# Demonstration of the NSTAR Ion Propulsion System on the Deep Space One Mission *

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**Deep Space 1 is the first interplanetary spacecraft to use an ion propulsion system for the primary delta-v maneuvers. The purpose of the mission is to validate a number of technologies, including ion propulsion and a high degree of spacecraft autonomy, on a flyby of an asteroid and a comet. The ion propulsion system has operated for a total of over 14,200 hours at engine power levels ranging from 0.48 to 1.94 kW and has completed the encounter with the asteroid Braille (1992KD) and the comet Borrelly. The system has worked extremely well after an initial grid short was cleared after launch. Operation on the DS 1 spacecraft has demonstrated all ion propulsion system and autonomous navigation functions. All propulsion system operating parameters are very close to the expected values. This paper provides an overview of the system performance from the first 14,200 hours of ion propulsion system operation in interplanetary space.**

## Introduction

NASA's New Millennium Program (NMP) is designed to flight validate high risk, high payoff technologies that will be required to execute future Earth orbital and Solar System exploration missions. A xenon ion primary propulsion system (IPS) is one of the key technologies being demonstrated on Deep Space 1 (DS1), the first of the New Millenium missions [1]. This spacecraft was launched in October, 1998, flew by the asteroid Braille (1992KD) in July, 1999 and the comet Borrelly in September, 2001. The validation objectives of DS1 include demonstrating the functionality and performance of the ion propulsion system in an environment similar to what will be encountered by future users, the compatibility of the IPS with the spacecraft and science instruments, and autonomous navigation and control of the IPS with minimum ground mission operations support. The in-space performance of the propellant feed system is discussed in reference [2] and preliminary results on the interactions of the IPS with the spacecraft and instrumentation are presented in [3]. This paper summarizes the results of validation activities associated with the engine performance and mission operations for the first 14,200 hours of engine operation.

## Overview of the NSTAR Ion Propulsion System

The flight ion propulsion system was delivered to DS1 by the NASA Solar Electric Propulsion (SEP) Technology Applications Readiness (NSTAR) program, a joint Jet Propulsion Laboratory/Glenn Research Center effort to develop NASA's 30 cm ion thruster technology for flight applications with industry participation from Boeing Electron Dynamics, Moog, Inc. and Spectrum Astro, Inc. The ion thruster uses propellant delivered by a Xenon Feed



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System (XFS) and is powered by a Power Processing Unit (PPU), which converts power from the solar array to the currents and voltages required by the engine. The XFS and PPU are controlled by a Digital Control Interface Unit (DCIU), which accepts and executes high level commands from the spacecraft computer and provides propulsion subsystem telemetry to the spacecraft data system. Planetary missions often require a wide throttling range to accommodate variations in solar array output power with distance from the Sun, so the NSTAR IPS was designed to operate over an engine power range of 0.5 kWe to 2.3 kWe. The development of the flight system is discussed in detail in references [4, 5].

DS1 also includes an autonomous system (AutoNav) to navigate the spacecraft to the next encounter target. This system contains an optimized trajectory that was computed on the ground and a catolog of ephemerides for a number of stars, asteroids, the planets and the DS1 target bodies. Periodically (one to three times per week) during a burn, the system automatically turns the spacecraft to optically observe the positions of a number of these bodies against the stellar background and calculates the spacecraft position. The heliocentric orbit is then determined and the trajectory propogated to the next target. Required course changes are generated by the maneuver design element and accomplished by varying the IPS thrust direction and duration. This technology dramatically reduces the need for mission operations support, as described below.

## The NSTAR Throttle Table

The DCIU is capable of operating the thruster at any one of 16 discrete throttle levels from a throttling table stored in memory. This table contains the setpoints for the PPU power supplies and the XFS pressures and can be modified by ground command. The NSTAR 16 level throttle table showing the entire range of operation is listed in Table (1). This throttle table contains the IPS setpoints required to operate the system over the required throttling range. A corresponding mission throttle table, listed in Table (2), which contains the flow rates, thrust and PPU input and output power levels is required to perform the mission trajectory design. The development of these throttle tables is described in this section.

Power throttling is accomplished by varying the beam voltage and current. The engine throttling envelope with lines of constant beam power is shown in Fig. (1). The boundaries of this envelope represent the maximum beam voltage and current capabilities, the minimum beam current (which is determined primarily by the minimum discharge current) and the beam voltage perveance limit. The NSTAR throttle table was designed to maximize the specific impulse, so the power is varied with beam current throttling over most of the range. The lowest power levels are achieved by operating at the minimum beam current and throttling the beam voltage.

The discharge chamber flow rate was selected to give the propellant utilization shown in Fig. (2). The propellant efficiency of 0.9 was chosen at high power levels as a compromise between maximizing total engine efficiency and minimizing double ion production, which can drive internal erosion rates. A propellant efficiency of 0.9–0.91 is maintained over most of the range. At the lowest powers the double-to-single ion current ratio is low, so lower propellant efficiencies were chosen to optimize performance and achieve the desired total power.

The neutralizer and cathode flow rates are approximately equal at each operating point and vary over the throttling range as shown in Fig. (3). The minimum flow rate was designed to prevent neutralizer operation in plume mode, which can cause excessive erosion of the orifice. End-of-life neutralizer characterization data from the 8200 hour Life Demonstration Test (LDT) of an engineering model thruster (EMT2) are shown on this plot as well [6]. As described in more detail below, after about 11,000 hours of operation the NSTAR diagnostics package on board the spacecraft detected higher fluxes of ions at higher energies than earlier in the flight. This appeared to be associated with throttling the neutralizer flow below a certain value. To avoid the possibility of damage to the neutralizer the throttle table was modified to allow higher flow rates at the lower throttle levels, as shown in Fig. (3). After this modification the ion flux and energy distribution returned to values observed earlier in the flight. The maxi-

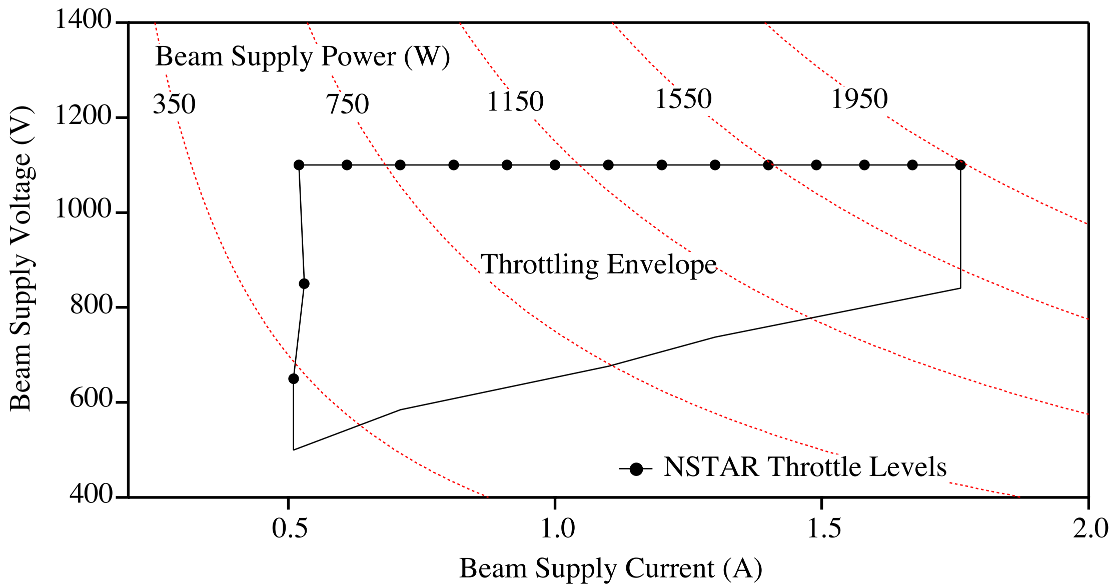


Figure 1: NSTAR power throttling strategy.

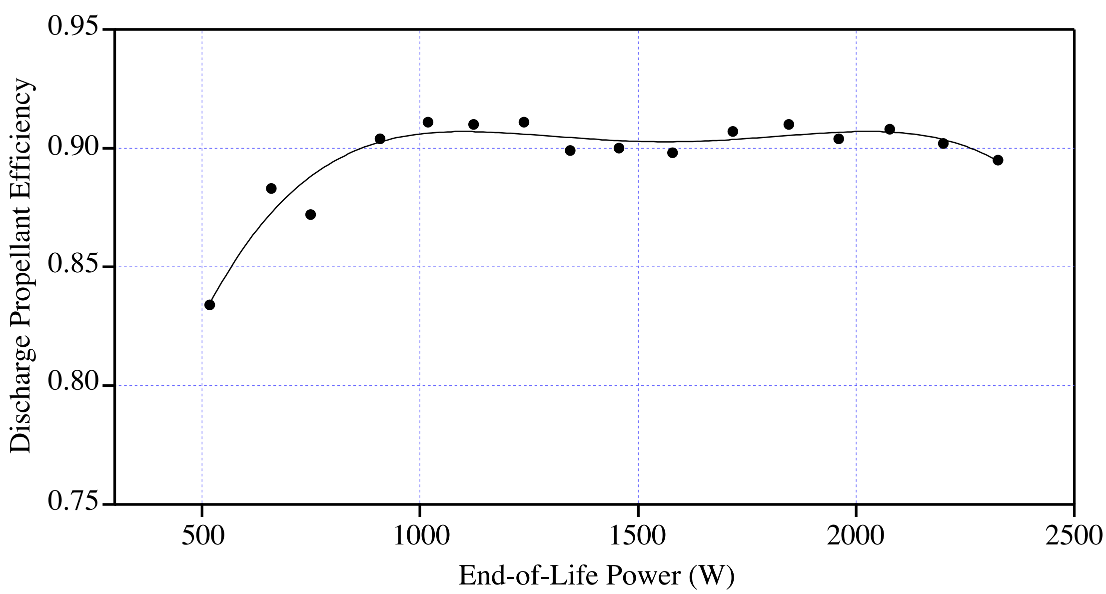


Figure 2: NSTAR ion thruster discharge propellant utilization efficiency.

