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VEN μ S Technological Payload

The Israeli Hall Effect Thruster Electric Propulsion System

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ABSTRACT

The VEN μ S technological payload, an electric propulsion system with two low power Hall thrusters, is overviewed. The principle of operation of the Hall thruster, its performance and characteristics, the development of the Israeli Hall effect thruster, and thruster ignition procedure are briefly reviewed. The Xenon flow controller, the propellant management assembly, and the power processing unit are also described. The complex and interdisciplinary nature of the propulsion system required for a successful operation of Hall thrusters in space is demonstrated. The VEN μ S electric propulsion system has to cope with some additional challenges: the tight volume and mass constraints, the low and variable available power, and the very large number of operating cycles. The different elements of the VEN μ S electric propulsion system are now in various stages of development and qualification.

1. INTRODUCTION

VEN μ S (Vegetation and Environmental New μ -Satellite) is a joint French-Israeli research satellite, scheduled to be launched by the end of 2009 with the Dneper launcher, carrying onboard scientific and technological payloads. The scientific payload is a multi-spectral camera whose mission is to monitor agricultural areas and water resources. The technological payload is a low power Hall thruster electric propulsion system (EPS) whose mission is to test this technology in space and to demonstrate enhancement of orbit control and orbit transfer capabilities. The EPS includes two small Hall thrusters, which are planned to operate for 2000 hours in thousands of short on/off cycles during the various phases of the mission. A typical operating cycle of a thruster will be 20 minutes on and 70 minutes off and the thruster input power will be varied between 300 and 550 Watts. The thruster specific impulse is expected to be between 1300 and 1500 sec. The development and integration of the EPS is lead by the Space Department of Manor, Rafael, while operational tests are being conducted at the electric propulsion test facility of Soreq.

The main advantage of electric propulsion is the high specific impulse which results in significant savings in propellant mass. These savings could lead to a reduction in mission costs, or to the extension of mission time, or could enable very large Δv missions that cannot be implemented practically with conventional propulsion. At the same time, due to the typical low thrust, electric thrusters are required to have a long operating lifetime, sometime in the thousand hours level. Small spacecraft and micro-satellites could also benefit from the advantages of electric propulsion in large Δv missions provided that high performance can be

obtained under the constraints of limited power and dry mass for the required long operating time. Among the various electric propulsion options, Hall thrusters represent by now a mature technology that combines high performance and compactness. Thrusters at various stages of development have demonstrated high specific impulse and efficiency values in a broad power range, from sub-kilowatt level and up to a few tens of a kilowatt. Typical specific impulse values are around 1500 sec with a thruster electric efficiency of around 50%. The operating lifetime of flight model thrusters is in the thousand hours range. Nevertheless, at sub-kilowatt power levels the performance of Hall thrusters tends to deteriorate as the power is reduced and/or the thruster becomes smaller. More severe is the sharp drop in the operating lifetime.

Electric propulsion R&D in Israel was initiated at Soreq in the 90's as a small scale research effort which focused on low power Hall thrusters. As part of this effort, a test facility, containing two cryo-pumped vacuum chambers and various electrical, magnetic, thermal, thrust and plasma diagnostics, was established. The research has included the design and construction of laboratory model Hall thrusters and an experimental parametric study of various magnetic and geometric configurations of the Hall thruster in a broad range of operating conditions [1-7]. Special emphasis was put on novel approaches to reduce the deterioration of performance and lifetime at low power [8,9]. The work was expanded also to the theoretical modeling of the Hall thruster and to spacecraft–thruster interaction issues through collaborations with H.I.T and the Technion [10-12]. A few years ago, Soreq joined Rafael/Manor in an effort to develop flight model thrusters and propulsion systems, based on the experience and knowledge gained at Soreq. Initially, this joint effort focused on a 600-700 Watts thruster and propulsion system. However, after the initiation of the VEN μ S program, effort has shifted to the 300-550 Watts power range.

