Performance Evaluation of a 20 kW Hall Effect Thruster

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Abstract: Within the framework of the European project HiPER, a 20kW Hall Effect thruster was studied, designed, manufactured and tested in the Pivoine test facility of the CNRS. The prototype, designated PPS-20k ML, was successfully operated from 2.6 kW to 23.5 kW with a discharge voltage ranging from 100 V to 500 V. The maximum performance was obtained at 500 V, with the following parameters: a thrust of 1050 mN, an Isp of 2700s, a total power of 22.4 kW and a total thruster efficiency of 60%. In the 5 kW to 22 kW range of total power, the total thruster efficiency was quasi-constant and equal to 60% with a 68% peak at 300 V and 5 kW. The cathode-to-ground voltage was about -16 V. An operating point for the PPS-20k ML HET was set with regard to the HiPER mission requirement: a thrust of 1 N and an Isp of 2500 s at 20 kW.

Nomenclature

B_r	=	Radial component of the magnetic induction field
I_d	=	Discharge current
P_{anode}	=	Power losses of the plasma on the anode
T_{eV}	=	Election temperature in eV
U_{CRP}	=	Cathode-to-ground voltage
Ud	=	Discharge voltage
V_b	=	Effective ion beam voltage

I. Introduction

DURING the last 40 years, the available electic power of spacecrafts increased significantly from about 1 kW to more than 20 kW for current platforms. Moreover, the continuous improvements of space solar arrays, such as the solar cell efficiency increases with time, have enhanced the expectations for use of high power electric propulsion. This availability of relatively high on-board power offers new opportunities for electric propulsion as primary propulsion for possible orbit-to-orbit transfer and exploration mission.

An ESA recent successfull mission implementing electric propulsion as the main propulsion was the SMART-1 lunar orbiter mission. The SMART-1 probe covered more than 100 millions kilometers, consuming only 82 kg of propellant, xenon in that case, and operated the 1.5 kW PPS-1350G Hall Effect Thruster (HET) developed and manufactured by Snecma. The PPS-1350G thruster reached a total flight duration of 4960 hours, demonstrating both

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flexible operation capabilities and a reliable implementation of the HET technology. The most powerfull HET currently in flight is the BPT-4000, a 4.5 kW HET operating for station keeping of the AEHF GEO communications satellite since 2010. Due to the failure of the satellite apogee chemical engine, the BPT-4000 thrusters were implemented for orbit tranfer. When considering higher powers, the most powerful electric propulsion thruster currently in progress is the VASIMR engine. Following ground testing of a 200 kW prototype, a future in-space test of a cluster of two 100 kW VASIMR thrusters is planed for 2014, on the International Space Station. These different examples illustrate the continuous development of high power electric propulsion and specially its implementation for in-space propulsion.

When focussing on HETs, it should be noticed that high power HETs were evaluated many years ago in test facilities, in Russia and in the USA. The Russian firm Fakel operated a SPT-290 at high power. The nominal discharge voltage was 600 V for a 1.5 N thrust, an Isp of about 3300 s and a lifetime estimated to 27 000 hours. Nevertheless, the most numerous developments were realized by the NASA and the test of four high power HETs. Firstly, a 50 kW thruster, the NASA-457M was constructed and operated. This thruster with a 457 mm outer diameter ceramic discharge chamber reached a power of up to 72 kW and a thrust of 2.9 N. The discharge Isp ranged from 1750 to 3250 s with a discharge efficiency between 46 and 65%. Later, a second version of the NASA-450M and a NASA-400M were tested. The thruster built lastly is the NASA-300M, defined by the following specifications: a 20 kW discharge power, a discharge current and voltage respectively of up to 50 A and 450 V. This HET was tested from 2.5 to 20 kW, with Ud varying from 200 to 600 V and Id from about 10 to 50 A, both with xenon and krypton propellants. The peak total efficiency and total Isp with xenon were respectively ~67% (Ud = 500 V and 20 kW) and 2920 s (Ud = 600 V and 20 kW). The peak thrust-to-power was ~ 70 mN / kW, achieved at Ud = 200 V.

