DEVELOPMENT AND TESTING OF A HIGH-POWER HALL THRUSTER

Olivier DUCHEMIN* and Pierre DUMAZERT†

SNECMA Moteurs
Space Propulsion Division, Spacecraft Propulsion and Equipment
Villaroche Nord, 77550 Moissy-Cramayel, France

S.D. CLARK‡ and D.H. MUNDY‡

Space Propulsion, QinetiQ, Farnborough, Hampshire, GU14 0LX, The United Kingdom

Abstract

This paper presents a prototype design of a high-power hall-effect thruster under development at SNECMA Moteurs. The PPS®X000 is a thruster with dual-mode capability that will be able to meet the propulsive needs of the next-generation geostationary satellites. Performance characterization testing has been carried out on a variety of magnetic configurations, allowed by the modular design of the thruster. The testing campaign has validated the thermal and magnetic models used in the preliminary design work, where both 2-D and 3-D finite element models have been implemented. The final configuration, power level and nominal operating points of the flight thruster have been determined after completion of the experimental parametric study. A flight model thruster will then be designed, built and tested as a continuation of the SNECMA development and qualification efforts to produce a complete flight-ready plasma propulsion system by end of 2005.

Introduction

Snecma Moteurs has been developing plasma propulsion technologies since 1992, based on a strong partnership with the Russian company FAKEL and the support of the French and European agencies CNES and ESA. The effort was first focused on the development and qualification of the PPS®1350, a 1.5 kW-class Hall Effect Thruster (HET) able to do meet both the requirements of North South Station Keeping for the current biggest geostationary satellites, and of primary propulsion for interplanetary missions.

The PPS®1350 was to be first demonstrated in flight on two plasma propulsion modules integrated on board the CNES *Stentor* technology demonstration satellite. Despite the loss of *Stentor* with the *Ariane 5* flight 157 launch failure, plasma propulsion based on Snecma Moteurs' PPS®1350 will soon have the opportunity to demonstrate its effectiveness by performing the orbit transfer on the ESA *SMART-1* lunar probe. The technology will benefit greatly from this flight experience, which is expected to confirm that Hall-effect-thruster-based electric propulsion sub-systems are very good candidates for both orbit-raising and station keeping on the next generation satellites. Although ion propulsion can provide a higher specific impulse and thus better performance for the particular needs of station keeping, the use of electric propulsion for *both* orbit topping *and* station keeping favors plasma propulsion solutions because of the higher thrust-to-power ratio reached by such systems: a larger fraction of the total transfer velocity increment ΔV can thus be performed by electric propulsion within the transfer time constraints generally imposed for commercial missions.

In order to allow yet larger space platforms with payload powers up to 25 kW to benefit from the use of plasma propulsion, Snecma Moteurs started, in 1999, the development of a new high power thruster – the PPS®5000 – with the support of CNES and in cooperation with the Russian company FAKEL. The development of this thruster is based on the so-called PPS®X000, a prototype model of a high power (up to 6 kW) Hall Effect Thruster with dual-mode capability able to meet the propulsive needs of the next-generation geostationary satellites.

* Senior Research and Development Engineer, Plasma Propulsion Section. Email: olivier.duchemin@snecma.fr
† Supervisor, Plasma Propulsion Section
‡ Senior Electronics Engineer, Electric Propulsion Group

Presented as Paper IEPC-03-0032 at the 28th International Electric Propulsion Conference, Toulouse, FRANCE, 17-21 March 2003.

This paper presents the main characteristics of the PPS®X000, a summary of the results of its characterization tests performed in the new QinetiQ LEEP2 test facility, and the development logic and schedule of the PPS®5000.

1. PPS®X000 design overview

The goals of the PPS®X000 technology demonstrator where twofold, namely :

a) to demonstrate, as close as possible to full scale, the feasibility and effectiveness of the technologies required for high power Hall-effect thrusters, *e.g.*, new coil wires, improved radial heat conduction within the internal coil, thermal drains to reduce inner coil temperature, and anode design adapted to high thermal loads; and
b) to allow experimental parametric testing on magnetic configuration, axial position of the gas distributor and ceramic discharge channel, discharge voltage and impact of ceramic wall erosion on performance stability.

The PPS®X000 design is based on Snecma Moteurs patents, consistent with a totally independent European design and product. The prototype, fitted with a 20-A HET cathode provided by QinetiQ, is shown in Figure 1. The thruster prototype currently consists of:

