JOURNAL OF PROPULSION AND POWER
Vol. 14, No. 5, September–October 1998

Status and Issues Related to In-Space Propulsion Systems

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In-space propulsion is required for multiple, critical, postlaunch functions for most space missions. The propulsion functions are broadly classified as insertion, on-orbit (including stationkeeping, repositioning, constellation management, etc.), and disposal/deorbit (if required). Orbit transfer represents a major additional element of in-space propulsion but only the onboard segment of in-space propulsion is treated herein. It is estimated that the mass of in-space propulsion will represent over 40% of the mass delivered by launchers and upper stages for all unmanned space missions over the next 10 years and, therefore, will exert a predominant influence on the capability and competitiveness of future space systems. Technical advancements of in-space propulsion are underway and include demonstrations of higher-performance (specific impulse) chemical and electric rocket systems. This paper will briefly 1) review the state-of-the-art in-space propulsion, 2) discuss some new technologies in advanced development, and 3) present the impact of selected advanced technologies on space missions.

Introduction

IN-SPACE propulsion is used after delivery by launchers and (if used) upper stages for many functions such as orbit insertion, on-orbit (including stationkeeping, repositioning, ultraprecise ephemeris control, and constellation management), and disposal/deorbit propulsion. Projections of future spacecraft were reviewed to determine the global drivers on in-space propulsion and typical results are shown in Fig. 1 and Table 1 (which do not include manned systems, some proposed constellations, and other national and experimental missions). Figure 1 shows worldwide spacecraft projections for the nine-year period from 1997 to 2005 (inclusive).[1] It is first noted that 90% of projected future missions will be in Earth space and that over 85% of all missions are expected to be in low earth orbit (LEO), middle Earth orbit (MEO), or geosynchronous Earth orbit (GEO). Table 1 includes rough estimates of average total spacecraft mass and in-space propulsion mass fraction for each orbit so that their overall impact could be projected. (Orbit-transfer stages are not included in the mass estimates of Table 1. Inclusion of those systems would have the effect of greatly increasing the propulsion masses for the higher-altitude missions.) It is seen that the mass of in-space propulsion is estimated to be over 40% of the total projected-launched-mass of unmanned systems, represents the single heaviest subsystem of launched mass, and significantly exceeds the masses of science and/or revenue-producing payloads or any other spacecraft bus subsystem. This feature is further illustrated in Fig. 2, which shows the mass fractions of in-space propulsion and all other spacecraft elements for some representative space systems. The in-space propulsion mass fractions range from about 10% for an experimental LEO spacecraft at low altitude, to 25% for a heavy LEO spacecraft just under the Van Allen belt, to over 80% (which includes orbit transfer propulsion) for the GEO system. For commercial communication satellites (COMSATS), apogee and north–south stationkeeping (N–S S/K) propulsion usually make up 50–60% of the total mass injected into geosynchronous transfer orbit (GTO).[2] Although somewhat uncertain, the fractional impact of in-space propulsion is expected to strongly increase because of the many emerging technical and programmatic drivers including new and/or significantly more demanding propulsion requirements such as the deorbit of some LEO spacecraft, longer life systems and repositioning, as well as nearly universal desires to decrease launcher sizes/costs. The overall predominance of in-space propulsion mass fractions, and the likelihood of their increased fractional impacts, implies that advanced in-space propulsion concepts offer significant potential for major improvements in the capabilities and competitiveness of many space systems.

Figure 1 illustrates the extreme range of future missions that results in an equally broad and diverse range of requirements for future in-space propulsion systems. The propulsion requirements are driven by considerations of acceleration and torque; wet mass; operations (including safety, storability, and contamination); technical maturity; costs; and many other factors. To meet future mission in-space propulsion requirements, the spacecraft propulsion community is advancing a suite of chemical and electric propulsion concepts: 1) Chemical = monopropellants and spacecraft bipropellant engines (SBEs), which include liquid apogee engines (LAEs) and the secondary combustion augmented thruster (SCAT); and 2) electric = resistojet, arcjet, pulsed plasma thruster (PPT), hall effect thruster (HET), and ion thruster (IT). This paper will briefly describe selected chemical and electric propulsion concepts that have reached a mature state of development, and discuss examples of technology impacts on example missions.

