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Ion Propulsion Development Projects in U.S.: Space Electric Rocket Test I to Deep Space 1

James S. Sovey,* Vincent K. Rawlin,* and Michael J. Patterson*
NASA John H. Glenn Research Center at Lewis Field, Cleveland, Ohio 44135

The historical background and characteristics of the experimental flights of ion propulsion systems and the major ground-based technology demonstrations are reviewed. The results of the first successful ion engine flight in 1964, Space Electric Rocket Test (SERT) I, which demonstrated ion beam neutralization, are discussed along with the extended operation of SERT II starting in 1970. These results together with the technologies employed on the early cesium engine flights, the applications technology satellite series, and the ground-test demonstrations, have provided the evolutionary path for the development of xenon ion thruster component technologies, control systems, and power circuit implementations. In the 1997–1999 period, the communication satellite flights using ion engine systems and the Deep Space 1 flight confirmed that these auxiliary and primary propulsion systems have advanced to a high level of flight readiness.

Introduction

KILOWATT-CLASS ion propulsion systems have found applications for spacecraft (S/C) north–south station keeping (NSSK), orbit insertion, and primary propulsion for deep space missions.[1,2] The ion engine operates at a specific impulse about eight times that of chemical thrusters, which are commonly used on communication satellites. The higher specific impulse operation saves enough propellant mass, vs chemical systems, to nearly double the transponder hardware on a communication satellite.[3] The electron-bombardment ion thruster development in the United States has evolved from the first laboratory tests of a 10-cm engine[4] to the first operational flights in 1997/1998.[2,5] Much of the early development of mercury ion engines is outlined in Refs. 6 and 7. Significant component improvements to the mercury, and then xenon, ion engines have taken place over the last 40 years. A roadmap of the component technology development is shown in Fig. 1. In the early 1960s, the wire grids were replaced by multiaperture grids.[8] Later in the mid-1960s, engine life extension was made possible by the incorporation of hollow cathodes for the neutralizer and main discharge.[9–11] The Space Electric Rocket Test (SERT) II flight was the major in-space demonstration of these technologies.[12] Major technology improvements in the 1970s were the development of high-perveance, dished grids,[13] methods to control spalling of sputter-deposited material in the discharge chamber,[14] and methods to provide deep-power throttling.[7] Mercury engines were developed with diameters ranging from 5 to 150 cm. A schematic of a divergent magnetic field ion engine is shown in Fig. 2. Endurance tests of these engines ranged up to 15,000 h to satisfy potential NSSK or primary propulsion requirements.

In the 1980 time frame, it was decided to replace the mercury propellant with xenon because xenon was less contaminating to spacecraft surfaces and ground-test operations were greatly simplified. In the 1980s and 1990s ring-cusp discharge chambers[15–17] were used instead of divergent-field chambers whose pole pieces, in the vicinity of the discharge chamber cathode, suffered severe ion erosion. The ring-cusp chamber, shown in Fig. 3, does not require pole pieces in the vicinity of the hollow cathode, and the boundary magnetic field device reduces the ion losses to the chamber walls.[18] Additionally, long-life, xenon hollow-cathode technology was enhanced by developments in the Space Station plasma contactor program, which focused on defining reliable processing, handling, and test procedures for the cathodes.[19] Ground tests of 13- and 30-cm-diam xenon engines demonstrated more than 8000 h of reliable operation.[5,20] The communication satellite and deep space operation of these engines, starting in 1997, confirmed the thrusters and power processing units (PPUs) are very mature technologies.

This paper focuses on gridded-ion engine development projects in the United States. Note that over the last three decades, very strong ion propulsion research and development programs have also been conducted in Japan and Europe.[21–23] In fact, Japan has flown an experimental ion propulsion system (IPS) in 1982 [Engineering Test Satellite (ETS-3)] and operational flights of IPS in 1994 (ETS-6) and 1998 Communications and Broadcasting Engineering Test Satellite (COMETS).[21] Additionally, this survey of ion propulsion development work does not include Hall Effect Thruster (HET) projects. The development of the HET, a nongridded-ion accelerator, has been pursued in many countries. In the HET, the xenon gas is ionized and accelerated in an electric discharge with crossed electric and magnetic fields. The HET is generally regarded as having a lower specific impulse but a higher thrust density than gridded-ion engines. The HET was developed by researchers in the former Soviet Union,[24] and the technology has been further developed in many other countries.[25]

Surveys of the history of electric propulsion systems have cataloged the evolution of IPS technology and generally described many of the experimental and operational flights.[23,25–27] The purpose of this paper is to provide more detail related to the IPS flights and major ground demonstrations of the technology. Background on system performance and in-space operation will be summarized, and the evolution of electron-bombardment ion thruster development in the United States will be discussed.

Experimental Flights of IPSs

The experimental flights of IPSs developed in the United States are summarized in Table 1. Some of the results indicated in Table 1 are expanded, and major results are described. Although there were major ground-test and development programs associated with each of the experimental flights, nearly all of the synopsized results reported here are associated with the endproduct, which is the flight test.

Program 661A, Test Code A

In November of 1961, Electro-Optical Systems (EOS) was awarded a contract by the U.S. Air Force to develop a 8.9-mN, cesium-contact ionization IPS for three suborbital flight tests. The

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*Aerospace Engineer, Power and On-Board Propulsion Technology Division, 21000 Brookpark Road. Member AIAA.

