

Qualification of Commercial XIPS[®] Ion Thrusters for NASA Deep Space Missions

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Electric propulsion systems based on commercial ion and Hall thrusters have the potential for significantly reducing the cost and schedule-risk of Ion Propulsion Systems (IPS) for deep space missions. The large fleet of geosynchronous communication satellites that use SEP, which will approach 40 satellites by year-end, demonstrates the significant level of technical maturity and spaceflight heritage achieved by the commercial IPS systems. A program to delta-qualify XIPS[®] ion thrusters for deep space missions is underway at JPL. This program includes modeling of the thruster grid and cathode life, environmental testing of a 25-cm EM thruster over DAWN-like vibration and temperature profiles, and wear testing of the thruster cathodes to demonstrate the life and benchmark the model results. This paper will present the delta-qualification status of the XIPS thruster and discuss the life and reliability with respect to known failure mechanisms.

I. Introduction

Solar Electric Propulsion (SEP) can provide advantageous performance for high Δ -v deep space missions. This is well illustrated by the DAWN mission presently flying where NASA NSTAR ion thrusters provide all of the post-launch Δ -v related to heliocentric transfer to the asteroids Vesta and Ceres, in addition to orbit capture, orbit transfer, de-orbit maneuvering around these asteroids and the capability of attitude control and reaction-wheel unloading during flight¹. Previous studies have demonstrated that U.S. commercial electric propulsion (EP) systems have advantageous performance in terms of power capability, throttle range, and efficiency compared to the systems NASA and ESA are flying to date, and the commercial systems also have the potential to significantly reduce the cost and schedule-risk of the Ion Propulsion System (IPS) for deep space missions^{2,3}. The fleet of geosynchronous communication satellites that use SEP increases every year and is now approaching 40 satellites and over 150 thrusters, which demonstrates a significant measure of technical maturity and flight heritage for these commercial IPS systems that can be applied toward NASA applications. The European Space Agency (ESA) has already implemented the commercially-produced PPS-1350 Hall thruster in its successful SMART-1 lunar mission⁴, demonstrating the viability of commercial station keeping EP hardware to perform in prime-propulsion, deep space applications.

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A methodology for qualifying commercial electric propulsion systems for deep space missions was recently published by Randolph⁵. A program to implement these processes for delta-qualifying XIPS[®] ion thrusters^{6,7} for NASA deep space missions is presently underway at JPL. This program includes modeling of the thruster grid life⁸ and cathode wear⁹, performing environmental delta-qualification tests on a 25-cm EM thruster over vibration and temperature profiles required for deep space applications like DAWN, and benchmark testing of the thruster and cathodes to validate the models and demonstrate the life of the thruster components. A discharge cathode assembly wear test^{10,11} has successfully completed nearly 11,000 hours to date, and the cathode discharge current level has been throttled down from the highest current point to lower power levels in two steps associated with operation at different throttling conditions. Modeling of the XIPS grids has demonstrated the benefits of the 3rd decel grid in essentially eliminating pits-and-groves erosion on the downstream face¹², and improved models of electron backstreaming provide good agreement with life test data. These models are being used to assess the thruster life for candidate deep space mission profiles. The results of the modeling and testing of the XIPS engine for deep space applications will be presented.

II. 25-cm XIPS Performance

The development¹³ and performance^{6,2,14,15} of the 25-cm XIPS thruster has been previously reported. The 25-cm thruster and Power-Processing Unit (PPU) are manufactured by L-3 Communications, Electron Technologies Inc. (ETI) in Torrance California. Photographs of the 25-cm thruster and PPU are shown in Figure 1. The XIPS gimbal, which was patented¹⁶ and is manufactured by Boeing, is shown in Figure 2. There are now 15 of the Boeing 702 communications satellites in orbit with four XIPS thrusters and two PPUs on each satellite (a total of sixty 25-cm thrusters and thirty PPUs operating in orbit to date). A comprehensive description of the XIPS thruster production and manufacturing process was provided by Chien⁷. The fact that the XIPS thruster and PPU are manufactured continuously by a commercial vendor at rates of up to four thrusters and two PPUs per month provides a strong indication of the robustness of the source supplier with respect to engine/power supply component availability and reproducibility for NASA applications. In addition, long range, multi-year procurement orders from non-NASA customers also help ensure the future availability of these same components with some reasonable assurances of cost-reproducibility.

In communications satellite applications where the EP system is used for orbit raising and station keeping, the performance of the 25-cm XIPS thruster is summarized¹⁴ in Table 1. The XIPS thruster normally operates in this case at two different power levels, with a thrust of 80 or 166 mN, an Isp between 3400 and 3600 s, and a total efficiency of over 67%. In all spacecraft applications, the total system efficiency, thrust and Isp of the EP system



Figure 1. Photograph of the 25-cm XIPS thruster and flight Power Processing Unit (PPU).

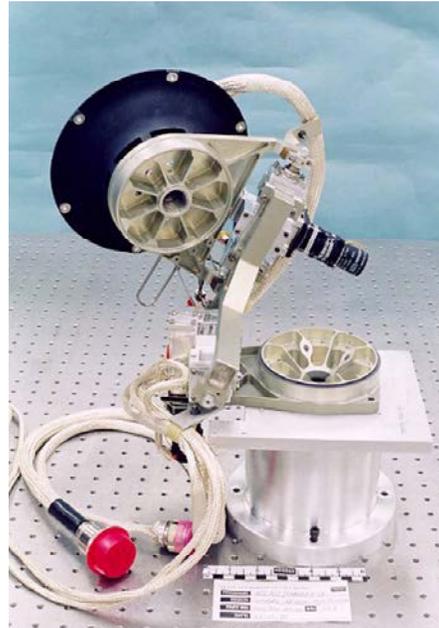


Figure 2. XIPS gimbal manufactured by Boeing.

versus the input power level to the Power Processing Unit (PPU) are the important parameters, and so the PPU efficiency must be taken into account in the performance specifications. The XIPS PPU parameters in the Boeing communications satellite application are summarized⁶ in Table 2.