In-Space Propulsion

Monopropellant Thrusters

Monopropellant hydrazine thrusters (MHTs) have been successfully used on Earth-orbit and planetary spacecraft for over 25 years for functions including attitude control, reaction control, stationkeeping, and orbit insertion.[3] Present suppliers of MHTs include Kaiser Marquardt (KM), Primex Aerospace Company (Primex), and TRW in the United States; Daimler-Benz Aerospace (DASA) and Societe Europeen de Propulsion (SEP) in Europe; and Ishikawajima-Harima Heavy Industries (IHI) in Japan. Table 2 lists features of representative MHTs to illustrate characteristics of that rocket class. The suppliers of the rockets are shown and, except as noted, the data were obtained from descriptive sales literature released by the cited organizations. Pressure-fed MHTs have been developed with nominal thrust levels from less than 1 N to >700 N, and typical devices can operate over a large (>4:1) thrust range as the inlet

Received Oct. 23, 1997; revision received April 15, 1998; accepted for publication April 29, 1998.

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Table 1 Projected Worldwide spacecraft[a]

Orbit	Number	Assumed spacecraft mass, kg	Assumed propulsion mass fraction	Total mass, kg	Total in-space propulsion mass, kg
LEO	984	500	0.15[b] (0.25)[c]	492,000	73,800[b] (123,000)[c]
MEO	66	1000	0.20	63,000	13,200
GEO	309	3000	0.55	927,000	509,850
Elliptical	47	2000	0.55	94,000	51,700
Lunar/solar	18	500	0.85	9,000	7,650
Planetary	21	500	0.85	10,500	8,925
Other	63	1000	0.20	63,000	12,600
Totals	1508	——	——	1,661,500	678,000[b] (727,000)[c]

[a]Teal Group in Aerospace American, November, 1996. [b]No deorbit. [c]With deorbit.

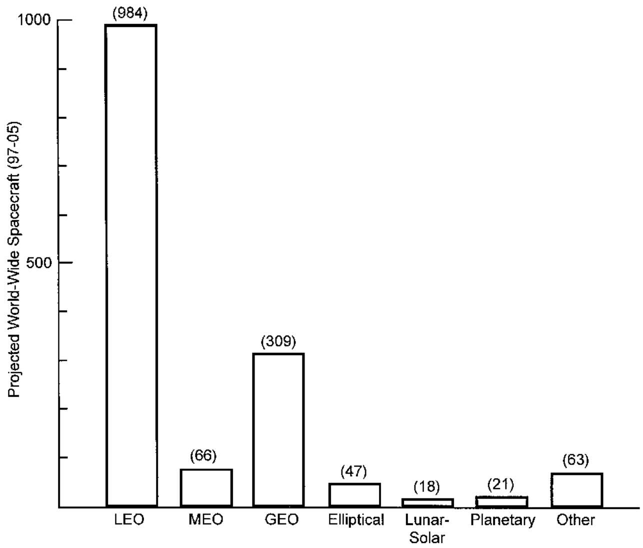

Fig. 1 Number of spacecraft from 1997–2005 (manned, Teledesic, and many experimental missions not included).

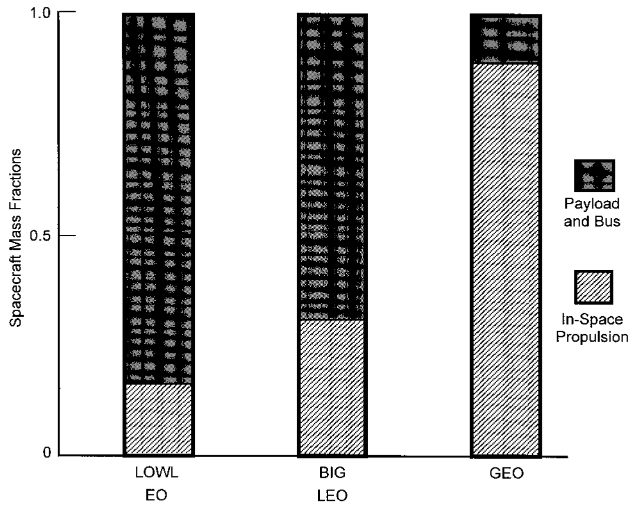

Fig. 2 Mass fractions of recent spacecraft.

pressure is varied. Table 2 also shows that as the thrust level increases 1) the specific impulse reaches a limit slightly below 240 s, 2) the total impulse increases slowly, and 3) the minimum impulse bit increases approximately with the thrust level. Pressure-fed monopropellant hydrazine rockets are very mature concepts with noteworthy success rates and mission lifetimes (including over 25 years on the Pioneer 10 spacecraft launched in 1972). Their established low cost, availability, reliability, and cleanliness will undoubtedly result in the use of MHTs into the indefinite future for missions with modest "delta V" and/or standard attitude/reaction control needs.