Table 1: Flight throttle table of parameters controlled by the DCIU.

| NSTAR Throttle Level | Mission Throttle Level | Beam Supply Voltage (V) | Beam Supply Current (A) | Accelerator Grid Voltage (V) | Neutralizer Keeper Current (A) | Main Plenum Pressure (psia) | Cathode Plenum Pressure (psia) |
|---|---|---|---|---|---|---|---|
| 15 | 111 | 1100 | 1.76 | -180 | 1.5 | 87.55 | 50.21 |
| 14 | 104 | 1100 | 1.67 | -180 | 1.5 | 84.72 | 47.50 |
| 13 | 97 | 1100 | 1.58 | -180 | 1.5 | 81.85 | 45.18 |
| 12 | 90 | 1100 | 1.49 | -180 | 1.5 | 79.29 | 43.80 |
| 11 | 83 | 1100 | 1.40 | -180 | 1.5 | 76.06 | 42.38 |
| 10 | 76 | 1100 | 1.30 | -180 | 1.5 | 72.90 | 41.03 |
| 9 | 69 | 1100 | 1.20 | -180 | 1.5 | 69.80 | 40.26 |
| 8 | 62 | 1100 | 1.10 | -180 | 1.5 | 65.75 | 40.26 |
| 7 | 55 | 1100 | 1.00 | -150 | 2.0 | 61.70 | 40.26 |
| 6 | 48 | 1100 | 0.91 | -150 | 2.0 | 57.31 | 40.26 |
| 5 | 41 | 1100 | 0.81 | -150 | 2.0 | 52.86 | 40.26 |
| 4 | 34 | 1100 | 0.71 | -150 | 2.0 | 48.08 | 40.26 |
| 3 | 27 | 1100 | 0.61 | -150 | 2.0 | 43.18 | 40.26 |
| 2 | 20 | 1100 | 0.52 | -150 | 2.0 | 39.22 | 40.26 |
| 1 | 13 | 850 | 0.53 | -150 | 2.0 | 39.41 | 40.26 |
| 0 | 6 | 650 | 0.51 | -150 | 2.0 | 40.01 | 40.26 |

Table 2: Flight throttle table of parameters used in mission analysis.

| NSTAR Throttle Level | Mission Throttle Level | PPU Input Power (kW) | Engine Input Power (kW) | Calculated Thrust (mN) | Main Flow Rate (sccm) | Cathode Flow Rate (sccm) | Neutralizer Flow Rate (sccm) | Specific Impulse (s) | Total Efficiency |
|---|---|---|---|---|---|---|---|---|---|
| 15 | 111 | 2.52 | 2.29 | 92.4 | 23.43 | 3.70 | 3.59 | 3120 | 0.618 |
| 14 | 104 | 2.38 | 2.17 | 87.6 | 22.19 | 3.35 | 3.25 | 3157 | 0.624 |
| 13 | 97 | 2.25 | 2.06 | 82.9 | 20.95 | 3.06 | 2.97 | 3185 | 0.630 |
| 12 | 90 | 2.11 | 1.94 | 78.2 | 19.86 | 2.89 | 2.80 | 3174 | 0.628 |
| 11 | 83 | 1.98 | 1.82 | 73.4 | 18.51 | 2.72 | 2.64 | 3189 | 0.631 |
| 10 | 76 | 1.84 | 1.70 | 68.2 | 17.22 | 2.56 | 2.48 | 3177 | 0.626 |
| 9 | 69 | 1.70 | 1.57 | 63.0 | 15.98 | 2.47 | 2.39 | 3136 | 0.618 |
| 8 | 62 | 1.56 | 1.44 | 57.8 | 14.41 | 2.47 | 2.39 | 3109 | 0.611 |
| 7 | 55 | 1.44 | 1.33 | 52.5 | 12.90 | 2.47 | 2.39 | 3067 | 0.596 |
| 6 | 48 | 1.32 | 1.21 | 47.7 | 11.33 | 2.47 | 2.39 | 3058 | 0.590 |
| 5 | 41 | 1.19 | 1.09 | 42.5 | 9.82 | 2.47 | 2.39 | 3002 | 0.574 |
| 4 | 34 | 1.06 | 0.97 | 37.2 | 8.30 | 2.47 | 2.39 | 2935 | 0.554 |
| 3 | 27 | 0.93 | 0.85 | 32.0 | 6.85 | 2.47 | 2.39 | 2836 | 0.527 |
| 2 | 20 | 0.81 | 0.74 | 27.4 | 5.77 | 2.47 | 2.39 | 2671 | 0.487 |
| 1 | 13 | 0.67 | 0.60 | 24.5 | 5.82 | 2.47 | 2.39 | 2376 | 0.472 |
| 0 | 6 | 0.53 | 0.47 | 20.6 | 5.98 | 2.47 | 2.39 | 1972 | 0.420 |

mum flow rate was chosen to match the discharge cathode flow rate used in a 1000 hour test of an engineering model thruster which demonstrated little cathode erosion compared to a previous 2000 hour test at a lower flow rate [7]. Subsequent tests suggest that other design changes were responsible for the erosion rate reduction, but the higher flow was maintained to be conservative.

The thrust in the mission throttle table is calculated from the engine electrical setpoints,

$$T = \alpha F_t J_b (V_s - V_g)^{1/2} \left( \frac{2M}{e} \right)^{1/2} \quad (1)$$

where $J_b$ is the beam current, $V_s$ is the beam power supply voltage, $V_g$ is the coupling voltage between neutralizer common and the facility ground or ambient space plasma, $M$ is the mass of a xenon ion

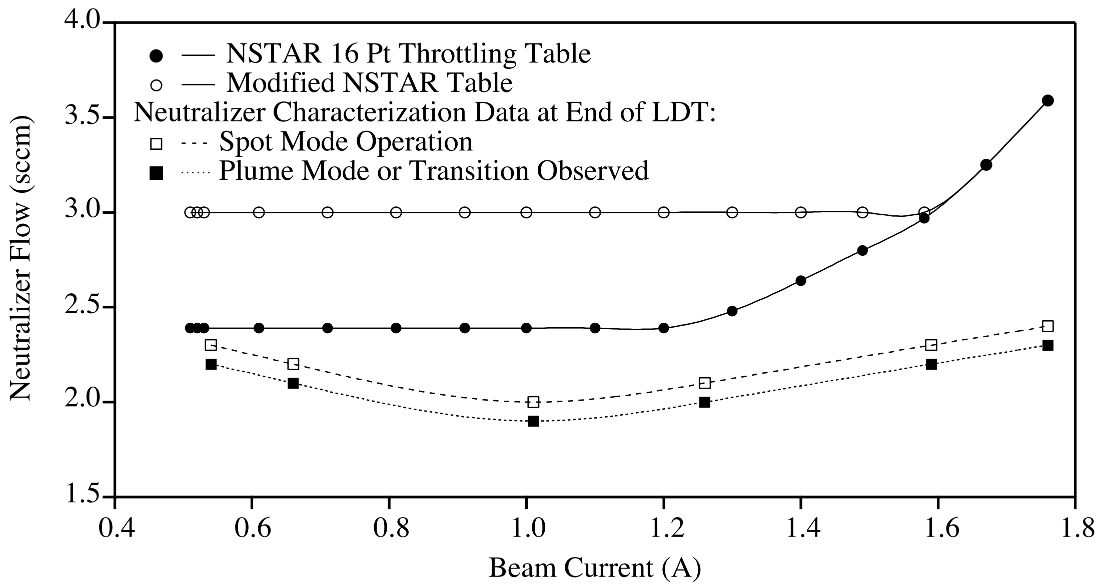


Figure 3: NSTAR cathode flow rates.

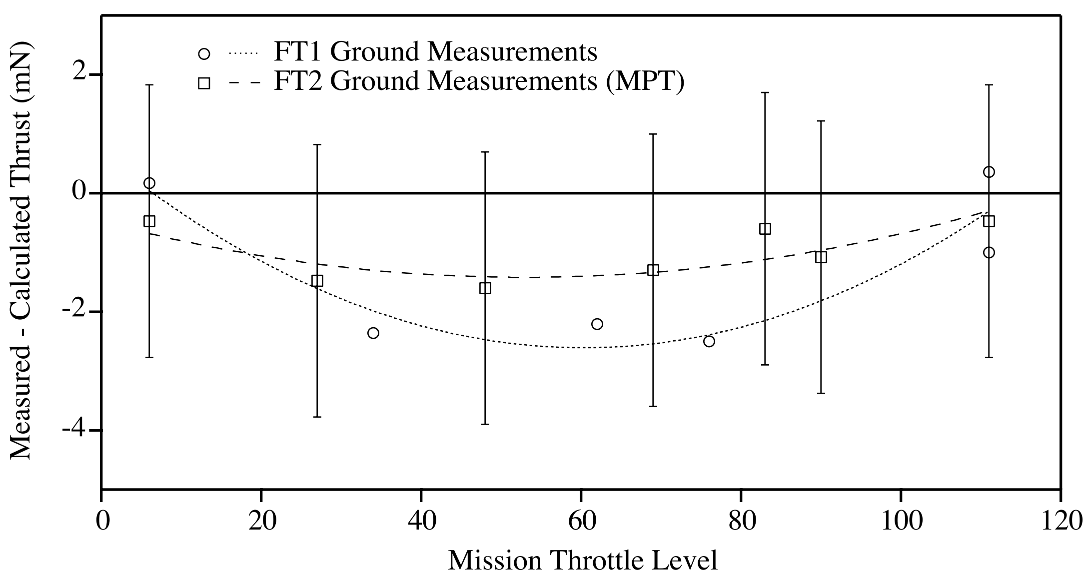


Figure 4: The difference between measured and calculated thrust over the NSTAR throttling range.

and $e$ is the charge of an electron. The factors $\alpha$ and $F_t$ correct for the doubly-charged ion content of the beam and thrust loss due to beam divergence [8]. A constant value of 0.98 for $F_t$ based on earlier 30-cm thruster ground tests and a value of $\alpha$ based on a curve fit to centerline double ion current measurements as a function of propellant utilization efficiency in a 30 cm, ring-cusp inert gas thruster [9] were used. Direct measurements of thrust with the flight thrusters yield values which are somewhat lower than the calculated values for intermediate throttle levels. The difference between the measured thrust and the table values is shown in Fig. (4). Recent measurements of the double ion content of the beam over the throttling range suggest that it is higher than the value used in calculating the parameter $\alpha$, so the throttle table overpredicts the thrust by 1–2 percent in the intermediate power range.

The power required for a given thrust level increases over the engine lifetime due to wear [8], so two tables representing beginning-of-life (BOL) and end-of-life (EOL) were developed. These have the same engine setpoints shown in Table (1) but different engine power levels. The BOL table was developed primarily through testing with engineering model thrusters and updated with data from preflight measurements with FT1. The EOL table was based largely on measurements from the 8200 hour test of EMT2. The power at the lowest throttle levels was extrapolated from performance curves obtained after about 6500 hours of operation. The extrapolations were based on sensitivity data, which were used to correct for slight differences in some of the controlled parameters. The difference between BOL and EOL engine power is plotted in Fig. (5). Additional measurements taken at some of these throttle levels after about 6900 hours of operation in the LDT are also shown. They suggest that the EOL power at some of the lower throttle levels is overestimated in the throttling table. BOL data obtained with the two flight thrusters demonstrates that their initial performance agrees well with the table values.

The PPU input power corresponding to a given engine power is determined by the PPU efficiency. The efficiency of the flight PPU was characterized as a function of input bus voltage and temperature in several ground tests, as shown in Fig. (6). The lowest measured values over this range of parameters were used to define the lowermost line in the figure. This conservative estimate of PPU efficiency was used to generate the PPU input powers in the throttle table.

To make finer steps in power throttling and more closely track the solar array peak power, a 112 point throttle table was also developed. Throttling between the 16 NSTAR throttle points is accomplished by varying the beam voltage to give steps approximately 20 W apart. A 16 point subset of this table is loaded into the DCIU to provide fine control over a restricted power range for a given mission phase.

## Post-Launch IPS Operation and Validation Activities

Operation of the ion propulsion system in the DS1 mission can be organized into several phases which are summarized in this section.