Table 1 Experimental flights of ion propulsion systems

	S/C										
Characteristic	Program 661A, test code A	SERT I		Program 661A, test code B	Program 661A, test code C	SNAPSHOT	ATS-4	ATS-5	SERT II	ATS-6	SCATHA P78-2
Sponsor	USAF	GRC		USAF	USAF	USAF	USAF/GSFC[a]	USAF/GSFC	GRC	GSFC	USAF/GSFC
Builder of IPS	EOS	GRC	Hughes	EOS	EOS	EOS	EOS	EOS	GRC, Westinghouse	EOS	Hughes (ion source)
Launch date	18 Dec. 1962	20 July 1964		29 Aug. 1964	21 Dec. 1964	3 April 1965	10 Aug. 1968	12 Aug. 1969	3 Feb. 1970	30 May 1974	30 Jan. 1979
Orbit, km	Suborbital	Suborbital		Suborbital	Suborbital	700	218 × 760	36,000	1,000	36,000	43,000 × 27,000
IPS type	Contact ionization	Electron bombardment	Contact ionization	Contact ionization	Contact ionization	Contact ionization	Contact ionization	Contact ionization	Electron bombardment	Electron bombardment	Electron bombardment
Propellant	Cesium	Mercury	Cesium	Cesium	Cesium	Cesium	Cesium	Cesium	Mercury	Cesium	Xenon
No. of thrusters	1	1	1	1	1	1	2	2	2	2	1
Thruster anode diameter, cm	~7	10	8	~7	~7	~5	5	5	15	8	3.6
Type of neutralizer	Wire filament	Ta wire	Ta wire	Wire filament	Wire filament in beam	Wire filament barium coated	Ta doped with Yttrium	Ta doped with Yttrium	Hollow cathode	Cesiated Ta	Ta doped with Yttrium
Beam power supply voltage, V	5,000	2,500	4,500	5,000	5,000	4,500	3,000	3,000	3,000	560	1,000–2,000
Power per thruster, kW	0.77	1.4	0.6	0.77	0.77	~0.4	0.02	0.02	0.85	0.15	0.03–0.045
Maximum thrust requirement, mN	8.9	28	5.6	8.9	8.9	~8.5	0.089	0.089	28	4.5	0.14
Specific impulse, s	7,400	4,900	8,050	7,400	7,400	5,100	6,700	6,700	4,200	2,500	350
Propellant mass, g	2			2	2		~0.05	~0.05	15	3.6	0.3
Maximum in-space operation time for one thruster	0 min	31 min	0 min	~19 min	~4 min	< 60 min	~10 h	No operation with a HV beam	~3,781 h	92 h	
Longest ground test, h	1,230						2,245		6,742, 5,169	2,614, 471 cycles	~600

[a]NASA Goddard Spaceflight Center.

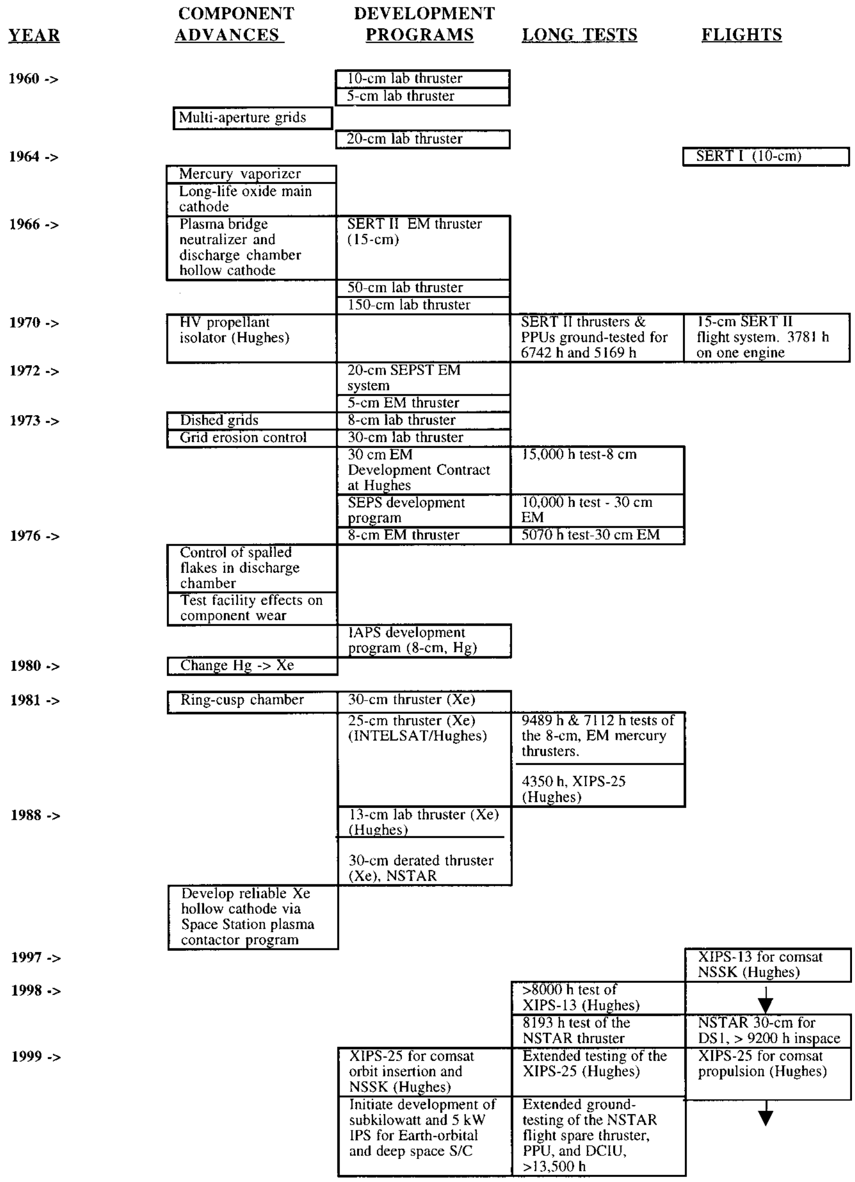

Fig. 1 History of electron-bombardment ion thruster development in the U.S. (all projects were NASA sponsored unless noted otherwise).

electric propulsion space tests were called Program 661A and were managed by the Air Force Space Systems Command in Los Angeles.[28–30] The flight objectives were to demonstrate in-space operation of the cesium ion engine and to obtain accurate measurements of engine performance.

The cesium contact engine incorporated an ionizer array of 84 porous tungsten buttons. The power level, thrust, and specific impulse were 0.77 kW, 8.9 mN, and 7400 s, respectively, in this engine, which had a beam extraction diameter of about 7 cm. The neutralizer was a wire filament, which was not immersed in the ion beam. Power to the PPU was supplied by 56-V batteries. The longest ground test was 1230 h.

The first suborbital flight test was launched on 18 December 1962. When the high-voltage power supplies were first turned on, intermittent high-voltage breakdowns occurred, and the beam power supply became inoperative. Postflight analysis indicated the high-voltage breakdowns were probably caused by pressure buildup in the PPU due to gas vented from the spacecraft batteries. The PPU high-voltage section was not adequately vented to keep the pressure low enough. Engine thrusting was not accomplished in this test.