Efforts are underway to extend the benefits of pressure-fed monopropellant hydrazine systems via use of monopropellant systems which use: 1) miniature pumps, or 2) propellants with increased performance (specific impulse and/or density) and/or more benign properties. Whitehead[4] developed and flight-tested a hydrazine system that employs a lightweight reciprocating pump that enables the use of high-pressure thrusters with low-pressure tanks. These features promise significant mass fraction benefits for selected applications, such as the Mars ascent, where small total impulse requirements imply 1) extremely high payoff for reduced dry mass but 2) mission energies below those where high-performance bipropellant chemical systems are appropriate. Warm gas thrusters (WGTs), which typically combine a combustion couple (such as H_2/O_2) in a diluent (such as N_2) in a single tank, are being pursued for both propulsion[5] and pressurization[6] applications. The specific impulses of safe WGT options appear limited to below those of hydrazine, so they are likely candidates as replacements for cold-gas systems in applications where safety and integration issues are overarching considerations. Efforts on higher-energy, benign propellants[7,8] have recently concentrated on combinations of hydroxyammonium nitrate (HAN), triethanolammonium nitrate (TEAN), and water, which have also been extensively developed for liquid gun applications. These options promise increases in both density (~40%) and specific impulse (~9%) over hydrazine with significantly reduced handling issues. HAN/TEAN, and other advanced monopropellant options such as nitromethane, offer multiple benefits. At this point, however, efforts to replace hydrazine are in R&D phases and will require considerable efforts prior to flight applications.

Spacecraft Bipropellant Engines

Spacecraft bipropellant engines (SBEs) are routinely used for performance-driven functions such as orbit insertions, orbit changes/maneuvers, and selected stationkeeping and altitude control applications. SBE suppliers include Atlantic Research Corporation (ARC), GenCorp Aerojet (Aerojet), KM, and TRW[9,10] in the U.S.; DASA, Royal Ordnance (RO), and SEP in Europe; and IHI in Japan. Most state-of-practice SBEs use the combination of 1) Earth-storable, liquid propellants [monomethyhydrazine (MMH) or hydrazine (N_2H_4) as fuels and nitrogen oxide mixtures (MON) or nitrogen tetroxide (N_2O_4) as oxidizers]; 2) niobium (Nb) chambers coated with silicide to prevent oxidation; and 3) inlet pressures of about 1.3–1.5 MPa. Table 2 and Fig. 3 show top-level characteristics of SBEs from several suppliers, which represent the highest performance options presently available or disclosed. The data were obtained from suppliers except where noted.

Table 3 illustrates state-of-the-art features, limits, and opportunities for SBEs that are, with an exception discussed later, pressure-fed and radiation-cooled (pump-fed SBEs are rarely felt to be appropriate and will not be discussed herein). With

Table 2 Representative monopropellant thrusters

Thrust, N	Specific impulse, s	Total impulse, N s	Minimum impulse bit, N s	Supplier
1.2–0.19	227–206	1.6×10^5	4.5×10^{-3}	Primex
1.0–0.25	234–222	1.1×10^5	2.3×10^{-2}	DASA[3]
4.45	230	3.1×10^5	——	KM
36	232	1.03×10^6	0.49	TRW
89	235	3.5×10^6	——	KM
721–116	237–232	5.6×10^6	4.5	KM

Table 3 Representative spacecraft bipropellant engines

	NTO/MMH				NTO/N_2H_4				
	Nb		Re		Nb		Ni	Re	Lox/N_2H_4
Inlet pressure, MPa	1.3	1.3	1.5	3.5	1.3	1.3	2.4	1.5, 3.4	1.4
Thrust, N	11	432	445	222	22	469	44	556, 222	890
Specific impulse, s	285	320	323	327	300	321	324	328, 337	353
Supplier	RO	RO	KM	TRW[9]	RO	TRW	TRW	TRW[9], TRW[9]	TRW[10]

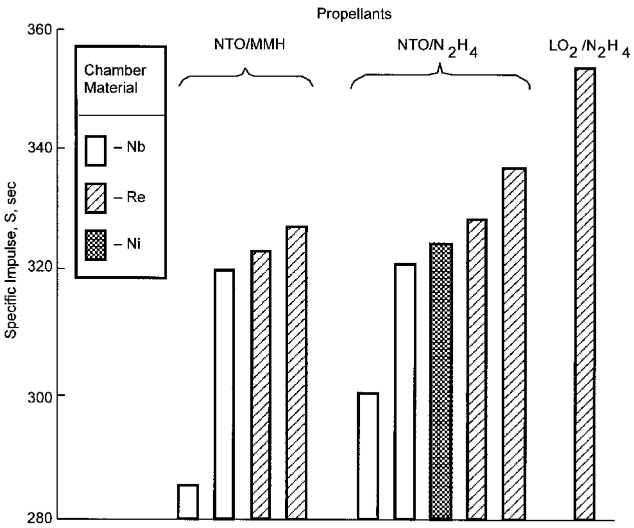

Fig. 3 Representative spacecraft bipropellant engines.