### Decontamination

The first IPS in-space activity was a bakeout of the downstream portion of the propellant feed system that occurred after launch. Prior to this the thruster axis was oriented away from the Sun. The spacecraft was turned so that the angle between the axis and the Sun was 30° to warm the thruster and feed system. This was done to help remove any residual contaminants in the portions of the feed system that had been exposed to air prior to launch. The cathode conditioning sequence was then executed to bakeout the cathode inserts. Finally, the discharges were operated for four hours at high power levels to further bakeout the engine prior to application of high voltage.

### Initial Start and Grid Short

The following day the first engine ignition occurred. Both cathodes ignited properly and the engine ran nominally at the minimum power point for 4.5 minutes before continuous recycling caused a thruster shutdown. A short between the grids was suspected, but at this point a failure of one of the high voltage supplies could not be ruled

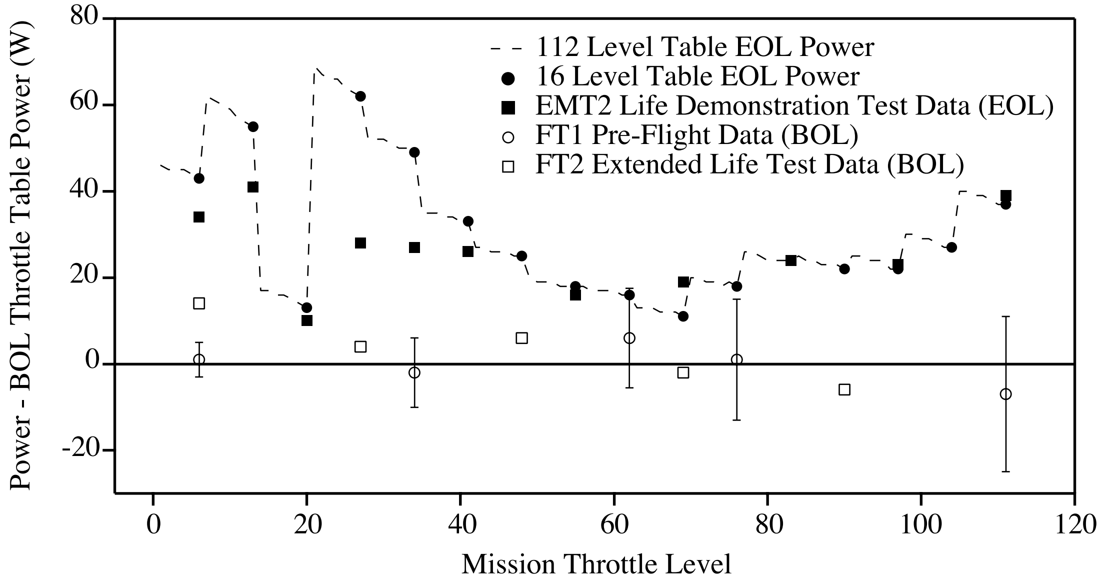


Figure 5: Difference between a given power level and the beginning-of-life power.

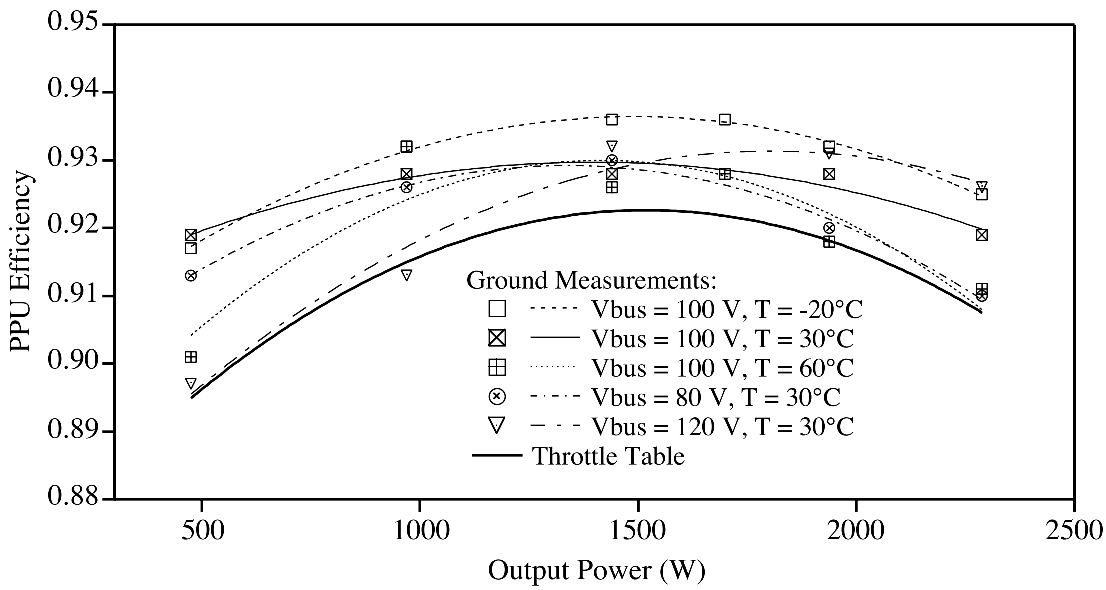


Figure 6: PPU efficiency measurements.

out. Fourteen additional start attempts were made under various engine thermal conditions (created by spacecraft turns toward or away from the Sun), but all ended in continuous recycling when the high voltage was applied.

### Troubleshooting

Delta-v maneuvers using the IPS were not required to encounter the target bodies until much later in the mission, so a detailed investigation of the problem was undertaken. Several options were identified, including attempting a grid clear command, thermally cycling the engine to force a mechanical separation of the grids that might dislodge a particle, running additional recycles and developing additional diagnostics to help identify the fault.

The NSTAR PPU is designed to deliver 4 A into a grid short to clear those that are not cleared by recycles, as outlined above. However, this system was designed primarily to clear thin molybdenum flakes generated by spalling of sputter-deposited films inside the discharge chamber after many thousands of hours of operation. Grid shorting this early in a mission was more likely due to particulates from the launch vehicle payload fairing or generated during the payload separation, which could be much larger than films from the discharge chamber. The risk of permanently welding a large particulate between the grids with the standard grid clear circuit was not known, so an experimental and theoretical effort to characterize the grid clear process was undertaken prior to using it under these circumstances. The results of this investigation are reported in [10].

Thermal and structural models of the ion optics were also coupled during this period to determine the mechanical effect of thermally cycling the grids. This modeling showed that significant transient changes in the grid spacing can be achieved by turning the spacecraft to heat or cool the grids. This technique was used to clear grid shorts on the SERT II flight experiment [11] and appeared to have the least risk. During the two week problem investigation period the spacecraft was turned several times to thermally cycle the grids.

The IPS is designed with hardware interlocks which prevent operation of the high voltage supplies before the discharges are ignited, so it was not possible to command these supplies on separately to test them. The DCIU software was modified to provide brief bursts of high speed data for various PPU electrical parameters during recycles to help diagnose which supplies were affected. Finally, a test involving operation of the discharge supply only with no flows (which is allowed by the system) was developed. If the grids are shorted, the accelerator grid voltage telemetry will change when the discharge open circuit voltage is applied, otherwise it remains close to zero. This is a clear discriminator between open circuits and shorts on the ion optics.

### Recovery Start

Thirty one days after launch the discharge-only test was executed and the results suggested that the grids were not shorted. Another start attempt was then made, primarily with the intent to gather high speed engine data during continuous recycling to help diagnose the fault. Fortunately the engine started properly this time, and has continued to run flawlessly since this point. Apparently the thermal cycling successfully cleared debris lodged between the grids.

### First Performance Test

Over the next two weeks the engine was operated at power levels ranging from 0.48 to 1.94 kW to characterize the BOL performance. This burn was used to contribute to the required spacecraft delta-v, but was not controlled by AutoNav. The throttle levels were dictated primarily by the validation objectives. This test was designated IPS Acceptance Test 1 (IAT1).

### Deterministic Thrusting for the Braille Encounter

IAT1 was followed by approximately 100 hours of thrusting at power levels ranging from 1.7 to 1.86 kW. These initial operations were executed with ground commands. These were followed by a coast period, then seven navigational burns (NBURNs) totalling approximately 900 hours of operation. These maneuvers were executed autonomously by the spacecraft and used automatic peak power tracking to determine the maximum achievable throttle level. This portion of the mission was on an outbound part of the trajectory, so the available array power dropped continuously and the engine was operated at power levels as low as 0.7 kWe. These burns completed the deterministic thrusting required for the encounter with asteroid 1992KD.

**Second Performance Test**

After another coast period a second throttling test was performed. This brief test, designated IAT2, was restricted to power levels ranging from 0.49 to 0.98 kW by total solar array power.

**Trajectory Correction Maneuvers for the Braille Encounter**

In addition to the deterministic burns, the IPS was used for some of the final trajectory correction maneuvers (TCMs) prior to the asteroid encounter. These burns each lasted 2–4 hours and placed the spacecraft on a trajectory which passed within 15 km of the asteroid. TCMs on the final day before the encounter were performed with the hydrazine attitude control thrusters. Certain spacecraft attitudes with respect to the Sun are not allowed because of the orientation of sensitive optical instruments or thermal control surfaces. If thrust is required in a direction disallowed by these constraints, the maneuver was decomposed into two burns in safe directions with a resultant thrust in the proper direction. All of these operations were executed autonomously with no ground intervention.

**Navigational Burns for the Comet Encounter Using AutoNav**

Two days after the Braille encounter, the IPS started another series of NBURNs under AutoNav control to achieve the proper trajectory toward the comet encounter. Twelve additional NBURNs totaling approximately 1700 hours of operation at power levels of 0.9 to 1.3 kWe were completed.

**Navigational Burns for the Comet Encounter Using MICAS**

In November, 1999 after about 3500 hours of IPS operation the DS1 spacecraft entered a safe mode when the stellar reference unit failed. This left the spacecraft without the ability to do complete attitude determination. To recover from this failure, the spacecraft software was modified to use the Miniature Integrated Camera and Spectrometer (MICAS), another one of the technologies being validated in this mission, as a star tracker. The system software was redesigned to use MICAS images of carefully chosen reference stars for closed loop control of the attitude control system. This solution proved to provide reliable spacecraft attitude control and in June, 2000 thrusting with the IPS was resumed. The modified attitude control approach did not allow the use of the autonavigation feature, so the trajectory was planned on the ground and thrust segments were executed between high gain antenna links with the Earth, which occurred approximately once a week. The IPS was controlled by the spacecraft during the periods between downlinks and was automatically throttled to track the available solar array power. An additional 40 NBURNS at power levels of 0.57 to 1.18 kWe were executed over the period from June, 2000 to May, 2001, for a total engine operating time of about 11100 hours. This completed the deterministic thrusting required for the Borrelly encounter.

**Thrust Vector Control With the IPS for Hydrazine Conservation**

After the long period in safe mode while the spacecraft software was being modified and tested, it appeared that the hydrazine attitude control system might not have sufficient propellant for the rest of the mission. The IPS is gimballed to provide pitch and yaw control, so when the engine is operating the hydrazine system is only used for roll control.