To operate a Hall thruster in space requires a complex and interdisciplinary propulsion system that supply and control the propellant flow and at the same time supply and control the appropriate electrical voltages and currents. The VEN μ S electric propulsion system is overviewed in section 2. Then, in section 3, the principle of operation of the Hall thruster, its performance and characteristics, the development of the Israeli Hall effect thruster, and thruster ignition procedure are briefly reviewed. The other main elements of the electric propulsion system are described in the following sections: the Xenon flow controller in section 4, the propellant management assembly in section 5, and the power processing unit in section 6.

2. OVERVIEW OF THE ELECTRIC PROPULSION SYSTEM

A block diagram of the electric propulsion system (EPS) is depicted in Fig. 1. It consists of two Israeli Hall Effect Thrusters (IHET-A and IHET-B), two Digital Xenon Flow Controllers (DXFC), one for each thruster, a Propellant Management Assembly (PMA), and a Power Processing Unit (PPU) which contains a Power Supplies Unit (PSU), a Thruster Selecting Unit (TSU) and a Sequencing & Control Unit (SCU). Two Filter Units (FU) connect the PPU to the thrusters. The EPS overall dry weight is 23 kg and its Xenon capacity is 16 kg where the VEN μ S mission requires only 14 kgs of Xenon. The following sections describe in details the functions and structure of each subassembly.

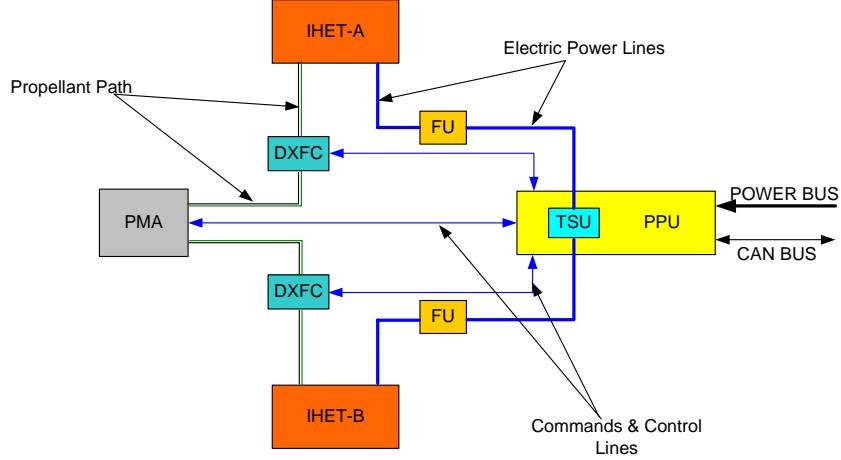


Fig. 1: The electric propulsion system block diagram

3. THE ISRAELI HALL EFFECT THRUSTER

3.1 Hall thruster principle of operation

The Hall thruster is a rocket engine in which thrust is generated as a reaction to the acceleration of quasi-neutral plasma by axial electric and radial magnetic fields in an annular ceramic channel. A conceptual Hall thruster is shown in Fig. 1. The propellant, Xenon gas, enters through holes in the annular anode to the channel where it is ionized. With no magnetic field we would have a strong axial current of electrons accelerated towards the anode under the influence of the applied voltage. The high electron mobility would prevent the existence of a significant electric field inside the plasma and thus no significant ion acceleration would occur. However, the radial magnetic field impedes the flow of electrons towards the anode. Then, the main electron motion is in the direction orthogonal to both the electric and magnetic fields, which results in a strong electron current in the azimuthal direction (Hall current) and thus enables the existence of a significant axial electric field inside the plasma. The electron motion is affected by the external magnetic field in such a way provided that the electron frequency of rotation in the magnetic field, $\omega_e = eB/m_e$, is larger than the effective collision frequency of the electrons with other particles and with the channel walls. Here B is the magnetic field strength, and e and m_e are respectively the charge and mass of the electron. Another requirement is that the electron radius of rotation, $l_{re} = v_e/\omega_e$, is much smaller than the channel length, where v_e is the electron velocity. In typical Hall thruster conditions, l_{re} is less than a millimeter while the channel length is a few centimeters. The much heavier ions are almost not affected by the magnetic field (in typical Hall thruster conditions the Xenon ion radius of rotation is in the meter range), and hence they are accelerated by the axial electric field towards the exhaust.