SERT I

The SERT I spacecraft was launched 20 July 1964 using a Scout launch vehicle.[31,32] This flight experiment had a 8-cm-diam cesium contact ion engine and a 10-cm-diam mercury electron bombardment ion engine and was the first successful flight test of ion propulsion. The cesium engine was designed to operate at 0.6 kW and provide 5.6 mN of thrust and a specific impulse of 8050 s. The cesium flow was controlled by a boiler and the porous tungsten ionizer electrode. The mercury ion engine provided flow control via a boiler and a porous stainless steel plug. A hot tantalum wire was used as the discharge cathode. Beam and accelerator power supply voltages were 2500 and 2000 V, respectively. The engine had a 1.4 kW power level with 28 mN of thrust at a specific impulse of about 4900 s. Each of the ion engines had a heated tantalum filament neutralizer.

The early part of the flight was dedicated to attempts to operate the cesium engine. The cesium engine could not be started because of a high-voltage (HV) electrical short circuit. The mercury engine was started about 14 min into the flight. The IPS was successfully operated for 31 min with 53 HV recycle events, which were handled by the PPU fault protection system. Each of the recycle events was only a few seconds duration. Major results from the test were the first demonstration of an IPS in space, effective ion beam neutralization, no electromagnetic interference (EMI) effects on other spacecraft systems, and effective recovery from HV electrical breakdowns. Thrust was measured or calculated using three independent measuring methods. In-space thrust, determined by both accelerometer and sun sensor data, agreed with the calculated thrust within 5%. The thrust was calculated from the beam current, beam voltage, doubly charged ion correction, and the beam-divergence correction.

Program 661A, Test Code B

Test code B was the second in the series of three suborbital flight tests of the EOS's 8.9-mN, cesium ion engine systems.[28,33] A Scout vehicle launched the payload on 29 August 1964. The launch was designed to provide about 30 min above an altitude of 370 km. After 7 min into the flight, the engine was operated with ion beam extraction. Full beam current of 94 mA was achieved about 10 min later. During the course of engine operation, an electric field strength meter was used to infer payload floating potential relative to space. Spacecraft potential was about 1000 V negative during most of the engine operation with the filament neutralizer. The absolute value of payload potential was about 10 times higher than anticipated, and it is suspected that there was inadequate neutralization of the ion beam. The contact ion engine operated for approximately 19 min until spacecraft reentry into the atmosphere.

In addition to withstanding the environmental rigors of space flight, the IPS demonstrated electromagnetic compatibility with other spacecraft subsystems and the ability to regulate and control a desired thrust level.

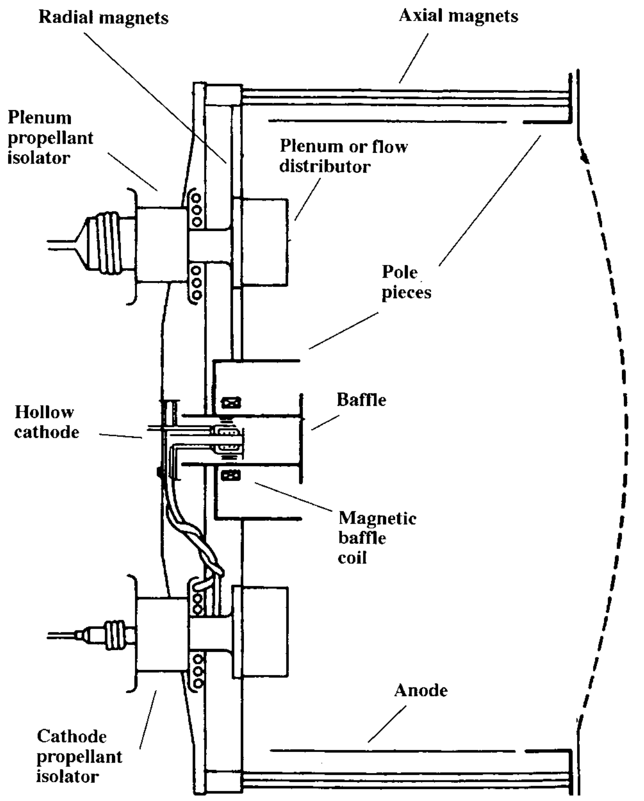

Fig. 2 Ion engine having a divergent-magnetic field discharge chamber.

Program 661A, Test Code C

The third and final IPS payload of the Air Force's program 661A was launched on 21 December 1964.[28,33] In this test, an additional wire neutralizer was incorporated and was immersed in the ion beam to provide a higher probability of adequate neutralization. The contact ion engine only achieved about 20% of full thrust before reentry into the atmosphere. The short test time was due to a very short burn of the Scout vehicle's third stage. The high voltage was applied to the engine 7 min into the flight, when the altitude was 490 km. Engine operation ended after 4 min when the altitude was only 80 km.

S/C Carrying SNAP 10A Nuclear Power System and Cesium Ion Propulsion System (SNAPSHOT)

On April 3, 1965 a Systems for Nuclear Auxiliary Power (SNAP) 10A nuclear power system was launched into a 1300-km orbit with a cesium ion engine as a secondary payload.[34–36] The ion beam power supply was operated at 4500 V and 80 mA to produce a thrust of about 8.5 mN. The neutralizer was a barium–oxide-coated wire filament. The ion engine was to be operated off batteries for about 1 h, and then the batteries were to be charged for approximately 15 h using 0.1 kW of the nominal 0.5-kW SNAP system as the power supply. The SNAP power system operated successfully for about 43 days, but the ion engine operated for a period of less than 1 h before being commanded off permanently. Analysis of flight data indicated a significant number of HV breakdowns, and this apparently caused sufficient EMI to induce false horizon sensor signals leading to severe attitude perturbations of the spacecraft. Ground tests indicated that the engine arcing produced, conducted, and radiated EMI significantly above design levels. It was concluded that low-frequency, < 1 MHz, conducted EMI caused the slewing of the spacecraft.