This reduces the hydrazine consumption by about a factor of ten, from 30 g/day to 3 g/day. The IPS was operated almost continuously since the restart in June, 2000 to conserve xenon. During parts of the trajectory that did not require deterministic burns, the engine was run at a low thrust level for thrust vector control.

### Trajectory Correction Maneuvers for the Borrelly Encounter

All final TCM's for the Borrelly encounter were successfully completed using the IPS. As a result, the spacecraft now has a reserve of over 3 kg of hydrazine to support further tests.

## In-Flight System Performance

One of the primary objectives of the flight validation activity is to verify that the system performs in space as it does on the ground. The parameters of interest to future mission planners are those in the mission throttle table: thrust and mass flow rate as a function of PPU input power. In this section the system power, thrust and mass flow rate behavior will be evaluated in terms of the throttle table.

### PPU Power Input Requirements

The PPU input power is determined by the PPU output power (engine power requirement) and the PPU efficiency. The difference between the in-flight engine input power and the BOL throttle table power is shown in Fig. (7). These power values are based on the individual power supply current and voltage telemetry readings. The flight electrical telemetry is calibrated to within $\pm 2$ percent of reading for the high voltage supply parameters and $\pm 2$ percent of full scale for the other parameters. In this paper the values have been corrected using the ground calibration data and are more accurate–typically the standard error is under 0.2–0.8 percent of full scale. The voltage measurements have also been corrected for flight cable line drops and represent the values that would be measured at the engine.

The total engine power consumed during the IAT1 throttle test and initial operations differed from the table values by only about 2 W on average, although the uncertainties are much larger than this, as shown by representative error bars on the figure. The engine power requirement increased by 12–15 W with time, however, as indicated by the data from the second major thrust period from 852 to 1802 hours of operation. During the third major thrust arc from 1802 to 3495 hours the engine power at throttle levels between 40 and 50 has increased to the EOL power values used in the throttle table, which is represented by the dashed line in Fig. (7). The subsequent 10700 hours of operation were primarily at lower power levels. At mission throttle levels between 25 and 40 the engine power increased by 30 to 40 W, but did not reach the EOL throttle table values. More recent operation at levels 15 and 10 for thrust vector control suggests that at the lowest power levels the engine was operating near the EOL throttle table values. This increased power demand is due primarily to increased discharge power losses, as discussed in the next section. This is a normal consequence of engine aging [8, 6], but does not imply that the engine is nearing end of life. In the 8200 hour Life Demonstration Test the engine power increased during the first 3000 hours and was relatively constant thereafter [6]. When the throttle table was modified after 12600 hours to allow higher cathode flow rates, the engine power decreased as shown in Fig. (7). As shown below, this is due to a lower discharge power resulting from higher discharge chamber flow rates.

In-flight measurements of the PPU efficiency suggest that it is higher than the conservative value assumed in the throttle tables, as shown in Fig. (8). These values are based on the total engine power and PPU high voltage bus current and voltage telemetry with an additional 15 W assumed for the low voltage bus input power. There is no telemetry for the low voltage bus, but ground testing showed a 15 W loss for all conditions. The efficiency is sensitive to the line voltage and the temperature, as the ground data show. The in-flight measurements should be compared with the solid line in the center of the preflight data and the highest dashed line. The range of uncertainty in these measurements encompasses the ground test data, but the in-space measurements ap-

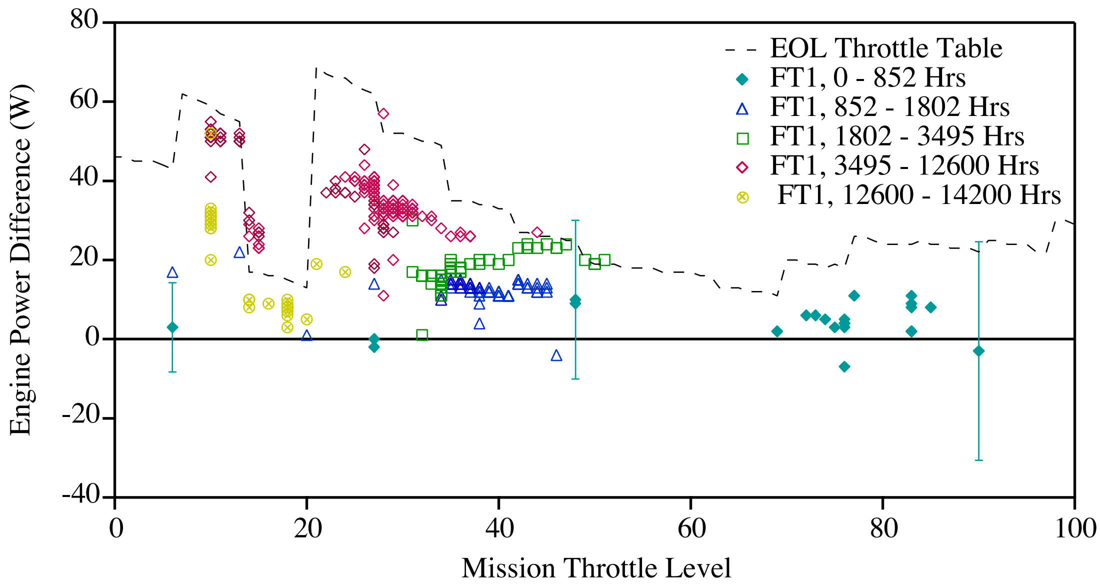


Figure 7: Difference between a given engine power level and the throttle table BOL values.

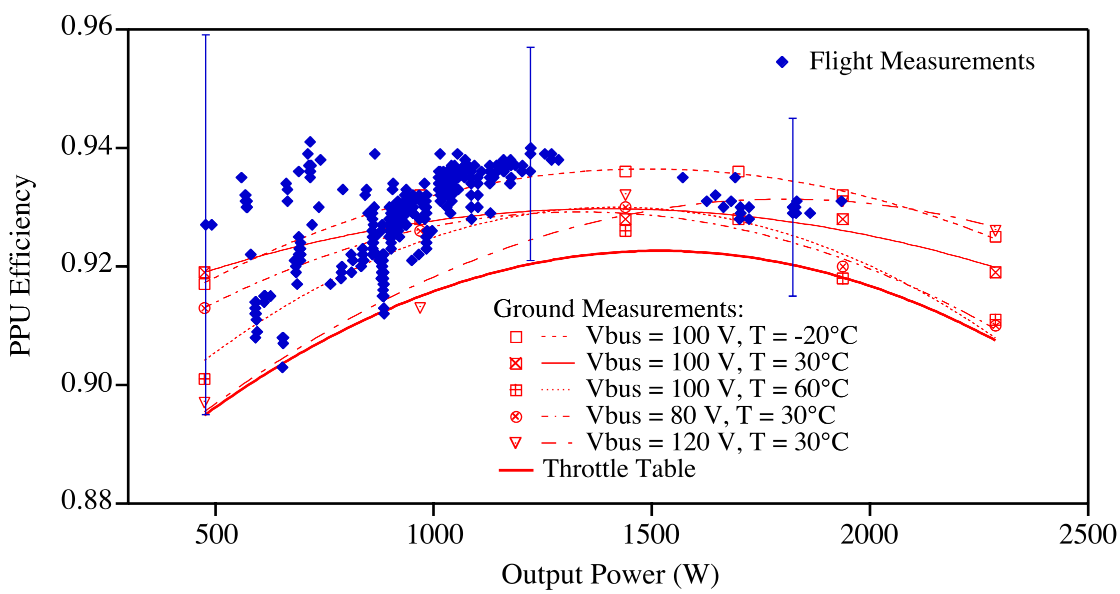


Figure 8: In-flight measurements of PPU efficiency compared to ground test data.

pear to be comparable to or higher than the ground measurements.

Because the PPU efficiency is higher than planned in the mission throttle table, it more than offsets the increased output power requirements observed so far in the actual flight. Figure (9) displays the difference between the observed PPU input power and the BOL input power from the throttle table. The input power required early in the mission was approximately 20 W lower than expected, because of the higher PPU efficiency. The data from the subsequent NBURNs show that the input power had exceeded the BOL throttle table value but had not reached the EOL levels. Over the last 1600 hours the PPU power has been close to the BOL value as a result of higher cathode flow rates.

**IPS Thrust**

The acceleration of the spacecraft is measured very accurately from changes in the Doppler shift of the telecommunications signals. With models of the spacecraft mass as a function of time, the Doppler residual data can be used to measure the thrust of the IPS with an uncertainty of less than 0.5 mN [12]. Preliminary thrust measurements were obtained in the first 852 hours of thrusting from IAT1 and during IAT2 at 1800 hours. The flight beam voltage and current values, which determine the thrust to a large extent, were slightly different from the setpoints in the table. The flight thrust measurements are therefore compared to the thrust calculated from the actual electrical parameters rather than the table values. The difference in the measured and calculated thrust is shown in Fig. (10), with the curve fits to similar data obtained with a thrust balance in ground tests. The ground and flight data agree well with the calculated values at low powers, but are lower at intermediate powers. The flight data suggest that the difference in true thrust and calculated thrust grows linearly with power and is up to 1.6 mN lower than expected at mission level 83 (1.82 kWe engine power). The error bars are based on the uncertainty in the measured thrust and do not include errors in the calculated thrust. This discrepancy is evidently due to an underestimate of the double ion content of the beam used in calculating the thrust values from the electrical parameters. More recent data on the double ion content yield better agreement with the measured values.

Although the actual thrust appears to be slightly lower than the throttle table value, at the beginning of the mission the overall system performance was still very close to the BOL throttle table level, in terms of thrust for a given PPU input power. Figure (11) shows that at the beginning of the mission the higher PPU efficiency largely compensated for the lower thrust. In this comparison, the thrust is within 0.5 mN of the table values. The gap between the two widens as the engine wears and the total engine power requirement for a given throttle level grows, however. The PPU input power required for the thrust levels measured during NBURN 0 has exceeded the EOL throttle table power for an equivalent thrust.

**Propellant Flow Rates**

The performance of the xenon feed system is discussed in detail in [2]. In general, the performance has been excellent, although the flow rates are slightly higher than the throttle table values. The mean value of the main flow is 0.05–0.14 sccm (about 0.4 to 1.0 percent) high, while that of the two cathode flows is 0.03 sccm (about 1.0 percent) high. This is in part intentional. As Fig. (12) shows, the XFS bang-bang regulators result in a sawtooth pressure profile. The control system is designed so that the minimum pressure in this sawtooth yields the throttle table flow rate values. In addition to this deliberate conservatism, there is a slight bias in both regulators because one of each of the three pressure transducers on the two plena had a slight offset after launch.

