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A Synopsis of Ion Propulsion Development Projects in the United States: SERT I to Deep Space I

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Abstract

The historical background and characteristics of the experimental flights of ion propulsion systems 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, are 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 Deep Space 1 flight confirmed that these auxiliary and primary propulsion systems have advanced to a high-level of flight-readiness.

Acronyms and Abbreviations

ASTRA 2A	A communication satellite built by Hughes for SES
ATS	Applications Technology Satellite
BBPPU	Breadboard Power Processing Unit
DARPA	Defense Advanced Research Projects Agency
DCIU	Digital Control and Interface Unit
DS1	Deep Space 1
EM	Engineering Model
EMI	Electromagnetic Interference
EOS	Electro Optical Systems
EWSK	East-West Stationkeeping
g	Acceleration due to Earth's gravity, 9.807 m/s^2
GRC	Glenn Research Center
GSFC	Goddard Space Flight Center
HS	Hughes Spacecraft
Hughes	Hughes Space and Communications Company
HV	High Voltage
IAPS	Ion Auxiliary Propulsion System
IPS	Ion Propulsion System
JPL	Jet Propulsion Laboratory
LeRC	Lewis Research Center
NSSK	North-South Stationkeeping
NSTAR	NASA Solar Electric Propulsion Technology Applications Readiness
PAS	PanAmSat Corporation
PPU	Power Processing Unit
rms	root mean square
RPM	Revolutions Per Minute
SATMEX	Satmex of Mexico Company
S/C	Spacecraft
SCATHA	Spacecraft Charging at High Altitude
SEPS	Solar Electric Propulsion System
SEPST	Solar Electric Propulsion System Technology
SES	Societe Europeenne des Satellites of Luxembourg
SERT	Space Electric Rocket Test
SIT	Structurally Integrated Thruster
SNAP	Systems for Nuclear Auxiliary Power
SNAPSHOT	Spacecraft carrying the SNAP 10A nuclear power supply and a cesium ion propulsion system
SPIBS	Satellite Positive-Ion-Beam System
ST4	Space Technology 4
USAF	United States Air Force
XIPS	Xenon Ion Propulsion System

Introduction

Kilowatt-class ion propulsion systems have found applications for spacecraft North-South stationkeeping, orbit insertion, and primary propulsion for deep space missions.[1,2] The first successful ion propulsion flight system was demonstrated in 1964 during the course of the SERT I flight.[3] Later on seven more ion propulsion systems and one ion source system were flown as space experiments using mercury, cesium, or xenon expellants. A total of six successful, operational flights of IPS are now in use for NSSK or primary propulsion. Articles on the origins of ion propulsion can be found in References 4 and 5.

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.[6-9] 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 Ion Propulsions Systems

The experimental flights of ion propulsion systems developed in the United States are summarized. Some of the results indicated in the Tables 1a and 1b 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 end product which is the flight test.

Program 661A, Test Code A

In November of 1961, Electro-Optical Systems was awarded a contract by the U. S. Air Force to develop a 8.9 mN, cesium contact ion propulsion system for three sub-orbital flight tests. The Electric Propulsion Space Tests were called Program 661A and were managed by the Air Force Space Systems Command in Los Angeles.[10-12]

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 hours.

The first sub-orbital flight test was launched on December 18, 1962. When the high-voltage power supplies were first turned-on, intermittent high-voltage breakdowns occurred, and the beam power supply became inoperative. Post-flight examination of the power supply 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 July 20, 1964 using a Scout vehicle.[3,13] This flight experiment had a 8 cm diameter cesium contact ion engine and a 10 cm 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 10 cm diameter mercury 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 V 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. Ion beam neutralization was provided by a heated tantalum filament.

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 electrical short circuit. The mercury engine was started about 14 minutes into the flight. The IPS was successfully operated for

31 minutes with 53 high-voltage recycle events which were handled by the PPU fault protection system. Each of the recycle events is only a few seconds duration. Major results from the test were the first demonstration of an IPS in space, effective ion beam neutralization, no EMI effects on other spacecraft systems, and effective recovery from electrical breakdowns. Thrust was measured using three independent measuring systems, and there were no major differences between in-space and ground test performance.