Nb chambers, the specific impulses of radiation-cooled, NTO/MMH SBEs are limited to slightly over 320 s and decrease as thrusts decrease to levels appropriate for stationkeeping and attitude control systems (ACS). This performance constraint is largely set by the chamber materials that impose temperature limits that require film cooling and other limiting design approaches. After considerable R&D,[11,12] rhenium coated with oxidation-resistant materials has emerged as the nearest-term advanced material to replace Nb. Rhenium (Re) chambers have been fabricated via both chemical vapor deposition (CVD) and powder metallurgy (PM) techniques and data on strength and life-cycle fatigue have been obtained.[12] Iridium (Ir) is used as an oxidation-resistant coating and both CVD and electrodeposition processes are employed to deposit Ir on the Re structure. Joints made by diffusion bonding, inertia welding, and explosive bonding have also been evaluated and validated.[13] Multiple research efforts are on-going to develop a lighter-weight alternative to Re. As yet, however, no substitutes have been demonstrated that offer the high temperatures and lifetimes achieved with coated Re rockets. It is first seen that the use of Re chambers results in improved specific impulse with NTO/MMH. Rhenium chambers basically allow the elimination of film cooling and, additionally, enable the use of higher-pressure combustion, neither of which can be achieved with Nb chambers.

Most present SBEs operate at inlet pressures of about 1.3–1.5 MPa as a result of a system-level trade between propellant and state-of-the-art titanium tank masses as the pressure is varied. The advent of lightweight propellant tank technologies, such as carbon-overwrapped, Al-lined concepts, is increasing the system optimum storage/inlet pressure. Table 3 shows that the simultaneous use of high pressures and Re chambers enabled a specific impulse of 327 s to be obtained with NTO/MMH propellant at a thrust level of 222 N.[9] An additional feature of increased-pressure SBEs is their decreased dimensions relative to present concepts. This characteristic may be of special impact in volume-limited situations, such as the orbit insertion of spacecraft that use small launchers. Data are also shown in Table 3 for NTO/N_2H_4 SBEs. This propellant combination is often used to leverage its modest specific impulse increase over NTO/MMH and, more importantly, the "dual-mode" system feature, wherein the hydrazine fuel can also be used for single monopropellant ACS thrusts and/or electric propulsion stationkeeping systems. The performance trends with NTO/N_2H_4 are comparable, for similar reasons, to those with NTO/MMH. NTO/N_2H_4 rockets have recently demonstrated slightly >320 s specific impulse with Nb chambers and have obtained up to 337 s specific impulse with the combination of Re chambers and increased pressures.

The use of high-temperature materials also has allowed the use of high-energy propellants with pressure-fed, radiation-cooled SBEs. Table 3 shows the LO_2/N_2H_4 SBE that demonstrated[10] a specific impulse of over 350 s at a thrust of 890 N. The engine was of research class, but thermal and chemical conditions were compatible with present Re engine technologies. Such high-specific impulses present new application opportunities for low-cost, pressure-fed SBE systems which, in particular, include high-energy Earth-orbit insertions and planetary missions. Some mission managers are considering the use of nontoxic, nonhypergolic propellants to benefit safety and operations. For example, O_2/ethanol is being evaluated for Shuttle Orbiter upgrades,[14] and hydrogen peroxide fuel combinations are again being considered for a wide range of applications. It should be noted that the use of O_2 will likely be limited in the near term to applications such as orbit insertion or Shuttle Orbiter ACS that do not require long-term storage. Multiple SBE concepts are being evaluated for such uses and it is likely that these missions will drive SBE designs somewhat different than those for performance-driven missions that are specifically optimized to reduce overall mission operation costs and risks.

Table 3 also includes, as the SBE with a nickel chamber, a novel approach to high performance with NTO/N_2H_4 called SCAT, shown in Fig. 4. The SCAT uses a standard catalyst bed to gassify hydrazine for use in either a monopropellant mode or as a fuel for a MON/N_2H_4 SBE. MON is gassified in a regeneratively cooled, nickel thrust chamber and the resultant gas–gas combustion enables combustion efficiencies of over

99%. The SCAT is extremely robust and flexible and has been recently flight qualified by TRW for operation over 1) thrust levels from 18 to 50 N, 2) inlet pressure variations of 6.7:1, 3) mixture ratios from 0.85 to 2.0, and 4) operation in monopropellant and bipropellant modes. Efforts are underway to fully validate bilevel thrust operation so that SCATs can perform both delta V and attitude control functions for a given mission. Analyses, anchored by the original SCAT development, have also indicated that a growth version SCAT, operating at 650 N, could achieve specific impulses of about 335 s. If that were realized it would represent the ultimate performance attainable from pressure-fed SBEs using classical Earth-storable propellants.