**Overall System Performance**

The propulsion system performance can be summarized in terms of specific impulse and efficiency. At the beginning of the mission the Isp was about 60 s lower and the engine efficiency was 2 to 2.5 percentage points lower than the throttle table

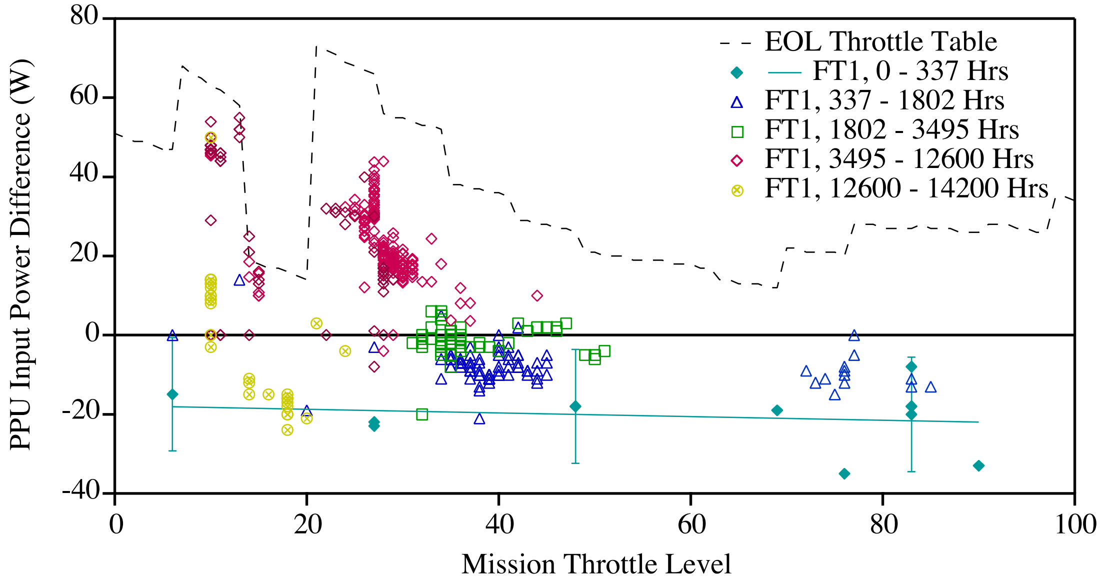


Figure 9: Difference between a given PPU input power level and the corresponding throttle table BOL value.

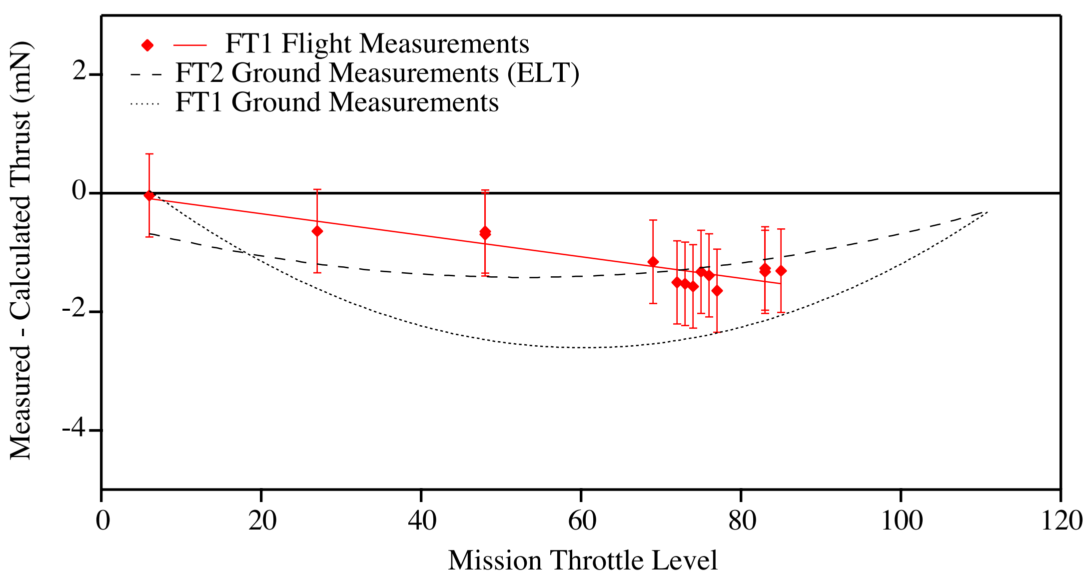


Figure 10: Difference between measured and calculated thrust in flight compared to ground measurements.

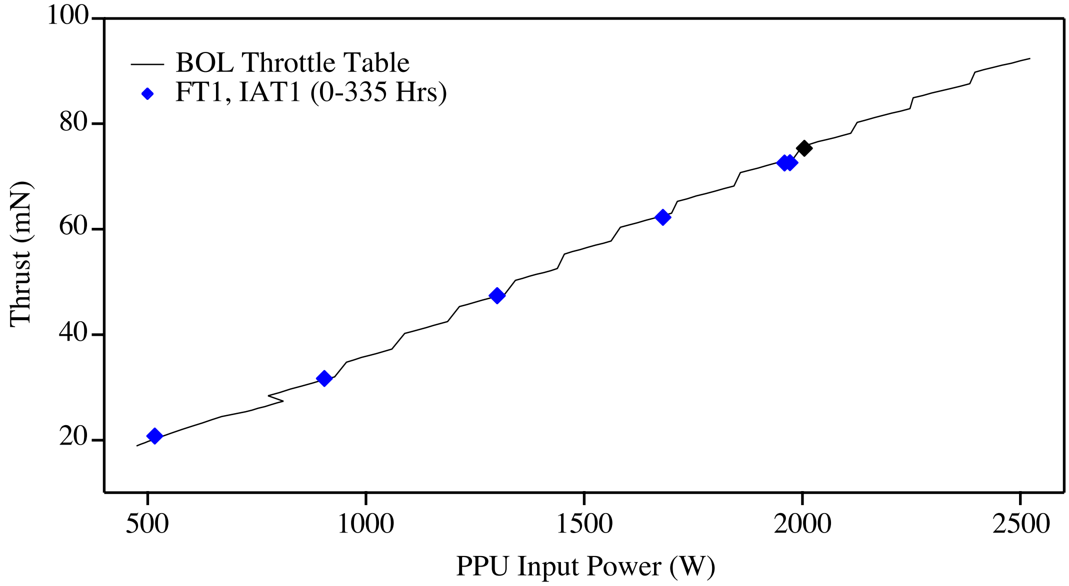


Figure 11: Measured thrust as a function of PPU input power compared to throttle table values.

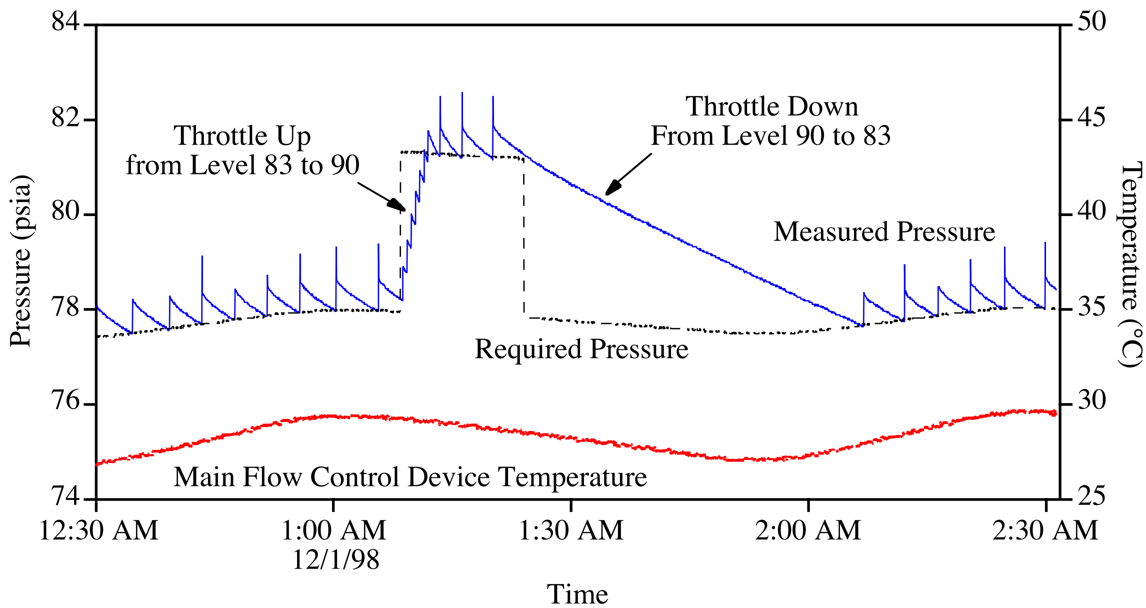


Figure 12: Example of flow rate throttling.

Table 3: Flight engine performance measured in space.

| NSTAR Throttle Level | Mission Throttle Level | PPU Input Power (kW) | Engine Input Power (kW) | Measured Thrust (mN) | Main Flow Rate (sccm) | Cathode Flow Rate (sccm) | Neutralizer Flow Rate (sccm) | Specific Impulse (s) | Total Efficiency |
|---|---|---|---|---|---|---|---|---|---|
| Measurements from IAT1 at beginning of mission. | | | | | | | | | |
| 0 | 6 | 0.501 | 0.478 | 20.797 | 6.05 | 2.50 | 2.43 | 1964 | 0.419 |
| 3 | 27 | 0.890 | 0.843 | 31.766 | 6.93 | 2.50 | 2.43 | 2776 | 0.513 |
| 6 | 48 | 1.286 | 1.222 | 47.298 | 11.42 | 2.50 | 2.42 | 2998 | 0.569 |
| 9 | 69 | 1.666 | 1.571 | 62.227 | 16.08 | 2.50 | 2.43 | 3068 | 0.596 |
| 11 | 83 | 1.944 | 1.823 | 72.561 | 18.63 | 2.75 | 2.67 | 3126 | 0.610 |
| 12 | 90 | 2.063 | 1.935 | 77.388 | 20.01 | 2.91 | 2.83 | 3114 | 0.611 |
| Measurements from IAT2 after 1800 hours of operation. | | | | | | | | | |
| 0 | 6 | 0.515 | 0.492 | 20.705 | 6.04 | 2.50 | 2.42 | 1958 | 0.404 |
| 1 | 13 | 0.670 | 0.626 | 24.234 | 5.88 | 2.49 | 2.41 | 2330 | 0.442 |
| 2 | 20 | 0.778 | 0.737 | 26.985 | 5.84 | 2.50 | 2.42 | 2598 | 0.467 |
| 3 | 27 | 0.910 | 0.860 | 31.460 | 6.91 | 2.51 | 2.43 | 2752 | 0.494 |
| 4 | 34 | 1.049 | 0.984 | 36.616 | 8.38 | 2.49 | 2.42 | 2854 | 0.521 |

values. This is a result of slightly lower thrust than predicted initially. The measured performance was still excellent, with a measured efficiency of 0.42 to 0.60 at Isp's ranging from 1960 to 3125 s over an engine throttling range of 478 to 1935 W. The measured mission planning performance parameters are summarized in Table (3).

## Engine Behavior In Flight

The engine behavior in space has been very similar to that observed in ground testing. The detailed operating characteristics of the engine are discussed in this section.