In 2008, the HiPER project ("High Power Electric propulsion: a Roadmap for the future"), co-funded by the European Union, was initiated under the space theme of the 7th Framework Programme. It is aimed at laying the technical and programmatic foundations for the development of innovative electric space propulsion technologies to fulfill future space transportation and space exploration needs. A complete description of the HiPER project can be found in references [1] and [2]. The main objectives of this European project are to drive a long term vision for mission-driven electric propulsion development, performing basic research and proof-of-concept experiments on some of key concepts. Among possible propulsion solutions, high-power Hall Effect thrusters were evaluated as a possible option. In order to progress in this field, six different European partners from the industry and the academic community (Alta – Italy, CNRS – France, IPPLM- Poland, Onera – France, Snecma – France and Tecnalia-Spain) addressed the main scientific and technological issues related to the design, manufacturing and testing of a 20 kW HET prototype. This thruster, designated PPS-20k ML, was designed and modeled during the first year of the HiPER project, manufactured during the second year and finally assembled and tested this year. A detailed description of the work realized during the two first years of the HiPER's HET workpackage is given in reference [3]. This publication summarizes the main results obtained recently and consisting in the manufacturing and testing of the 20 kW HET prototype.

II. Design of the PPS-20k ML Hall Effect Thruster

Following the mission requirements, specified by a dedicated HiPER working group, the design a 20 kW thruster was proposed in the perspective of a cluster of 20 kW HETs. This choice was driven by several advantages offered by clustering the thrusters: test of a down-sized cluster with enough confidence when scaling up to large power, flexibility of thrust orientation by operating a part of the thrusters, reliability in case of the failure of one thruster, availability of test facilities in Europe ...

Nevertheless, when considering the development of high power Hall Effect thrusters, several technological and scientific challenges are to be addressed. The starting point of the design of the HET is the estimation of the main geometrical parameter of the thruster by setting scaling laws. These scaling laws were discussed in numerous publications and are often based on the definition of non-dimensional numbers and algebric equations, relating theses numbers. This set of equations are completely determined by fixing proportionaly coefficient, generally by post-processing a data base collecting performance measurements of tested HETs from about 100 W to several kW.The available thruster sizing and test results are however often incomplete so the elaboration of scalings laws with a wide validity range is questionable. Moreover, very limited data are available for high power thrusters and large scale HET parameters are often extrapolated from few kW test results. Due to these limitations and uncertaintainties, a modular and dismountable HET prototype was proposed in our project. In parallel and in order to improve our design approaches, a 1D code predicting the plasma discharge in HETs and developed by the IPPLM, was improved [11]. A refined approach describing the electron behavior was implemented as well as a better description of the plasma wall interactions.

A. Anode and Neutral distributor

In standard Hall Effect thrusters, the anode works both as a distributor for the neutrals and an anode by collecting the electrons of the discharge. The delivery of the neutrals is often realized by an annular array of small diameter orifices. Based on the location and spacing of these orifices, the axial velocity and azimuthal uniformity of the neutrals can be significantly altered, changing the propellant utilization and current to the anode. Small changes in neutral uniformity can affect the thruster during operation, by decreasing its efficiency, modifying the plume symmetry and altering the discharge stability. Another potential limit to an anode proper functioning is the heating of the anode, both by Joule effect and by an excessive heat flux due to the back-streaming electrons. The average electron power lost to the anode can be estimated by the following equation: $P_{anode} \approx 2\ T_{eV}\ Id \approx 0.02\ V_b\ Id$. For a 20 kW thruster, the power applied to the anode can be estimated to about 360 W in comparison to approx. 25 W for a 1.5 kW. This estimation could vary significantly when changing the neutral gas flow rate and discharge voltage because of the non-linear discharge behavior. Moreover, the thermal fluxes are also set by the design of the anode.

In order to study possible improvements of the thermal margins, two different options were set for the design of the 20 kW thruster. First, a standard anode, combining gas distribution and electron collection was manufactered. Second, an alternative design was proposed by splitting the gas distribution from electron collection. For both options, the axial location of the gas distributor and anode can be varied. Consequently, the length of the discharge channel was increased in order to offer a wide range of possible anode to thruster exit distances, from 20 to 75 mm.