Electric Propulsion

The five electric-propulsion (EP) concepts shown in Table 4 and Fig. 5 have achieved operational status, and many programs are underway to increase the number and types of missions served by EP.[15–24] The following will briefly highlight the characteristics of mature EP systems that have become operational, or for which near-term flight programs are firmly planned, and comment on the potentials of various classes of EP systems. Table 4 and Fig. 5 show key characteristics of selected, mature EP systems. To mitigate the effects of mission specifics, the system specific mass (in kg/kW) only includes the mass of the thruster and power processor (the masses of the propellant subsystem, gimbals, and other mission specifics are not included).

Resistojets have been used for north–south stationkeeping (N–S S/K) and orbit insertion of, respectively, GEO and LEO COMSATS[25] in the U.S. and for orbit control and ACS functions on Russian spacecraft.[26] Propellant temperatures are fundamentally determined by material limits in resistojets, which implies modest (propellant specific) maxima for specific impulses, which are around 300 s for the 0.5–1 kW class resistojets developed by Primex.[18] Resistojets have several desirable features including values of thrust/power far higher than other EP options (because of their high efficiencies and modest specific impulses), the lowest EP system dry masses (primarily because of the lack of a requirement for a power processor), and uncharged/benign plumes. These features will continue to make resistojets attractive for low-to-modest energy applications, particularly where power limits and/or thrusting times, and/or plume impacts are mission drivers. In addition, resistojets can operate on a wide variety of propellants that led to their proposed use as a propulsion/waste gas management concept on the Space Station[27] and operated on hydrogen for Earth-orbit insertions.[28]

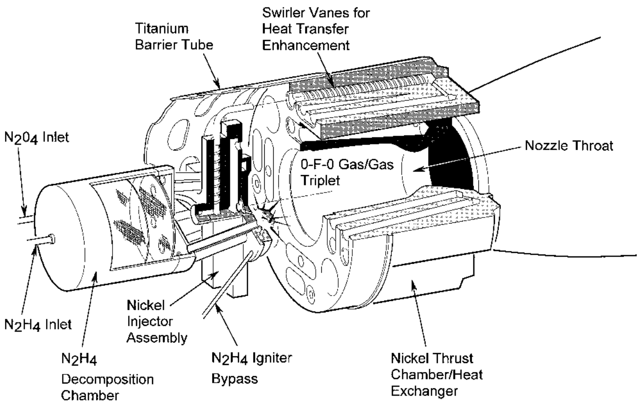

Fig. 4 Cutaway of SCAT.

Table 4 shows three low-power arcjets. The two hydrazine concepts are produced by Primex and are being used for N–S S/K on large COMSATS.[25] The ammonia version[16] was supplied by the Institut für Raumfahrt Systeme (IRS) and the Institut für Thermodynamik und Technische Gebaudeausrustung (ITT) in Germany and will soon be used for orbit insertion (both orbit raising and inclination change) of a German amateur radio spacecraft. Arcjets enable a significant ($\sim 2\times$) increase in specific impulse over resistojets and maintain some desirable features of resistojets such as use of standard propellants and relatively low dry masses. It is seen that the increased specific impulse, coupled with relatively low efficiencies of about 0.3 to 0.4, lead to significant decreases ($\geq 6\times$) of thrust/power relative to resistojets. In addition, as complex plasma/arc phenomena must be controlled, arcjets require relatively complex power conditioning that results in dry masses about twice those of resistojet systems. Significant efforts including development of novel materials were necessary[19] to define and validate the 600-s, hydrazine arcjet. It is likely, therefore, that 600–650 s represents the upper range of specific impulses that can be expected of low-power arcjets using storable propellants. Arcjets do provide major mass benefits for many spacecraft, are relatively simple to integrate, and are the least complex and costly of any plasma propulsion device. For those reasons, low-power arcjets can be expected to undergo evolutionary improvements and be used well into the future for a variety of medium- to high-energy propulsion functions. Table 4 also shows a 26-kW, ammonia arcjet that operates at a specific impulse of 800 s and will be flown in 1998. The arcjet is part of a flight system called ESEX, built under a U.S. Air Force program[20] by a TRW, Primex, and CTA Systems team, which includes the arcjet, supporting subsystems, and a