Table 1. 25-cm XIPS thruster performance in Communication Satellite applications⁶.

| | Low Power Station Keeping | High Power Orbit Raising |
|---------------------------------|------------------------------|-----------------------------|
| Active grid diameter (cm) | 25 | 25 |
| Thruster Input Power (kW) | 2.0 | 4.2 |
| Average ISP (seconds) | 3420 | 3550 |
| Thrust (mN) | 80 | 166 |
| Total Efficiency (%) | 67 | 68.8 |
| Mass Utilization Efficiency (%) | 80 | 82.5 |
| Electrical Efficiency (%) | 87.1 | 87.5 |
| Beam Voltage (V) | 1215 | 1215 |
| Beam Current (A) | 1.45 | 3.05 |

Table 2. Typical parameters of the 25-cm XIPS power processing unit⁶.

| Parameter | Performance Low Power | Performance High Power |
|------------------------|--------------------------|---------------------------|
| Total Input Power (kW) | 2.3 | 4.5 |
| Bus Input Voltage (V) | 100 | 100 |
| Efficiency (%) | 91 | 93 |
| Size (cm) | 20.6 x 54.1 x 35.3 | |
| Mass (kg) | 21.3 | |

For deep space applications where throttling of the engine power is required, the XIPS ion thruster performance from 0.4 to 5 kW input power levels has been reported⁶. The total thruster efficiency versus input power to the PPU from this work is shown² in Figure 3. Even though the NSTAR and XIPS thrusters have a similar electrical design and common development heritage, the XIPS based IPS has demonstrated over twice the throttling range as NSTAR and slightly higher efficiency over this range. This information, along with the thrust and Isp variation with PPU

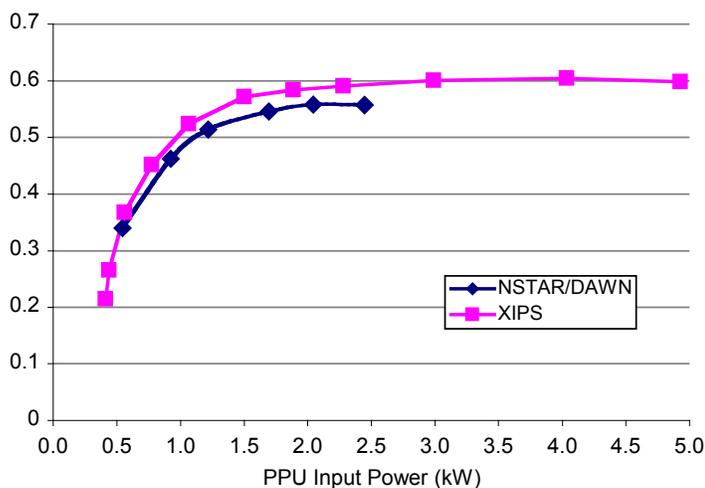


Figure 3. Total efficiency of the 25-cm XIPS thruster and the NSTAR thruster as used on DAWN versus PPU input power.

input power, is used for mission planning and performance analysis. Curve fits to the data for thrust, Isp and efficiency versus input power used in the present JPL mission studies² are shown in Figure 4. These plots include data on the BPT-4000 Hall thruster³ and the NEXT ion thruster¹⁷ for comparison. The XIPS thruster has significant performance advantages over the NSTAR thruster, and costs significantly less² than the higher performing NEXT thruster.

The mission performance benefits achieved using XIPS thrusters has been evaluated for several reference missions². Figure 5 shows the burnout mass,

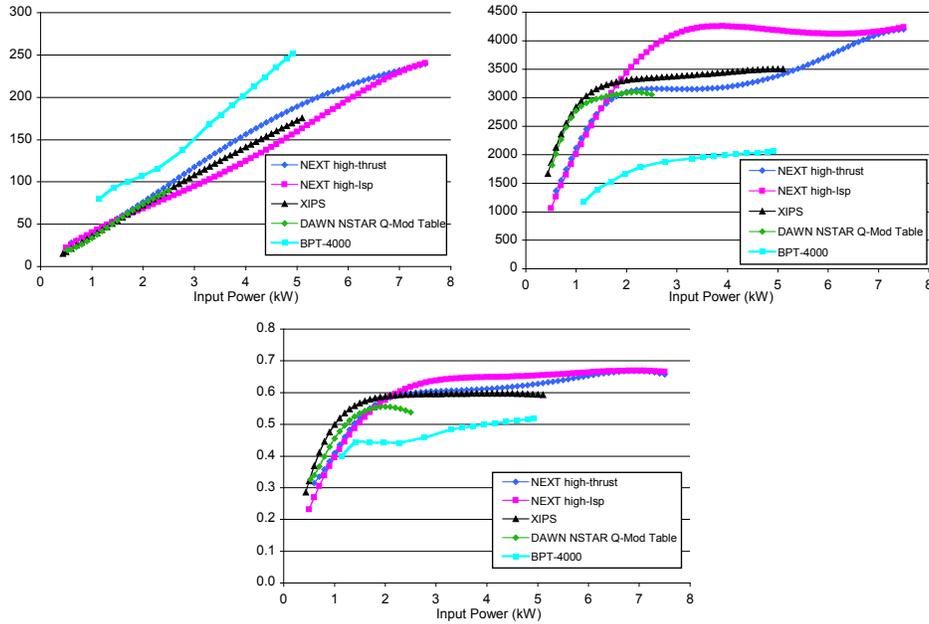


Figure 4. Curve fits to the data² for thrust, Isp and efficiency for the 25-cm XIPS, 30-cm NSTAR, and 36-cm NEXT ion thruster and the BPT-4000 Hall thruster versus PPU input power level.

defined as the total mass of the spacecraft at its final destination including the payload, propulsion system, and residual propellant, for two of the reference missions analyzed. The higher power level and greater throttling range of the XIPS engine produces a higher delivered mass than a spacecraft using an NSTAR engine, and also provides the same burn out mass as an ion propulsion system using twice the number of NSTAR thrusters. The lower cost of the XIPS engine compared to the infrequently built NSTAR thruster, in addition to the possibility of using few engines, reduces the cost of the IPS system².