B. Ceramic of the discharge channel

For a PPS-1350 thruster, the internal diameter of the outer ceramic is 100 mm. In order to manufacture this channel, a ceramic ingot is generally made by the hot press of boron nitride or borosil powders and the thruster discharge channel is then machined from the ingot. When increasing the power of HETs, the size of the discharge channel needs to be increased and technological limitations of the hot press process are to be asssessed. The evaluations of the discharge channel manufacturing limitations and also of alternative processes were realized by Tecnalia; see reference [12] for a detailed description. Finally, a 320 mm ceramic was manufactured for the HiPER project. In order to be able to vary the anode and neutral distributor axial locations in the discharge channel, a lengthnen ceramic channel was made.

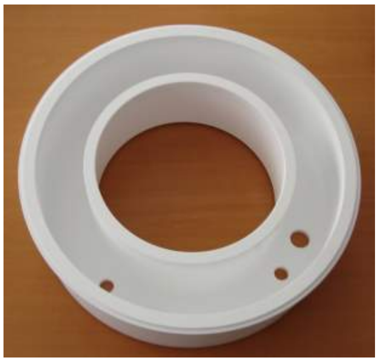

Figure 1. Ceramic of the discharge channel

Another important issue is the interaction between the plasma and the ceramic of the discharge chamber. In fact, in Hall Effect thrusters, the plasma is confined by the ceramic and both ions and electrons interact with the walls. Ions impinging the ceramic produce an erosion of the discharge channel, limiting the lifetime of the thruster. Electrons are susceptible to modify the plasma sheath on the wall and by consequence the discharge behavior and thruster performance. In order to identify the most efficient ceramic composition, Onera performed erosion and secondary electron emission yield measurements of several ceramics samples. Results of their work are described in references [8] and [9].

C. Magnetic circuit

The confinement of the plasma in a Hall Effect thruster is made by a magnetic lens in the thruster exit plane. This magnetic field can be generated by an electromagnet or permanent magnets. When considering high power Hall Effect thruster, the self-induced magnetic field produced by the Hall current can also be high enough and distort the magnetic map. This phenomenon is however not well understood nowadays and is covered by the discharge optimization when testing a thruster. Consequently, in order to perform the magnetic optimization of the discharge, the retained option was an electromagnet in order to be able to vary the coil current during the thruster tests. For the external magnetic circuit, a total of eight coils were

Figure 2. Internal annular coil

implemented. For the internal circuit, a single annular coil was manufactured. Figure 2 shows the photograph of the internal annular coil. The external and internal circuits can be powered independently.

After assembling the prototype, the magnetic field of the thruster was measured for different coil currents and these data were compared to 2D and 3D magnetostatic modeling. A good agreement between predictions and measurement was obtained with a maximum discrepancy of about 10 %. The azimuthal homogeneity of the magnetic field was also evaluated and variations less than 1.5 % were obtained. The tested range of coil currents confirmed the capabilities of the electromagnet to obtain the B_r specified by the scaling laws.

D. Thermal field

Even if up to 50 to 70% of the electric power of a HET is converted into thrust, the power losses to the wall of the discharge chamber and to the anode can be significant when considering high power thrusters. The temperature map of the thruster is actually a balance between the conduction - radiation thermal losses of the thruster walls and the fluxes due to the plasma (radiation and particle interactions through the plasma sheath) in the discharge chamber. For a PPS-1350 HET, the thermal map is the following: approx. 700°C for the ceramic walls, 600°C for the anode, 300°C for the coils and 300°C for the thruster body. For a high power thruster, the absolute power losses are increased but the increase in size of the HET offers also opportunities to improve the cooling of the thruster because of higher radiative surfaces. As a precise prediction of the plasma fluxes on the thruster walls is not easy due to the very complex physical phenomena involved in the discharge of a HET, the PPS-20k ML prototype was equipped with several thermocouples. The temperature of the coils, magnetic circuit and thruster body were measured in order to check the thermal compatibility of the different thruster parts. A thermal camera was also used during the testing campaign in order to evaluate the temperature evolution of the ceramic.

It is also important to notice that the thermal transient phase of a thruster can vary significantly as a function of its size. A 1.5 kW type HET will be thermally stabilized after approx. 1 hour. When increasing the power and consequently the size of the thruster, several hours are needed to reach a steady temperature map, because of the higher thermal inertia of the thruster.