Program 661A, Test Code B

Test Code B was the second in the series of three suborbital flight tests of the Electro Optical Systems' 8.9 mN, cesium ion engine systems.[10,14] A Scout vehicle launched the payload on August 29, 1964. The launch was designed to provide about 30 minutes above an altitude of 370 km. After 7 minutes into the flight, the engine was operated with ion beam extraction. Full beam current of 94 mA was achieved about 10 minutes 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 ten 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 minutes until spacecraft re-entry 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.

Program 661A, Test Code C

The third and final IPS payload of the Air Force's Program 661A was launched on December 21, 1964.[10,14] 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 re-entry 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 minutes into the flight when the altitude was 490 km. Engine operation ended after 4 minutes when the altitude was only 80 km.

SNAPSHOT

On April 3, 1965 a SNAP 10A nuclear power system was launched into a 1300 km orbit with a cesium ion engine as a secondary payload.[15-17] 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 one hour, and then the batteries were to be charged for approximately 15 hours 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 hour before being commanded off permanently. Analysis of flight data indicated a significant number of high-voltage breakdowns which apparently caused sufficient EMI to induce false horizon sensor signals which created 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.

ATS-4

Two cesium contact ion engines were launched aboard the ATS-4 spacecraft on August 10, 1968. Flight test objectives were to measure thrust and to examine electromagnetic compatibility with other spacecraft subsystems.[6,18,19] The 5 cm diameter 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 5 setpoints from 18 μN to 89 μN. Thrusters were configured so they could be used for East-West stationkeeping. Prior to launch, a 5 cm cesium thruster was life tested for 2245 hours at the 67 μN thrust level.[20]

During the launch process the Centaur stage did not achieve a second burn, and the spacecraft remained attached to the Centaur in a 218 km by 760 km orbit. It was estimated that the pressure at these altitudes was between 1 x 10^{-6} Torr and 1 x 10^{-9} Torr[16]. Each of the two engines was tested on at least two occasions each over the throttling range. Combined test time of the two engines was about 10 hours over a 55 day period. The spacecraft re-entered the atmosphere on October 17, 1968.

The ATS-4 flight was the first successful orbital test of an ion engine. There was no evidence of IPS electromagnetic interference 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.[18]

ATS-5

A flight IPS, identical to the one flown on ATS-4, was launched on ATS-5 on August 12, 1969. The purpose of this flight was to demonstrate NSSK of a geosynchronous satellite.[21,22] 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 revolutions per minute, and this caused an effective 4g 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 high-voltage 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 hours and 5169 hours duration. A prototype version of the SERT II spacecraft was ground-tested for a period of 2400 hours with an operating ion engine. The spacecraft was launched into a 1000 km high polar orbit on February 3, 1970.[23] In addition to diagnostic equipment and related IPS hardware, the spacecraft had two identical 15 cm diameter, mercury ion engines and two PPUs. 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 hours and 2011 hours. 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 hours during the course of the mission. Additionally, the IPS was electromagnetically compatible with all other spacecraft systems.

ATS-6

The purpose of the ATS-6 flight experiment was to demonstrate NSSK of a geosynchronous satellite using two cesium ion engine systems.[21,22,24,25] Thruster development tests included a lifetest of 2614 hours 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 May 30, 1974. One of the ion engines operated for about one hour and the other for 92 hours. 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 high voltage power supplies. This failure mechanism was verified through a series of ground tests.[25]

The IPS operation demonstrated an absence of EMI related to spacecraft systems, verified predictions of spacecraft 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. Further, tests indicated that "differential charging was reduced by the neutralizer when operated in spot mode and eliminated by operation of the ion engine."[22]

SCATHA, P78-2

The SCATHA spacecraft had two charged particle injection systems one of which was the Satellite Positive-Ion-Beam System (SPIBS).[26,27] This was a xenon ion source which included some of the technologies used in thrusters; however, the 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 1 keV or 2 keV. Neutralization was accomplished by a tantalum filament. The specific impulse is low because there was no attempt to optimize the propellant efficiency. The SPIBS system was ground-tested for a period of 600 hours. The SCATHA spacecraft was launched January 30, 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."[28] The SPIBS ion source discharged the SCATHA spacecraft 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 ion propulsion ground-test demonstrations in the United States. The projects described in this section involve ion propulsion systems that were never flown, or in the case of the XIPS-25, a pre-flight development program that is ongoing. Only those systems that included a structurally integrated thruster or an engineering model class thruster and an advanced PPU are described here.

