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Advanced chemical and low power electric propulsion offer attractive options for small satellite propulsion. Applications include orbit raising, orbit maintenance, attitude control, repositioning, and deorbit of both Earth-space and planetary spacecraft. Potential propulsion technologies for these functions include high pressure Ir/Re bipropellant engines, very low power arcjets, Hall thrusters, and pulsed plasma thrusters, all of which have been shown to operate in manners consistent with currently planned small satellites. Mission analyses show that insertion of advanced propulsion technologies enables and/or greatly enhances many planned small satellite missions. Examples of commercial, DoD, and NASA missions are provided to illustrate the potential benefits of using advanced propulsion options on small satellites.

Introduction

The current emphasis on cost reduction and spacecraft downsizing has forced a reevaluation of technologies with critical impact on spacecraft mass. For many commercial, scientific, and DoD near-Earth missions, on-board propulsion is the predominant spacecraft mass. Therefore, high performance propulsion systems offer substantial leverage for reducing injected mass requirements. Additional issues resulting from the emphases on use of smaller launch vehicles, new spacecraft architectures, and the costs associated with ground testing and handling toxic or hypergolic propellants have also led to the consideration of alternative propulsion technologies.

Small spacecraft require propulsion for a wide range of on-orbit functions, including orbit raising and adjustment, drag make-up and stationkeeping, sun-synchronous orbit maintenance, and satellite orientation control. In addition, new communications and remote sensing markets and requirements for constellation maintenance and deorbit are emerging which will increase propulsion requirements for small satellites. This diverse set of propulsion functions results in a wide range of propulsion requirements. Figure 1 shows the total impulse required by a number of planned NASA, DoD, and commercial small spacecraft. The values range from a low of 1.4×10^4 N-s for the HETE spacecraft[1] to a high of 2.5×10^6 N-s for the Vesta asteroid rendezvous mission.[2] Commercial spacecraft, not identified by name in the figure because of their proprietary nature, also require a wide range of total impulses. These propulsion requirements result in the typical small satellite mass breakouts shown in Fig. 2. For all cases shown the propulsion system wet mass is the largest mass spacecraft subsystem, and thus improvements in this subsystem have potential for large satellite mass reductions.

On-board propulsion options include both advanced chemical and electric propulsion technologies. Advanced chemical engines, using nitrogen tetroxide with either monomethyl hydrazine or hydrazine propellants and liquid oxygen with hydrazine propellant, have been successfully tested using high temperature Ir/Re combustion chambers at thrust levels between 5 and 400 N.[3] A new effort is underway to reduce the engine volume now required to achieve specific impulses between 320 and 350 s. This effort is directed at developing high pressure chemical rocket systems, and includes propellant feed system, pump, and combustion chamber technologies.

Near-term electric propulsion options for small, power limited spacecraft include very low power arcjets, Hall thrusters, and pulsed plasma thrusters (PPTs). While the planned spacecraft power range, shown in Fig. 3, is quite large, there is a clear need for electric propulsion systems requiring less than 500 W of power. 1.8 kW arcjets are currently flying on AT&T's Telstar 401 satellite, and arcjets have been successfully operated at power levels below 100 W. However, arcjet performance was found to degrade substantially at power levels below 400 W.[4] Hall thrusters have been flown

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on over 60 Soviet and Russian spacecraft.[5] PPTs, which use solid cloroflourocarbon propellant, have been operational on several spacecraft for over 20 years.[6] PPTs have several unique features which make them attractive for small satellite missions, including simplicity, use of inert, non-toxic propellants, and the ability to operate over a wide input power range at constant performance via changes in pulse frequency.[6]

The renewed emphasis on small, power- and volume-limited spacecraft has opened up a series of opportunities for application of advanced on-board propulsion technologies. Results of the study presented in this paper show that significant improvements in payload mass, reduced spacecraft mass and volume, and enhanced mission capabilities can be achieved by replacing the current propulsion systems with new high performance chemical or electric systems. This paper reviews the status of these small satellite propulsion options, and provides examples of commercial, DoD, and NASA missions for which advanced propulsion offers significant benefits.

Propulsion Options

Advanced Chemical Rockets

Low thrust chemical rockets are currently used on almost all space missions, and development of both Earth-storable and space-storable concepts is continuing.[3] Earth-storable propellants include nitrogen tetroxide as an oxidizer, with monomethyl hydrazine or anhydrous hydrazine as fuels. Space storable propellants include liquid oxygen as an oxidizer with hydrazine or nontoxic hydrocarbons, such as liquid methane, ethane, and ethanol, as fuels. Rocket chambers are presently fabricated from niobium (C-103) with a fused silica coating (R-512A or R-512E) for oxidation protection. Improved performance and lifetime for small chemical rockets are sought through the introduction of higher temperature materials to eliminate fuel-film cooling and its associated combustion inefficiency, and improved component designs to optimize performance and reduce system mass and volume. Elimination of fuel-film cooling also reduces spacecraft contamination issues. The most promising material under development is iridium-coated rhenium. Component tests of designs optimizing performance have indicated that gains of 10 to 20 s specific impulse (I_{sp}) are possible with Earth-storable propellants. Further gains of 5 to 10 s I_{sp} are expected with designs which operate at high chamber pressure such that frozen flow losses in the nozzle are minimized. Components designed for space-storable propellants are expected to provide an additional 15 to 20 s I_{sp} over Earth-storables due to the more energetic nature of these propellants.

Performance and life tests of 22, 62, and 440 N thrust class rockets using Ir/Re chambers have been conducted with both nitrogen tetroxide/monomethyl hydrazine and nitrogen tetroxide/hydrazine propellants.[7,8] Both steady-state and pulsed testing were performed, and thermal management issues were successfully addressed. Performance and life results are shown in Table 1.

High pressure chemical systems have been developed recently for short-lived DoD missions. NASA is sponsoring a program to develop long-lived systems which will leverage advances made in DoD and industrial programs. High pressure tests of small rockets will be used to determine their combustion chamber efficiency when designed with high temperature materials. These materials may offer the thermal margin necessary to withstand the increased heat fluxes associated with high pressure rocket chambers, without paying a performance penalty for film cooling. Operation at high pressure also allows a reduction in size of rockets, which is potentially of value to small satellites.

Recent efforts to improve the performance of small chemical rockets have focused the use of the more energetic space storable propellants. These propellants can be passively stored in space, within mission constraints, without active cooling or refrigeration. Based on system analysis, liquid oxygen and hydrazine were chosen for rocket development at TRW using their pintle injector design. Tests to date have produced a maximum I_{sp} of 350 seconds based on 200:1 area ratio nozzle.[9,10] In addition, a facility is under construction at NASA's Lewis Research Center to test liquid oxygen/hydrocarbons to explore nontoxic propellant options.