In parallel to the thruster design, manufacturing and assembly, a 50 A cathode was developed by the Ukrainian National Aerospace University. A detailed description of the test results of the cathode is given in reference [7]. This cathode was mounted on the centerline of the prototype. A photograph of the thruster with its cathode, before its integration in the test facility, is shown in figure 3.

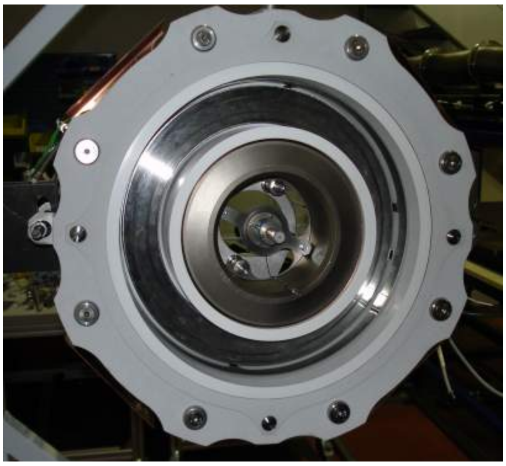

Figure 3. Photograph of the PPS-20k ML HET

III. Testing of the PPS-20k ML Hall Effect Thruster Prototype

The PPS-20k ML thruster was tested in a cryogenically pumped cylindrical vacuum chamber called Pivoine. This test facility is located in the ICARE laboratory of the CNRS, located in Orléans, France. In the next paragraph, a brief description of the test facility is given.

E. The Pivoine test facility

The Pivoine test facility was built in 1997 with the support of the French space agency CNES, the “Région Centre” council and Snecma. This facility is dedicated to research activities in the field of HET propulsion and enables different research teams to test HETs, to develop new diagnostics and to characterize experimentally the discharge of Hall Effect thrusters. In 2007, an upgraded version of the Pivoine test facility was built. The current version is made of two

Figure 4. The Pivoine test facility

cryogenic stages, optimized for xenon as a propellant. The first stage enables to test 5 mg/s HETs at a pressure of 2.10^{-5} mbar. With both stages operating, 21 mg/s HET tests can be performed at 2.10^{-5} mbar. The test facility is equipped of different diagnostics, like a thrust balance, plasma probes and several acquisition systems. A cylindrical test port was mounted along the main axis of the chamber. A photograph of the test facility is shown in figure 4.

In order to test the 20 kW HET prototype, a new thrust balance was designed and specially fabricated. This stand was designed based on a thrust stand used in previous test campaigns. It is a pendulum stand, calibrated by a serie of two different weights. The estimated error of the measured thrust is about ± 1.5 % for a thrust ranging from 300 to 1500 mN.