Applications Technology Satellite-4 (ATS-4)

Two cesium-contact ion engines were launched aboard the Applications Technology Satellite-4 (ATS-4) spacecraft on 10 August 1968. Flight-test objectives were to measure thrust and

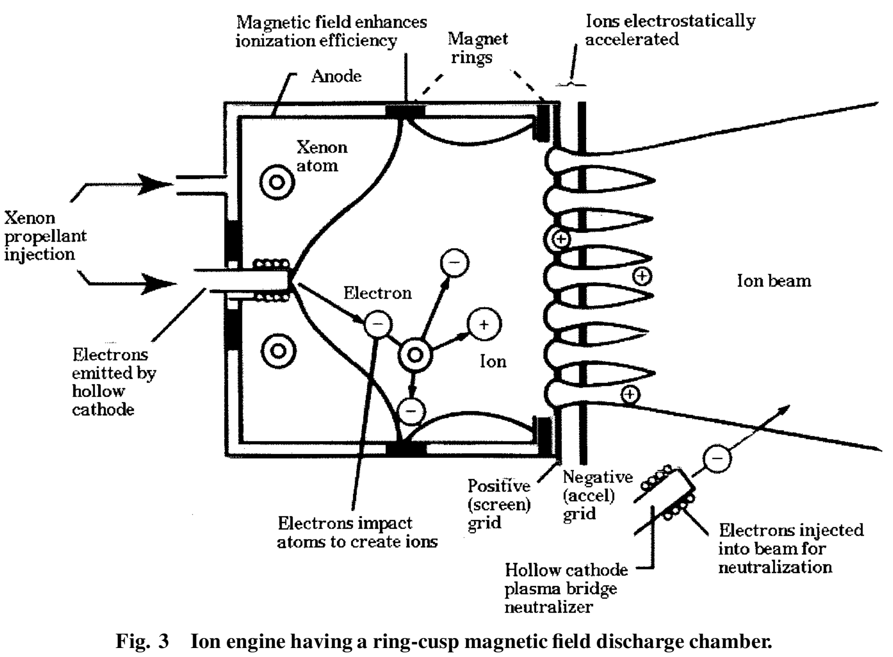

Fig. 3 Ion engine having a ring-cusp magnetic field discharge chamber.

to examine electromagnetic compatibility with other spacecraft subsystems.[26,37,38] The 5-cm-diam thrusters were designed to operate at 0.02 kW and provide about 89-μN thrust at about 6700-s specific impulse. Thrusters had the capability to operate at five setpoints from 18 to 89 μN. Thrusters were configured so they could be used for east–west stationkeeping (EWSK). Before launch, a 5-cm cesium thruster was life tested for 2245 h at the 67-μN thrust level.[39]

During the launch process, the Centaur stage did not achieve a second burn, and the spacecraft remained attached to the Centaur in a 218 × 760 km orbit. It was estimated that the pressure at these altitudes was between 1.3×10^{-4} and 1.3×10^{-7} Pa (Ref. 35). Each of the two engines was tested on at least two occasions over the throttling range. Combined test time of the two engines was about 10 h over a 55-day period. The spacecraft reentered the atmosphere on 17 October 1968.

The ATS-4 flight was the first successful orbital test of an ion engine. There was no evidence of IPS EMI related to spacecraft subsystems. Measured values of neutralizer emission current were much less than the ion beam current implying inadequate neutralization. The spacecraft potential was about −132 V, which was much different than the anticipated value of about −40 V (Ref. 37).

ATS-5

A flight IPS, identical to the one flown on ATS-4, was launched on ATS-5 on 12 August 1969. The purpose of this flight was to demonstrate NSSK of a geosynchronous satellite.[40,41] Once in geosynchronous orbit, the spacecraft could not be despun as planned, and thus the spacecraft gravity-gradient stabilization could not be implemented. The spacecraft spin rate was about 76 rpm, and this caused an effective 4-g acceleration on the cesium feed system. The high-g loading on the cesium feed system caused flooding of the discharge chamber, and normal operation of the thruster with ion beam extraction could not be performed. The IPS was able to be operated as a neutral plasma source, without HV ion extraction, along with the wire neutralizer to examine spacecraft charging effects. The neutralizer was also operated by itself to provide electron injection for the spacecraft charging experiments.

SERT II

The SERT II development program, which started in 1966, included thruster ground tests of 6742- and 5169-h duration. A prototype version of the SERT II spacecraft was ground tested for a period of 2400 h with an operating ion engine. The spacecraft was launched into a 1000-km-high polar orbit on 3 February 1970.[12] In addition to diagnostic equipment and related IPS hardware, the spacecraft had two identical 15-cm-diam, mercury ion engines and two PPUs. The ion engine is shown in Fig. 4. Flight objectives included in-space operation for a period of 6 months, measurement of thrust, and demonstration of electromagnetic compatibility. The thruster maximum power level was 0.85 kW, and this provided operation at a 28-mN thrust level at 4200-s specific impulse. Flight data were obtained from 1970 to 1981 with an ion engine operating intermittently in one of three different modes, namely, HV ion extraction, discharge chamber operation only, or just neutralizer operation.

Major results were that two mercury engines thrusted for periods of 3781 and 2011 h. Test duration was limited due to shorts in the ion optical system. Thrust measured in space and on the ground agreed within the measurement uncertainties. Up to 300 thruster restarts were demonstrated. A PPU accumulated nearly 17,900 h during the course of the mission. Additionally, the IPS was electromagnetically compatible with all other spacecraft systems.