The chemical propulsion options anticipated for small satellites and their estimated performance are given in Table 2. Component masses used in the analysis are given in Table 3. Tank masses were derived from an empirical relationship[11] using the operating pressure given in Table 2. The pressurant tank was assumed to be fiber overwrapped and to operate at 3.44×10^7 Pa (5000 psia). Vendor data indicated that overwrapped tanks were half the weight of state-of-art tanks. Many of the other state-of-art component masses are also given in Ref. 11. The lightweight component masses were obtained from commercial vendors. Typical monopropellant and bipropellant propulsion systems dry masses were derived for comparative analysis using these data and the system schematic shown in Fig. 4. The results are summarized in Table 4, and were used in the mission analyses presented below.

Very Low Power Arcjets

A highly simplified schematic of an arcjet thruster system is shown in Fig. 5. In operation, an arc is initiated between the cathode and the converging secion of the anode and is forced by the propellant flow through the throat to seat diffusely in the diverging section of the nozzle which also functions as the anode of the device. The arcjet electrodes are made from tungsten alloys. Current arcjets use hydrazine propellant so as to be compatible with flight qualified propellant feed systems, and the propellant is passed through a catalyst decomposition bed before entering the thruster. The arcjet power processing unit (PPU) must ignite the discharge and reliably operate the thruster in both the period of transition immediately following startup and in the steady state mode. Operating voltages are on the order of 100 V.[12,13] Flight arcjets have been built for power levels of 1.4 and 1.8 kW, and current development efforts are focused on both increasing the I_{sp} to 600 s at 2.0 kW and decreasing the operating power level to between 400 and 800 W. Typical thruster performance during steady-state operation at power levels between 400 and 800 W ranges from 26 to 41 percent efficiency at between 320 and 530 s I_{sp}.

For the mission analyses presented below the arcjets were assumed to operate at 500 W. The arcjet mass, including the catalyst bed, controller, and structure, was set to 1.0 kg, and the PPU efficiency and mass were 90 % and 1.6 kg, respectively. An additional 1.44 kg was assessed to each thruster/PPU set to account for feed system, cabling, and thermal control The hydrazine tankage fraction was taken as 7 %, which is typical of dual-mode propulsion systems.[14] Dry mass contingencies were set to 15 %, which is consistent with the high state of development of flight arcjets.

Hall Thrusters

A simplified schematic of a Hall thruster system is shown in Fig. 6. Briefly, xenon propellant is ionized in the chamber and then accelerated by an axial electrostatic field created by a radial magnetic field which retards the flow of electrons from the external hollow cathode to the anode. While only a single power supply is required in steady-state, thruster ignition requires additional power supplies to preheat the cathode and ignite the discharge. The PPU sequences the power supplies properly to ignite the thruster and transition to steady-state operation.[15,16] The discharge supply is a voltage-regulated power supply connected to the thruster anode and cathode through the electromagnet. In this configuration, the discharge current excites the electromagnet, setting up the radial magnetic field. The discharge current is a function of xenon flow through the thruster, and the PPU maintains closed loop flow control by regulating the discharge current. At the nominal 700 W operating point, the discharge supply output voltage is 300 VDC and the discharge current is about 2.5 A. The cathode heater power supply produces a 12 ADC current at a maximum voltage of 8 VDC. Both breadboard[15] and flight-like power processors[16] have been developed and successfully integrated with 1.4 kW Hall thrusters in the U. S.

Hall thrusters operating at 700 W have been flight qualified in Russia.[17] The nominal operating point for this system is 1600 s I_{sp} and 50 % efficiency. For the mission analyses, the Hall thruster and PPU weights were set to 7.1 kg and 5.7 kg, respectively. Note the thruster mass includes controller and structure. An additional mass of 1.2 kg was assessed to each thruster/PPU set to account for the feed system, cabling, and thermal control. The xenon propellant tankage fraction was taken to be 15%, and 30% contingency was used on the propulsion system dry mass.

Pulsed Plasma Thrusters

Pulsed plasma thrusters rely on the Lorentz force generated by an arc passing from anode to cathode and the self-induced magnetic fields to accelerate a small quantity of cloroflourocarbon propellant.[6] Thruster I_{sp} ranges from 300 to 2000 s, depending on the thruster geometry, operating condition, and propellant choice.[6] Operational PPT power levels range from 5 to 30 W, though they have been extensively tested at 150 W.[18]

Pulsed electromagnetic thruster systems consist of the accelerating electrodes, energy storage unit, power conditioning unit, ignitor supply, and propellant feed system. A typical PPT system schematic is shown in Fig. 7. During operation, the energy storage capacitor is first charged to between 1 and 2 kV, and the ignition supply is then activated to generate a low density plasma which permits the energy storage capacitor to discharge across the face of the cloroflourocarbon propellant bar. The peak arc current level is typically between 2 and 15 kA, and the arc duration is between 5 and 20 μsec.[18] The pulse cycle can be repeated at a rate compatible with the available spacecraft power, and typical missions require over 10^7 pulses.[19] The propellant feed system consists of a negator spring which pushes the solid cloroflourocarbon bar against a stop on the anode electrode. The ability to use the same thruster over a wide range of spacecraft power levels without sacrificing performance is one of the advantages of pulsed thrusters.

Flight PPT systems were developed and flown between 1964 and 1982. Typical flight unit power conditioner efficiencies were near 85 %,[18] yielding system efficiencies between 6 and 13 % depending on the discharge energy level. A flight qualified 30 W PPT system, including the PPU, controls, structure, thermal control, and the propellant storage and feed system, was built in 1974 with a dry mass of 5.85 kg.[18] While no new PPT technology work has been done since approximately 1975, a new NASA effort has been initiated to bring these systems to current state-of-art in electronics, energy storage, and propellant technologies. For this study the PPT efficiency was assumed to be 15 % at 1000 s I_{sp}. The thruster system dry mass was 4.5 kg, and 30 % contingency was used.

Mission Analysis

Mission analyses were performed using a spreadsheet code called Solar Electric Propulsion Spacecraft System and Mission Analyzer (SEPSSMA). This code permits parametric modelling of the spacecraft and mission to establish the final spacecraft mass as a function of thrusting time. The code can also evaluate specific mission scenarios assuming spacecraft mass, power, and full power thrust time. This study included the effects of both atmospheric drag and shading, and neglected array degradation.

The parametric portion of the code models the power and propulsion systems using a specific-mass/power-level combination.[20] The Edelbaum velocity increment of an orbital maneuver was used.[21] The delivered mass, excluding the power and propulsion systems, was calculated as a function of thrusting time to allow the user to find the thrusting time yielding the desired net mass/trip time combination. The I_{sp} maximizing the net mass for the input thrust time was determined analytically.[20]

For a specific mission analysis, system inputs such as launch mass, support system masses, contingency, etc. could be varied. The mission model used 'analytical' steps to assess shading, radiation fluence, and atmospheric drag over the mission. Extra thruster systems were added if their lifetime, which was input by the user, was exceeded.