### Engine Ignitions

A total of 175 successful engine ignitions have occurred in the first 14200 hours of the mission with only one failure to achieve beam extraction due to the initial grid short discussed above. The data from these ignitions are reviewed here. The nominal heater current value is 8.5 A; the actual cathode and neutralizer heater currents in-flight have been constant and within about 2 percent of the setpoint value. The uncertainty in these measurements is about $\pm 2$ percent. The peak heater voltage is a function of the heater impedence, current and temperature. The flight telemetry shows that the heater voltage increases in any rapid sequence of ignitions because the conductor is hotter at the beginning of each consecutive start. The data show that the heater voltage is also higher when the initial thruster temperature is higher. The scatter in the peak voltages under similar temperature conditions is low and very similar to that observed in ground tests.

The time required for the cathodes to ignite after the initial heating phase and application of the high voltage ignitor pulses is also a function of the initial temperature, with 20–80 s delays in neutralizer ignition observed for the lowest temperatures. Delays of up to 86 s were also observed during ground thermal tests at the lowest temperatures [13], and are not considered to be a concern. The discharge cathode typically ignites 5–6 s after successful neutralizer ignition, which reflects delays in the start sequence. Its ignition delay may be shorter because it has a slightly higher heater current and because it automatically goes through a longer heat phase when the neutralizer ignition is delayed.

### Throttling Characteristics

The throttling sequences were in all cases executed properly by the DCIU after receiving ground commands. An example of the throttling sequence

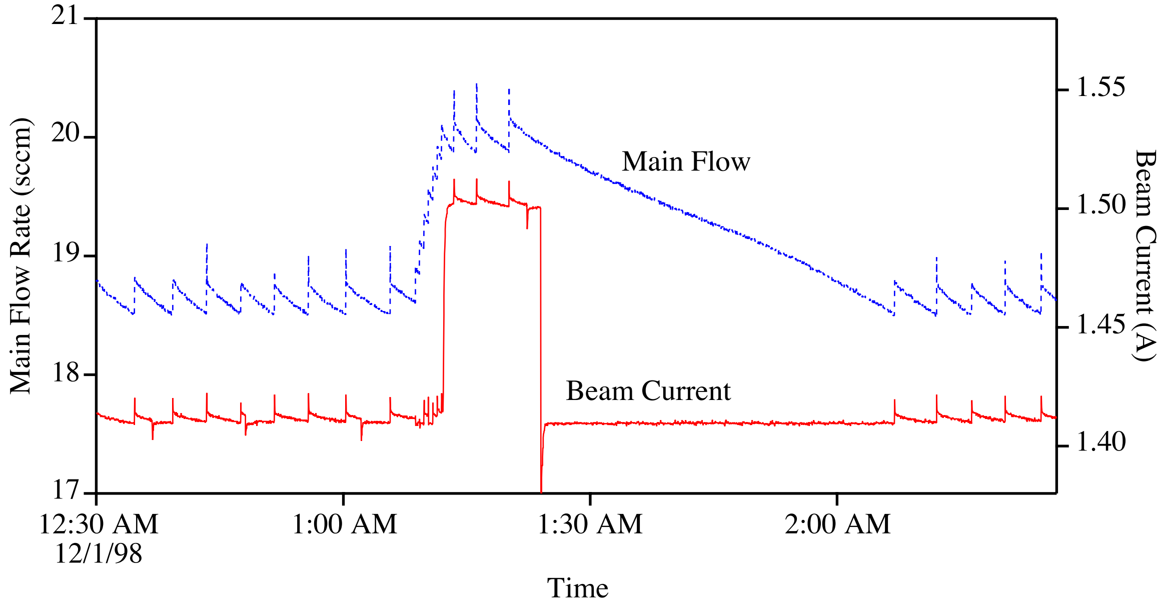


Figure 13: Example of throttle-up and throttle-down sequences.

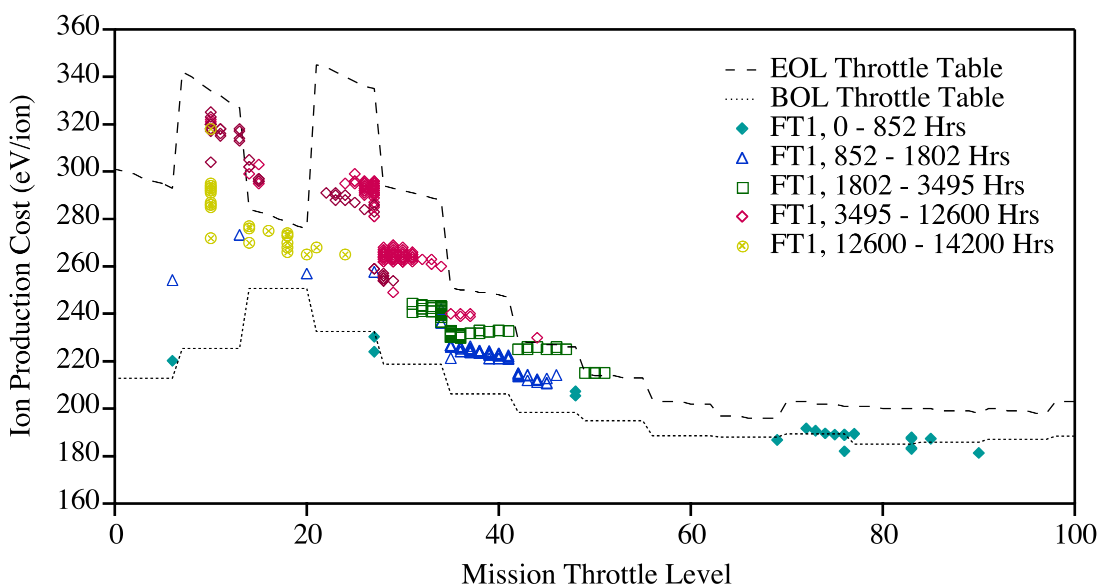


Figure 14: Discharge losses measured in flight compared to throttle table values.

is shown in figures (12) and (13). The software onboard the spacecraft is also designed to autonomously throttle the engine and track the peak power available from the array. The Navigation Manager software recalculates the throttle level every 12 hours and commands the IPS to the proper throttle level via the IPS Manager. The Navigation Manager uses models of the solar array power, spacecraft power consumption and the trajectory to calculate the throttle level. If the solar array output cannot supply the demands of the spacecraft and the IPS, power is drawn from an auxiliary battery. The battery algorithm will autonomously throttle the engine if the battery discharge rate and charge drop below a prescribed threshold. This new level will be maintained until the Navigation Manager resets the throttle level again. This function was successfully demonstrated in all of the NBURNs, which were accomplished without requiring ground control over the detailed engine operation.

**Steady-State Setpoint Accuracy**

As mentioned above, the flight flow rates are slightly higher than the throttle table setpoints. In addition, the beam current is approximately 1 percent high over a range of 0.51 to 1.49 A. The beam current is controlled in flight to within $\pm 2$ mA by varying the discharge current. Variation in the beam current is driven primarily by the flow rate sawtooth, as shown in Fig. (13). The neutralizer keeper current is within one percent of the setpoint. The accelerator grid voltage is 1.1 percent higher than the setpoint at the full power operating point. The beam voltage is on average about 0.3 percent lower than the full power setpoint. The offsets in beam power supply settings result in slightly higher beam power levels than the throttling tables assume, although this is largely offset by lower neutralizer power. All of these parameters are well within the specified tolerances.

**Discharge Performance**

As indicated in the previous section, the difference between the total engine power and the throttle table values is dominated by the discharge power difference. The discharge performance is summarized in terms of the ion energy cost in Fig. (14). The standard error of these measurements is 1.5 percent. This plot shows the beginning- and end-of-life discharge loss as a function of mission throttle level. The data from early in the DS1 mission are quite close to the throttle table values except in the middle of the range (throttle levels 40-60), where the flight data are higher. This appeared to be true of the ground measurements as well, suggesting that the BOL throttle table discharge power is low by about 10 W in this range. The data from the subsequent thrust periods indicate that the discharge losses increased with time as a consequence of engine wear [8, 6]. The lowest throttle levels are particularly sensitive to engine wear [8] and show the largest increases in flight, up to 95 W. The data from the thrust arc from 1802 to 3495 hours show that the discharge losses increased to the EOL throttle table values in the mid-power range. During the period up to 12600 hours of operation the discharge loss increased to the EOL values for the lowest power levels, although the losses at throttle levels between 25 and 40 were still below the EOL values. Data obtained since the cathode flow rates were increased show a decrease of up to 30 W in the ion production cost.

The discharge voltage and current are compared with the throttle table values in figures (15) and (16). The voltages measured in flight are typically within 2 percent of the throttle table voltages. The ground test data are also plotted in this figure and tend to be slightly higher, although some of these measurements have not been corrected for voltage drops in the ground facility power cables. There is very little drift in the discharge voltage over most of the flight, which is consistent with long duration ground test data [14, 6, 7]. For the last 1600 hours the voltage has been about 1 V lower than before as a result of higher cathode flow rates. The discharge current is also close to the BOL table values initially, with the exception of measurements at mission level 48. This is in the range where the table values appear to underestimate true BOL behavior. Unlike the voltage, the discharge current increased with time driving the

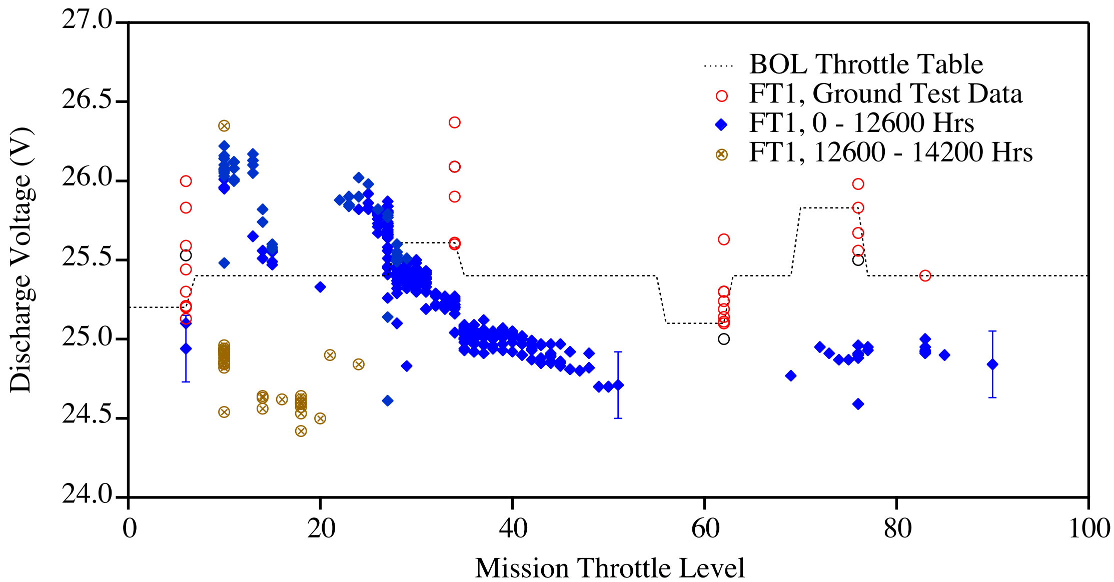


Figure 15: Discharge voltage measured in flight compared to throttle table and ground test measurements.