- one anode block; and
- one cathode

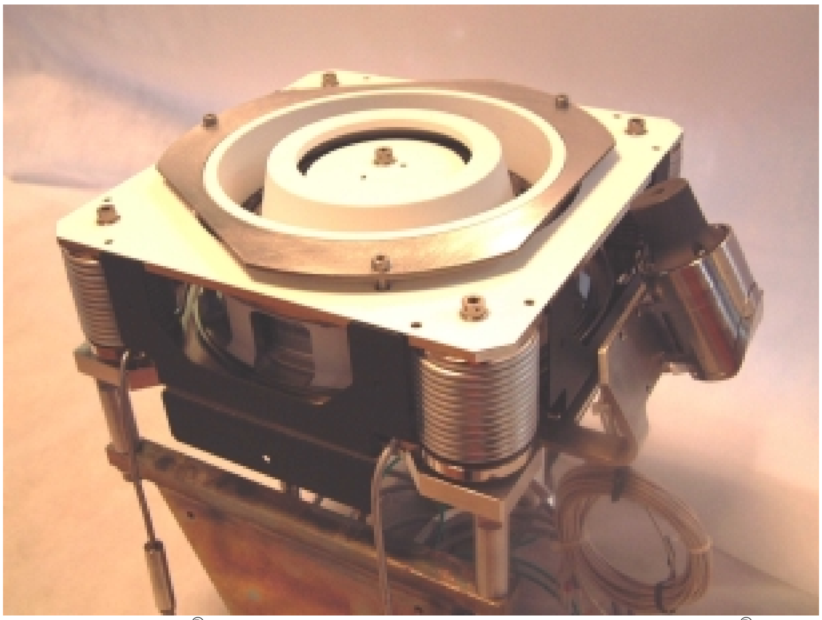


***Figure 1** - PPS®X000 : a technological prototype for the PPS®5000*

The cathode itself is attached by four screws to the cathode bracket, which is itself screwed onto the anode block. Cathode position was changed for certain tests during the characterization test campaign by means of a cathode spacer placed between the cathode bracket and the anode block.. The anode block, in turn, includes:

- an anode / gas distributor : an annular unit made of a material with a low coefficient of thermal expansion. This provides the benefit of a coefficient of thermal expansion close to that of the ceramic material making up the annular discharge channel. The discharge voltage is applied on the anode, which is also connected to the propellant line and provides uniform xenon distribution into the discharge chamber through orifices machined in the anode;
- a discharge chamber : an annular, U-shaped unit made of ceramic that confines the plasma discharge within electrically insulating walls;
- a magnetic system : this consists of one internal and four external coils, along with the magnetic pole pieces, which are electrically independent of the discharge circuit.

The PPS®X000 was designed with the goal of allowing a wide operating range, which is represented in Figure 2. This figure was established using the calculated predictions of a thruster performance model. The operating range is bounded by the limitations associated with thruster design thermal limitations (> 6 kW) as well as plasma discharge dynamics. – upper and lower limits on voltage as well as mass flow rate (or discharge current density).

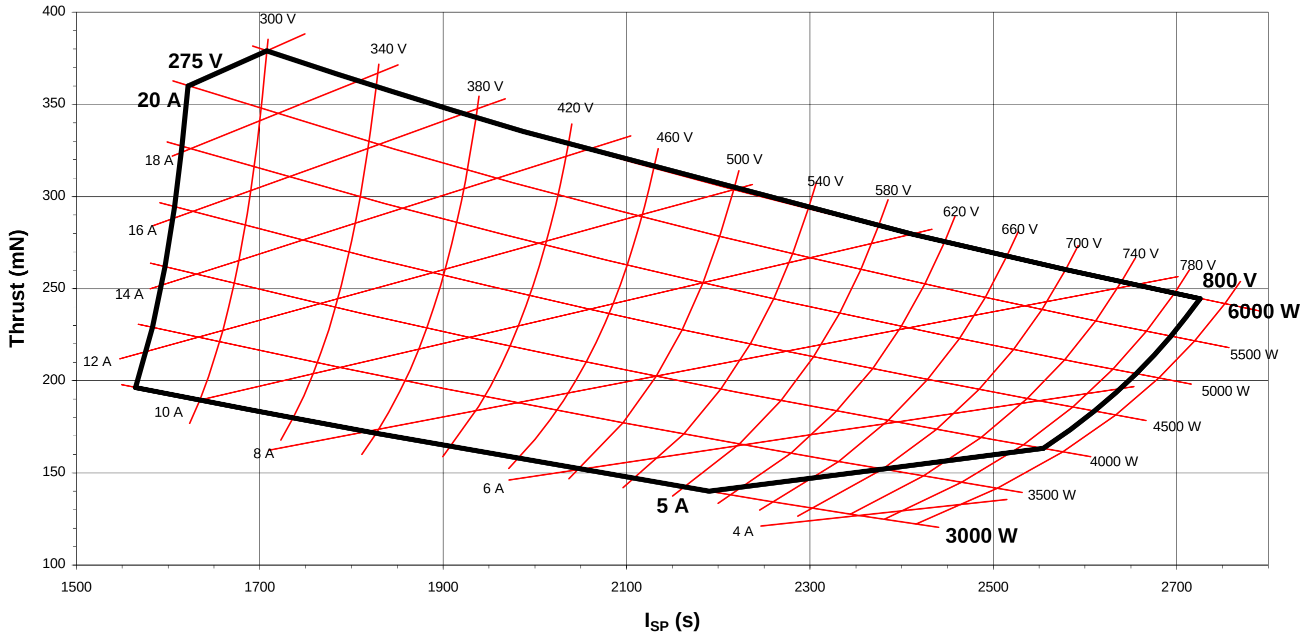


***Figure 2** - PPS®X000 design operating range.*
An operating range wide enough to propose high I_{sp} for station keeping and high thrust for orbit raising

Because of its versatile design, over 3450 distinct functional configurations can be assumed by the PPS®X000. While of course a smaller number was of practical interest, careful inspection and design of experiment led to a choice of seven configurations to be experimentally tested for evaluation, to fit within the calendar and budgetary constraints. In particular, two basic magnetic configurations were investigated during the tests.

The first – or reference—magnetic configuration is directly derived from that of the PPS®1350. The second – referred to as "advanced magnetic lens" – magnetic configuration features a magnetic induction field such that the flux lines, shaped like "magnetic lens," are pushed further downstream of the thruster exit plane. This magnetic configuration was estimated to have the potential of increasing thruster operational lifetime by decreasing ceramic wall erosion.