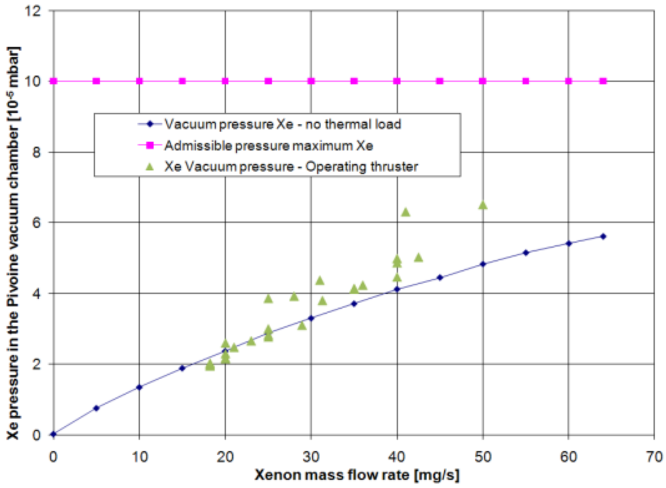

Figure 5. Pumping capabilities of the Pivoine vacuum chamber

Moreover, the xenon feed system was upgraded to reach a total xenon mass flow rate of up to 80 mg/s. The discharge power was provided by two (500V, 30 A) supplies in parallel. The maximum available discharge voltage and discharge current were respectively set to 500 V and 50 A. A new filter unit was also used, consisting of an 8 µF capacitor, a 100 Ω resistance and a 340 µH inductor. As the Pivoine test facility was built for the testing of 5kW HETs, preliminaries pumping tests were performed with no thruster operating in order to verify the pumping capabilities for a 20kW prototype. Moreover, during the functioning of the PPS-20k ML thruster, the vacuum pressure was monitored and continuously controlled. The Xenon pressures measured during the test campaign are summarized by figure 5. The maximum admissible pressure during the firing of HETs is generally equal to $10.\ 10^{-5}$ mbar. With no thruster in operation, the vacuum Xe pressure was less than $6.\ 10^{-5}$ mbar at 60 mg/s. During the tests of the PPS-20k ML, good vacuum conditions were obtained up to 10 kW. For discharge powers higher than 10 kW, the thruster was shut down when the Xe vacuum pressure reached $6.\ 10^{-5}$ mbar because of the high thermal fluxes heating up the cryogenic panels of the pumping system. At high power, good operating conditions could be obtained during 40 minutes.

F. Results and Thruster performance

The PPS-20k ML HET was tested over a range of input powers from 2.6 to 23.5 kW by varying the anode mass flow rate from 5 to 42.5 mg/s and the discharge voltage from 100 to 500 V. Over this range of operating conditions, the thruster discharge was stable. Figure 6 shows the thruster operating at low power, 15 kW and 23.5 kW.

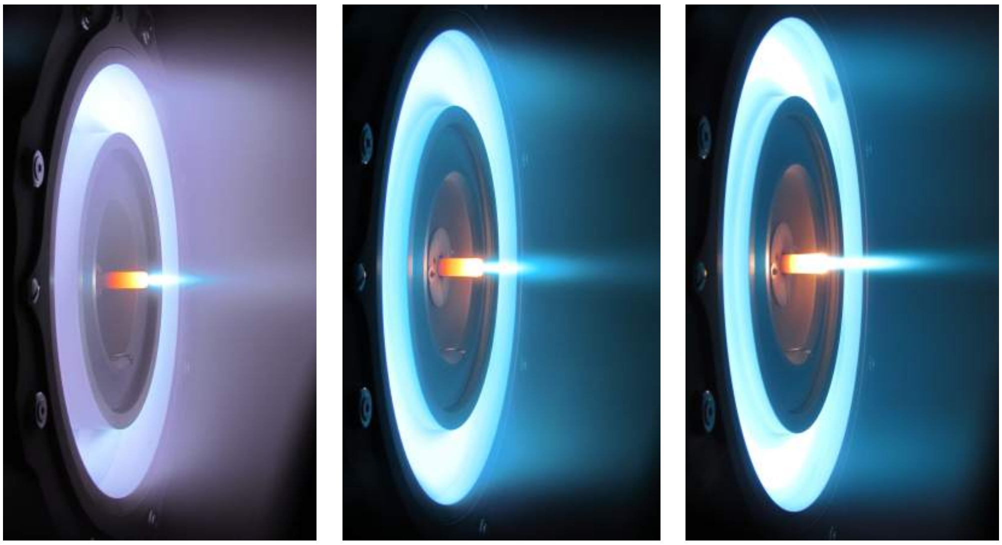

Figure 6. PPS-20k ML at low power, 15 kW and 23.5 kW

The test results of the PPS-20k ML thruster are summarized by figures 7 to 11. In order to assess the thruster performance, the total efficiency and total power of the HET were evaluated, i.e. that the performance of the thruster was calculated by considering the anode plus the cathode xenon mass flow rates as well as the discharge power plus the power supplied to the coils. Moreover, the currents of the inner and outer electromagnets were modified in order to optimize the magnetic field topology for some operating points. This magnetic optimization couldn't be achieved for all the tested discharge powers and the ratio between the inner and the outer coil current was generally kept constant.

As mentioned in the previous paragraph, due to the pumping limitation of the vacuum facilities, steady thermal conditions couldn't be achieved during the campaign for powers higher than 10 kW. Due to the large thermal inertia of the thruster, no attempts were made to establish complete thermal equilibrium prior to measuring performance data, except for one operating point (about 10 kW and Ud = 400 V). For these test conditions, a complete magnetic field optimization was performed and a maximum temperature of about 290°C was reached by the inner coil, which was the hottest part of the thruster during all the test campaign.