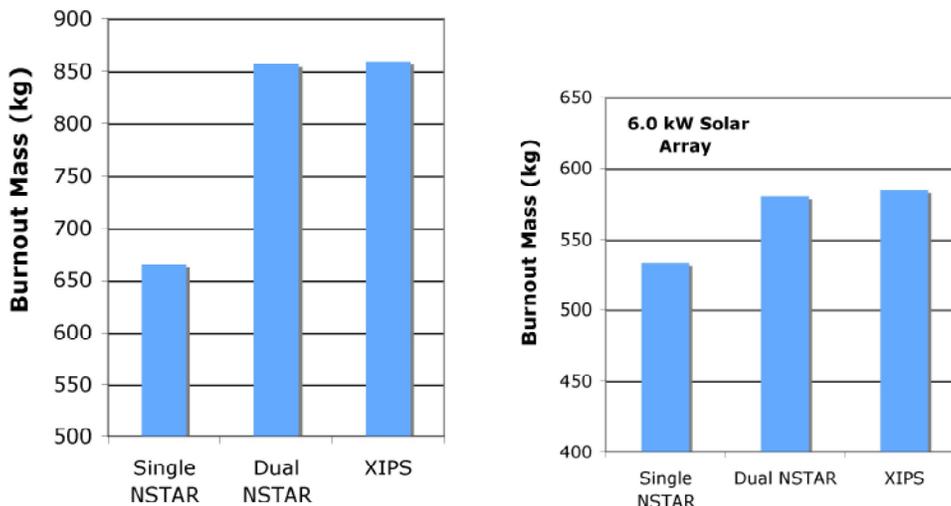


Figure 5. Calculated² burnout mass for a Near Earth Asteroid Sample Return Mission (left) and a Comet Rendezvous Mission (right) showing the benefit of the higher performance of XIPS compared to NSTAR.

III. Qualification

A methodology for qualifying commercial electric propulsion systems for deep space missions was recently published by Randolph⁵, and a JPL Standard for Life Qualification for thrusters is in preparation¹⁸. For delta-qualifying the XIPS ion thrusters, there are several topics that are presently being addressed. These are:

- Environmental
 - Dynamic
 - Thermal
- Mission Assurance
 - Life
 - Materials analysis and certification
 - Reliability
- Electromagnetic Compatibility

It is common at JPL to use the NASA Deep Space-1 and DAWN parameters for reference in defining the requirements for the above items. The potential use of larger launch vehicles in potential missions that might consider electric propulsion, such as the Delta IV and the Atlas V series of launch vehicles, will expand some of the vibration requirements for the IPS systems. Each of these qualification topics will now be discussed and assessed with respect to delta-qualifying XIPS thrusters for NASA deep space missions.

A. Dynamic Environment

Requirements for the dynamic environment associated with launch or pyro-shock (related with mechanism deployment) are largely determined by the launch vehicle and spacecraft configuration. Table 3 shows the present JPL requirements for random vibration, and represents the reference parameters for the XIPS vibrate delta-qualifications. Different launch vehicles will have somewhat different requirements, which are generally enveloped for the various Delta- IV and Atlas V launch-vehicles under consideration for cost-capped missions by the “Generic” levels shown in Table 3. The assemblies must be designed and verified to have primary mode frequencies above 80 Hz. The random vibration inputs for the cases in Table 3 must be applied to the part or assembly under test in each of three orthogonal axes, and the vibration amplitude distribution should be Gaussian.

Table 3. Random vibration levels for EP system components

| Component | DAWN IPS Requirement (Grms) | Generic Requirement (Grms) |
|------------------------------|-----------------------------|----------------------------|
| Thruster | 8.4 | 10 |
| Gimbal | 8.4 | 10 |
| Power Processing Unit | 8.4 | 14 |
| Propellant Management System | 13 | 13 |

For random vibrate, the XIPS thruster and PPU have been tested by industry to slightly different levels than indicated in the table. Figure 6 shows the thruster and PPU vibration levels versus frequency for DAWN, the Generic specification and the XIPS 25-cm thruster and PPU qualifications both In-plane to the launch axis and Normal to the axis. The thruster partially meets the random vibration requirements over the mid-frequency regime for normal mode vibration and the PPU meets or exceeds the requirements over most of the frequency domain for normal mode vibration. While the values that the XIPS engine has been qualified to are slightly different than the generic specification, the difference is not considered significant. A re-test of the 25-cm thruster to verify compliance with these vibration specifications is underway at L-3 Communications, and will be completed shortly. Likewise, the XIPS PPU qualification levels are also slightly different than the DAWN and generic specifications, and in some cases exceed the requirement. These differences are also not considered significant, and further testing is planned to demonstrate the PPU's ability to handle the generic level vibration requirements.