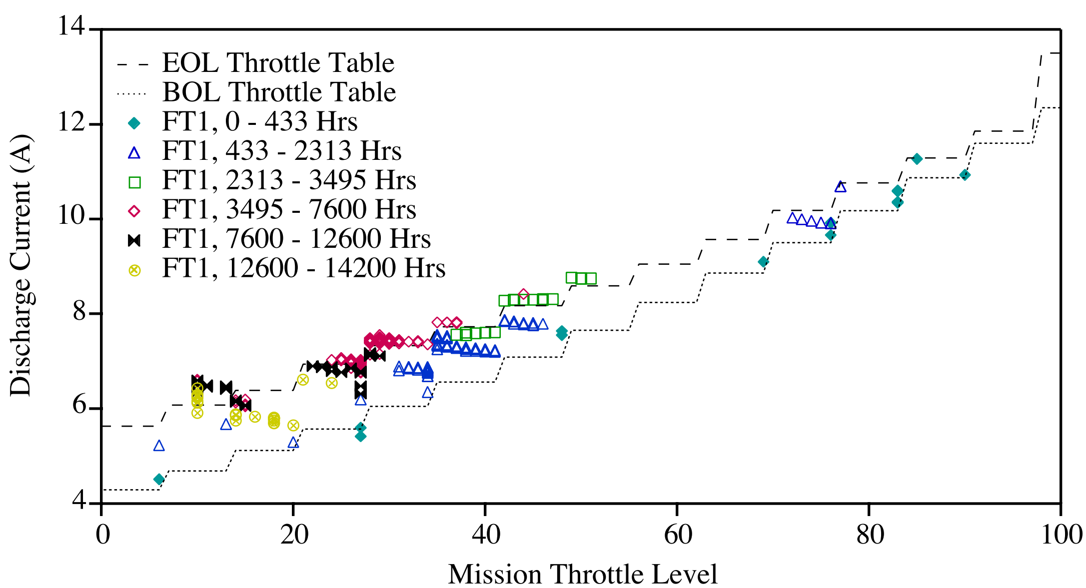


Figure 16: Discharge current measured in flight compared to throttle table values.

discharge power toward the EOL values. Since the cathode flow rates were increased, the discharge current has dropped slightly.

Data on the sensitivity of discharge losses, voltage and current to small variations in flow rates and beam current from the on-going Extended Life Test were used to examine the effect of setpoint errors on the flight discharge parameters. The effects compete, and result in negligible changes in these parameters due to the small flow and beam current errors.

**Ion Optics Performance**

The ion optics appear to be performing very well so far in flight. The accelerator grid impingement current as a function of beam current is compared to ground test data in Fig. (17). The standard error of these measurements is about 0.03 mA. The data obtained in the ground test facilities are higher because they include a contribution from charge exchange reactions with residual tank gas. The initial impingement current levels in space were about 0.4 mA lower at 0.51 A and 1.7 mA lower at 1.5 A compared to pre-flight measurements in the JPL endurance test facility, which operates at pressure levels of 2–5$\times10^{-4}$ Pa (1.5–4$\times10^{-6}$ Torr) over the full throttle range. Accelerator grid erosion measurements obtained in long duration tests in this facility are therefore conservative. Data obtained in VF 5 at NASA GRC, which has a residual gas pressure about 3 times lower than that at JPL, show impingement currents which are about 0.4 mA greater than the initial space values. The impingement current increased slightly after the first 430 hours of operation and is now comparable to the lowest values measured on the ground. The ratio of impingement current to beam current is shown as a function of beam current in Fig. (18). This parameter, which is used in some probabilistic models of accelerator grid erosion [15, 16, 17] ranges from 0.19 percent at 0.51 A to 0.3 percent at 1.5 A with a standard deviation of 0.012 percent.

Only 172 high voltage faults have occurred during the entire 11200 hours of engine operation (excluding those that occurred as a result of the initial grid short) and in the last 3700 hours there has been only one recycle event. There has been no evidence of electron backstreaming. The discharge loss has consistently increased slightly when the accelerator grid voltage is raised from -250 V after ignition to the throttle setpoint, which is the nominal behavior. This transition is monitored for decreases in the discharge loss, which could signal the loss of electron backstreaming margin.

**Neutralizer Performance**

The neutralizer power consumption has been 4–7 W lower than the BOL throttle table values due to a lower neutralizer keeper voltage, shown in Fig. (19). This power savings roughly compensates for a higher beam power demand due to the beam current offset. The voltage dropped by about 0.5 V over several days before many of these data were taken in IAT1. The IAT1 data show that at that point in the mission the keeper voltage was up to 2 V less than the ground test values. This difference is not yet understood. The voltage decreased with time over the first 4155 hours as it typically does in ground tests [7, 6, 14]. During operation at lower power levels over the next 8500 hours, however, the neutralizer keeper voltage increased by 1–2 V.

In May, 2001 after 11000 hours of operation the IPS Diagnostics System (IDS) detected changes in the plasma environment generated by the engine. The IDS package includes a retarding potential analyzer (RPA) sensor (originally built for the IAPS mission) to characterize the charge exchange ion environment due to IPS operation on DS1. The orientation of the RPA is such that ions from both the zone downstream from the accelerator grid and from the neutralizer are detected. For most of the DS1 mission, the RPA detected ions from the neutralizer start-up, followed shortly by a substantial charge-exchange ion current generated by the IPS beam. The data typically showed that the charge-exchange ion current gradually drops to a value less than one-half the initial current as the cathode plenum pressure bleeds down from the start-up flow condition to steady-state thrust condition. Typical transients are shown by solid lines in Fig. (20). In May 2001,

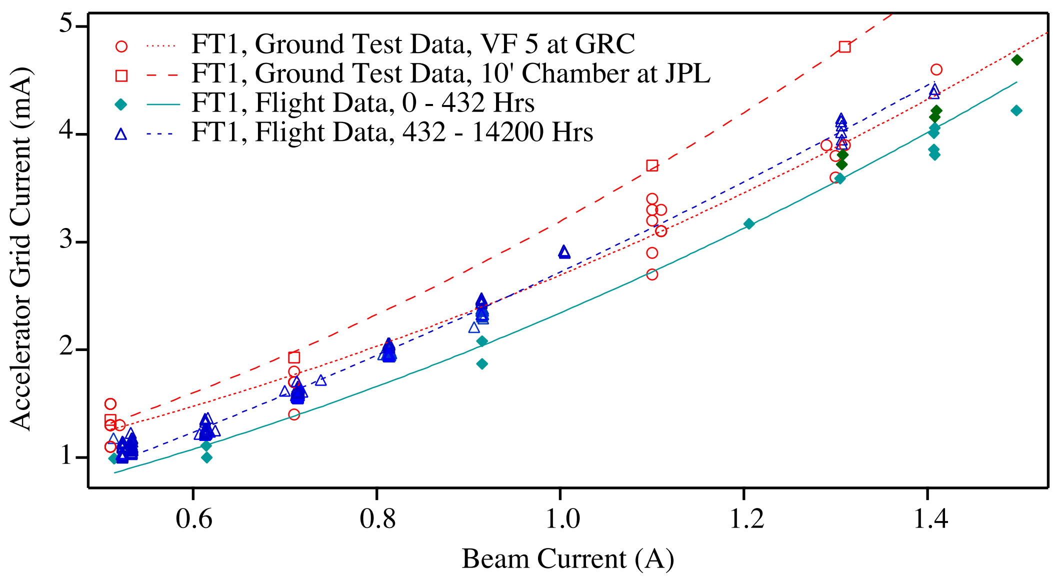


Figure 17: Accelerator grid impingement current measured in space compared to ground test measurements.

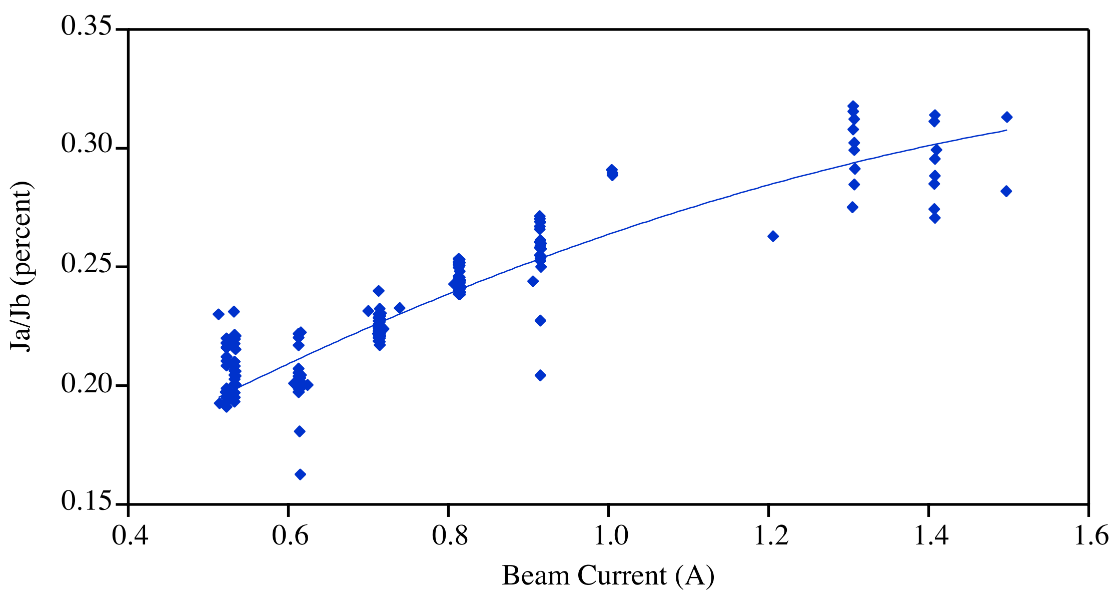


Figure 18: In-space ratio of accelerator grid impingement current to beam current.

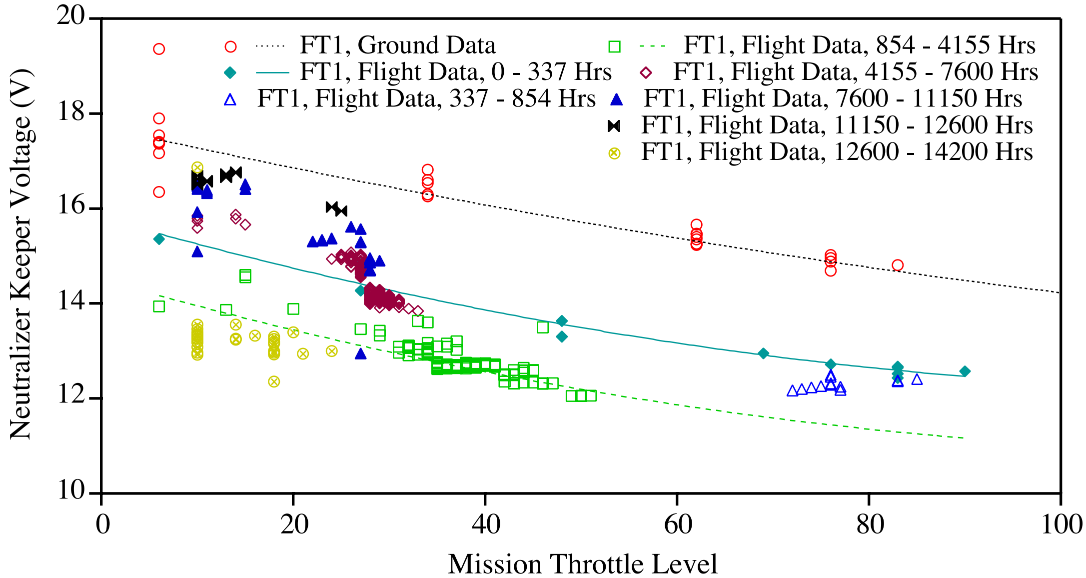


Figure 19: Neutralizer keeper voltage measured in space and in ground tests.