Great attention in the PPS®X000 design was devoted to reducing the thermal loads induced by the high power operating conditions and, in turn, internal thruster temperatures. This was achieved by several design features, and more specifically:

- large open-view areas built into the outer magnetic screen in order to allow cooling of the discharge chamber walls and gas distributor by direct radiation to space;
- copper thermal drains that conduct heat from the center of the thruster – mainly heat dissipated by the inner coil – to radiators located on the lateral sides of the thruster.

The PPS®X000 thermal design has been validated through extensive use of a 3-D finite-element thermal model including 20000 elements, as well as through the recently completed characterization tests. The effectiveness of all specific solutions implemented on the thruster has thus been demonstrated, with, in particular, measured temperatures on the ceramic chamber walls as well as coils at 6 kW very similar to the ones on the PPS®1350 at 1.5 kW.

2. The next step: PPS®5000 development

The main results of the PPS®X000 characterization tests are discussed in Paragraph 3 of the present paper. These results, along with additional functional modeling laying particular emphasis on thruster lifetime assessments [1], will be used to select the best magnetic and functional configuration in compliance with the requirements of the next generation satellites to be developed in Europe.

The PPS®X000 prototype will then be submitted to a (approximately) 1000-hours partial life test in order to validate by testing the lifetime capability assessments of the chosen configuration. The PPS®5000 configuration, power level and nominal operating points will be determined after completion of this test, when the environmental requirements become final.

A flight model thruster will then be designed, built and tested as a continuation of the SNECMA development and qualification efforts to produce a complete flight-ready plasma propulsion system (including power processing unit) by end of 2005.

3. Performance testing

Test Facility

Because of the lack of European testing facilities able to accommodate thrusters in the power range – and associated mass flow rates – of the PPS®X000 within the calendar constraints of this program, coupled with the willingness of QinetiQ to commission the Large European Electric Propulsion (LEEP 2) facility by 2002, a strong partnership was built between Snecma Moteurs and QinetiQ in the area of electric thruster testing. As a result, and after initial low-power functional tests of the PPS®X000 technological demonstrator in the Snecma LI-C facility at Villaroche, France, the characterization testing campaign was successfully carried out in QinetiQ's LEEP 2 facility between October and December 2002.

The LEEP 2 test facility has been fully described elsewhere [2] and only the most relevant details will be given here. The configuration of the vacuum chamber can be seen in schematic form in Figure 3. The largest section of the chamber is 3.8 m in diameter, and the small section is 2.6 m in diameter.

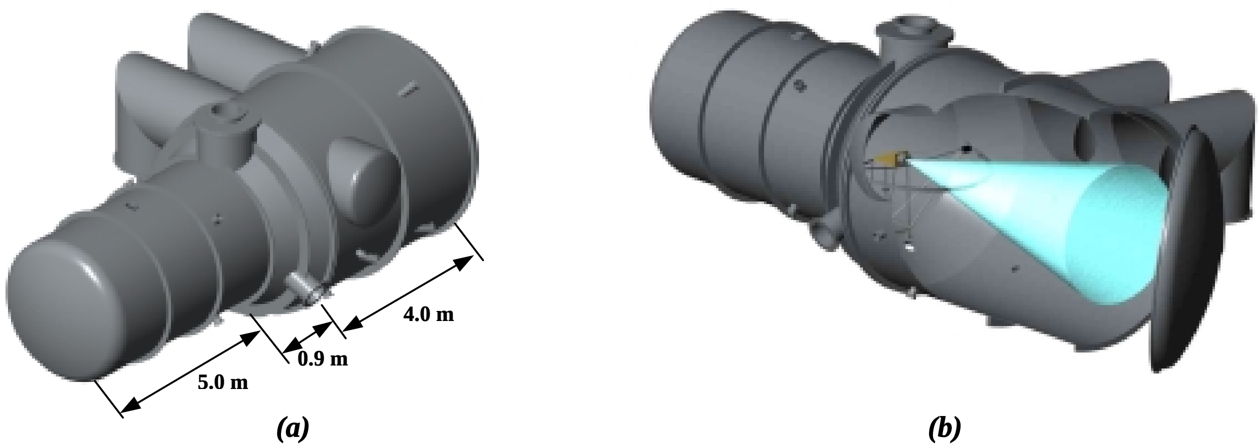
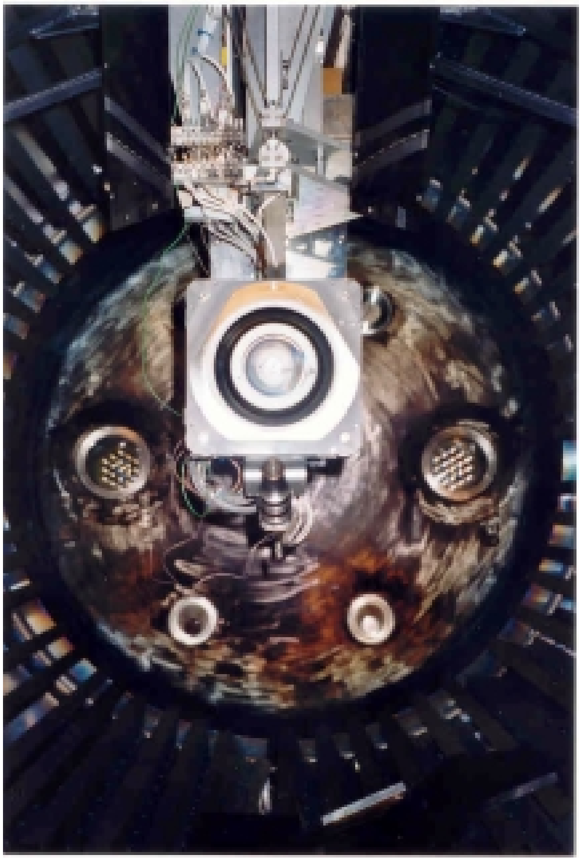


Figure 3 *- LEEP2 vacuum test facility chamber at QinetiQ.*
Rear quarter view, (a); and front quarter view with thruster and beam probe arm configurations, (b).