The thrust versus the total power for various discharge voltages are plotted in figure 7. A maximum thrust of 1050 mN was obtained for the highest tested total power and 1000 mN were obtained for 19.3 kW. Ud was varied from 100 to 500 V but for discharge voltages equal or less than 200 V low performances of the thruster were measured.

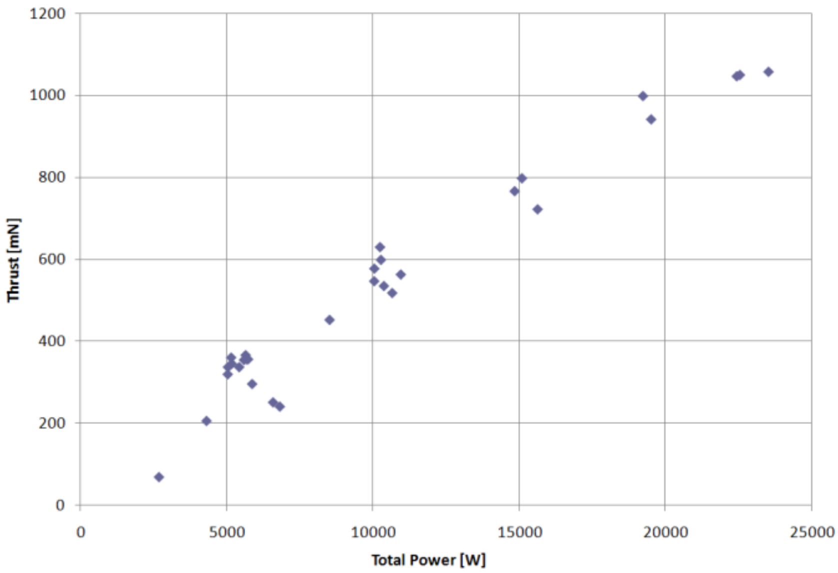

Figure 7. Thrust versus Total Power for various discharge voltages

The variation of the discharge specific impulse as a function of the discharge voltage is shown in figure 8. As expected, the Isp increased with the discharge voltage and a maximum Isp of about 2700 s was reached. At Ud = 200, 300, 400 and 500 V, evolutions of the xenon mass flow rate and modifications of the magnetic field of the thruster lead to small variations of the Isp.

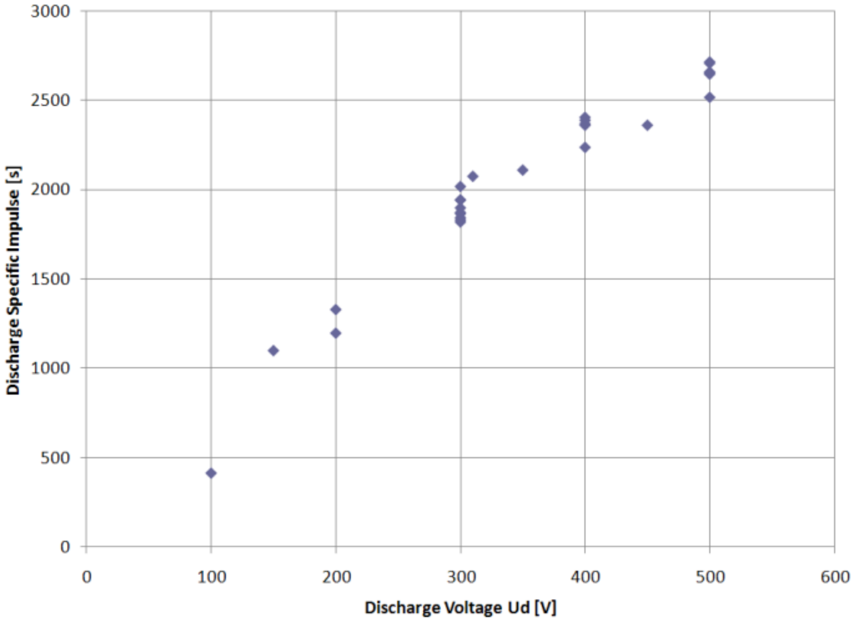

Figure 8. Discharge specific impulse versus discharge voltage Ud

The total efficiency, taking into account the cathode xenon mass flow rate and the coil power is presented in figure 9 for various discharge voltages. Only test results with Ud higher than 200 V are summarized in this figure. For Ud = 200 V, the total efficiency of the thruster was about 38%. The highest total efficiency was 68% at 300 V for a 5 kW total power. For 20 kW total power, the total thruster efficiency was equal to 60 %. In the 5 kW to 22 kW range, a quasi-constant total efficiency of about 60% was measured.