The analytical trajectory used in SEPSSMA is simplified and does not provide optimal trajectories. Comparisons of selected cases with results from the numerical optimizer SECKSPOT show the deviation is insignificant for the purposes of this study. The calculations assumed that the orbit was quasi-circular during the transfer, that the thrust magnitude and angle were constant during each revolution, and that the thrust-to-weight ratio was 0.01 or less. The model also assumed that the shading caused by the Earth was cylindrical, calculated drag using a 1992 average atmosphere calculated using the results in Ref. 22 with solar panels always perpendicular to velocity (worst case drag), and the radiation model used 45° inclination fluence data.[23] Data for silicon solar array cells were used for all spacecraft. Finally, the impacts on the attitude control system were not assessed.

Small Satellite Missions

Four missions were selected to illustrate the capabilities of advanced propulsion for small satellites. These were orbit raising and maintenance for a 70 kg class LEO commercial spacecraft, orbit raising and stationkeeping for DoD Tactical Satellites (TACSATs), orbit raising and deorbit for NASA's Total Ozone Mapping Spectrometer - Earth Probe (TOMS-EP), and planetary ΔV maneuvers for NASA's Mars Upper Atmosphere Dynamics, Energetics, and Evolution (MAUDEE) spacecraft. For each case the minimum spacecraft modifications possible were made to accommodate the new propulsion system.

Commercial LEO Small Spacecraft

These spacecraft include small communications and remote sensing platforms ranging in mass from 60 to 100 kg.[24] These spacecraft, which would be launched on Pegasus class launch vehicles, have power levels ranging from 50 to 300 W. The very low power requirements of PPTs make them the only suitable electric propulsion candidate for this class of very small satellite. For this analysis a constant 240 W of power (except in shadow) was baselined for the two PPTs placed on each spacecraft. Table 5 compares the number of 68 kg spacecraft which could be launched using a Pegasus XL if each used PPTs or a hydrazine auxiliary propulsion system (HAPS) final stage. By accepting trip times on the order of a few months, it was found that the PPTs could raise the satellite orbit and greatly increase useable payload compared to that delivered directly by the Pegasus launch vehicle. As shown in the table, using the PPTs permits launch of four spacecraft per launch vehicle to any altitude below 3000 km, and substantially increased the mass margin over that obtained using the HAPS stage. The HAPS upper stage could only launch a single spacecraft to a final orbit of 3000 km. These benefits could yield significant launch cost reductions for some missions.

Similarly, maintaining a 100 kg spacecraft in an accurate sun-synchronous orbit for 5 years requires a total PPT system wet mass of 8 kg and a power consumption of 2.5 W. This compares to a propulsion system wet mass of 24 kg for a monopropellant hydrazine system, yielding a savings of 16 kg per satellite. As with the orbit raising mission, this mass savings could be especially significant for cases in which multiple satellites will be launched on a single launch vehicle, as has been proposed for several LEO constellations.

Communication TACSATs/DoD

Tactical Satellites were introduced by Rosen[25] to satisfy the DoDs need for small, capable spacecraft that can be rapidly deployed. A geosynchronous (GEO) communication TACSAT was proposed which would perform backup duties for DSCS III. The projected GEO beginning-of-life mass and lifetime were 455 kg and 10 years, respectively. The satellite would also have a rapid on-orbit repositioning capability to permit rapid response or to provide a larger coverage area. The proposed TACSAT payload would consist of two 40 W DSCS III transponders. This study assumed that the station keeping 'box' was 0.1° wide and that two 90° repositions were required per year for the ten year life. The duration of each repositioning maneuver was two weeks.

Both electric and advanced chemical thrusters can be used to augment the TACSAT capabilities. Assuming the TACSAT is three-axis stabilized and has a payload power level of 1.5 kW, either hydrazine arcjets or xenon Hall thrusters could be added to the satellite to perform the north/south (NSSK) and east/west station keeping (EWSK) as well as to provide rapid on-orbit repositioning. The arcjet configuration assumed that eight 500 W arcjets could be placed on the satellite, two on each east/west face and two on each north/south face, canted 17° to avoid plume impingement of the arrays. Only four 700 W Hall thrusters would be placed on the satellite, two on each north/south side canted 45° to avoid plume impingement of the arrays. The thrusters would operate in pairs on each face. Four PPUs would provide thruster power. Existing geostationary guidance, navigation, and control would be reconfigured to allow for daily NSSK/EWSK burns and repositioning spirals. Battery power would be used to power the thrusters for NSSK/EWSK burns to enable constant communications payload operations. The payload was assumed inactive during reposition thrusting.

The benefits of advanced propulsion for this spacecraft were evaluated in two ways. First, use of electric propulsion for NSSK, EWSK, and repositioning was examined assuming that the initial GTO spacecraft mass was kept constant. For this case, any benefit resulting from reduction of the baseline propulsion system wet mass would be used to augment the payload or increase the spacecraft lifetime. The latter would be achieved by increasing the propellant load beyond that needed for the 10 year life. For this scenario, the initial propulsion system wet masses of the arcjet and Hall thruster equipped communications TACSAT were found to be 140 kg and 130 kg, respectively. This compared with a state-of-art bipropellant repositioning/station keeping fuel mass of 200 kg (assuming 310 s I_{sp} chemical thrusters for repositioning and 285 s I_{sp} thrusters for station keeping[26]). Thus, the propulsion system mass savings resulting from use of arcjets or Hall thrusters were 60 kg and 70 kg, respectively, which could be used to either increase the payload or the spacecraft lifetime, since the initial wet mass was kept constant. Note that the chemical system dry mass and attitude control propellant were left intact for this analysis.

The second benefits analysis incorporated both electric propulsion for stationkeeping and repositioning and the use of advanced chemical propulsion systems (lightweight or pump fed Ir/Re bipropellant) for the apogee insertion. The benefits for these cases were established using the TACSAT characteristics given above, but instead of keeping the initial GTO mass constant the benefits of using advanced propulsion were calculated in terms of reduced GTO mass. This reduced mass could be used either to reduce the required launch vehicle size or increase the payload mass.

Figure 8 shows the required initial GTO mass for each combination of electric and advanced chemical thruster. Electric propulsion station keeping and repositioning reduces the GTO mass by approximately 100 kg. By adding advanced chemical systems for apogee insertion, a total mass reduction of 200 kg or more may be achieved. While the Hall thruster option is best with both the S.O.A. and lightweight bipropellant apogee engines, the spacecraft mass is reduced sufficiently with the pump-fed bipropellant engine that the arcjets lower dry mass yields the lowest mass spacecraft. In terms of launch vehicles, the baseline chemical propulsion TACSAT fits in the Delta 7920 launch vehicle (1300 kg to GTO[27]). Adding electric propulsion would allow use of the Delta 6920 (900 kg to GTO[27]). While Taurus and Pegasus do not have the GTO capability required (125 kg and 375 kg, respectively) development of a launch vehicle with capabilities between the Taurus and Delta classes would permit significant savings if electric and advanced chemical propulsion are utilized by TACSAT class spacecraft.