Shock design requirements are dependent on the launch vehicle, device or assembly configuration and type of pyrotechnic device. Pyro-shock requirements are usually defined in terms of shock response spectrum (SRS), which represents the structurally transmitted transients from pyrotechnic devices used for launch vehicle separation and deployments of various mechanisms. The shock pulse should have a time history that is oscillatory and decays to less than 10% of its peak value in about 20 ms, and must be applied at the assembly interface in each of three

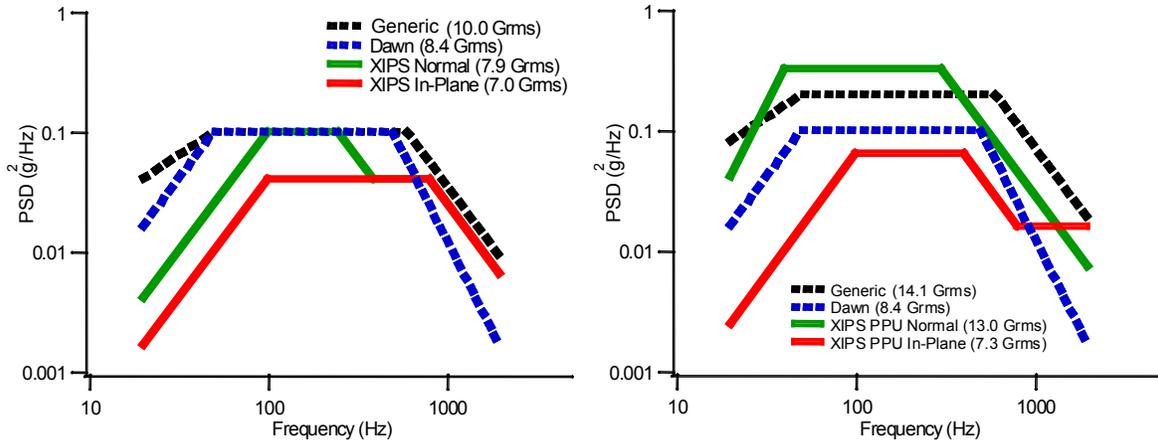


Figure 6. Random vibration specification and qualification levels for the XIPS thruster and PPU.

orthogonal axes. An example of the SRS shock requirements for thrusters and PPUs is given in Figure 7. Again, the XIPS qualification level is slightly different, but this is not considered significant. Additional testing and an evaluation of the PPU shock requirements are planned to verify this.

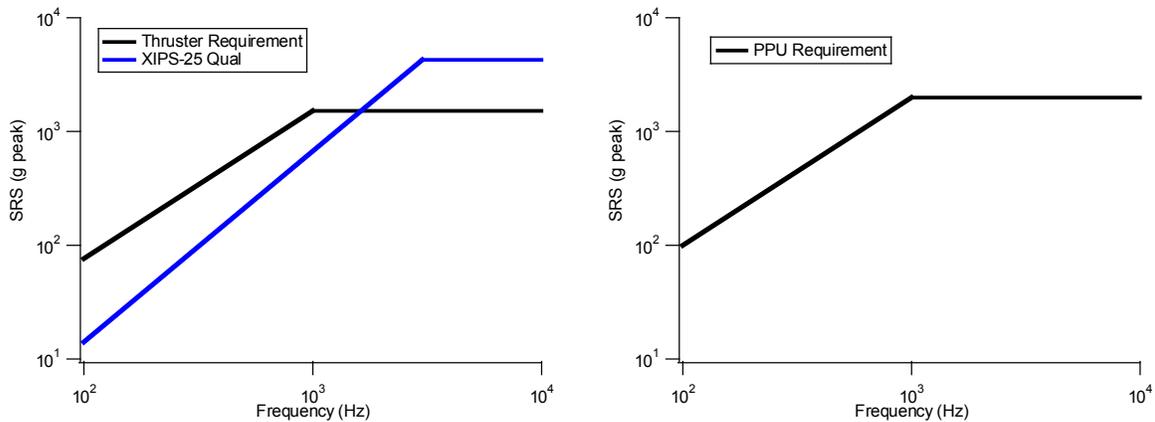


Figure 7. Shock requirements for the thruster and PPU versus frequency.

B. Thermal Environment

The qualification requirements for the thermal environment are complicated by the different spacecraft configurations and different test conditions in the various systems deployed to date. Thruster components located inside the spacecraft, such as the Power Processing Unit (PPU), Digital Control Interface Unit (DCIU) and Xenon Flow System (XFS), will have requirements and environments dominated by the spacecraft bus parameters. The active and passive thermal systems of the spacecraft typically control the temperature of these components. Once the requirements of the SEP components internal to the spacecraft are specified, any difference between the NASA specification and the commercial specification are usually straightforward to handle by the inclusion of strategically placed heaters or modification to the conduction and radiator design. The externally mounted SEP components, specifically the thruster and gimbal assembly, may have appreciably different thermal environments between Earth-orbit applications and deep space missions. Thermal environments associated with geosynchronous orbits are usually dominated by the solar radiation on the thruster and gimbal (at 1 AU) and during eclipse sessions for heating and cooling extremes. In addition, north-south station keeping is often accomplished by SEP burns on a daily basis¹³, which results in much more thermal cycling of the thruster than is experienced in deep space missions. In addition, deep space missions may utilize Venus or Earth gravity assists, which can expose the thruster/gimbal

assembly to multiple mission operation timelines with heat fluxes equal to or significantly exceeding the geosynchronous solar heat flux at 1 AU.

Another factor complicating any delta-qualification for thermal environments is related to differences between the location of the temperature measurements and temperature reference-points on different thrusters. For example, the NSTAR ion thruster reference point for the operating temperature is on the front mask of the thruster, which has a full view to both deep space and the sun depending on the spacecraft orientation. In addition, this location is relatively close to the first magnet ring, which is the hottest part in the thruster. In contrast, the reference location for the XIPS ion thruster and the BPT-4000 Hall thruster is on the rear interface to the gimbal mount, which does not have such a view and experiences direct conduction to the gimbal assembly. The XIPS and Hall thruster reference location is also significantly farther away from the hottest parts of these thrusters, which will change the indicated temperatures compared to the NSTAR/DAWN case. As observed by Randolph⁴, this situation illustrates how the thermal environment of externally mounted components (the thruster and gimbal) must be evaluated on an individual basis within the complete framework of the spacecraft thermal configuration. It also dictates the need for a good thermal model of the thruster and gimbal to relate the measured thermal conditions in testing to the spacecraft environment for the specific mission.