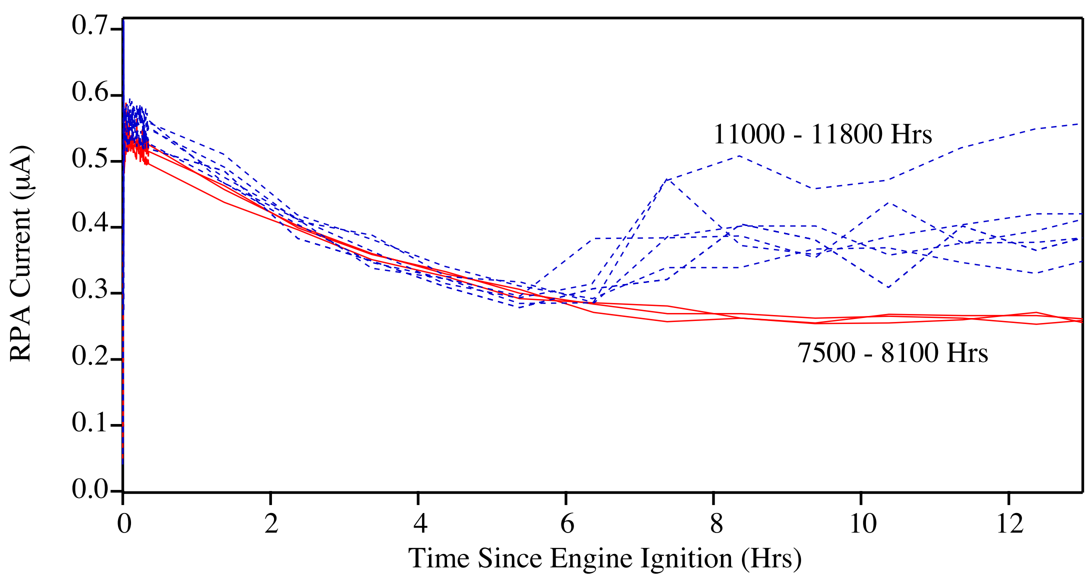


Figure 20: Total ion current collected by the retarding potential analyzer as a function of time after engine ignition.

this RPA total ion current signature was observed to change. Approximately 6 hours after start-up, the ion current increased rather than leveling out at the typical steady-state value. Over the course of the month, this increase became more pronounced, until the RPA ion currents even exceeded the initial start-up value as shown by the dashed lines in Fig. (20).

The RPA also provides ion energy distribution information. Prior to the changes observed in May, the ion energy distribution remained constant during the cathode plenum bleed down, with only the total current changing. Figure (21) shows the energy distribution 2 minutes after ignition and 10 hours later. In data taken after 11000 hours, the energy distribution 10 hours after start-up reveals a higher energy component. These transitions in behavior appeared to occur when the neutralizer flow rate dropped below a certain threshold value, suggesting that the source of the ions could be from the neutralizer plume, rather than from the charge-exchange zone in the ion beam.

In the 8200 hour wear test of an engineering model NSTAR engine, the neutralizer experienced some orifice erosion and had little flow rate margin for the spot-to-plume mode transition by the end of the test [6]. The IDS results might indicate that the flight neutralizer was approaching this margin. To prevent the possibility of accelerated wear of the neutralizer by high energy ions, the DS1/IPS operations team specified higher steady state cathode flow rates in a new throttle table. IDS measurements after this change demonstrated ion current magnitudes and energy distributions that were similar to those observed early in the flight. The neutralizer keeper voltage also dropped back to the previous values, as shown in Fig. (19).

There is no instrumentation on the DS1 spacecraft that allows the true neutralizer coupling voltage to be easily determined. The voltage of neutralizer common with respect to the spacecraft ground is metered, and the behavior is shown in Fig. (22). The magnitude of this potential has increased slowly over the mission and is now about zero. To properly compare this with the ground measurements of coupling voltage, the spacecraft potential with respect to the ambient plasma must be known. Data from the Plasma Experiment for Planetary Exploration (PEPE), another experiment on DS1, suggest that the spacecraft potential is within 8 V of the ambient potential. The coupling voltage in space therefore appears to be less than 8 V. This is somewhat lower than that observed in ground tests, which is typically 12-14 V.

## Mission Operations

Although the total thruster operating time so far has been orders of magnitude longer than that required by impulsive propulsion systems, the mission operations demands have been minimal. This is largely due to the successful implementation of a high degree of spacecraft autonomy.

Autonomous navigation has significantly reduced the demands on the navigation and trajectory design teams and spacecraft control of the IPS relieves the ground controllers considerably. In the initial phase of the mission a number of propulsion engineers were involved in mission operations and validation. However, the final NBURNs have become sufficiently routine at this point that very little workforce is assigned to this area. The flight data dissemination and analysis has also been largely automated. During the initial, nearly continuous Deep Space Network coverage the spacecraft telemetry was displayed in real-time on a website that could be accessed by the flight team. Data are also stored in the JPL ground data system and automatic queries generate files that are sent via ftp to all flight team members. A series of macros in Igor Pro software are used to automatically load, analyze and plot these data. The success in reducing mission operations requirements with automation is an extremely significant result, because the fear of excessive operations costs has been a major barrier to the acceptance of ion propulsion for planetary missions. This flight demonstrates that mission operations costs for SEP-driven spacecraft are similar to those for conventional spacecraft or possibly less in cases where ion propulsion results in shorter trip times.

## Conclusions

The test of ion propulsion on the Deep Space 1 mission has been extremely successful so far. All

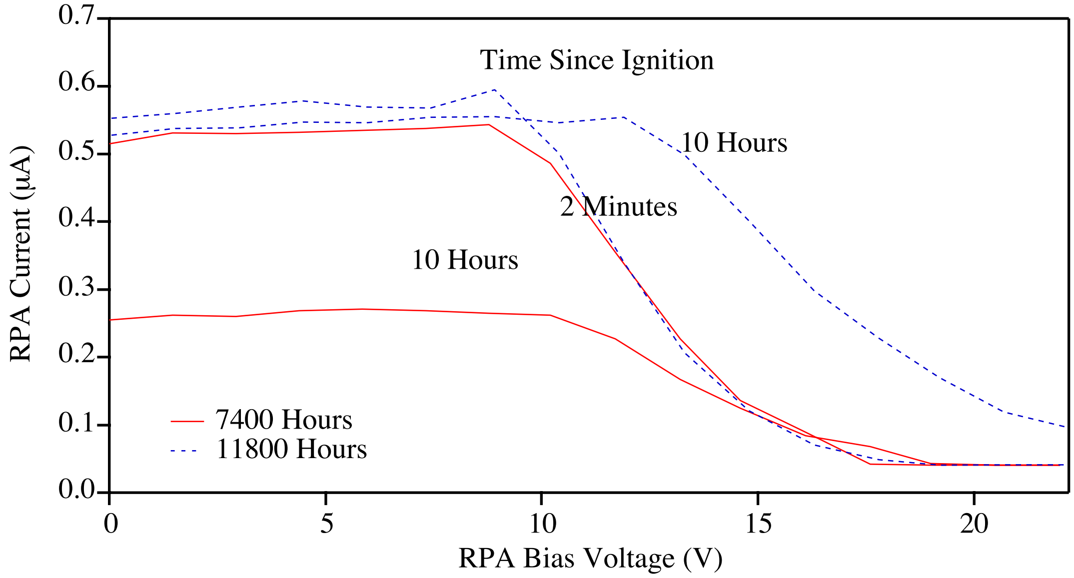


Figure 21: Ion energy distributions measured with the retarding potential analyzer.

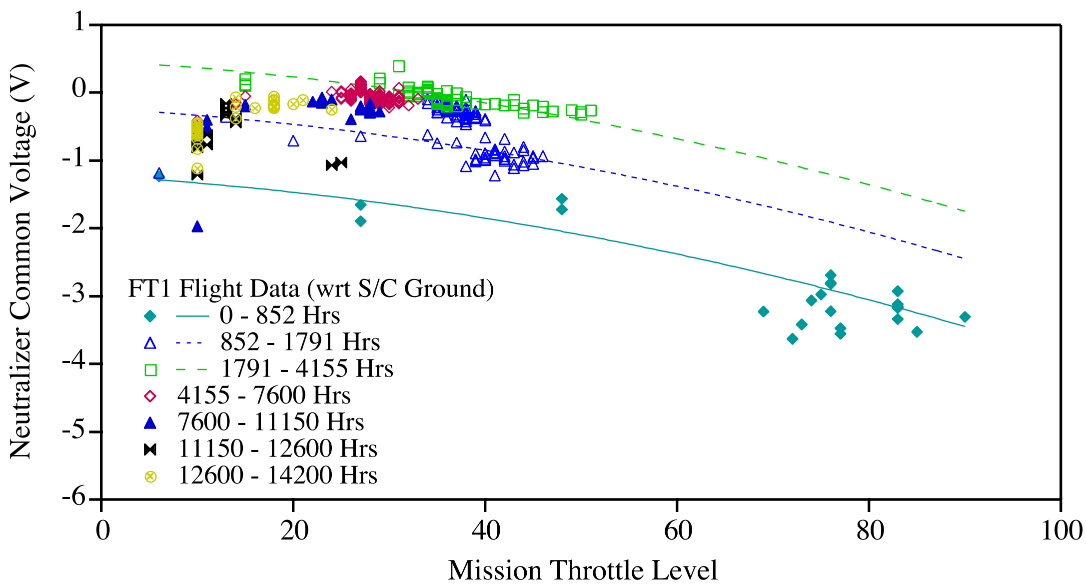


Figure 22: Neutralizer common voltage measured with respect to spacecraft ground in space.

normal IPS functions and some of the fault recovery modes have been demonstrated over a total of 14200 hours of operation. The differences between system performance and engine operating characteristics in space and in ground tests have been very small. The thrust is slightly lower than the throttle table values at the higher power levels because the double ion content of the beam is somewhat higher than initially estimates. The PPU efficiency appears to be higher than the conservative values assumed in the throttling tables. Measurements obtained with the NSTAR diagnostics package after about 11000 hours of operation indicated that the neutralizer was producing more high energy ions. To minimize the possibility of neutralizer erosion the throttle table was modified to allow higher neutralizer flow rates at low power levels. Subsequent diagnostics measurements confirm that this eliminated the high energy ions observed previously. Fully autonomous navigation and operation of the IPS have been demonstrated, achieving the goal of minimizing the required ground support for low thrust propulsion systems. This flight validates ion propulsion technology for use on future interplanetary spacecraft and has provided a wealth of information for future mission and spacecraft designers.

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