TOMS-EP/NASA

The TOMS-EP spacecraft is directed at measuring the characteristics of the Earth's ozone layer.[28] The baseline mission is to be launched using a Pegasus XL into a 275 x 350 km orbit which is then raised using the Orbit Adjust System to a sun synchronous circular orbit at an altitude of 955 km. The fixed solar arrays are sized to provide a maximum of approximately 500 W while not in shadow. The low available power limits the electric propulsion system options to PPTs.

For this mission several operational and system modifications would be necessary for using PPTs. To ensure sufficient power for thruster operation, the 500 W arrays would need to be rotated about their axis during the transfer instead of being fixed. In addition, sufficient chemical propellant was kept to raise the initial orbit to a 400 km circular x 99.3° inclination parking orbit and perform all the baseline mission's attitude control thrusting. The 400 km altitude was selected to ensure that drag did not exceed one sixth of the thrust. The orbit's right ascension was chosen so that the TOMS spacecraft arrived in the final sun synchronous orbit with the proper ascending orbit crossing between 11 a.m. and noon local solar time after the 80 day transfer. Thus, the right ascending node of the parking orbit is 91.2° behind the desired initial sun synchronous ascending node. This results in almost direct solar illumination of the solar arrays for the entire transfer, assuming the arrays are rotated.

For this analysis, the final orbit and the masses of all spacecraft subsystems except propulsion were fixed. Four 200 W PPTs were placed on the spacecraft to replace the existing Orbit Adjust System. The assumed PPT power level is 50 W higher than the 150 W for which extensive testing has been performed. To complete the planned TOMS-EP mission, the PPTs would be placed in pairs on the sides of the spacecraft pointing through the center of mass. One set of thrusters would perform the transfer while both sets would alternate to circularize the final orbit. This configuration preserves the normal attitude control setup planned for TOMS-EP. By keeping the payload pointed in the nadir direction the PPT thrusters would always be pointed in a circumferential direction (perpendicular to the radius vector and in the orbit plane), which closely optimizes thrusting and greatly simplifies the guidance, navigation, and control requirements. On arrival at the final altitude the orbital eccentricity would be removed with either the chemical or PPT system similar to the original TOMS mission.

Results of the mission analysis are shown in Table 6. Using PPTs increases the baseline payload from 35 kg to 55 kg, an increase of nearly 60 %. The total transfer time is 107 days, of which 79 days are spent thrusting.

No other orbit maintenance is required for the TOMS mission, though additional maneuvers such as repositioning, orbit raising, and deorbit are possible for a relatively small amount of additional PPT fuel. While the deorbit is not provided for in the baseline TOMS mission, new NASA guidelines require that all spacecraft below 2000 km must end their mission with a perigee of 500 km or less to ensure a timely deorbit disposal of the spacecraft.[29] If the baseline chemical TOMS spacecraft were required to change its perigee to 500 km, an additional 12 kg of fuel (an impulsive maneuver of 120 m/s at 220 s I_{sp}) would be required which would reduce the useable payload to only 23 kg.

For the PPT equipped spacecraft only an additional 4.1 kg of fuel would be needed to spiral down to a 500 km circular orbit, which still leaves a payload mass of 51 kg. Thus, even with deorbit, PPTs provide for 16 kg (46 %) extra payload over the baseline mission without deorbit. With full power the deorbit transfer would require 52 days including shadow time (38 days of thrusting). Table 7 presents the results of the baseline TOMS-EP mission with a deorbit requirement and the benefits gained by using the PPTs.

Mars Orbiter

The MUADEE[30] Discovery class mission was used to illustrate the benefits of using advanced chemical engines for planetary missions. The MUADEE spacecraft is a simple spinner design based on the Pioneer Venus spacecraft. MUADEE will explore and sample Mars' upper atmosphere by flying through it.

While several launch opportunities exist for MUADEE, this study used the baseline launch date of 04/08/01. The Mars-capture ΔV was assumed to be 1472 m/s to insert MUADEE into a 40,000 km by 600 km orbit. The periapsis would then be lowered to ~130 km (23.3 m/s ΔV) to begin science operations. Both of these maneuvers (a total of 1495.3 m/s ΔV) would to be performed by the bipropellant system.

The baseline bipropellant system which performs the Mars insertion and orbit acquisition consists of three canted 410 N thrusters at 300 s I_{sp}. Other small ΔV maneuvers are performed by 220 s I_{sp} small thrusters. Assuming a 90% efficiency and the 300 s I_{sp}, the baseline science payload for MUADEE is 49 kg. By removing the baseline Mars capture engines and replacing them with equivalent thrust level advanced engines (lightweight and pump-fed Ir/Re bipropellant) the capture and initial orbit maneuver fuel may be reduced significantly. Electric propulsion options were not considered for the Mars orbit maneuvers because they did not provide significant benefits for the very small ΔVs required. The resulting spacecraft mass breakdown comparisons for the various bipropellant options are shown in Fig. 9. The substitution of the lightweight and the pump fed Ir/Re chemical engines allows for a payload enhancement of ~40 kg and ~80 kg, respectively. This mass could be utilized to add science instruments, provide more maneuvering fuel, and/or increase lifetime.

Summary

Advanced propulsion options for Earth-space and planetary small satellites include advanced chemical systems, very low power hydrazine arcjets, xenon Hall thrusters, and cloroflourocarbon propellant pulsed plasma thrusters (PPTs). Earth and space storable propellant chemical engines have been demonstrated with specific impulses of 330 s and 350 s, respectively, and efforts to develop low volume, lightweight Ir/Re bipropellant systems are underway. Low power arcjets have been extensively tested at power levels between 500 and 800 W, and yield specific impulses between 320 and 510 s at 26 to 42 % efficiency. Hall thrusters providing 1600 s I_{sp} at 50 % efficiency have been flight qualified in Russia at a power level of 700 W. PPTs providing between 1000 and 2000 s I_{sp} at between 8 and 15 % efficiency were flight qualified in the mid-1970's at power levels between 1 and 150 W. Current development efforts are directed toward both bringing the PPT power technology to today's standards and improving PPT performance.