The thermal specifications for the SEP hardware on DAWN are listed in Table 4. The larger specified range of the minimum and maximum temperatures for the external components, the thruster and gimbal, is indicative of their more direct exposure to space as described above. The PPU is mounted internally to the spacecraft, and temperature control over the specified range is provided directly by the spacecraft thermal system in order to meet the requirements for this electronic component. The NSTAR PPU has a relatively low maximum operating temperature (<45 °C), which is a challenge to satisfy in the spacecraft thermal-system design. A direct comparison of the present qualification levels for the measured reference temperature locations of the XIPS and DAWN thruster, gimbal and PPU is shown in Figure 8. The XIPS thruster has a higher qualification temperature associated with its higher power operation. The low qualification temperature for the XIPS thruster, -50°C, is a requirement of the Boeing application, and not a limitation of the thruster. XIPS thrusters have been operated below -100 °C without problem, and no issues with low temperature exposure are expected since the thruster shares materials and design heritage from NSTAR. The XIPS PPU was qualified at 89 °C and can operate at up to about 80 °C, which is significantly easier for the spacecraft to provide compared the NSTAR PPU limitation. The non-operating PPU low temperature specification from DAWN is not considered an issue for the XIPS PPU. The XIPS gimbal has a smaller temperature range than the DAWN gimbal, which need only be evaluated if the mission profile and thermal model require this large a range. Finally, Figure 8 also shows the number of thermal cycles required for DAWN for the thruster, gimbal and PPU, and the XIPS qualification number. The daily stationkeeping requirement for thruster operation dictates many more cycles than experienced in deep space prime propulsion applications, and the commercial EP components are qualified well in excess of the NASA requirement.

Table 4. Temperature ranges specified for the DAWN IPS components.

| Component | Minimum Temperature (°C) | Maximum Temperature (°C) |
|---------------------------------------|--------------------------|--------------------------|
| Thruster | -108 | 158 |
| Power Processing Unit (operating) | -20 | 55 |
| Power Processing Unit (non-operating) | -40 | 70 |
| Gimbal | -108 | 120 |

C. Life

The life of ion thrusters is of concern for deep space missions due to the high throughput and long operation time typically required of most missions². The concerns for thruster life are centered on the potential failure mechanisms in the engine and the related wear-out mechanisms that lead to failure. Since ion thrusters have been in development for about 50 years and have accumulated over 100,000 hours of life testing to date, the failure mechanisms for this type of engine are very well known. A historical survey of the identified failure mechanisms and life tests has been produced by Brophy and Polk¹⁸. The credible failure mechanisms that still exist for ion thrusters, i.e. the ones that have not been designed out of the thruster or eliminated by manufacturing process modifications, and their causes are listed below and grouped in terms of the three main components of an ion thruster:

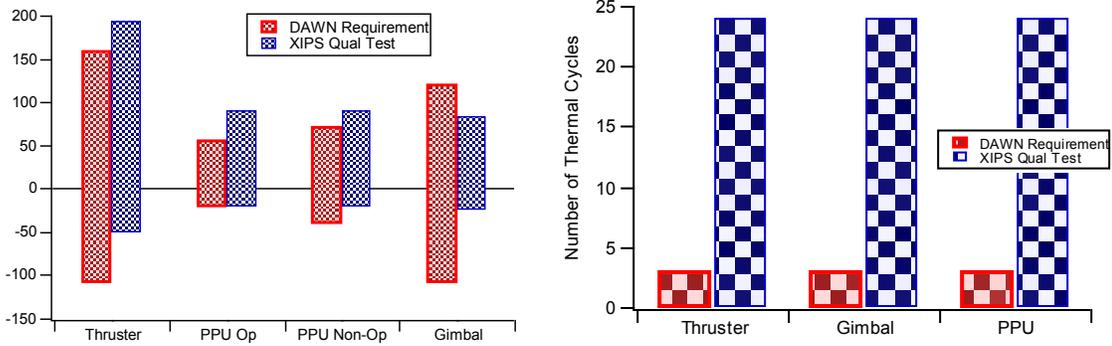


Figure 8. Thermal specifications and thermal cycling levels for the XIPS thruster, gimbal and PPU compared to the DAWN requirements.

Ion Optics

1. Electron Backstreaming
 - a) accelerator grid barrel erosion caused by charge exchange ions
 - b) pits and groves erosion from downstream sputtering of the accel grid
 - c) rogue hole formation from delaminated flakes or grid-manufacturing-produced slivers
 - d) grid gap closure due to long term stress release or loss of curvature due to material removal
2. Grid Shorting
 - a) flakes from the discharge chamber
 - b) flakes from launch debris
 - c) flakes from delamination of sputtered material from the other grids
3. Grid Damage or Structural Failure
 - a) excessive arc energy from PPU
 - b) structural failure from ion sputtering (screen and accel)
4. Electrical Breakdown
 - a) material deposition on insulators
 - b) surface modification of the grids versus time (field emission site evolution, flake evolution, etc.)
 - c) grid gap changes versus time (see 1d above)
 - d) plasma screen shorts to high voltage electrodes due to thermally-induced gap closure or flakes

Hollow Cathodes

1. Heater Failure
 - a) insulation degradation
 - b) conductor failure (“filamentation” and/or burn-out)
 - c) sputter erosion of sheath
2. Thermionic Emitter Failure
 - a) Barium depletion
 - b) Insert poisoning
 - c) Tungstate formation
 - d) Insert surface morphology modification
3. Electrode Erosion
 - a) Cathode orifice erosion
 - b) Keeper orifice or face erosion
4. Shorts or Electrical Breakdown
 - a) Cathode to keeper shorts due to flakes or deposits in the gap
 - b) Insulator deposition or coating
5. Cathode Orifice Plugging
 - a) Low power operation of neutralizer cathode