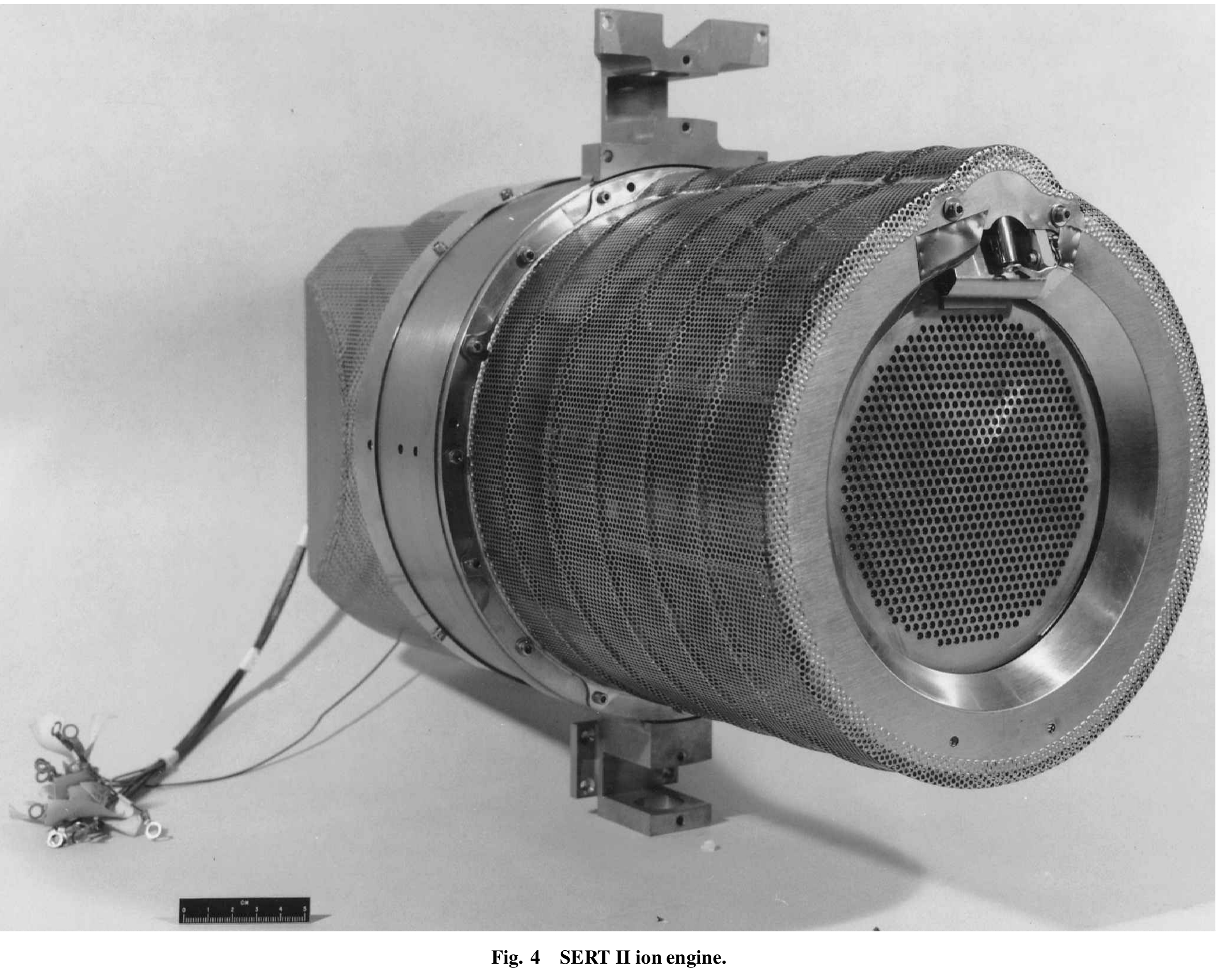

Fig. 4 SERT II ion engine.

Table 2 Major IPS ground demonstrations

Characteristic	Project name				
	SEPST	SIT-5	SEPS	IAPS	XIPS-25
Sponsor	JPL	GRC	GRC	GRC	INTELSAT
Builder of thruster	JPL	Hughes	Hughes	Hughes	Hughes
Builder of PPU	Hughes/TRW	——	TRW	Hughes	Hughes
Integrator of IPS	JPL	——	GRC	Hughes	Hughes
Project duration	1968–1972	1969–1972	1972–1980	1974–1983	1985–1988
Propellant	Mercury	Mercury	Mercury	Mercury	Xenon
Thruster diameter, cm	20	5	30	8	25
Type of neutralizer	Hollow cathode	Hollow cathode	Hollow cathode	Hollow cathode	Hollow cathode
Beam power supply voltage, V	2,000	1,600	1,100	1,200	750
Power per thruster, kW	2.5	0.072	2.6	0.13	1.3
Maximum thrust, mN	88	2.1	128	5.1	63
Specific impulse, s	3,600	3,000	3,000	2,500	2,800
Longest ground test, h	1,300	9,715	10,000	15,040, 9,489, 7,112	4,350, 3,850 cycles

ATS-6

The purpose of the ATS-6 flight experiment was to demonstrate NSSK of a geosynchronous satellite using two electron-bombardment ion engine systems with cesium propellant.[40,41–43] Thruster development tests included a lifetest of 2614 h and 471 cycles. Thruster input power was 0.15 kW, which resulted in a thrust of 4.5 mN at a specific impulse of 2500 s. The ATS-6 was launched on 30 May 1974. One of the ion engines operated for about 1 h and the other for 92 h. Both of the engines failed to provide thrust on the restarts due to discharge-chamber cesium flooding. The feed system flooding problem caused overloading of the discharge and HV power supplies. This failure mechanism was verified through a series of ground tests.[43]

The IPS operation demonstrated an absence of EMI related to spacecraft systems, verified predictions of spacecraft (S/C) potential with engines operating, and demonstrated compatibility with the S/C star tracker. It was found that the ion engines or just the neutralizer could discharge large negative spacecraft potentials at all times. Furthermore, tests indicated that "differential charging was reduced by the neutralizer when operated in spot mode and eliminated by operation of the ion engine."[41]

S/C Charging at High Altitude (SCATHA), P78-2

The S/C Charging at High Altitude (SCATHA) had two charged-particle injection systems, one of which was the Satellite Positive-Ion-Beam System (SPIBS).[44,45] This was a xenon ion source, which included some of the technologies used in thrusters; however, the small discharge chamber was not performance optimized as was done with ion engines. Maximum operating power was 0.045 kW, and the ion source could produce a thrust of about 0.14 mN at a specific impulse of 350 s. Ions could be ejected at about 30 eV with only the ion source discharge operating. With HV applied to the ion extraction system, 1-keV or 2-keV ions could be extracted. Neutralization was accomplished by a tantalum filament. The specific impulse was low because there was no attempt to optimize the propellant efficiency. The SPIBS system was ground tested for a period of 600 h. The SCATHA was launched 30 January 1979 and placed in a near geosynchronous orbit. Ion beam operations were performed intermittently over a 247-day period.

The SCATHA flight demonstrated that "a charged spacecraft, and the dielectric surfaces on it, could be safely discharged by emitting a very low energy (<50 eV) neutral plasma—in effect 'shorting' the spacecraft to the ambient plasma before dangerous charging levels could be reached."[46] The SPIBS ion source discharged the SCATHA from a potential of −3000 V using as little as 6 μA of ion beam current.

Major Ground-Based Demonstrations of IPS

Table 2 contains brief descriptions of the major electron-bombardment ion propulsion ground-test demonstrations in the United States. The projects described in this section involve IPSs that were never flown. Only those systems that included a structurally integrated thruster or an engineering model class thruster and an advanced PPU are described here.

Solar Electric Propulsion System Technology (SEPST)

The objective of the Solar Electric Propulsion System Technology (SEPST) program at the Jet Propulsion Laboratory (JPL) was to demonstrate a complete breadboard IPS that would be applicable to an interplanetary spacecraft.[47,48] The focus of this program was directed toward thruster performance improvements, PPU and control technology, and power matching and switching. Most of the program efforts were conducted in the late 1960s and early 1970s. The 20-cm-diam mercury ion engine first employed a thermally heated oxide cathode and later on used a hollow cathode. Maximum thruster power was 2.5 kW, which enabled thrusting at 88 mN and a specific impulse of about 3600 s. Three basic servoloops were demonstrated, and they were similar in concept to the two loops used in the SERT II technology. Servoloops included an ion beam current to main vaporizer loop, a discharge voltage to cathode vaporizer loop, and a neutralizer keeper voltage to neutralizer vaporizer loop. The closed loops, to first order, maintained the thrust level, the propellant efficiency, and the floating potential from neutralizer common to facility or S/C ground.