Four example missions were used to illustrate the potential benefits of using advanced propulsion on small, power limited spacecraft. These missions were orbit raising and maintenance of 100 kg class LEO commercial satellites, apogee insertion, repositioning and stationkeeping of a DoD communication TACSAT, orbit raising and deorbit of NASA's Total Ozone Mapping Spectrometer - Earth Probe mission, and providing the Mars-capture ΔV and in-orbit maneuvering for NASA's Mars Upper Atmosphere Dynamics, Energetics and Evolution spacecraft. For each case, significant mass savings were obtained using advanced propulsion technology. While electric propulsion increased the trip time for orbit transfer missions, in all cases use of advanced propulsion systems either greatly increased the payload capability, increased the number of spacecraft per launch vehicle, or allowed significant extensions of the spacecraft operational lifetime.

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Table 1 Performance and life data on Ir/Re rockets.

Thrust Class, N	Propellants	Area Ratio	Performance, sec	Total Operating Time, hr	Total Cycles
22	NTO/MMH	150:1	310	1.7	100,311
62	NTO/MMH	75:1	305	0.2	263
440	NTO/MMH	286:1	321	6.2	93
550	NTO/N_2H_4	200:1	330	-	-

Table 2 Chemical propulsion system options.

Option	Tank Pressure, MPa (psia)	Rocket Chamber Pressure, MPa (psia)	Specific Impulse, sec
S.O.A. bipropellant	1.79 (260)	0.69 (100)	315
Lightweight Ir/Re bipropellant	1.79 (260)	0.69 (100)	330
Pumped Ir/Re bipropellant	0.344 (50)	2.75 (400)	345

Table 3 State-of-art and advanced chemical propulsion system component masses.

Component	S.O.A. Weight, kg	Lightweight, kg
Propellant tank	$1.2+35.0V_p(m^3)/N_p$*	$0.6+17.5V_p(m^3)/N_p$
He tank	$0.6+260V_{He}(m^3)/N_{He}$**	$0.3+130V_{He}(m^3)/N_{He}$
Axial thruster	3.36	3.36
ACS thruster	0.21	0.21
Pyro valves	0.15	0.04
Manual valves	0.08	0.02
Check valves	0.16	0.02
Latch valves	0.26	0.04
Relief valves	0.45	0.02
Filters	0.23	0.05
Regulator	0.84	0.07
Lines and fittings	2.50	2.50
Pressure transducers	0.10	0.10
Temperature transducers	0.01	0.01
Residuals	$0.03V_p(m^3)$	$0.03V_p(m^3)$
Structure	10% of component mass	10% of component mass
Contingency	10% of dry mass	10% of dry mass
He pressurant	$59.8V_{He}(m^3)$	$59.8V_{He}(m^3)$
Pump (if used)	5.80	2.30
Propellant tank (pumped)	$1.2+6.7V_p(m^3)/N_p$	$0.6+3.4V_p(m^3)/N_p$

*N_p: number of propellant tanks
**N_{He}: number of helium tanks

Table 4 Chemical propulsion system dry masses.

Option	Dry Mass, kg
S.O.A. monopropellant	$18.9+42.4V_p(m^3)+315V_{He}(m^3)$
Lightweight monopropellant	$12.3+21.1V_p(m^3)+157V_{He}(m^3)$
Pumped monopropellant	$15.2+4.1V_p(m^3)+157V_{He}(m^3)$
S.O.A. bipropellant	$23.2+42.4V_p(m^3)+315V_{He}(m^3)$
Lightweight Ir/Re bipropellant	$14.3+21.1V_p(m^3)+157V_{He}(m^3)$
Pumped Ir/Re bipropellant	$19.7+4.1V_p(m^3)+157V_{He}(m^3)$

Table 5 Performance comparison of pulsed plasma thrusters and monopropellant hydrazine thrusters for 68 kg commercial LEO spacecraft.

Initial/Final 28.5° Orbit Altitudes, km	PPT Trip Time (with shading), days	# PPT Equipped Spacecraft per Pegasus XL* & Mass Margin	# 68 kg Spacecraft per Pegasus XL using HAPS Upper Stage & Mass Margin
400 / 1000	55	4 S/C & 51 kg	4 S/C & 31 kg
400 / 2000	131	4 S/C & 31 kg	2 S/C & 54 kg
400 / 3000	198	4 S/C & 19 kg	1 S/C & 7 kg

*Assumes 68 kg spacecraft mass plus wet PPT system mass, and 240 W power available for PPTs, 1000 s Isp, and 15% efficiency.

Table 6 TOMS - EP baseline and PPT equipped masses without deorbit requirement.

Element	Baseline Spacecraft Element Mass, kg	PPT Version Element Mass, kg
Spacecraft dry mass less payload	197	207
Chemical fuel	55	15
PPT fuel	-	8.3
Science payload	35	55
Total launch mass	287	287

Table 7 TOMS-EP baseline and PPT equipped masses when deorbit to 500 km is included.

Element	Baseline Spacecraft Element Mass, kg	PPT Version Element Mass, kg
Spacecraft dry mass less payload	197	272
Chemical fuel	78	15
PPT fuel	-	12.4
Science payload	23	51
Total launch mass	287	287

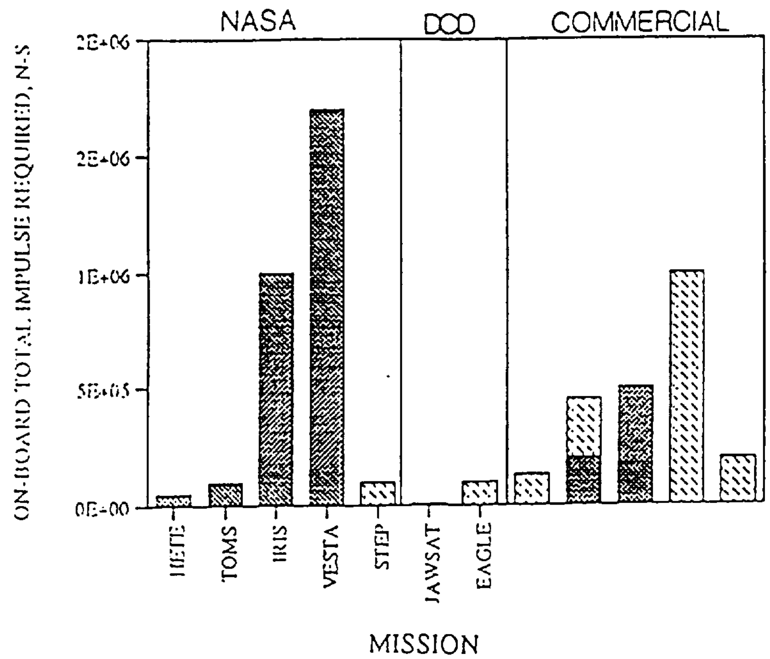

Fig. 1 Total impulse ranges required by planned small satellite missions.