SEPST

The objective of the Solar Electric Propulsion System Technology program at JPL was to demonstrate a complete breadboard IPS that would be applicable to an interplanetary spacecraft.[29,30] 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 diameter 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 servo-loops were demonstrated, and they were similar in concept to the two loops used in the SERT II technology. Servo-loops 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 spacecraft ground.

PPU development centered around the beam power supply. The beam power supply had 8 inverters and had an efficiency of 89% to 90% over a bus voltage range from about 53 V to 80 V.[30] The PPU was integrated with the thruster, 2:1 power throttling with closed-loop control was demonstrated, and HV recycle algorithms were developed. Initial BBPPU efficiencies were about 84%-86%, and subsequent experimental BBPPUs had efficiencies of 88%-90%. The experimental breadboard PPUs, which provided 2.5 kW, had a specific mass of 5.4 kg/kW. Later work in the 1970s focused on the SEPS program which developed a 30 cm mercury engine system which is described in a subsequent section.

SIT-5

A 5 cm diameter mercury ion engine, called SIT-5, was developed circa 1970 for attitude control and NSSK of geosynchronous satellites.[31-33] 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 degree vectoring capability were baselined. The engine was successfully random vibration tested at 19.9g rms. The 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 hours. The envelope was about 31 cm long by 12 cm diameter. 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 hours at a beam voltage of 1300 V, a thrust of 1.8 mN, and a specific impulse of 2500 s.[34,35] During the initial 2023 hours, the thruster was operated with a translating screen grid thrust vector system, and the thruster had an electrostatic thrust vector system for the remainder of the test. The electrostatic beam vector grids were operated at 5 degrees deflection for about 120 hours and at either 2 degrees or 4 degrees deflection for 1770 hours. There were an number of grid shorts that were successfully cleared by the application of 200 V to 400 V at currents from 6 mA to 70 mA. These tests were helpful in the later definition of grid-clear circuits for the IAPS, XIPS, and NSTAR thrusters.

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

SEPS

The Solar Electric Propulsion Stage program was started in the early 1970s with a goal to provide a primary ion propulsion system 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.[36,37] The SEPS program included the development of 25 kW solar arrays, PPUs, thermal control systems, gimbals, throttleable/long-life 30 cm diameter ion thrusters, and mercury propellant storage and distributions systems. This multi-Center, multi-Contractor effort was ongoing for about 10 years with a NASA investment of approximately $30 million dollars. 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 bi-module consisting of two thrusters, two PPUs, a propellant system, a gimbal system, thermal control, and supporting structure.[38,39] 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.[5,39] The thruster/PPU was capable of throttling down to 1.1 kW. More detailed references related to the development and test of the SEPS bi-module hardware can be found in Reference 37.

One of the early engineering model thrusters was tested for 10,000 hours over an input power range of 0.8 kW to 2.4 kW.[40] Endurance tests of these 30 cm ion engines confirmed the need for spalling control of sputter-deposited discharge chamber coatings,[40,41] and for the need of low sputter-yield materials for the cladding of pole-pieces and baffles.[42] 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.[43]

Subsequent to these EM thruster tests, a total of seven advanced engineering model thrusters were tested in segments including 3,940 hours and 5,070 hours and a total test time of 14,541 hours.[42] Ninety five percent of the test was implemented using either breadboard or brassboard PPUs which were of the series-resonant inverter design.[42,44]

IAPS

The Ion Auxiliary Propulsion System project and other preflight technology work took place in the 1974 to 1983 timeframe.[45] 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 7,000 hours and 2500 on/off cycles. The 8 cm diameter, 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 kg, 6.85 kg, and 4.31 kg, respectively.[46] 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 Air Force technology satellite.[47] The flight of the Teal Ruby spacecraft was canceled by the Air Force due to lack of funding.

During the course of the technology and preflight programs there were a number of endurance test performed. A laboratory-type 8 cm engine was tested for 15,040 hours and 460 cycles at the 0.14 kW level.[48] An engineering model IAPS engine and PPU were successfully tested for 9,489 hours and 652 cycles.[49] The thruster and PPU were located in the same vacuum chamber during this test. In another test, an engineering model thruster was operated at full-thrust for 7112 hours and had 2571 restarts.[50] There were no major changes in thruster performance, and no life-limiting degradation effects were observed. A single PPU was used to run two tests and had operated for 14,000 hours without malfunction.