PPU development centered around the beam power supply. The beam power supply had eight inverters and had an efficiency of 89–90% over a bus voltage range from about 53 to 80 V (Ref. 48). The PPU was integrated with the thruster, 2:1 power throttling with closed-loop control was demonstrated, and HV recycle algorithms were developed. Initial breadboard power processing unit (BBPPU) efficiencies were about 84–86%, and subsequent experimental BBPPUs had efficiencies of 88–90%. The experimental BBPPUs, which provided 2.5 kW, had a specific mass of 5.4 kg/kW. Later work at NASA John H. Glenn Research Center at Lewis Field (GRC) in the 1970s focused on the development of 30-cm-diam ion engine, which operated at derated power levels compared to the SEPST engine. The 30-cm-diam thruster system, using mercury propellant, was brought to engineering model status under the solar electric propulsion system (SEPS) program, which is described in a subsequent section.

Structurally Integrated Thruster-5 (SIT-5)

A 5-cm-diam mercury ion engine, Structurally Integrated Thruster-5 (SIT-5), was developed around 1970 for attitude control and NSSK of geosynchronous satellites.[49–51] The thruster input power was 0.072 kW, and it provided a thrust of 2.1 mN at a specific impulse of 3000 s. Electrostatic thrust vectoring grids with a ±10-deg vectoring capability were baselined. The engine was successfully random vibration tested at 19.9-g rms. The dry mass of the thruster and mercury storage and feed system was 2.2 kg.

The propellant system could store 6.8 kg of mercury, which could provide operation at full power for approximately 30,000 h. The envelope was about 31 cm long × 12 cm diam. The SIT-5 development program focused on the thruster and feed system development; there was no PPU technology effort.

Hollow-cathode component tests demonstrated over 2800 simulated duty cycles. A separate test of the SIT-5 thruster was conducted for 9715 h at a beam voltage of 1300 V, a thrust of 1.8 mN, and a specific impulse of 2500 s (Refs. 52 and 53). During the initial 2023 h, the thruster was operated with a translating screen grid thrust vector system. For the remainder of the test, the thruster had an electrostatic thrust vector system. The electrostatic beam vector grids were operated at 5-deg deflection for about 120 h, at either 2- or 4-deg deflection for 1880 h, and with no deflection for 5690 h. There were a number of grid shorts that were successfully cleared by the application of 200–400 V at currents from 6–70 mA. These tests were helpful in the later definition of grid-clear circuits for the Ion Auxiliary Propulsion System (IAPS), Xenon Ion Propulsion System (XIPS), and NASA Solar Electric Propulsion Technology Applications Readiness (NSTAR) thrusters.

The SIT-5 mercury propellant system was successfully tested for a period of 5400 h in an independent test.

SEPS

The SEPS program was started in the early 1970s with a goal to provide a primary IPS capable of operating at a fixed power for Earth orbital applications or over a wide power profile such as would be encountered in planetary missions. One of the potential planetary targets was an encounter with the comet Enke.[54,55] The SEPS program included the development of 25-kW solar arrays, PPUs, thermal control systems, gimbals, throttleable/long-life 30-cm-diam ion thrusters, and mercury propellant storage and distributions systems. This multicenter, multicontractor effort was ongoing for about 10 years with a NASA investment of approximately $30 million. Because of funding limitations, a planetary flight program was not carried out; rather, a ground-based technology demonstration was pursued.

The thrust subsystem was a bimodule consisting of two thrusters, two PPUs, a propellant system, a gimbal system, thermal control, and supporting structure.[56,57] This module would be a basic building block of a electric stage with simple interfaces. The 30-cm thruster was designed for 2.6-kW input power with 128-mN thrust and a specific impulse of about 3000 s (Refs. 7 and 57). The thruster/PPU was capable of throttling down to 1.1 kW. More detailed references related to the development and test of the SEPS bimodule hardware may be found in Ref. 55.

One of the early engineering model thrusters was tested for 10,000 h over an input power range of 0.8–2.4 kW (Ref. 58). Endurance tests of these 30-cm ion engines confirmed the need for spalling control of sputter-deposited discharge chamber coatings[14,58] and for the need of low sputter-yield materials for the cladding of pole pieces and baffles.[59] Other tests indicated that very small concentrations of nitrogen in the vacuum facility could significantly reduce wear on the upstream surface of the screen grid compared to that expected in space.[60]

Subsequent to these engineering model (EM) thruster tests, a total of seven advanced EM thrusters were tested in segments, including two at 3940 and 5070 h long, with a total test time of 14,541 h (Ref. 59). Either breadboard or brassboard PPUs of the series-resonant inverter design[59,61] were used in 95% of the tests.

IAPS

The IAPS project and other preflight technology work took place in the 1974–1983 time frame.[62] Flight-test objectives were to verify in space the thrust duration, cycling, and dual-thruster operations required for stationkeeping, drag makeup, station change, and attitude control. This implied demonstration of overall thrusting times of 7000 h and 2500 on/off cycles. The 8-cm-diam, mercury ion engine input power was 0.13 kW, and the thrust was 5.1 mN at a specific impulse of 2500 s. The masses of the flight thruster-gimbal-beamshield unit, the PPU, and the digital controller were 3.77, 6.85, and 4.31 kg, respectively.[63] The system stored 8.63 kg of mercury, and the propellant storage and feed system weighed 1.56 kg. The IAPS successfully completed all flight qualification tests and was installed on an U.S. Air Force technology satellite.[64] The flight of the Teal Ruby spacecraft was canceled by the U.S. Air Force (USAF) due to lack of funding.

During the course of the technology and preflight programs, there were a number of endurance tests performed. A laboratory-type 8-cm engine was tested for 15,040 h and 460 cycles at the 0.14 kW level.[65] An engineering model IAPS engine and PPU were successfully tested for 9489 h and 652 cycles.[66] The thruster and PPU were located in the same vacuum chamber during this test. A third endurance test was conducted using another engineering model thruster and PPU. This hardware was operated at full thrust for 7112 h and had 2571 restarts.[67] No major changes in thruster performance and no life-limiting degradation effects were observed in this test.