High vacuum pumping is achieved using 2×600 mm diameter cryogenic pumps augmented by a large xenon cryogenic pumping array. In order to condense xenon on the pump panel at the vacuum pressures specified, a temperature of < 50 K is required. At such low temperatures the cooling power is limited, therefore radiant heat from the walls of the vacuum chamber, must be restricted. This is achieved by the use of LN_2 cooled baffles. The solution adopted for the xenon pumping system achieved an effective pumping speed of about 27000 litres per second during the tests with the thruster in operation at 6 kW, providing a maximum operating pressure lower than 1.3×10^{-4} mbar , and lower than 1×10^{-6} mbar with the thruster and propellant mass flow disabled.

A flat, graphite clad ion beam target was located on the door of the facility. Graphite tiles were fixed to a series of pipes through which cooling water is passed during thruster operation. Before thruster operation, the graphite is baked to accelerate outgassing by passing hot water through the pipes which heats the graphite above 100ºC. This approach has been demonstrated to eliminate the issue of severe target outgassing when the thruster is activated.



Thrust Balance

The chamber is equipped with a single axis thrust balance. In operation, the control system generates an equal and opposite thrust to that generated by the test thruster such that deflection of the pendulum is nulled. In order to minimize the potential for thermal drift the entire device is manufactured from low thermal expansion material. To overcome the inherent problem of null-point drift due to residual thermal and mechanical drift, and low frequency vibration, a second, identical pendulum balance (with a dummy mass to represent the thruster) and control system is included. Its output is subtracted from the output of the balance carrying the thruster. Although balance specifications included a thrust accuracy of ± 2.5mN over a range of 1 – 500mN, difficulties in providing proper shielding and sensor signal conditioning from thruster discharge perturbations resulted in an effective accuracy of ± 10mN during the PPS®X000 tests. This was found to be the leading term in the error analysis. The balance system has been modified to reduce this problem in future testing. A photograph of the thruster – fitted with a High-Power cathode provided by LABEN/Proel, Italy – attached to the thrust balance can be seen in Figure 4.

***Figure 4** - Photograph of the thrust balance with a PPS®X000 thruster fitted*

The protective covers normally fitted to the thrust balance have been removed to show the construction of the balance system.

Electrical Power and Data Acquisition

The overall architecture of the power processing rack (PPR), bench filter unit (BFU), propellant supply system (PSS) and data acquisition system (DACS) is shown in Figure 5. The DACS records thruster-related data including data from the thrust balance and propellant feed system. In addition, automatic data acquisition is also obtained from a facility monitor which acquires data relating to the vacuum facility status and operating conditions.

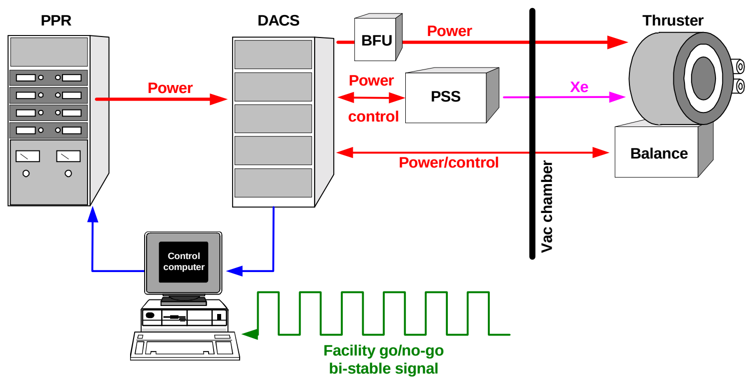


***Figure 5** - Electrical Power and Data Acquisition Architecture*

Propellant Supply System

The thruster was operated using a laboratory control system. Control of the propellant gas mass flow rate was achieved by means of proportional-valve, digital mass flow controllers (MFC). Each MFC is configurable between 0 to 30 mg s^{-1} with an accuracy better than 1% of the setpoint, and was calibrated on xenon at the factory.

Thruster performance

The thruster was started following a typical automatic start-up procedure. The start sequence begins by flowing xenon propellant through each of the two feed lines for 10 minutes before cathode ignition (t_0). Typical flow rates of 6 mg/s for the anode and 0.6 mg/s for the cathode were used. Five seconds after the flow rate commands are sent, a magnet current of 6A is sent to the thruster coils. Ten minutes before sending the cathode ignition command by applying a 30—60-V voltage step to the keeper, cathode heater current is ramped up to either 3A or 10A, depending on the cathode in use. A voltage of 300V is then applied to the anode. At this stage, the automatic sequence continuously checks for a possible thruster *self-ignition*, *i.e.*, a spontaneous discharge struck between the hot cathode and the live anode. In the event of a self ignition, the automatic start sequence detects the anode current and throttles the engine to its normal stabilisation point. At t_0 then, the start-up voltage step is applied to the cathode and electrons start flowing to the keeper (starting) electrode. Once keeper discharge is struck the plasma generated by the cathode bridges the gap between the cathode and the discharge channel, and the discharge is transferred to the anode. Upon detection of the resulting anode current, the automatic start sequence then throttles the engine to a discharge current close to 6A and ramps up the coil current to 13A before switching back to manual control mode.