XIPS-25 (1.3 kW)

This XIPS-25 program, conducted by Hughes Research Laboratories, developed thrusters, BBPPUs, and a feed system pressure regulator for possible NSSK of 2500 kg class communication satellites.[51] The 25 cm diameter, 3-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 thruster types 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 4,350 hours of testing and 3850 cycles which is equivalent to about 10 years of NSSK. The Hughes Space and Communications Company subsequently pursued development of XIPS-13 (0.44 kW) and XIPS-25 (4.2 kW) systems, instead of the 1.3 kW XIPS-25 system, for NSSK and orbit insertion applications.

XIPS-25 (4.2 kW)

A 25 cm diameter xenon engine system is being developed by the Hughes Space and Communications Company for NSSK, EWSK, attitude control, and momentum dumping for its HS 702 spacecraft.[52-54] The ion thrusters provide stationkeeping at a cost of only 5 kg per year. Additionally, the IPS will be 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. It is planned that the HS 702 spacecraft use four XIPS-25 engines and two PPUs. Only two of the four thrusters are required to perform the stationkeeping and momentum control functions. Hughes has not yet launched the HS 702 spacecraft with the XIPS-25. The spacecraft has an end-of-life solar array power capability of about 15 kW.

Each thruster has an input power of 4.2 kW and provides 165 mN thrust at 3800 s specific impulse. XIPS hardware is currently under extended tests at the Hughes-Torrance, CA 6.1 m diameter by 12.2 m long vacuum facility.

Evolution of Electron-Bombardment Ion Thruster Development in the United States

Figure 1 shows a "roadmap" of the history of electron-bombardment ion thruster development in the United States from the first tests of a 10 cm engine[55] to the first operational flights in 1997/1998.[2,54] Much of the history of the early development of mercury ion engines was outlined in Reference 5. The following remarks only focus on the insertion of component improvements to the mercury and xenon ion engines. In the early 1960s the wire-grids were replaced by multiaperture grids.[56] Later, in the mid-1960s engine life extension was made possible by the incorporation of hollow cathodes for the neutralizer and main discharge.[57-59] The SERT II flight was the major in-space demonstration of these technologies.[23] Major technology improvements in the 1970s were the development of high-perveance, dished grids[60], methods to control spalling of sputter deposited material in the discharge chamber[41], and methods to provide deep-throttling.[5] Mercury engines were developed with diameters ranging from 5 cm to 150 cm. Endurance tests of these engines extended from about 4,000 hours to 15,000 hours.

In the 1980 timeframe it was decided to replace the mercury engines with xenon engines because xenon was less contaminating to spacecraft surfaces, and ground-test operations were greatly simplified. In the 1980s and 1990s ring-cusp discharge chambers[61,62] 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 chambers do 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.[63] 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.[64] Ground tests of 13 cm and 30 cm diameter xenon engines demonstrated more than 8,000 hours of reliable operation.[54,65] The communication satellite and deep space tests of these engines, starting in 1997, confirmed the thrusters and PPUs are a very mature technology.

Operational Flights of Ion Propulsion Systems

In 1997/1998, a new era of ion propulsion for spacecraft 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. These were the first operational uses of IPS by United States industry and government.

PAS-5, Galaxy VIII-i, ASTRA-2A, SATMEX 5, PAS B

As shown in Table 3, the Hughes Space and Communications Company has launched five operational communications satellites each employing four-0.44 kW xenon ion thrusters for NSSK.[52-54] The high specific impulse IPS reduces the propellant requirements, versus chemical systems, by 300 kg to 400 kg, thus allowing incorporation of more communications hardware aboard the spacecraft or reduction in launch size and cost. The IPS consists of two fully redundant strings each consisting of two thrusters and one PPU. Two daily "burns" of 5 hours each are generally required for the NSSK function. Typical spacecraft lifetime is about 15 years.

Approximate masses for a thruster and PPU are 5.0 kg and 6.8 kg, respectively.[66] 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 was Hughes' first customer for the XIPS-13 propulsion system on PAS-5. This was the first successful, operational spacecraft employing IPS and was launched August 27, 1997 from Kazakhstan on a Russian Proton rocket. Since then, four more spacecraft are operational using the XIPS-13 system namely, Galaxy VIII-i, ASTRA-2A, SATMEX 5, and PAS 6B.