XIPS-25 (1.3 Kilowatt)

This Xenon Ion Propulsion System (XIPS-25) program developed thrusters, BBPPUs, and a feed system pressure regulator for possible NSSK of 2500-kg class communication satellites.[68] The 25-cm-diam, three-grid, xenon ion engine input power was 1.3 kW with a thrust level of 63 mN and a specific impulse of 2800 s. Three versions of the thruster were developed, namely, a laboratory type, an advanced development model, and an engineering model. Performance tests indicated that the later models inherited virtually identical performance. A BBPPU with greatly reduced parts count, over SEPS designs, was built and tested. Overall PPU efficiency was 90%, and the flight-packaged specific mass was estimated to be 8 kg/kW. A 15-month wear test was conducted using the laboratory model thruster, a BBPPU, and a flight-type regulator. The hardware successfully completed 4350 h of testing and 3850 cycles, which is equivalent to about 10 years of NSSK. Instead of using the 1.3-kW XIPS-25 system, the Hughes Space and Communications Company subsequently pursued development of XIPS-13 (0.44 kW) for NSSK and the XIPS-25 (4.2 kW) for combined orbit insertion and NSSK applications, which are described in the following section.

Operational Flights of IPSs

In 1997/1998, a new era of ion propulsion for S/C began with the deployment of communication satellites using an IPS with 0.44-kW thrusters for auxiliary propulsion and a deep space mission using a 2.3-kW thruster for primary propulsion. These were the first operational uses of IPS by United States industry and government.

Communication Satellites

XIPS-13

As shown in Table 3, the Hughes Space and Communications Company has launched 10 operational communications satellites each employing four 0.44-kW xenon ion thrusters for NSSK.[3,5,69] The high specific impulse IPS reduces the propellant requirements, vs chemical systems, by 300–400 kg, thus allowing incorporation of more communications hardware aboard the spacecraft or reduction in launch vehicle size and cost. The IPS consists of two fully redundant strings each consisting of two thrusters and one PPU. Two daily burns of 5 h each are generally required for the NSSK function. Typical S/C lifetime is about 15 years.

Approximate masses for a thruster and PPU are 5.0 and 6.8 kg, respectively.[70] Overall IPS dry mass for the spacecraft is about 68 kg. The PPU contains seven power modules for the beam, accelerator, discharge, two keepers discharges, and two heaters. Overall PPU efficiency of a BBPPU was 88%.

PanAmSat Corporation (PAS) was the first customer for the XIPS-13 propulsion system. The PAS-5 was the first successful, operational spacecraft employing IPS and was launched 27 August 1997 from Kazakhstan on a Russian Proton rocket. On 28 July 2000, the 10th S/C using the XIPS-13 was launched on a Sea Launch rocket.

Table 3 Operational flights of IPSs

Characteristic	S/C HS 601[a]	S/C DS1-NASA	S/C HS 702[b]
Builder of IPS	Hughes	Hughes	Hughes
Launch dates	27 Aug. 1997–28 July 2000	24 Oct. 1998	21 Dec. 1999 and 21 Nov. 2000
Orbit, km	36,000	Orbits sun	36,000
IPS type/propellant	Electron bombardment/xenon	Electron bombardment/xenon	Electron bombardment/xenon
No. of thrusters	4	1	4
Thruster diameter, cm	13	30	25
Beam power supply voltage, V	750	650–1,100	1,200
Power per thruster, kW	0.44	0.50–2.3	4.5 maximum
Maximum thrust, mN	18	92	165
Specific impulse, s	2,590	1,900–3,100	3,800
Propellant mass, kg	>100	82	——
Maximum in-space operation time for one thruster	——	9,241 h as of 17 Feb. 2001	——
Longest ground test, h	>8,000	8,193	——

[a]HS 601 S/C: PAS-5, Galaxy VIIIi, satellite built by Hughes (ASTRA) 2A–Societe Europeenne des Satellites of Luxembourg (SES), SATMEX 5/Satmex of Mexico Co., PAS 6B, ASTRA 1H-SES, DIRECTV 1R-DIRECTV, Galaxy XR, Galaxy IVR, PAS 9.
[b]HS702 S/C: Galaxy XI, PAS 1R, ANIK F1-Telesat Canada.

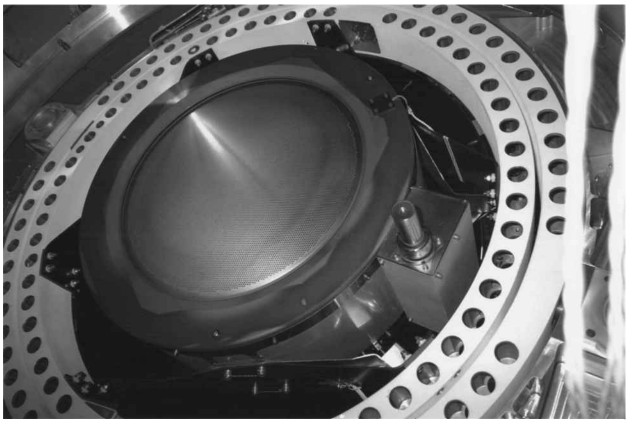

Fig. 5 DS1 ion engine mounted on a gimbal.

XIPS-25

A 25-cm-diam xenon engine system has been developed for NSSK, EWSK, attitude control, and momentum dumping for the Hughes S/C HS 702.[3,5,69] Each thruster has an maximum input power of 4.2 kW and provides up to 165-mN thrust at 3800 s specific impulse. The ion thrusters provide stationkeeping at a cost of only 5 kg/year. Additionally, the IPS is capable of boosting the communication satellite's 14,500-km perigee of the initial elliptical orbit to a circular geosynchronous orbit. Chemical propellant savings could be as much as 450 kg. The HS 702 spacecraft uses four XIPS-25 engines and two PPUs. Only two of the four thrusters are required to perform the stationkeeping and momentum control functions. The XIPS-25s were launched aboard the Galaxy XI spacecraft on 21 December 1999, the PAS-1R spacecraft on 15 November 2000, and the ANIK F1 S/C on 21 November 2000. These S/C have an end-of-life solar array power capability of about 15 kW.