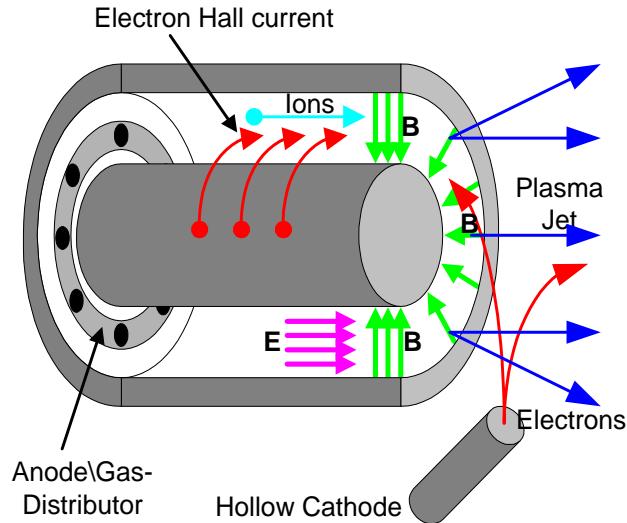


Fig. 2 Schematic drawing of the Hall effect thruster.

Due to collisions with other particles and with the channel walls some electrons do diffuse across the radial magnetic field, giving rise to a small axial electron current towards the anode. It is these electrons that ionize by impact the propellant atoms emerging from the anode. The length of the accelerating channel has then to be large enough to allow the ionization of a substantial fraction of the propellant flow. On the other hand, efficient thruster operation requires to minimize the distance over which acceleration takes place in order to reduce, as much as possible, energy losses due to collisions of accelerated particles with the channel walls. A solution to these seemingly two contradictory requirements is obtained by concentrating most of the voltage drop near the channel exhaust. By that the propellant is ionized in the region closer to the anode where the electric field is small ("ionization region") while effective acceleration takes place over a short distance near the exhaust ("acceleration region"). Such a voltage drop is produced by controlling the electron mobility using a magnetic field distribution that is minimal near the anode and increases towards the exhaust where it reaches its maximal value. Such a field profile is generated with a specially designed complex magnetic circuit [2] consisting of internal and external coils, iron core rods, external and internal concentric pole pieces and additional magnetic elements.

The function of the external hollow cathode is to close the electrical circuit of the anode discharge and to supply electrons in order to neutralize the emerging flow of the accelerated ions. The electron emitting element in the cathode is a low work function insert. The proper operation of the hollow cathode requires that some propellant, typically of the order of 10% of the main flow through the anode, would flow through it. The heat generated due to the discharge inside the cathode helps in keeping the emitter at the high temperature ($>1000^{\circ}\text{C}$) required for the emission of the discharge current.

3.2 Hall thruster performance and characteristics

Like any rocket engine, electric thrusters are characterized by the thrust, T , and the specific impulse, $\text{Isp} = T/mg$, where m is the propellant mass flow rate and g is the acceleration of gravity at sea level. Since spacecraft are typically power limited systems, electric thruster

performance is characterized also by the thruster efficiency, $\eta = \frac{T^2}{2\dot{m}P_e}$. Here, $P_e = I_d V_d$ is the input electric power, where I_d and V_d are, respectively, the discharge current and voltage.