Deep Space 1

The NSTAR program provided a single string, primary IPS to the Deep Space 1 spacecraft. The 30 cm ion thruster operates over a 0.5 kW to 2.3 kW input power range providing thrust from 19 mN 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 LeRC and JPL of 2000 hours, 1000 hours, and 8193 hours using engineering model thrusters.[2,65] The flight-set masses for the thruster, PPU, and DCIU were 8.2 kg, 14.77 kg, and 2.51 kg, respectively[67]. 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 was comprised of two segments which 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 by Hughes, and the DCIU was built by Spectrum Astro, Inc. The feed system development was a collaborative effort between JPL and Moog, Inc.[68]

As of April 27, 1999, the primary thrusting of the NSTAR engine system required to encounter the asteroid 1992KD was completed. The thrusting time at the end of April was 1764 hours. Thruster input power levels were varied from 0.48 kW to 1.94 kW. A total of 11.6 kg of xenon was expended. As shown in Table 4, the NSTAR engine already has demonstrated the largest propellant throughput in space as compared to a SERT II engine that expended about 9 kg of mercury. Propellant throughput is a signature of total impulse capability. Nearly 70 kg of xenon remains on the DS 1 spacecraft for a possible mission extension. It is intended that the DS 1 spacecraft will pass within 10 km of the asteroid 1992KD in July 1999. If an extended mission is approved, DS 1 will encounter comets Wilson-Harrington and Borrelly in the year 2001.

Next Generation Ion Propulsion Technologies

Over the next decade, it is expected that there will be many communications spacecraft employing the XIPS-13 and XIPS-25 propulsion systems. Additionally, the Space Technology 4 spacecraft will be developed by JPL for a flight to the comet Tempel 1, and a small vehicle will be sent to the comet surface for scientific measurements. The ST4 spacecraft will use three NSTAR ion engine/PPU subsystems for primary propulsion.

In the next few years, new IPS technologies will be developed by NASA for higher thrust density 30 cm ion engines and sub-kilowatt, smaller engines both of which have application to planetary and Earth-orbital spacecraft. Some of the near-term work, shown in Figure 2, 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-flowrate neutralizers are also needed for a wide class of thrusters which operate at low power levels or are throttled over a wide range of input power. Design approaches and manufacturing technologies which provide reduced ion engine and PPU mass and cost are receiving significant attention in order to enable or enhance planetary and small-body missions using relatively small launch vehicles.

Conclusions

The historical background and characteristics of the experimental flights of ion propulsion systems 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, are 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 Deep Space 1 flight confirmed that these auxiliary and primary propulsion systems have advanced to a high-level of flight-readiness.

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Table 1a. Experimental Flights of Ion Propulsion Systems

Spacecraft	**Program 661A, Test Code A**	**SERT I**		**Program 661A, Test Code B**	**Program 661A, Test Code C**	**SNAPSHOT**
Sponsor	USAF	NASA LeRC		USAF	USAF	USAF
Builder of IPS	EOS	LeRC	Hughes	EOS	EOS	EOS
Launch date	12.18.62	07.20.64		08.29.64	12.21.64	04.03.65
Orbit, km	Suborbital	Suborbital		Suborbital	Suborbital	700
IPS type	Contact ionization	Electron bombard-ment	Contact ioniza-tion	Contact ionization	Contact ionization	Contact ionization
Propellant	Cesium	Mercury	Cesium	Cesium	Cesium	Cesium
No. of thrusters	1	1	1	1	1	1
Thruster anode diameter	~7 cm	10 cm	8 cm	~7 cm	~7 cm	~5 cm
Type of neutralizer	Wire filament	Ta wire	Ta wire	Wire filament	Wire filament in beam	Wire filament, barium coated
Beam power supply voltage	5000 V	2500 V	4500 V	5000 V	5000 V	4500 V
Power per thruster	0.77 kW	1.4 kW	0.6 kW	0.77 kW	0.77 kW	~0.4 kW
Maximum thrust	8.9 mN	28 mN	5.6 mN	8.9 mN	8.9 mN	~8.5 mN
Specific impulse	7400 s	4900 s	8050 s	7400 s	7400 s	5100 s
Propellant mass	2 g			2 g	2 g	
Maximum in-space operation time for one thruster	0 min.	31 min.	0 min.	~19 min.	~4 min.	<60 min.
Longest ground test	1230 h					
Comment	HV power supply failed due to contamination from gases vented from batteries. Ref. 10	IPS and neutraliza-tion demonstra-tion. Ref. 3	Cesium engine had a HV short.. Ref. 3	Stable operation of the IPS. S/C potential ~1000 V at full thrust.. Ref. 14	Failed 3rd stage burn shortened operation. Obtained ~20% of full thrust.. Ref. 14	Continuous arcing at HV terminals induced EMI to the S/C systems. Ref. 16

