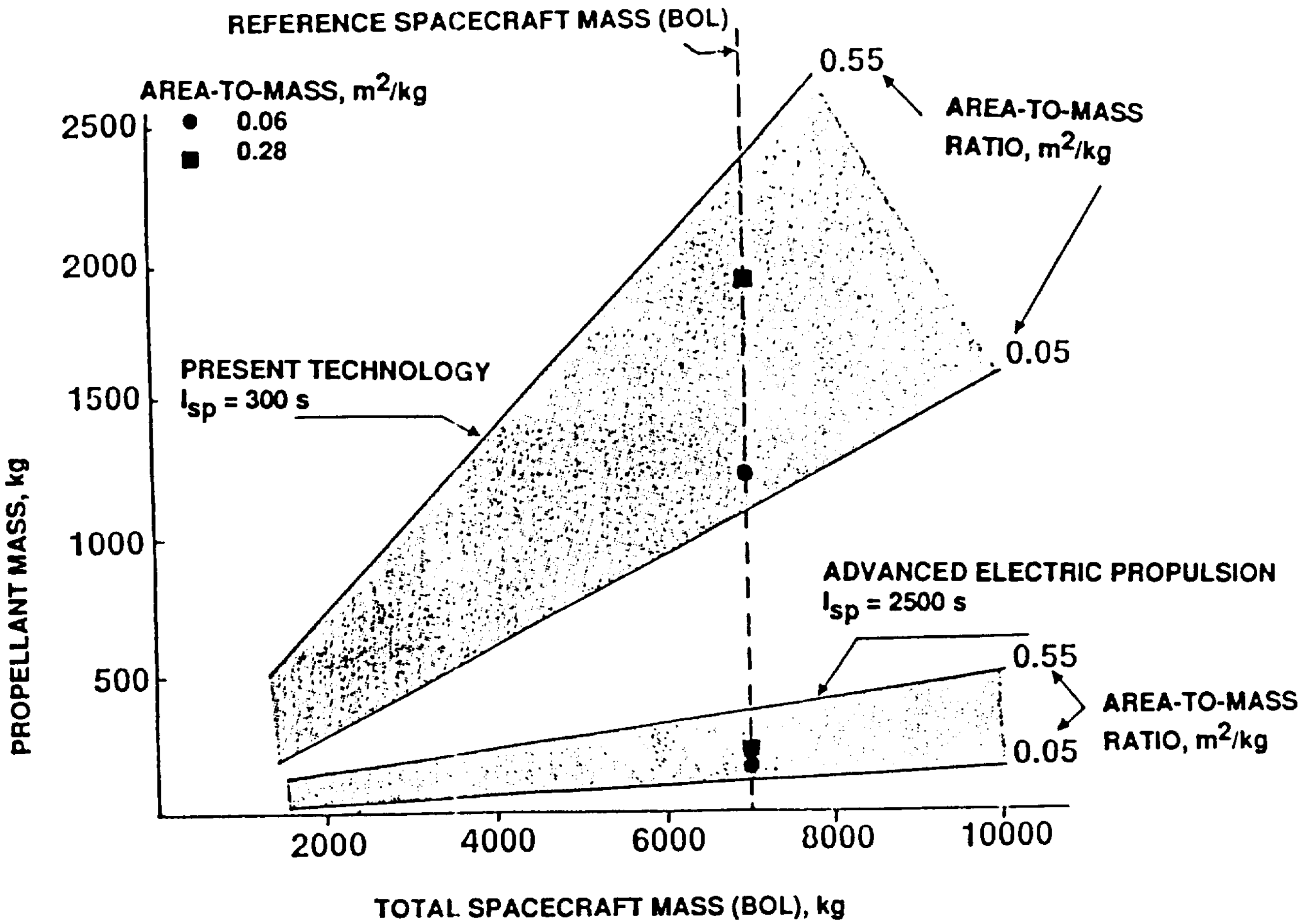



FIGURE 9. - COMPARISONS OF STATIONKEEPING PROPELLANT REQUIREMENTS FOR 10 YEAR LIFE IN GEO (ref. 8)

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16. Abstract

Mission requirements and mass savings applicable to specific low Earth orbit and geostationary Earth orbit platforms using three highly developed propulsion systems are described. Advanced hypergolic bipropellant thrusters and hydrazine arcjets can provide about 11% additional instrument payload to 14,000 kg LEO platforms. By using electric propulsion on a 8,000 kg class GEO platform, mass savings in excess of 15% of the beginning-of-life platform mass are obtained. Effects of large, advanced technology solar arrays and antennas on platform propulsion requirements are also discussed.

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