Table 4 Characteristics of selected electric propulsion flight systems

Concept	Specific impulse, s	Input power, kW	Thrust/power, mN/kW	Specific mass, kg/kW	Propellant	Supplier
Resistojet	296	0.5	743	1.6	N2H4	Primex [18]
	299	0.9	905	1	N2H4	Primex [18]
Arcjet	480	0.85	135	3.5	NH3	IRS/ITT [16]
	502	1.8	138	3.1	N2H4	Primex [19]
	>580	2.17	113	2.5	N2H4	Primex [19]
	800	26[a]	——	——	NH3	TRW, Primex, CTA[20]
PPT	847	<0.03[b]	20.8	195	Teflon	JHU/APL[21]
	1200	<0.02[b]	16.1	85	Teflon	Primex, TiNIIMASH, NASA[23]
HET	1600	1.5	55	7	Xenon	IST, Loral, Fakel[22]
	1638	1.4[a]	——	——	Xenon	Primex, TiNIIMASH, NASA[23]
	2042	4.5	54.3	6	Xenon	SPI. KeRC
IT	2585	0.5	35.6	23.6	Xenon	HAC[24]
	2906	0.74	37.3	22	Xenon	MELCO, Toshiba[17]
	3250	0.6	30	25	Xenon	MMS[16]
	3280	2.5	41	9.1	Xenon	HAC, NASA[15]
	3400	0.6	25.6	23.7	Xenon	DASA[16]

[a]Thruster input power. [b]Power dependent on pulse rate.

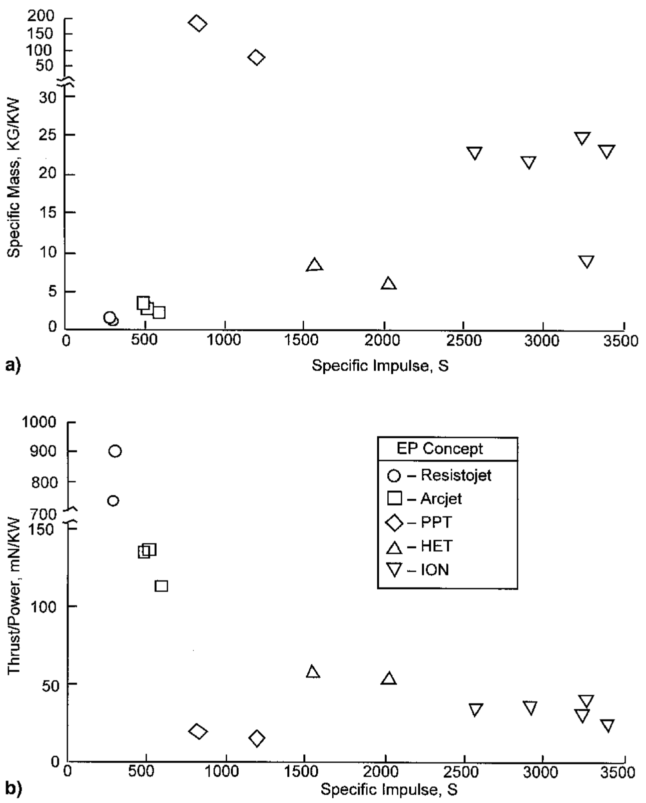

Fig. 5 Key characteristics of mature electric propulsion systems: a) specific mass and b) thrust/power.

diagnostic suite to evaluate plume and EMI effects. The increased specific impulse relative to low-power arcjets is largely a result of a reduction of losses associated with low-Reynolds-number flows that are fundamental penalties for low-power (smaller) arcjets. The space test will represent a major ($\geq 10\times$) increase in power level of flight-demonstrated EP devices and will address integration and mission issues critical to potential users of high-power (orbit-transfer-class) electric propulsion.

Pulsed plasma thrusters (PPTs) are inherently pulsed devices and versions that operate at about 847 s specific impulse [built by the Johns Hopkins University, Applied Physics Laboratory (JHU/APL)] that have successfully maintained precision control of three NOVA spacecraft for many years.[21] PPTs feature very small ($\leq 4 \times 10^{-4}$ N s) impulse bit capability, use of a solid propellant (Teflon®), and the ability to operate at near constant performance over large power ranges. An improved-version PPT that operates up to 1200 s specific impulse (Table 4), is now being developed by a NASA/Primex team and a flight test in 1999 is planned[25] on the Earth Observer 1 spacecraft to demonstrate propulsive ACS. The characteristics of PPTs will likely limit their power operating range to under a few hundred watts and, as suggested by Table 4, they have large dry masses. Within their operating capability, however, PPTs promise a combination of low-power, high-specific-impulse, and small-impulse bit that is unique. It is anticipated that PPTs will find uses for ACS and for modest energy delta V applications for small spacecraft, where the power and thrust limitations of PPTs are acceptable and/or desirable.