Table 1b. Experimental Flights of Ion Propulsion Systems

Spacecraft	ATS-4	ATS-5	SERT II	ATS-6	SCATHA P78-2
Sponsor	USAF/NASA GSFC	USAF/NASA GSFC	NASA LeRC	NASA GSFC	USAF/NASA GSFC
Builder of IPS	EOS	EOS	NASA LeRC, Westinghouse	EOS	Hughes (ion source)
Launch date	08.10.68	08.12.69	02.03.70	05.30.74	01.30.79
Orbit, km	218x760	36,000	1000	36,000	43,000x27,000
IPS type	Contact ionization	Contact ionization	Electron bombardment	Electron bombardment	Electron bombardment
Propellant	Cesium	Cesium	Mercury	Cesium	Xenon
No. of thrusters	2	2	2	2	1
Thruster anode diameter	5 cm	5 cm	15 cm	8 cm	3.6 cm
Type of neutralizer	Ta doped with Yttrium	Ta doped with Yttrium	Hollow cathode	Cesiated Ta	Ta doped with Yttrium
Beam power supply voltage	3000 V	3000 V	3000 V	560 V	1000 V to 2000 V
Power per thruster	0.02 kW	0.02 kW	0.85 kW	0.15 kW	0.03 to 0.045 kW
Maximum thrust	0.089 mN	0.089 mN	28 mN	4.5 mN	0.14 mN
Specific impulse	6700 s	6700 s	4200 s	2500 s	350 s
Propellant mass	~0.05 kg		15 kg	3.6 kg	0.3 kg
Max. in-space operation time for one thruster	~10 h	No operation with a HV beam.	~3781 h	92 h	
Longest ground test(s)	2245 h		6742 h, 5169 h	2614 h, 471 cycles	~600 h
Comment	S/C was in a low altitude parking orbit due to a Centaur stage failure. First successfull orbital test of an ion engine. No EMI to S/C subsystems. Ref. 18	S/C had a 76 RPM spin=rate. This produced a 4g field which compromised the cesium feed system and precluded normal operation. Ref. 21	One ion engine operated 3781 h until the neutralizer tank was depleted. The other engine had a grid short which limited operation to 2011 h. Ref. 23	One thruster operated for ~1 hour and the other for 92 hours. Further thrusting was terminated due to a feed system "flooding" problem. No EMI. Ref. 24	Operations were performed intermittantly over a 247 day period. Ref. 26

Table 2. Major Ion Propulsion System Demonstrations

Project Name	SEPST	SIT-5	SEPS	IAPS	XIPS-25	XIPS-25
Sponsor	NASA JPL	NASA LeRC	NASA	NASA LeRC	INTELSAT	Hughes
Builder of thruster	JPL	Hughes	Hughes	Hughes	Hughes	Hughes
Builder of PPU	Hughes/TRW	--	TRW	Hughes	Hughes	Hughes
Integrator of IPS	JPL	--	LeRC	Hughes	Hughes	Hughes
Project duration	1968 to 1972	1969 to 1972	1972 to 1980	1974 to 1983	1985 to 1988	Ongoing in 1999, preflight
Propellant	Mercury	Mercury	Mercury	Mercury	Xenon	Xenon
Thruater diameter	20 cm	5 cm	30 cm	8 cm	25 cm	25 cm
Type of neutralizer	Hollow cathode	Hollow cathode	Hollow cathode	Hollow cathode	Hollow cathode	Hollow cathode
Beam power supply voltage	2000 V	1600 V	1100 V	1200 V	750 V	~1400 V
Power per thruster	2.5 kW	0.072 kW	2.6 kW	0.13 kW	1.3 kW	4.2 kW
Maximum thrust	88 mN	2.1 mN	128 mN	5.1 mN	63 mN	165 mN
Specific impulse	3600 s	3000 s	3000 s	2500 s	2800 s	3800 s
Longest ground test	1300 h	9715 h	10,000 h	15,040 h, 9489 h, 7112 h	4350 h, 3850 cycles	Ongoing in 1999