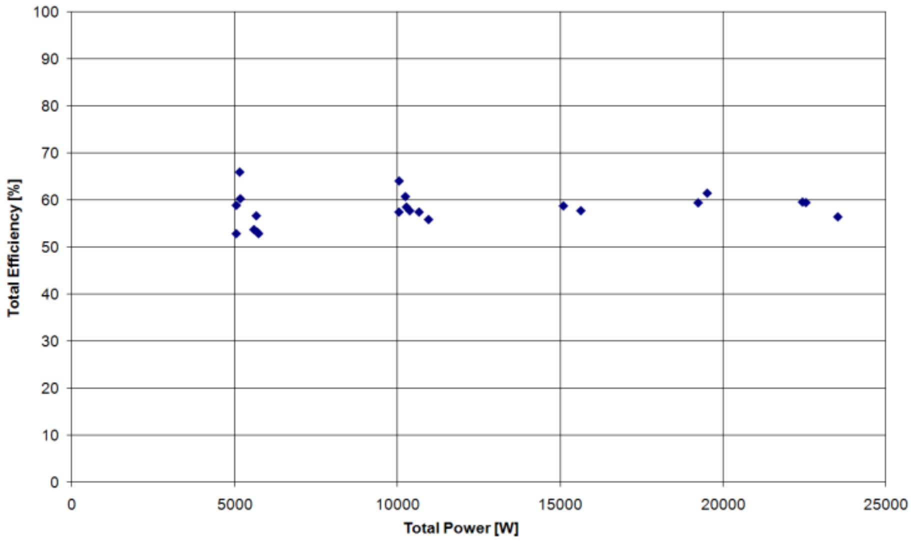

Figure 9. Total Efficiency versus Total Power for various discharge voltages

Figure 10 shows the variation of the thrust-to-total power versus the thruster total power for discharge voltages above 200 V. The highest thrust-to-total power was about 70 mN / kW at 300 V and 5 kW. In the 10 kW to 20 kW range, this ratio varied between 50 to 60 mN / kW.

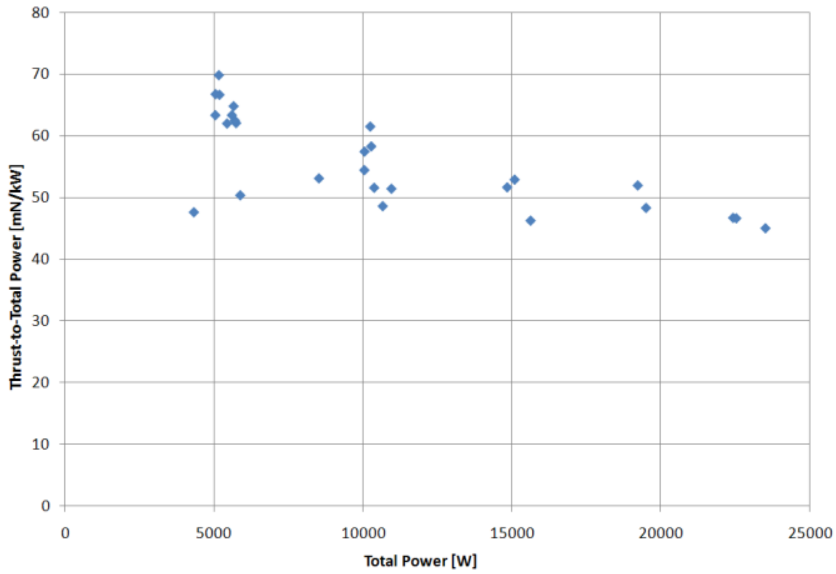

Figure 10. Thrust-to-Total Power versus Total Power for various discharge voltages