In order to relate the Hall thruster performance to the ionization and acceleration processes we define three additional parameters. Since the thermal velocity of Xenon neutrals is very low compared to that of the accelerated ions, the propellant utilization can be defined as the ionized fraction of the mass flow, $\eta_p = \frac{m}{e} \frac{I_i}{\dot{m}}$, where m is the Xenon ion mass and I_i is the ion current at the thruster exhaust. Then, the specific impulse is equal to $\eta_p v_i / g$, where $v_i = (2e\eta_v V_d/m)^{1/2}$. η_v is the voltage utilization representing the effective fraction of the applied voltage used for acceleration. Factors affecting η_v include ionization costs, cathode voltage drop and energy losses by channel wall collisions. Finally, the current utilization is given by $\eta_l = I_i/I_d$. Then the thruster efficiency can be written as $\eta = \eta_p \eta_v \eta_l$.

A working point of a given Hall thruster is determined by the discharge voltage, the mass flow rate, and the coil current (magnetic field strength). Once V_d and \dot{m} are fixed, the coil current, I_c , has to be large enough in order to minimize the axial electron current and as a result maximize the current utilization. However, if I_c is too large, current oscillations, mainly in the tens of kilohertz frequency range, starts to be significant and as a result the average value of I_d rises again. Thus the dependence of I_d on I_c has typically a minimum [2,6] which usually is the optimal coil current representing the maximal current utilization and hence the maximal thruster efficiency for the specific V_d and \dot{m} .

Experiments show [2,6] that over a wide range of parameters I_d is proportional to \dot{m} while almost independent of V_d . Thus, by varying both V_d and \dot{m} we can in principle operate a Hall thruster of a given size over a wide range of power levels, where the upper power level is usually limited by heat load constraints. Nevertheless, the ability to reduce the discharge voltage is quite limited because it will directly reduce the specific impulse and to some extent also the voltage utilization. For that reason, the more preferred approach to vary the thruster power is to change the mass flow rate. However, a general characteristic of Hall thrusters is that as \dot{m} is reduced the propellant utilization tends to deteriorate due to the reduced chance of ionization by impacting electrons of the lower density propellant in the channel. The lower propellant utilization at lower power levels leads indeed to a degradation of the specific impulse and thruster efficiency. As a consequence, the nominal power level (or range) of a Hall thrusters of a given size is usually relatively close to the upper allowed power level.

3.3 The development of the Israeli Hall Effect Thruster for the Venus mission

The Israeli Hall effect thruster (IHET) for the Venus mission is a flight model thruster developed cooperatively by Rafael and Soreq based on parametric investigations conducted at the electric propulsion test facility of Soreq with Soreq built laboratory model Hall thrusters [1-9]. These laboratory model thrusters have a flexible and modular design which allows to change their configuration relatively easy. Designed to operate in the sub-kilowatt range these thrusters have a channel diameter of 7-7.5 cm. Many thruster configurations including various channel lengths and profiles and magnetic profiles were investigated at discharge voltages of 100-500V, and propellant flow rate of 0.7-3.5mg/s, corresponding to power levels of 100-1000 Watts [1-9]. Different channel ceramics were also tested. One important result from these studies was the experimental demonstration that by extending the

channel length one can partially compensate for the degradation in the propellant utilization at reduced flow rates (power levels) [3,5,6,9].

The transition from a laboratory model to a flight model has been done through the development of engineering model thrusters (EM). Fig. 3 depicts an engineering model IHET. On the left side one can see the flight hollow cathode manufactured by Electric Propulsion Laboratory, Inc. (EPL). The annular ceramic channel and the annular anode are clearly seen. Around the channel one can see also some details of the magnetic circuit: the external and internal poles in the front near the channel exhaust plane, three of the four external coils, and an external screen (large cylinder around the channel) that helps to shape the magnetic field profile. Other details of the magnetic circuit, like the forth external coil and an additional internal coil are hidden in this photograph.