Table 3. Operational Flights of Ion Propulsion Systems

Spacecraft	**PAS-5**	**Galaxy VIII-i**	**ASTRA-2A**	**Deep Space 1**	**SATMEX 5**	**PAS 6B**
Sponsor	PanAmSat	PanAmSat	SES	NASA LeRC/JPL	Satmex	PanAmSat
Builder of IPS	Hughes	Hughes	Hughes	Hughes	Hughes	Hughes
Launch date	08.27.97	12.08.97	08.29.98	10.24.98	12.05.98	12.21.98
Orbit, km	36,000	36,000	36,000	Orbits sun	36,000	36,000
IPS type	Electron bombardm't	Electron bombardm't	Electron bombardm't	Electron bombardm't	Electron bombardm't	Electron bombardm't
Propellant	Xenon	Xenon	Xenon	Xenon	Xenon	Xenon
No. of thrusters	4	4	4	1	4	4
Thruster diameter	13 cm	13 cm	13 cm	30 cm	13 cm	13 cm
Type of neutralizer	Hollow cathode	Hollow cathode	Hollow cathode	Hollow cathode	Hollow cathode	Hollow cathode
Beam power supply voltage	750 V	750 V	750 V	650 V to 1100 V	750 V	750 V
Power per thruster	0.44 kW	0.44 kW	0.44 kW	0.50 kW to 2.3 kW	0.44 kW	0.44 kW
Maximum thrust	18 mN	18 mN	18 mN	92 mN	18 mN	18 mN
Specific impulse	2590 s	2590 s	2590 s	1900 s to 3100 s	2590 s	2590 s
Propellant mass	>100 kg	>100 kg	>100 kg	82 kg	>100 kg	>100 kg
Maximum in-space operation time for one thruster				1764 h as of 04.27.99		
Longest ground test	>8000 h			8193 h		

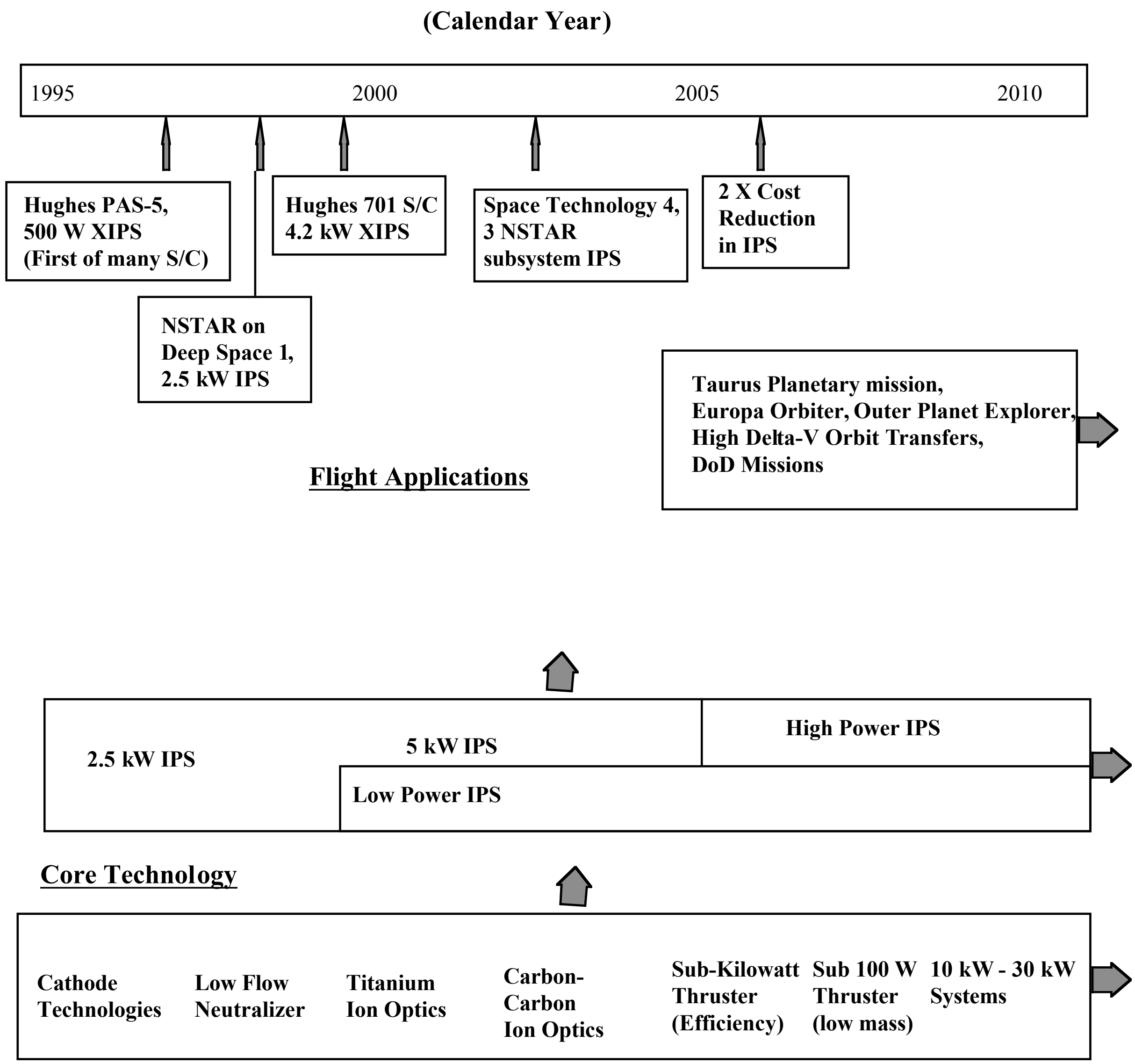
Table 4. Comparison of the Propellant Throughput Capability of the SERT II and Deep Space 1 Ion Propulsion Systems