The cathode mass flow rate was varied from 0.86 mg/s to 1.1 mg/s. The cathode-to-ground voltage variations are shown in figure 11. In order to achieve a good HET efficiency, a small cathode-to-ground voltage is necessary to minimize the energy loss mechanism associated with poor cathode coupling. During our tests, the cathode-to-ground voltage varied from -14.3 to -23 V with a mean value of about -16 V. In order to avoid electrical insulation losses of the cathode igniter, a shield against backsputtering was fixed to the base of the cathode during the test campaign.

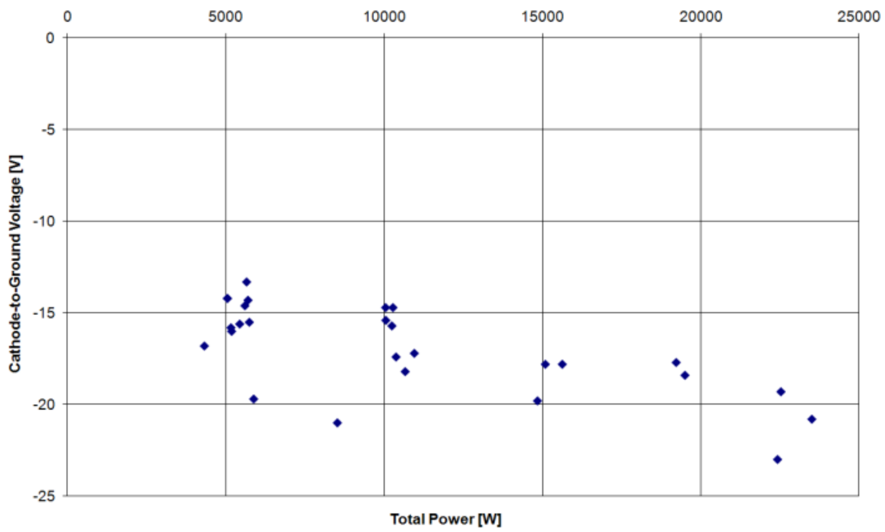

Figure 11. Cathode-to-Ground Voltage versus Total Power for various discharge voltages

Considering the mission requirement defined within the HiPER project, the following specification for the high power HET was set: a 1 N thrust, an Isp of about 2500 s for a 20 kW discharge power.

Following the PPS-20k ML characterization campaign, the following operating point is proposed for our 20kW HET thruster with regard to the HiPER requirements:

	PPS-20k ML Operating Point		
	Measured Performances		Proposed HiPER Point
Thrust	1000 mN	1050 mN	***1020 mN***
Isp	2400 s	2700 s	***2500 s***
Discharge voltage	400 V	500 V	***450 V***
Xenon total mass flow rate	43 mg/s	41 mg/s	***42 mg/s***
Total electrical power	19.3 kW	22.4 kW	***20.5 kW***
Total thruster efficiency	60 %	60 %	***60 %***

As was previously mentioned in paragraph II, the PPS-20k ML HET is a dismountable thruster. Neither complete parametric studies nor magnetic optimizations could be performed during this first test campaign. Consequently, possible improvements of the thruster performance deserve future test campaigns.

IV. Conclusion

Within the framework of the HiPER European project, dedicated to high power electric propulsion for exploration, a 20 kW Hall Effect thruster was studied, designed, manufactured and finally tested. The test campaign was conducted in the Pivoine vacuum facility of the CNRS. The prototype, designated PPS-20k ML, was successfully operated from 2.6 to 23.5 kW. The xenon mass flow rate was varied from 5 mg/s to 42.5 mg/s and the discharge voltage from 100 V to 500 V. A maximum thrust of 1050 mN was obtained with an Isp of about 2700 s. The maximum total thruster efficiency was 68% at 300 V. In the 5 kW to 22 kW range, a quasi-constant total efficiency of about 60 % was measured with a thrust-to-total power varying from 50 to 60 mN / kW and a cathode-to-ground voltage of about -16 V. These tests results will be used to finalize the Mission analysis studies, initiated in the HiPER project.

Acknowledgments

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