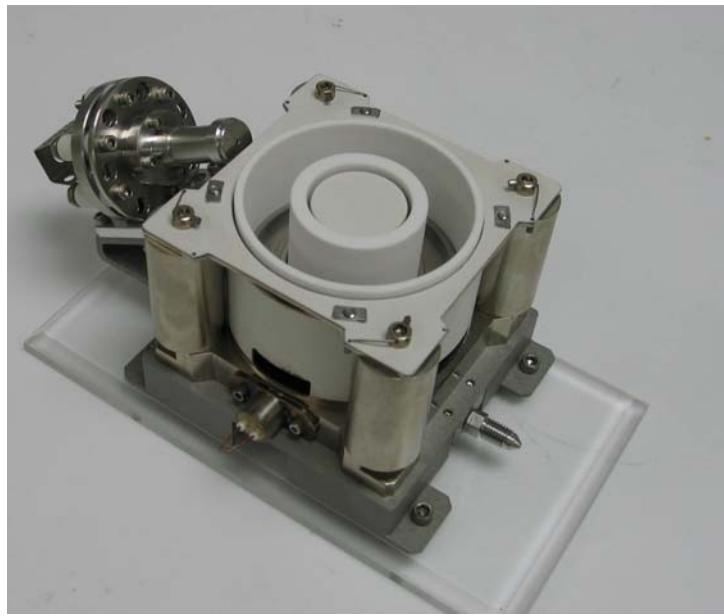


Fig. 3: An engineering model Israeli Hall effect thruster.

The EM thrusters have been used to address two main issues: 1. improvement of the mechanical design in order for the thruster to sustain the mechanical load during the launch phase; 2. the materials and structure of the thruster have to sustain the required operating lifetime (2000h for Venus). The mechanical redesign of the thruster included the change of the channel material. In the laboratory thrusters the channel was made of Boron-Nitride (BN) sintered ceramic. This material is a good electrical isolator, has a high usage temperature, and has a very good thermal shock resistance. However, it has a poor mechanical strength. In the EM thrusters it was replaced by a BN/SiO₂ composite which has a much better mechanical strength. However, as thruster operation experiments have shown, thruster performance is somewhat lower with this ceramic.

The next step was the characterization of the temperatures developed inside the thruster during operation. For that purpose, an EM thruster was built with six thermocouples embedded in different thruster body locations [13]. In addition, a thermal imaging camera was used for the measurement of channel surface temperatures [13]. It was found that the highest

steady state body temperatures were obtained at the internal pole and inside the internal coil, close to 400°C at an operating power of 600 Watts and 350°C at 300 Watts [13]. Channel surface temperature near the exhaust was measured to be about 600°C at 600 Watts and only about 25°C less when the power is reduced to 430 Watts [13]. It took about 1.25 hours for the thruster body to reach 95% of its steady state temperature at 600 Watts and about 1.5 hours at 300 Watts [13]. The temperature measurement results were used to check and calibrate a 3D thermal model of the thruster that take into account both conduction and radiation [13]. This model has been used to get an insight about the heat loads in different thruster parts and as a helping tool in the proper choice of materials that could sustain these loads. In addition, the thruster thermal model was successfully incorporated into the satellite's thermal model (SIND/G) for the design of satellite's thermal control.

The most critical thermal issue in the flight model thruster development has been that of the magnetic coil's wire whose insulation has to sustain a temperature of close to 400°C (internal coil) under vacuum conditions without deterioration or evaporation and, at the same time, is subject to tight volume constraints. The solution found, a copper wire with a relatively thin glass fiber insulation, already demonstrated its potential to cope with these requirements in hundreds of hours of thruster operation tests.

As could be deduced from the description of thruster operation in section 3.1 the anode plays also the role of the propellant distributor in the channel, were the flow emerges from a set of small orifices around the anode perimeter. The propellant is fed to the anode through a narrow pipe and is distributed through its internal structure in order to obtain a flow as azimuthally homogeneous as possible. Azimuthal homogeneity is required in order to minimize thrust vector errors and excitation of undesired azimuthal plasma waves. A computer flow analysis and simulation was performed in order to verify that the anode design indeed provides even gas distribution through all the orifices.

The performance of the EM IHET for Venus (almost final version), after about 400 hours of operation is presented in Table 1.