HETs and ITs represent the highest performance EP options and characteristics of mature versions of both concepts are shown in Table 5. HETs were developed and flown on dozens of Russian space missions for various functions and are under intense development for use on other nations' spacecraft. At present, flight-type HETs have been produced by Fakel Enterprise (Fakel), Keldysh Research Center (KeRC), and Ts-NIIMASH, all of Russia, and quite aggressive HET R&D programs are in-place in Europe.[16] Japan,[17] and the U.S.[15] Table 4 lists three HET concepts to illustrate state-of-the-art. The 1600 s specific impulse (*Isp*) concept was developed to flight-ready status[22] by a team including International Space Technology, Inc. (IST), Loral, and Fakel; the 1638-s *Isp* device was built, qualified, and delivered for a flight test (at reduced levels of power and specific impulse) by a NASA, Primex, Ts-NIIMASH team; and the high-power HET is being built by Space Power Inc. (SPI), KeRC, and TRW for a 1999 flight test on a Russian GEOSAT.[25] In addition, two versions of 1.5 kW-class HETs traceable to the Fakel concept are planned to provide N–S S/K for nine years on the French Stentor spacecraft,[16] which will be launched in late 1999. Details of the systems were, however, not available for inclusion in Table 4.

Five mature ion thrusters are also shown on Table 4. The 2585-s *Isp* system was built by the Hughes Aircraft Company (HAC) and is operational on a commercial COMSAT launched in 1997. The 2906-s *Isp* concept was built by a team of Mitsubishi Electric Corporation (MELCO) and Toshiba Corporation of Japan and was flown on the ETS-VI spacecraft in 1994. An orbit insertion issue prevented the system from performing its planned N–S S/K function but in-space characterizations were performed in 1995 and an identical system will soon be flown on the Japanese COMETS spacecraft. The 3250- and 3400-s *Isp* systems were built in Europe by teams headed by, respectively, Matra Marconi Space (MMS) and DASA. These devices have been baselined for N–S S/K on the European Space Agency's Artemis spacecraft to be launched in 2000. The 3280-s, 2.5-kW device is the highest power, mature ion thruster for which data are available, and it is planned for use[25] on NASA's New Millennium Deep Space 1 mission to multiple planetary bodies.

HETs and ITs are the highest specific impulse options available to mission planners and many analyses have been conducted to evaluate their use for high-energy mission. Comparisons of the two devices are difficult because of the relative lack of maturity of devices built to comparable powers and standards. ITs operate reliably at higher specific impulses than HETs and their performance and specific mass are deeply penalized by operation at specific impulses less than about 2500 s (because of the constraints imposed by the ion optics systems). On the other hand, HET systems perform at values of thrust/power 30% or more larger than those of ITs and are considerably lighter, but HET operations above about 2500 s will pose major lifetime, or redesign, challenges. Both concepts eject high-velocity, charged plumes and present approximately the same issues regarding spacecraft integration. Both HETs and ITs have been demonstrated to provide extreme benefits for emerging space missions and the choice between them will likely be quite mission specific. In general, however, ITs become increasingly beneficial as mission energies increase and HETs appear optimum for many, time-constrained situations, typical of Earth-space missions.

In-Space Propulsion Impacts

Advanced in-space propulsion provides benefits for many space missions, and numerous studies have been performed to quantify impacts in specific situations. The following text will briefly discuss the leverage of advanced propulsion for two different missions that use a variety of launch sites and vehicles.

MEO Mission

Studies were conducted to evaluate the MEO payload delivery potential of different chemical in-space propulsion concepts. A Russian Soyuz launch vehicle was assumed that was launched from Plesetsk and delivered about 5100 kg to a 200 $\times$ 1700 km parking orbit at an inclination of 51.8 deg. Payload delivery to an approximately 10,000 km altitude was calculated for several chemical insertion systems (Table 5). Mission analyses were performed that were based on detailed primary and ACS propulsion subsystem definitions and accounted for,

Table 5 MEO mission in-space propulsion trades

	In-space propulsion injection propulsion options			
	Baseline	1	2	3
Injection propulsion				
Engines	2, N_2O_4/N_2H_4	2, N_2O_4/N_2H_4	2, N_2O_4/N_2H_4	8, SCATs
Thrust/engine, N	~450	~450	~450	~50
Specific impulse, s	314.5	318	328	326
ACS	SCATs	N_2H_4	SCATs	SCATs
Injected mass benefit over baseline case, kg	—	21	30	96