	Propellant Throughput
SERT II	
Propellant type	Mercury
Longest ground test	~16 kg
Maximum thruster propellant throughput in-space	Estimated to be ~9 kg
NSTAR/Deep Space 1	
Propellant type	Xenon
Longest ground test	87.5 kg
Maximum thruster propellant throughput in-space as of 04.27.99	11.6 kg
DS 1 propellant throughput capability for the primary and extended mission	82 kg

Figure 1. History of electron-bombardment ion thruster development in the United States. (All projects were NASA sponsored unless noted otherwise.)

YEAR	COMPONENT ADVANCES	DEVELOPMENT PROGRAMS	LONG TESTS	FLIGHTS
1960 ->		10-cm lab thruster		
		5-cm lab thruster		
	Multi-aperture grids			
		20-cm lab thruster		
1964 ->				SERT I (10-cm)
	Vaporizer			
	Long-life oxide main cathode			
1966 ->	Plasma bridge neutralizer and discharge chamber hollow cathode	SERT II EM thruster (15-cm)		
		50-cm lab thruster		
		150-cm lab thruster		
1970 ->	HV, propellant isolator (Hughes)		SERT II thruster & PPU ground-tested 6742 h	15-cm SERT II flight system
1972 ->		20-cm SEPST EM system		
		5-cm EM thruster		
1973 ->	Dished grids	8-cm lab thruster		
	Grid erosion control	30-cm lab thruster		
		30 cm EM Development Contract at Hughes	15,000 h test-8 cm	
		SEPS development program	10,000 h test - 30 cm EM	
1976 ->		8-cm EM thruster	5070 h test-30 cm EM	
	Control of spalled flakes in discharge chamber			
	Test facility effects on component wear			
		IAPS development program (8-cm, Hg)		
1980 ->	Change Hg -> Xe			
1981 ->	Ring-cusp chamber	30-cm thruster (Xe)		
		25-cm thruster (Xe) (INTELSAT/Hughes)	9489 h test of the 8-cm, EM mercury thruster	
1988 ->		13-cm lab thruster (Xe) (Hughes)		
	Develop reliable Xe hollow cathode via Space Station plasma contactor program			
1997 ->				XIPS-13 for comsat NSSK (Hughes)
1998 ->			>8000 h test of XIPS-13 (Hughes)	
			8193 h test of the NSTAR thruster	NSTAR 30-cm for Deep Space 1
1999 ->		XIPS-25 for comsat orbit insertion and NSSK (Hughes)	Extended testing of the XIPS-25 (Hughes)	
		Initiate development of subkilowatt and 5 kW IPS for Earth-orbital and deep space S/C	Extended ground-testing of the NSTAR flight spare thruster, PPU, and DCIU	

Figure 2. Ion propulsion technology roadmap for Earth-orbital and planetary applications



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