Discharge Chamber

1. Magnet Degradation due to time at temperature
2. Insulation Failure
 - a) sputter deposition on stand-off insulators
 - b) wire failure

While the XIPS ion thruster, like all ion thrusters, are susceptible to these failure mechanisms, the 25-cm XIPS thruster is a second-generation design based on the initial work of producing, qualifying and flying the 13-cm XIPS thrusters in the 1990's. Extensive design upgrades, process improvements and the manufacturing and testing of over 70 of the 25-cm thrusters to date have resulted in the elimination of some of these failure modes and mitigation of many others. The XIPS thrusters have also undergone several life tests, the results of which are summarized⁷ in Table 5. The assessment of the 25-cm XIPS thruster life with respect to the above failure mechanisms will be discussed next.

Table 5. Life test results⁷ of the XIPS 13 cm and 25 cm ion thrusters.

| Thruster | Power Level | Operation Hours | Cycles |
|-------------|-------------|-----------------|--------|
| 13 cm (Q1) | 0.5 kW | 16,146 | 3,275 |
| 13 cm (Q20) | 0.5 kW | 21,058 | 3,369 |
| 25 cm | 4.2 kW | 2,680 | 324 |
| 25 cm | 2.0 kW | 13,350 | 13,654 |

1. Ion Optics

The XIPS grids and ion optics assembly are designed and fabricated to eliminate many of the failure modes listed in the previous section. First, XIPS uses a three-grid geometry in that a third “decel” grid is incorporated into the assembly. This grid physically shields the negatively biased accel grid from backstreaming charge-exchange-produced ion bombardment from the beam that causes the characteristic “pits and groves” erosion of the accel grid in two-grid designs. The decel grid is biased near the neutralizer cathode-common potential, which is within 50 V of the spacecraft potential. This reduces the back-flowing ion energy impinging on the decel grid to the order of tens of volts, compared to hundreds of volts if allowed to bombard the accel grid. The low backstreaming ion energy reduces ion sputtering of the downstream face of the grids to negligible levels⁸, which eliminates Failure Mechanism 1b and mitigates 1d under Ion Optics above. Figure 9 shows the downstream Pits and Groves erosion rates calculated by Wirz⁸ using the JPL CEX code for the 3-grid XIPS system and the 2-grid NSTAR system. The 3rd grid in the XIPS system reduces the downstream grid face erosion to negligible levels.

To mitigate other ion optics failure mechanisms, the XIPS grids are manufactured with a thermal stress relief step after forming of the “dome” radius of curvature (ion thruster grids are domed to survive launch vibration and provide stable gaps during thermal expansion). The grids are also mounted on flex-structures⁷ that permit radial expansion due to heating of the grids during

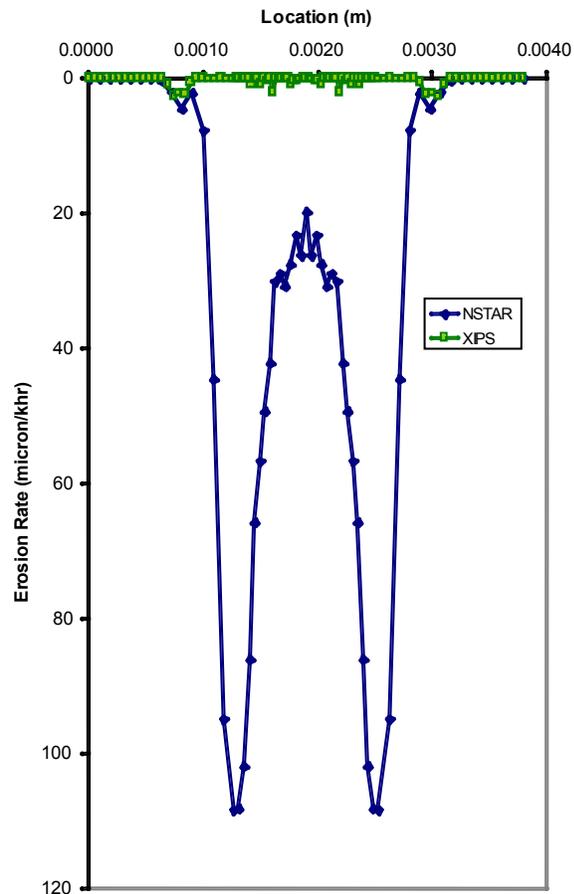


Figure 9. Calculated Pits and Groves erosion rates⁸ showing the XIPS 3rd grid effectiveness compared to the 2-grid NSTAR erosion rate.

operation without causing the grid gap to change. The stress-relieving manufacturing step and the flex-mounting of the grids improves the grid gap stable over time and temperature, which largely eliminates Failure Mechanism 1d under Ion Optics above. Direct grid gap measurement data is to be collected on life tested grids at L-3 in the coming months and should provide further evidence of this projected grid gap stability

Rogue hole formation and electrical shorting has been observed during development of the XIPS thrusters due to delamination of sputter-deposited materials on the grids. This was the primary failure mechanism of the first 13-cm XIPS thruster life test⁷ after about 16,000 hours of operation. This failure mechanism normally occurs near the end of the life when a significant amount of grid material has been eroded and deposited on a facing grid to create sufficiently thick layers that will delaminate from the surface. This failure mechanism has been mitigated by a modification of the surface processing of the grids to enhance adhesion of the sputter-deposited material. The success of this technique was demonstrated by the successful completion of over 21,000 hours by the second 13-cm life test⁷, and by the successful operation of the NSTAR thruster (which was manufactured by the same vendor and used the same process) for over 30,000 hours in the Extended Life Test¹⁹. It is also possible to cause rogue hole formation by improper etching of the screen grid such that the cusps is undercut and can peel off to form a flake or sliver across the screen aperture. This problem has been mitigated by detailed inspection of the grids after manufacturing and the utilization of sufficient burn-in time to observe undercutting of the cusp that might cause sliver formation leading to rogue hole formation.