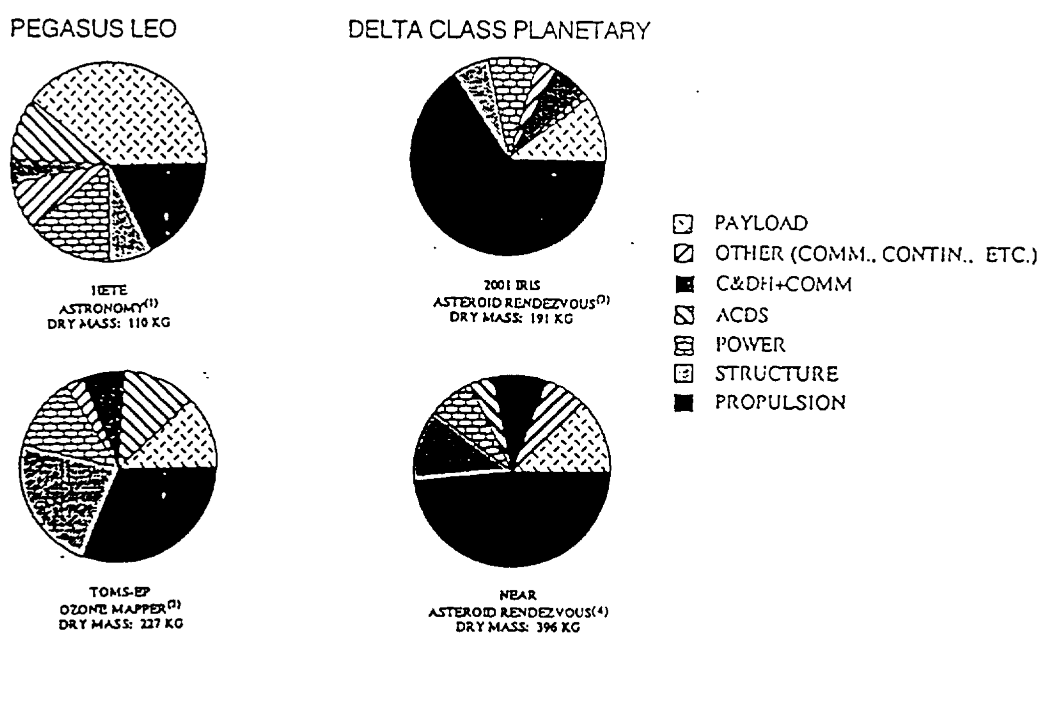

Fig. 2 Mass breakouts of some small spacecraft planned for the Pegasus and Delta launch vehicles.

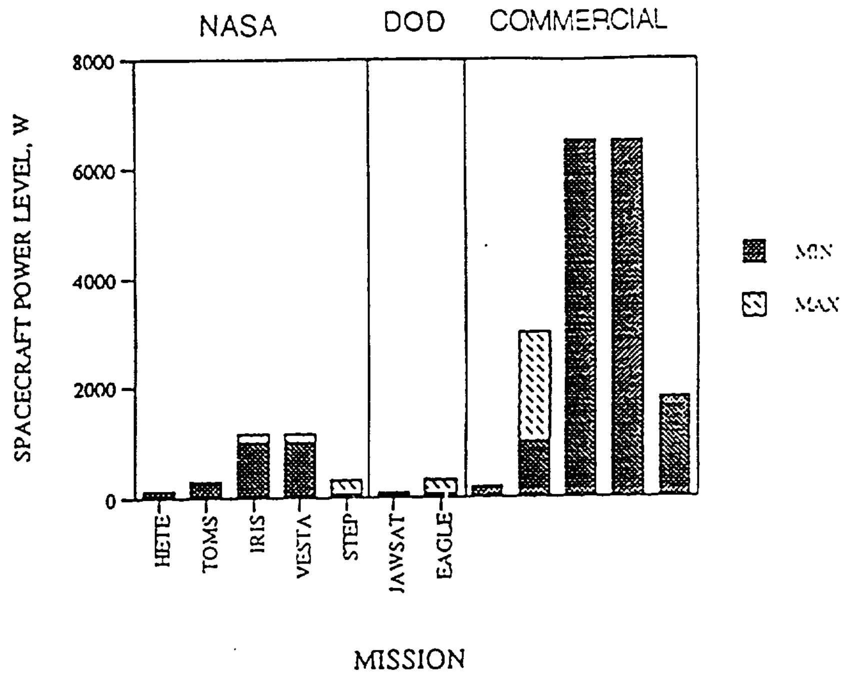

Fig. 3 Planned small spacecraft power range.

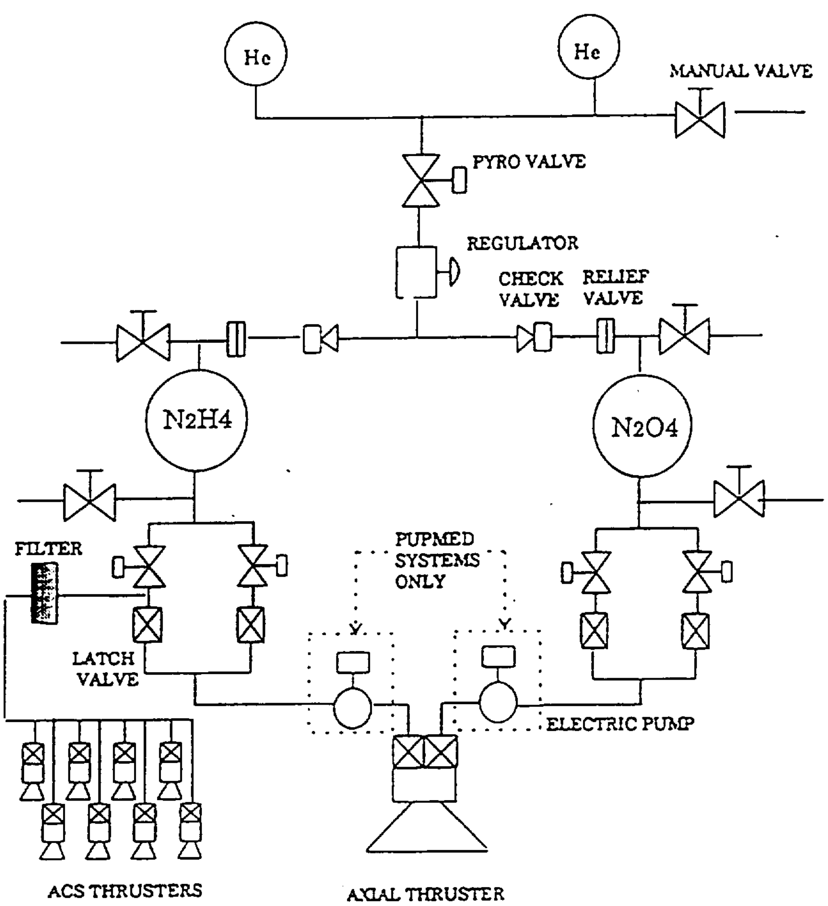

Fig. 4 Chemical propulsion system schematic used for dry mass analysis.