For each configuration in the experimental parametric study, the thruster was first left to outgas in a 2-kW, low power mode for a minimum of three hours *and* until no anode mass flow rates were necessary to maintain a discharge current constant within the measurement accuracy (less than one percent). The engine was then throttled to points A, B, C and D for a systematic characterization of thruster operation, as represented schematically in Figure 6. This included the acquisition of performance data as well as optimization of the magnetic coil current. In addition and for points A and D, three extra sets of experiments were conducted, namely: a) varying the magnet current balance between internal and external coils by ± 15%; b) varying cathode mass flow rate within ± 15% of its nominal value – typically around 5% of the anode mass flow rate; and c) increasing background xenon pressure in the test chamber by up to 50%. This was achieved by leaking xenon gas through an independent feed line to the chamber, with an inlet located on the loadlock chamber above the main vessel. The pressure was measured at the downstream end of the main chamber cylindrical section.

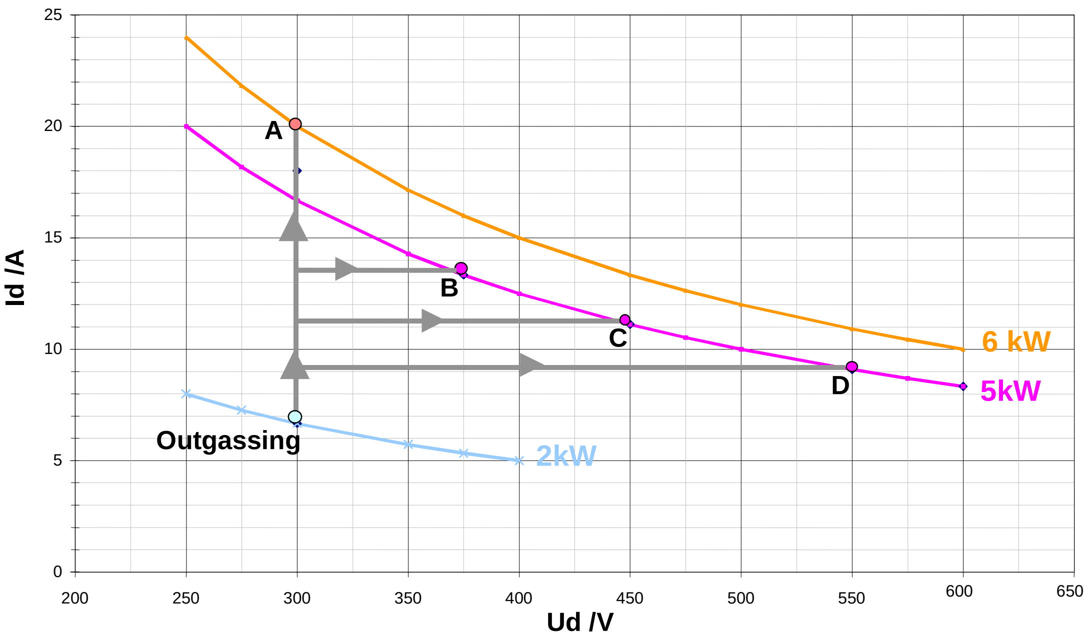


Figure 6 – *PPS®X000 outgassing and characterisation points*

A bench filter unit (BFU) was used throughout the tests between the anode power supply and the thruster. The BFU consisted of a 6-µF capacitor placed between anode positive and cathode common, together with a 0.35-mH inductance placed in parallel with a 100-Ω resistance on the anode positive line, on the power supply side of the capacitor.

Pictures of the thruster during operations are shown in Figure 7, and typical measured performance data are represented in Figure 8. In addition to the PPS®X000 thrust *vs.* specific impulse data, the nominal operating point for the PPS®1350 (4.28A and 350V of discharge current and voltage, respectively) is also shown for reference. The shaded area represents the portion of the design operating range (Figure 2) visited during the tests, and the performance points corresponding to the discharge conditions represented in Figure 6 are shown with the error bars associated with the experimental uncertainty, essentially from the thrust measurement. Also shown is the performance corresponding to the additional point D^+, *i.e.*, at 5 kW and 585V of discharge power and voltage, respectively. This point, which was not part of the systematic characterization, was visited only with the thruster in the reference configuration, to which all data mentioned here are related.

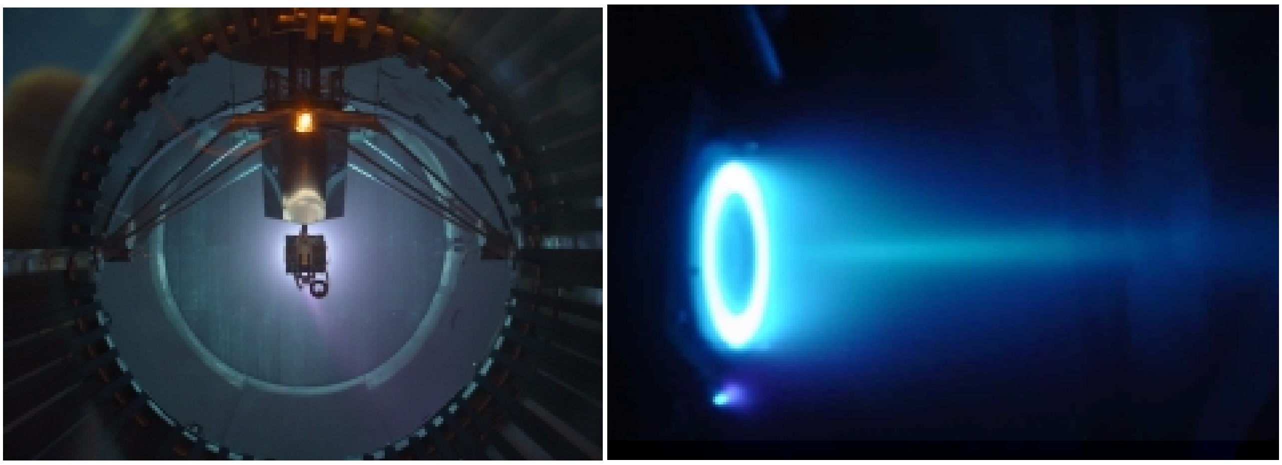


Figure 7 – *PPS®X000 during operations*

Engine seen from the back of the vacuum chamber, left picture; and from a side view port, right picture.