The XIPS thrusters avoid grid shorting failure mechanisms in items 2 through 4 under Ion Optics above in the same manner as in the NSTAR thruster. The discharge chamber wall is manufactured to retain sputter-deposited material using the same textured surface-material as in NSTAR. The PPU has a grid-clearing circuit that provides high current pulses to melt and open flakes or whiskers from launch debris or spalled material between the grids. However, the PPU is also designed to limit the energy deposited in the grids to avoid damaging the surfaces. The insulators are all shadow-shielded to avoid material deposition and electrical leakage. The high reliability reported⁷ for the 25-cm XIPS thrusters on orbit illustrate the reliable and robust ion optics design and manufacturing techniques.

Finally, the life of the ion optics and the ultimate throughput capability of the thruster is primarily determined by electron backstreaming (Ion Optics failure mechanism #1) limitations due to accel grid barrel erosion and enlargement, and ion sputtering of the screen and accel grids that leads to structural failure (#3b). The XIPS life tests described above monitored electron backstreaming margin and accel grid erosion. Detailed modeling of this effect using the CEX ion optics code is underway at JPL and is reported in another paper at this conference¹¹. This work will benchmark the ion optics code results with the XIPS life test results to provide accurate life predictions for the thruster, which appears at this date to predict that the XIPS thruster will meet or exceed the life/throughput of the NSTAR thruster. Evaluation of the sputtering of the screen grid is also the subject of another paper at this conference²⁰, and the results will be benchmarked against the XIPS life test results later this year to provide accurate predictions of the grid life.

2. *Hollow Cathodes*

The XIPS discharge and neutralizer hollow cathodes are essentially identical to those used in the NSTAR engine with changes only in the heater and orifice sizes. The XIPS heater has been fully qualified²¹ and tested for a large number of thermal cycles and extended on-time durations. By reducing or eliminating keeper wear (described below) to provide protection of the heater sheath, heater failure (Item 1 under Hollow Cathode failure mechanisms) is an unlikely failure mechanism for the XIPS thrusters.

Potential failure mechanisms related to the thermionic emitter depletion and poisoning have been extensively investigated computationally^{22,23} and experimentally^{24,25} at JPL. The rate at which barium evaporates and the insert is depleted²⁶ in the XIPS discharge hollow cathode, assuming the worse case situation where the barium does not recycle to the surface, is shown in Figure 10 as a function of the cathode discharge current. The cathode operation time and total thruster throughput (shown in the figure) are extremely large for this cathode, and typically exceed mission requirements by a factor between 5 and 10. In reality, barium is recycled in the insert region²³, which greatly extends the life of this cathode against this failure mechanism. Barium depletion is not anticipated to be an issue for the thruster life in deep space applications.

Insert poisoning, tungstate formation and insert surface modifications are potential failure mechanisms related to impurities in the feed gas. Extensive investigations of these effects^{24,25} have been performed. In general, if the normally specified xenon purity level (99.9995%) is provided, based on the results from the ELT tests¹⁹ these failure mechanisms do not occur. The recent experimental investigations are aimed at determining the maximum impurity level that the hollow cathode can tolerate, but this is not an issue for the XIPS qualification effort.

Erosion of the cathode orifice and keeper electrode of the discharge and neutralizer hollow cathodes has also been extensively investigated at JPL. The neutralizer cathode orifice erosion observed in the NSTAR ELT test¹⁹ and the 13-cm XIPS life tests²⁷ has been explained by a comprehensive model published by Mikellides²⁸. Figure 11 shows the measured neutralizer orifice erosion from the LDT²⁹ test of the NSTAR engine and the calculated profile from the model²⁸. The wear stops at about this point because the larger orifice reduces the neutral density and collisionality, which reduces the plasma potential and bombarding ion energy to negligible levels.

Erosion of the discharge cathode orifice is typically small¹⁵ because the ion flux is significantly lower than in a neutralizer cathode and the ion energy is low. This is not considered a failure mechanism for the NSTAR or XIPS cathodes based on the negligible orifice erosion observed in the various life tests of these engines. Keeper erosion, however, is potentially an issue for any ion thruster. The discharge cathode keeper in the ELT test of the NSTAR thruster completed eroded away¹⁹, and significant erosion was observed in the 8200 hr LDT²⁹. Measurements of the keeper wear rate by surface-layer activation³⁰ indicate significant erosion is observed and expected, especially at higher power levels. The cathode keeper wear mechanism has been investigated using a 2-D plasma code by Mikellides³¹ and experimental measurements of the energetic ion generation by Goebel³². This information was applied directly to assessing the XIPS keeper erosion rate⁹. Figure 12 shows the calculated⁹ and measured¹¹ keeper erosion after 5500 hours of wear testing at JPL for two values of the amplitude of the plasma potential oscillations (the term “A”) that produce the energetic ions that cause ion-sputtering erosion. The code predictions for the orifice diameter erosion brackets the experimental data, while the wear if the face exceeds the present prediction. The numerical results produce reasonable agreement with the observed keeper erosion, sufficient to say that the mechanism is understood. This amount of keeper erosion is not deemed¹¹ to be an issue for the cathode life or performance.