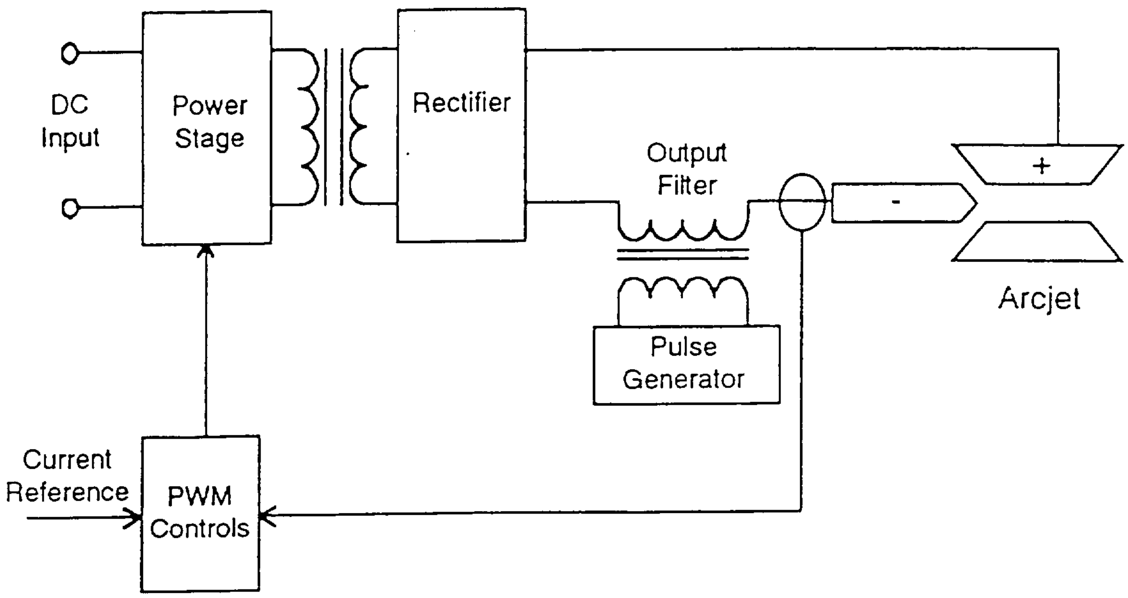

Fig. 5 Arcjet system schematic. Propellant storage and feed not shown.

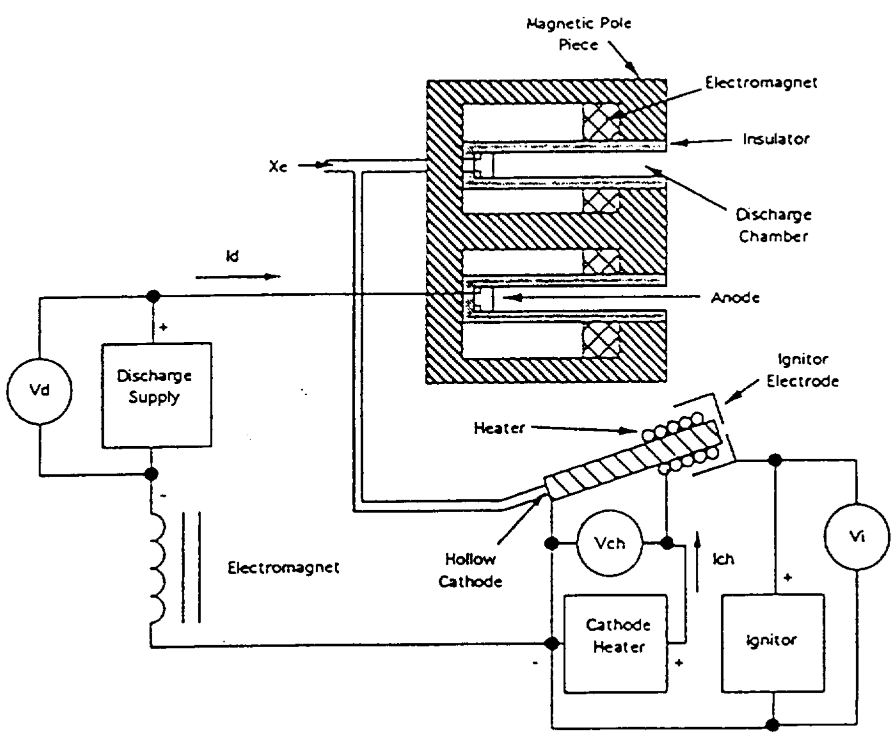

Fig. 6 Hall thruster system schematic. Propellant storage and feed systems not shown.

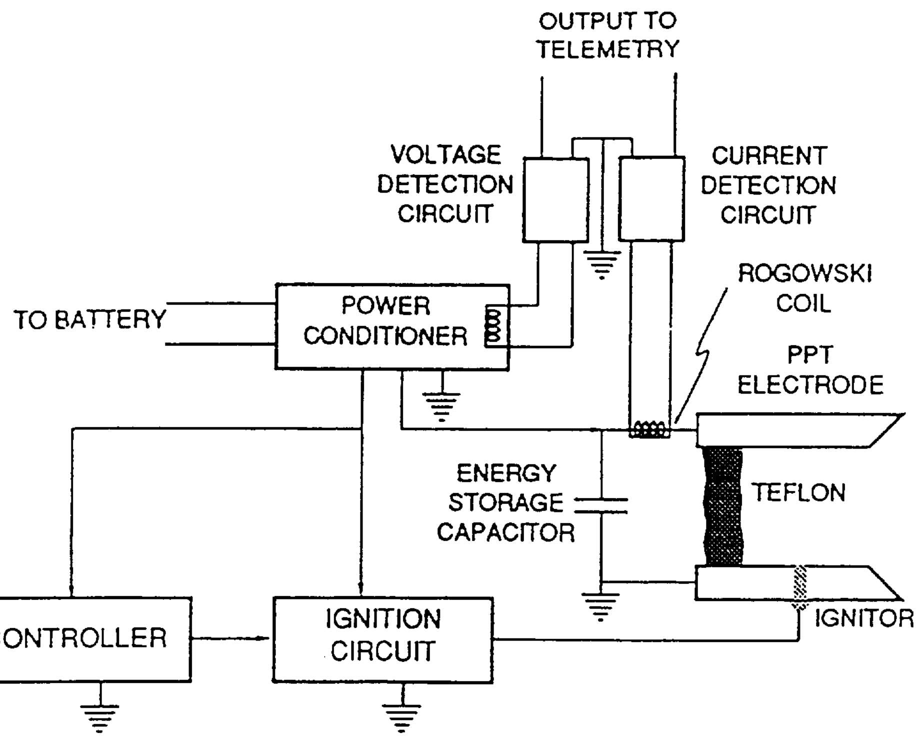

Fig. 7 Pulsed plasma thruster system schematic. Propellant feed system is negator spring pushing teflon bar into interelectrode region.