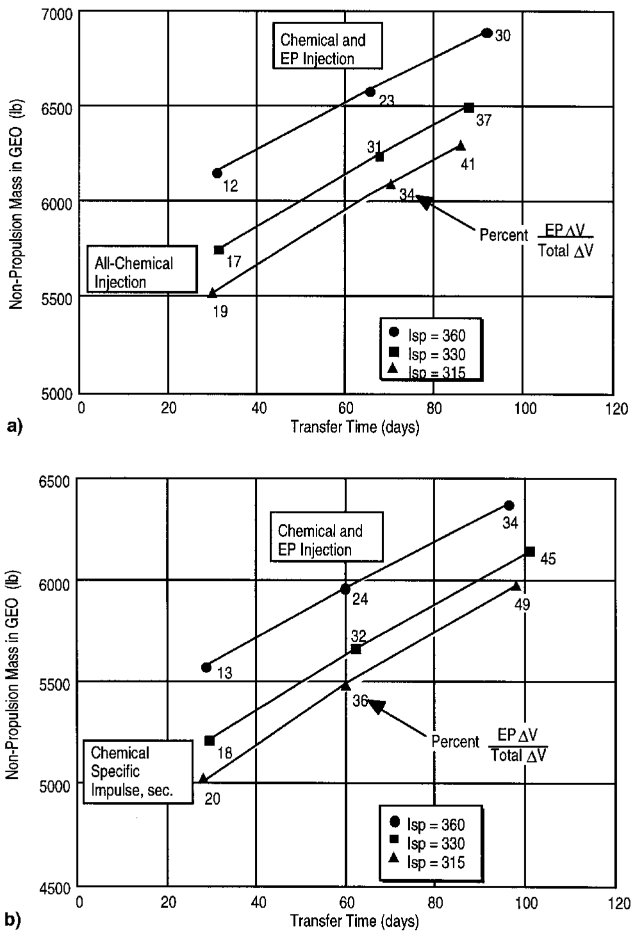

Fig. 6 GEO mission payloads: a) long march, and b) Atlas IIARS.

among other things, three sigma variations in launch vehicle dispersions, thrust vector uncertainties, and c.g. uncertainties. Table 5 shows the baseline insertion propulsion systems that assumed two N_2O_4/N_2H_4 450N-class rockets that operated at a specific impulse of 314.5 s. The additional mass delivered to the 10,000-km orbit is shown for the three advanced in-space propulsion cases. It is seen that in this case the use of SCATs for both delta V and ACS provided the maximum payload increment of about 75 kg. In this case, the ability to use the SCATs for ACS as well as delta V overcame the higher specific impulse of the 328-s engine. (It should be noted that complete trades must include features beyond those of mass, including reliability, degradation, and cost issues as appropriate.)

GEO Missions

GEO missions offer opportunities to combine advanced chemical and electric in-space propulsion for significant benefits. Figure 6 shows GEO payloads, for missions launched from the U.S. and China (with Atlas and Long March launchers, respectively), as a function of the insertion time required for the apogee propulsion maneuver. The analyses follow the approach used in Ref. 29. The short insertion times reflect the use of all-chemical apogee propulsion and the values of specific impulse of the chemical engines reflect state-of-the-art and projected values for Earth-storable and high-energy (LO_2/fuel) propellants. It is seen that the use of improved chemical in-space propulsion can result in about 15% more GEO payload mass. As discussed in Ref. 29, redesign of the mission to use of electric propulsion for the final phase of apogee injection can result in significant additional benefits in modest times. In Fig. 6 a Hall-effect thruster, which operated at about 10 kW, was assumed. Importantly, the power for the electric apogee injection phase is within that available for many GEO spacecraft and suffers little degradation when used in this fashion. It is seen that the use of electric propulsion can result in an additional 10–15% GEO payload benefit for insertion times less than about two months. The combination of advanced electric and chemical in-space propulsion provides major mission mass benefits that can be leveraged in a variety of ways in different situations.

Conclusions

In-space propulsion represents a major mass penalty for the majority of space missions. Use of available mission projections indicates that in-space propulsion will represent between 40 and 50% of all mass delivered by launchers and upper stages in the next 10 years (neglecting manned mission projects). Several advanced chemical and electric concepts are being developed that allow users to meet mission requirements and significantly reduce the mass penalties now attendant to in-space propulsion. These advanced concepts offer combinations of higher performance; and/or simpler (reduced cost) systems; and/or operations benefits via the use of benign propellants. Analyses indicate that significant benefits are available for a broad set of international missions of a variety of classes.

Acknowledgment

Presented as Paper 2.1 at the 3rd International Symposium on Space Propulsion, Beijing, People's Republic of China, August 11–13, 1997.

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