Deep Space 1

The NSTAR program provided a single string, primary IPS to the Deep Space 1 (DS1) spacecraft.[2] The 30-cm ion thruster, shown in Fig. 5, operates over a 0.5–2.3 kW input power range providing thrust from 19 to 92 mN. The specific impulse ranges from 1900 s at 0.5 kW to 3100 s at 2.3 kW. The flight thruster and PPU design requirements were derived with the aid of about 50 development tests and a series of wear tests at NASA GRC and JPL of 2000, 1000, and 8193 h using engineering model thrusters.[2,20] The flight-set masses for the thruster, PPU, and digital control and interface unit (DCIU) were 8.2, 14.77, and 2.51 kg, respectively (H. G. Gronroos, NSTAR Project Office at JPL, private communication, May 1998). About 1.7-kg mass was added to the PPU top plate to satisfy the DS1 micrometeoroid requirements. The power cable between the thruster and PPU comprised two segments that were connected at a field junction. The thruster cable mass was 0.95 kg, and the PPU cable mass was 0.77 kg. The xenon storage and feed system dry mass was about 20.5 kg. A total of 82 kg of xenon was loaded for the flight. Thrusters and PPUs were manufactured for NASA GRC by Hughes Electronics, and the DCIU was built by Spectrum Astro, Inc. The feed system development was a collaborative effort between JPL and Moog, Inc.[71]

The DS1 spacecraft was launched on 24 October 1998. In-space testing and the IPS technology demonstrations were completed within the next three months.[72] By 27 April 1999, the primary thrusting of the NSTAR engine system, required to encounter the asteroid Braille, was completed. The thrusting time at the end of April was 1764 h. Thruster input power levels were varied from 0.48 to 1.94 kW. On 26 July 1999 DS1 obtained spectrometer data and images of Braille 15 min after the flyby.

The DS1 mission was extended to continue a thrusting profile until the encounter with the comet Borrelly in September 2001. By 17 February 2001, the ion engine had accumulated 9,241 h of thrusting. The NSTAR ion engine has already demonstrated a propellant throughput in excess of 30 kg. For comparison purposes, a SERT II ion engine expended about 9 kg of mercury. Propellant throughput is an approximate signature of total impulse capability. After the encounter with comet Borrelly, the ion engine will have operated for more than 14,000 h.

Next-Generation Ion Propulsion Technologies

Over the next decade, it is expected that there will be many communications S/C employing the XIPS-13 and XIPS-25 propulsion systems. Additionally, advanced ion propulsion is a strong candidate for many deep space missions including Comet Nucleus Sample Return, Titan Explorer, Venus Sample Return, Neptune Orbiter, Saturn Observer, Europa Lander, and Mars sample return missions.

In the next few years, new IPS technologies will be developed by NASA for higher thrust ion engines and also subkilowatt, smaller engines, both of which have application to planetary and Earth-orbital S/C. Some of the near-term work, shown in Fig. 6, involves development of titanium and carbon–carbon ion optics, which will provide significant lifetime improvements compared to the baseline molybdenum grid systems. Low-power and low-flow-rate neutralizers are also needed to improve efficiency for a wide class of thrusters that operate at low-power levels or are throttled over a wide range of input power. Design approaches and manufacturing technologies that provide reduced ion engine and PPU mass and cost are receiving significant attention to enable or enhance planetary and small-body missions using relatively small launch vehicles.

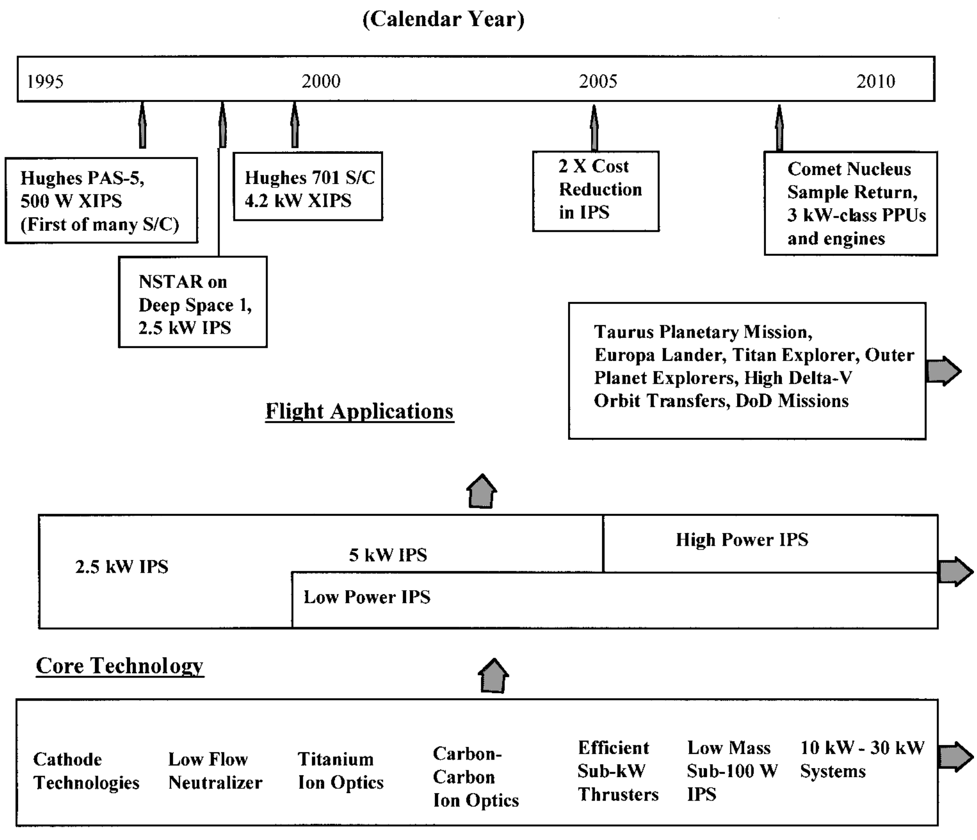

Fig. 6 Ion propulsion technology roadmap for Earth-orbital and planetary applications.

Concluding Remarks

The historical background and characteristics of the experimental flights of IPSs and the major ground-based technology demonstrations were reviewed. The results of the first successful ion engine flight in 1964, SERT I, which demonstrated ion beam neutralization, were discussed along with the extended operation of SERT II starting in 1970. These results together with the technology employed on the early cesium engine flights, the ATS series, and the ground-test demonstrations have provided the evolutionary path for the development of xenon ion thruster component technologies, control systems, and power circuit implementations. In the 1997–1999 period, the communication satellite flights using ion engine systems and the DS1 flight confirmed that these auxiliary and primary propulsion systems have advanced to a high level of flight readiness.

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