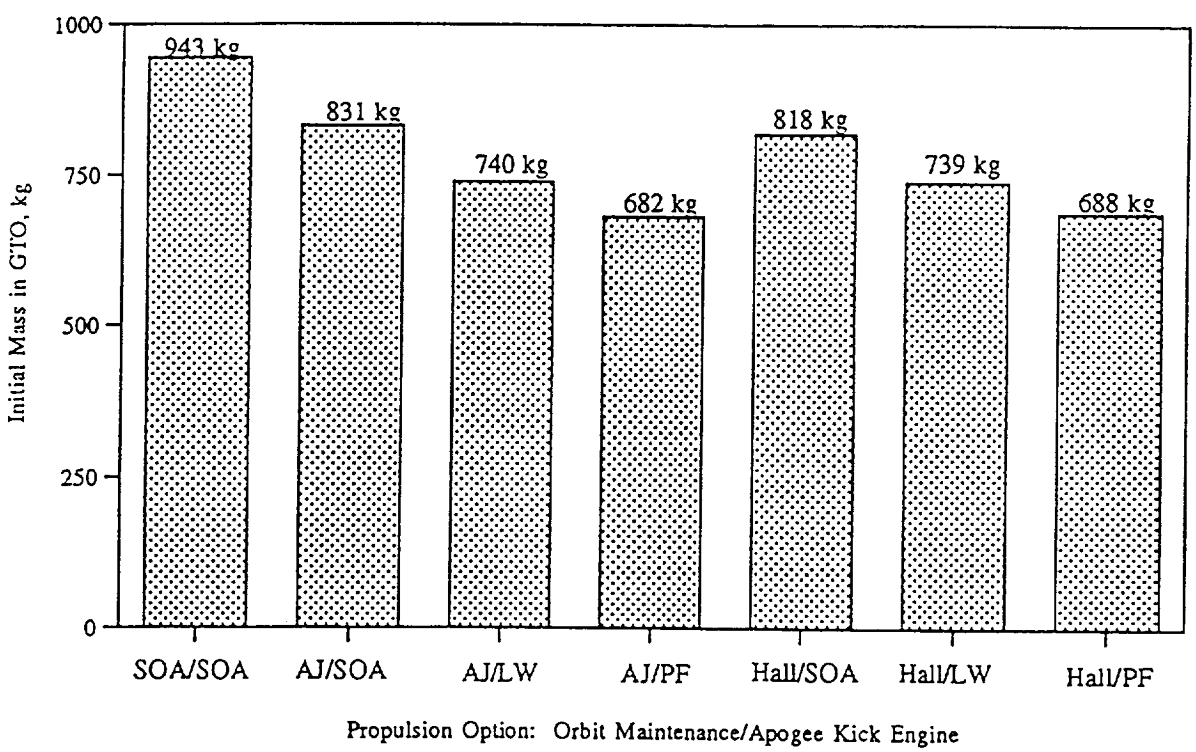

Fig. 8 Required GTO mass for a Communication TACSAT with various propulsion option combinations. SOA = state-of-art bipropellant, AJ = arcjet, LW = lightweight bipropellant, PF = pump-fed bipropellant, and Hall = Hall thruster.

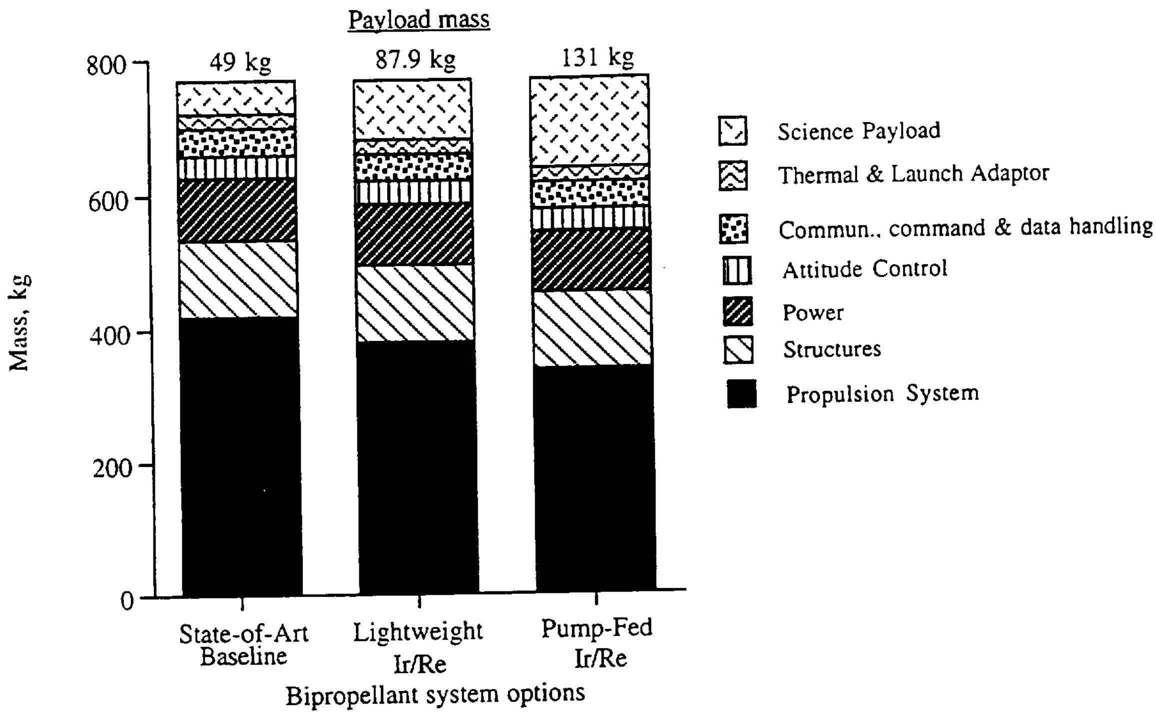

Fig. 9 System mass comparisons for MAUDEE using state-of-art and advanced bipropellant chemical propulsion systems. State-of-art baseline data from Ref. 30.

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13. ABSTRACT *(Maximum 200 words)*

Advanced chemical and low power electric propulsion offer attractive options for small satellite propulsion. Applications include orbit raising, orbit maintenance, attitude control, repositioning, and deorbit of both Earth-space and planetary spacecraft. Potential propulsion technologies for these functions include high pressure Ir/Re bipropellant engines, very low power arcjets, Hall thrusters, and pulsed plasma thrusters, all of which have been shown to operate in manners consistent with currently planned small satellites. Mission analyses show that insertion of advanced propulsion technologies enables and/or greatly enhances many planned small satellite missions. Examples of commercial, DoD, and NASA missions are provided to illustrate the potential benefits of using advanced propulsion options on small satellites.

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