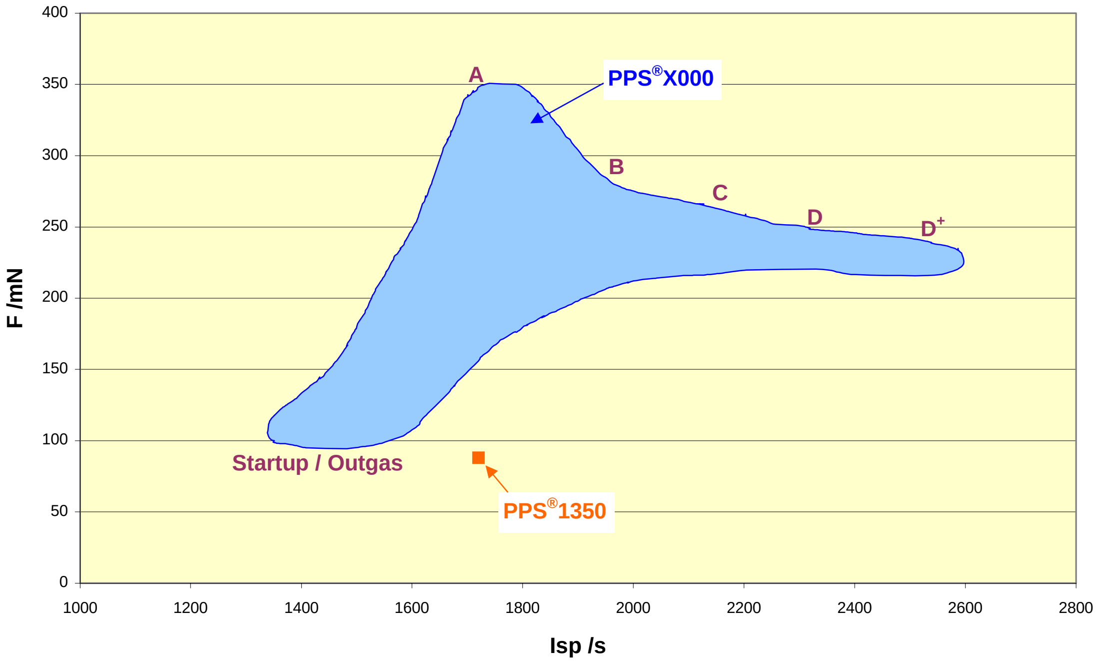


Figure 8 – *Thrust vs. I_{sp} in reference configuration*

Finally, Figures 9 to 11 show thrust, total specific impulse, and total (thrust) efficiency as a function of anode mass flow rate and discharge voltage. In Figures 9 and 10 and at 300V of discharge voltage, cathode mass flow rate m_c was adjusted as a function of anode mass flow rate m_a in three steps as follows: m_c=0.59mg/s for m_a<9.26mg/s; m_c=0.73mg/s for 9.26mg/s<m_a<11.3mg/s; and m_c=0.98mg/s for m_a>11.3mg/s. Cathode mass flow rate was kept constant at 0.6mg/s over the full range of m_a for the points corresponding to 450V. In Figure 11, cathode mass flow rate is 0.6mg/s. Finally, magnet current was kept constant at 20A for all data sets in Figures 9 to 11.