However, conversion of the keeper material from Mo to Ta, as was done in the NSTAR engines in DAWN, was shown to mitigate this issue⁹ and cause the erosion to become negligible for deep space missions applications. The wear testing of the discharge cathode is continuing, and has exceeded 10,500 hours to date at two different throttle levels without issue. Based on the modeling and wear test results, and with a flight-heritage material changes available to essentially eliminate the erosion, keeper erosion is not a viable failure mechanism for the XIPS thrusters for deep space missions.

The potential cathode failure

QuickTime™ and a
TIFF (Uncompressed) decompressor
are needed to see this picture.

Figure 10. Barium depletion time calculated for the XIPS discharge hollow cathode as a function of discharge current level showing extensive cathode life.

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are needed to see this picture.

Figure 11. Neutralizer orifice profile from the LDT test²⁸ (open circles) and calculated by the JPL model²⁷ (solid line) illustrating erosion mechanism is well understood.

mechanisms associated with shorts or electrical breakdown (#4) have been mitigated by design of the XIPS cathodes. The insulator is completely shadow shielded and the extensive flight heritage and numerous life tests have demonstrated the effectiveness of this design. The possibility of a short between the cathode and keeper electrode has been mitigated by two features. First, the gap is over twice that used in the NSTAR engine such that significantly larger flakes must be generated to bridge the gap. Second, the XIPS discharge cathode uses a keeper power supply that can provide up to 1 A of current between the cathode and keeper. In the even of a short, this power supply clears the short by passing the current through the flake to melting it. Providing higher current levels is under consideration to completely eliminate this potential issue.

Finally, neutralizer cathode orifice plugging (#5) was observed in the NSTAR ELT¹⁹ at low current levels. This is a potential failure mechanism for neutralizer and discharge cathodes. It is not expected to occur in the XIPS cathodes because the orifices are much larger than in the NSTAR cathodes, and both cathodes utilize active keeper current power supplies to raise the emitter current level, which is known to eliminate the plugging problem. Nevertheless, wear testing of XIPS discharge and neutralizer cathodes is underway at JPL at lower power levels to characterize this potential mechanism and determine the operating parameters that eliminate it.

3. Discharge Chamber

The potential failure mechanisms for the discharge chamber have been mitigated by the XIPS design. The magnets are rated to withstand a temperature of almost 100 °C above their measured temperature during qualification and acceptance testing at the highest power operation. Manufacturer specifications for these magnets indicate that degradation due to time at the qualification temperatures will not occur, eliminating this as a potential failure mechanism. In addition, the wire used in the XIPS thruster is the same as that used in the NSTAR engine, and has been fully qualified for use at the highest power levels of the thruster. The wiring and wire harness is professionally installed and mounted⁷ to avoid exposure to the hottest regions of the thruster, sharp edges or excessive voltage.

D. Materials analysis and certification and E. Reliability

The XIPS thruster has the same heritage as the NSTAR thruster, is manufactured by the same vendor (L-3 Communications ETI), and uses the same materials and manufacturing processes except as previously noted for grid stress relieving. The same is true for the PPU, which are both manufactured by L-3. A complete assessment of the materials and certifications is planned for next year, but these areas are not envisioned to be an issue for qualifying the thrusters because of the NSTAR heritage. The reliability of the thruster and PPU is beyond anything demonstrated by NSTAR due to the large number of thrusters manufactured and flown. With 60 thrusters operating in orbit and another dozen completed awaiting flight, compared to 4 NSTAR thrusters on DS1 and DAWN and one flight spare, the manufacturing variability and reliability statistics of the XIPS thruster are extremely well known. Likewise, there are 30 XIPS PPUs operating in orbit, compared to 3 NSTAR PPUs, and the reliability of this unit is

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Figure 12. XIPS discharge cathode keeper erosion profiles from the 5500-hour wear test¹¹ and the model⁹, where two amplitudes for the plasma potential oscillation levels are used to bound the ion energy.

also well known. This illustrates one of the large advantages of using commercial EP hardware for NASA missions in that a large sample size exists to understand the manufacturing and performance variability.

E. Electromagnetic Compatibility

Electromagnetic compatibility (EMC) requirements are a large part of the qualification of any spacecraft hardware. The electronics box represented by the thruster PPU is typically required to meet the parameters specified in MIL-STD-461, "Electromagnetic Emission and Susceptibility Requirement for the Control of Electromagnetic Interference" modified for specifics of the flight program. The flight subsystems must be tested and qualified for the following EMC areas:

- Radiated Emissions
- Radiated Susceptibility
- Conducted Emissions
- Conducted Susceptibility

As pointed out by Randolph⁵, commercial space hardware is tested against EMC requirements in the same way as in NASA missions and the results evaluated accordingly. Radiated emissions for electric thrusters is typically evaluated with respect to the payload on a spacecraft-case basis. Radiated emissions from electric Hall thrusters and their PPUs have not been found to be a problem³³. The EMC data for the XIPS thrusters and PPUs is held by Boeing Satellite Systems, and no issues with on-orbit performance in communication satellites operating from L to Ka-band has been reported to L-3 Communications. The successful use of XIPS electric propulsion systems on a wide variety of communications satellites with the thrusters operating at the same time as various high bandwidth channels with complex modulation schemes indicates that EMC should not be an issue for delta-qualification. This will be assessed in detail later as the EMC qualification documentation becomes available from Boeing.

IV. Conclusion

In this paper, the status of the qualification effort for using XIPS ion thrusters and PPUs in NASA deep space missions has been described and discussed. Several of the efforts to delta-qualify the XIPS thruster are still in work; namely the environmental testing and life analysis. Wear testing of the discharge and neutralizer cathodes is continuing to provide data to benchmark the erosion and life codes, and to demonstrate the life of the cathodes. Analysis of the valid thruster failure mechanisms for the XIPS thruster indicate that many of these mechanisms have been designed out or mitigated. The thruster life/throughput will be determined by grid wear and electron backstreaming, which is continuing to be analyzed to provide accurate predictions for mission trajectories.

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