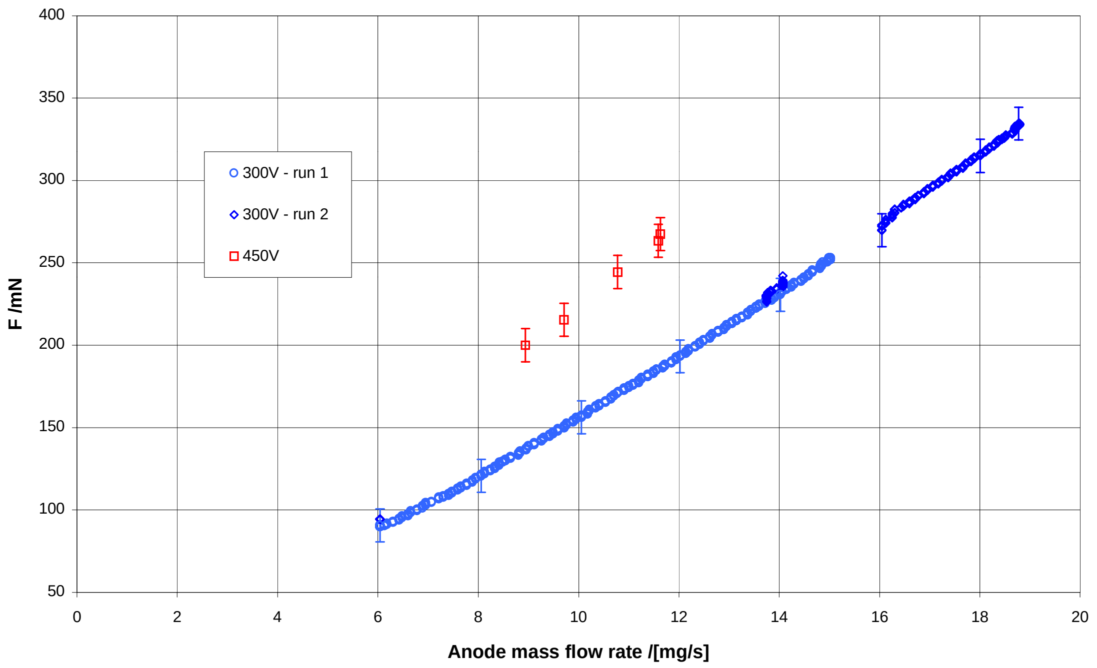


Figure 9 – *Thrust vs. anode mass flow rate*

Magnet current was kept constant at 20A for all data above. Cathode mass flow rates were: at 300V, 0.5mg/s for m_a<9.26mg/s, then 0.73mg/s up to m_a=11.3mg/s, then 0.98mg/s for m_a>11.3mg/s; at 450V, 0.6mg/s

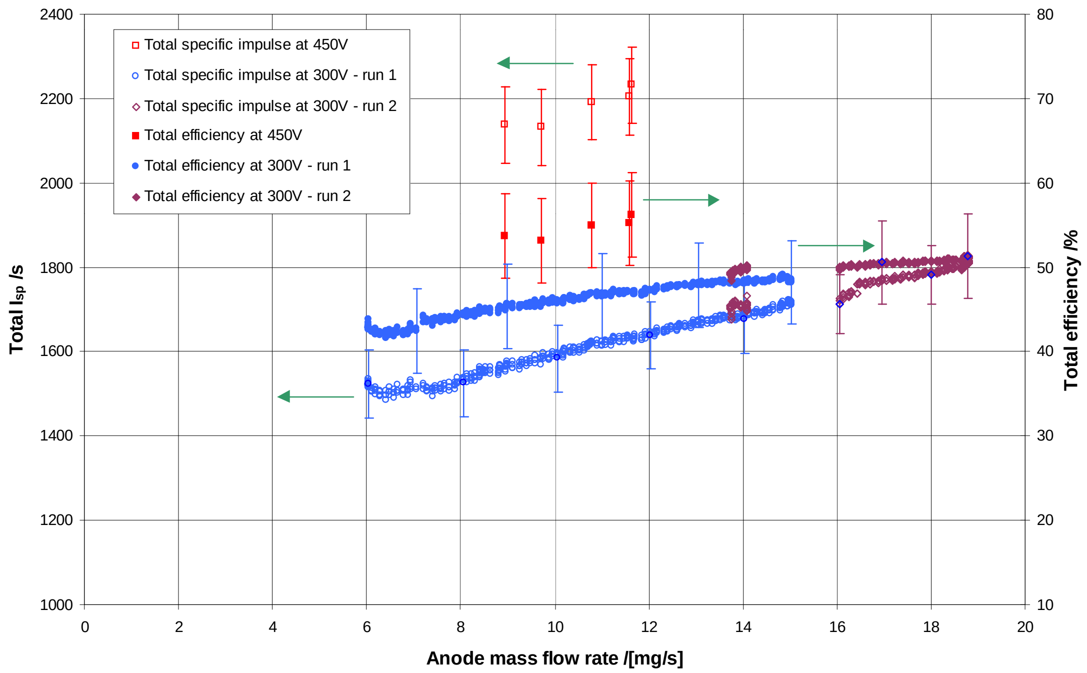


Figure 10 – *Total specific impulse and total efficiency vs. anode mass flow rate*

Magnet current was kept constant at 20A for all data above. Cathode mass flow rates were: at 300V, 0.5mg/s for m_a<9.26mg/s, then 0.73mg/s up to m_a=11.3mg/s, then 0.98mg/s for m_a>11.3mg/s; at 450V, 0.6mg/s

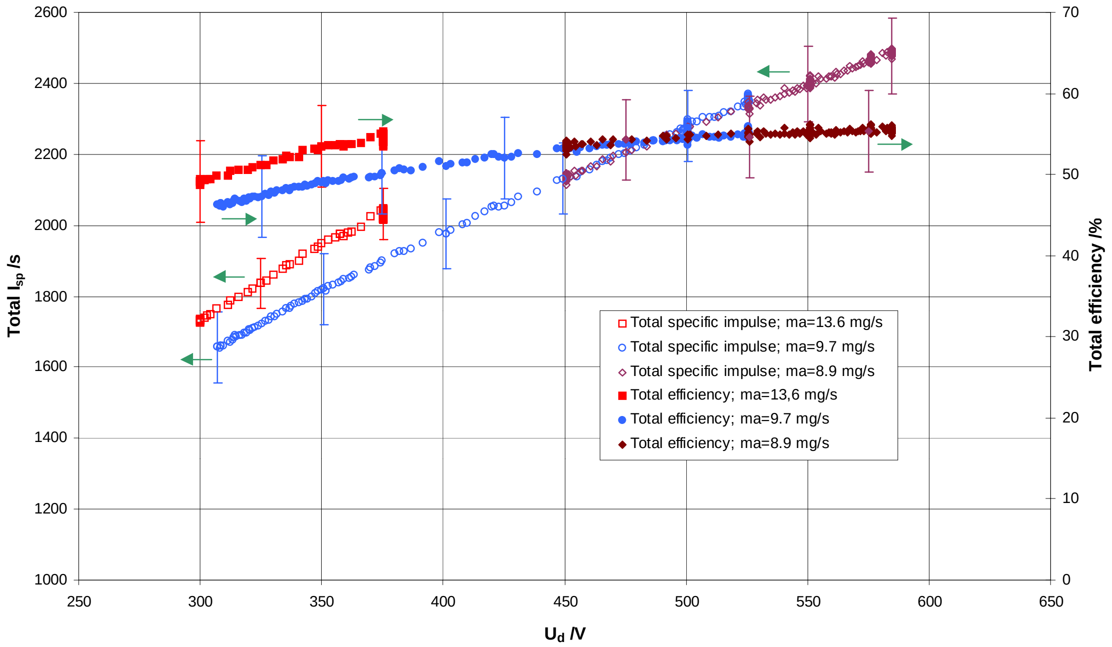


***Figure 11 –** Total specific impulse and total efficiency vs. discharge voltage*

Magnet current and cathode mass flow rate were kept constant at 20A and 0.6mg/s, respectively, for all data above

Discussion

The main performance points are summarized in Table 1. While assessment of the best configuration was not always possible by mere comparison of the performance parameters (thrust, specific impulse and efficiency) themselves because the difference was usually less than the experimental uncertainty, a “best” thruster configuration could be established based on thruster stability criteria. As an example, some configurations were designed to suggest and amplify end-of-life thruster behavior, which yielded an indication of robustness of operating conditions to discharge chamber wall erosion. Other configurations were discarded because they exhibited an increased amplitude of discharge current oscillations in the 400—500V range of discharge voltage. Finally, the chosen configuration featured a surprisingly flat discharge current-voltage characteristic, with a variation of discharge current less than 1 % between 300V and 550V, while other configurations showed an increase of discharge current at high voltage, consistent with other reports [3] and generally attributed to either an increase in axial electron conductivity or increased production rate of multiply-charged ions – detrimental to thruster efficiency in both cases.

It should be noted that discharge current sensitivity to magnet current was less than 2 % over the range of thruster discharge stability, *i.e.*, *above* a lower bound as low as 6A depending on discharge current and over the whole range of magnetization allowed by the magnetic circuit, *i.e.*, up to 21A. Performance data were not corrected for facility background pressure effects, as the large facility pumping speed kept the ratio of ingested flow – from the random flux of vacuum chamber particles – to anode flow below 2% under all operating conditions.

***Table 1 –** PPS®X000 performance summary*

Disch. voltage U_d (V)	Disch. current I_d (A)	Disch. power P_d (kW)	Thrust F (mN)	Spec. impulse Isp (s)	Anode spec. impulse Isp_a (s)	Thrust efficiency η (%)	Anode efficiency η_a (%)
300	20.0	6.0	335 ± 10	1769 ± 53	1864 ± 56	47.4 ± 4.6	49.9 ± 3.0
375	13.4	5.1	271 ± 10	1929 ± 71	2026 ± 75	49.8 ± 5.2	52.3 ± 3.9
450	11.2	5.1	256 ± 10	2120 ± 83	2215 ± 86	52.1 ± 5.0	54.4 ± 4.3
550	9.4	5.3	233 ± 10	2304 ± 99	2402 ± 103	49.4 ± 5.6	51.5 ± 4.5
585	8.4	5.1	232 ± 10	2480 ± 107	2649 ± 114	55.6 ± 6.6	59.4 ± 5.2

As a final note, the discharge voltage limit of 585V during this characterization test campaign corresponded to an practical limit from a facility power supply and cabling standpoint. Further characterization testing up to 800V is expected by the end of the first semester of 2003, before the PPS®X000 undergoes a 1000-h partial lifetime test.

Conclusion

Characterization testing was conducted on the PPS®X000, a technological demonstrator for a 5—6-kW Hall-effect thruster: the PPS®5000. The total cumulated testing time is 187 hours, with over 160 on/off cycles. This engine demonstrated a thrust capability of about 335 mN at 6kW and 300V of discharge power and voltage, respectively, as well as a specific impulse in excess of 2400s at 5kW and 585V. Additional characterization testing activities of the PPS®X000 up to discharge voltages of about 800V will be carried out within the next two months, before the thruster is to undergo a 1000-h, cycled endurance test to assess operational lifetime capability. In addition to the performance parameters measured during the tests, thruster temperature data were collected in order to calibrate a finite-element thermal model. Maximum ceramic temperatures lower than 560°C measured at 6kW of discharge power suggest that a total lifetime capability greater than 7×10^6 N.s is well within reach for the future PPS®5000.

Acknowledgments

The work described in this paper was performed with the support of the Centre National d'Études Spatiales (CNES), France.

References

1. Duchemin, O., Dumazert, P., Carichon, S., Boniface, C., Garrigues, L. and Bœuf, J-P., "Performance Characteristics of the PPS®5000 During a 1000-Hour Endurance Test." In 39th AIAA/ASME/SAE/ASEE Joint Propulsion Conference, Huntsville, AL, 2003. Submitted.
2. Simpson, H.B., Wallace, N.C., Clark, S.D. and Mundy, D.H., "A Description of the New 3.8m Diameter High Power Electric Propulsion Test Facility at QinetiQ, Farnborough, UK." In 28th *International Electric Propulsion Conference*, Toulouse, France, 2003. IEPC-03-0213.
3. Pote, B. and Tedrake, R., "Performance of a High Specific Impulse Hall Thruster." In 27th *International Electric Propulsion Conference*, Pasadena, CA, 2001. IEPC-01-35.