Table 1: Performance of the EM IHET, after about 400 hours of operation

Mass flow(mg/s)	V _d (V)	I _d (A)	P _e (W)	T(mN)	Isp(sec)	Eff.(%)
1.17	300	0.97	291	14.9	1300	32.5
1.62	300	1.45	435	23.8	1500	40
1.95	300	1.82	546	30	1570	42.5

Channel walls erosion by colliding energetic ions in the acceleration region is usually regarded as the main mechanism of thruster degradation which limits the operating lifetime. Once the channel material is completely eroded at the exhaust the magnetic poles become directly exposed to the plasma, which may cause electrical shortening of the accelerating voltage and thruster malfunction. Even before reaching this point, but when the walls are already very thin as a result of erosion, the temperature of the magnetic poles could rise to a level at which their magnetic properties would deteriorate causing a severe degradation of thruster performance. Thus, after all other mechanical, thermal and flow design issues seems

to be solved satisfactory, channel erosion remains one of the two lifetime challenges to be verified by a thruster operation endurance test (lifetest) of 2000 hours. Nevertheless, erosion studies using laser profilometry after extended tests of hundreds of operating hours with EM thrusters indicates the ability of the flight model IHET to endure such a test.

The second challenge, which is quite unique to the Venus mission compared to other EP mission (e.g. Smart-1), is the very large number, in the thousands, of on/off cycles of thruster operation. The critical element in this regard is the cathode. The cathode manufacturer, Electric Propulsion Laboratory Inc. (EPL)[14], have already qualified the Venus IHET cathode for 2000 on/off cycles in a simple cathode – anode setup. The ability to obtain thousands of on/off cycles with this cathode as part of the IHET will be verified during the lifetest.

3.4 Thruster ignition procedure

Every time the thruster is turned on a set of synchronized actions have to be implemented. In space, this procedure has to be performed automatically. Most of required ignition steps result from the need to condition the hollow cathode. For that purpose the hollow cathode structure includes an integrated heater and an ignition electrode, called the keeper, which covers the cathode. Once propellant flows through the cathode, a relatively high current (11A in the intended flight model cathode) is passed through the heater for 2-5 minutes in order to heat the active element to the high temperature required for obtaining a sufficient thermionic emission of electron current. The typically 60 Volts applied between the keeper and the cathode emitter results in the creation of plasma channel and initiation of electron current between them when the emitter is hot enough. Then these initial plasma and current serves to ignite the main discharge between the cathode and the anode.

Figure 4 depicts a typical thruster ignition procedure performed with the EM IHET and similar to the one that is planned to be implemented onboard the VEN μ S spacecraft. It shows the anode voltage (green, Volts/300), current (red, A) and propellant flow (turquoise, mg/s), magnetic coil current (blue, A), heater current (yellow, A/15), keeper current (purple, A), and thrust (black, Nx100) vs. time (sec). The left side of this graph begins after about 140 seconds of cathode heating and when the anode voltage and propellant flow rate are already on. 20 seconds later the keeper and main discharge start. At this point the magnetic coils are not yet energized in order to facilitate the creation of the main discharge. In order to prevent breakdown in the very low resistance non-magnetic plasma and protect the thruster and especially the cathode, the main (anode) power supply changes its operational mode from voltage regulated to current regulated. In this mode the value of the discharge (anode) current is determined by a preset of the main power supply. Only when the coils are energized, the thruster enters into the proper discharge mode, the main supply returns back to voltage regulated mode and the anode voltage rises back to 300 Volts. At this point the discharge (anode) current decreases to a value determined by the propellant flow. From Fig. 4 we can see the immediate appearance of thrust at this point. The oscillatory behavior of the thrust reading in the first seconds after thrust initiation is a result of the pendulum type [2] thrust-stand response to the abrupt thrust jump and does not represent any real thruster phenomenon. To ensure smooth transition from the intermediate current regulated mode, thruster operation

always starts at the same low power working point. Then the heater current is turned off. Only after the control verifies that the thruster operates properly at this point, a rise to the desired working point (if needed) is performed by changing simultaneously the values of the propellant flow rate and the coil's current. The keeper supply is also turned off.

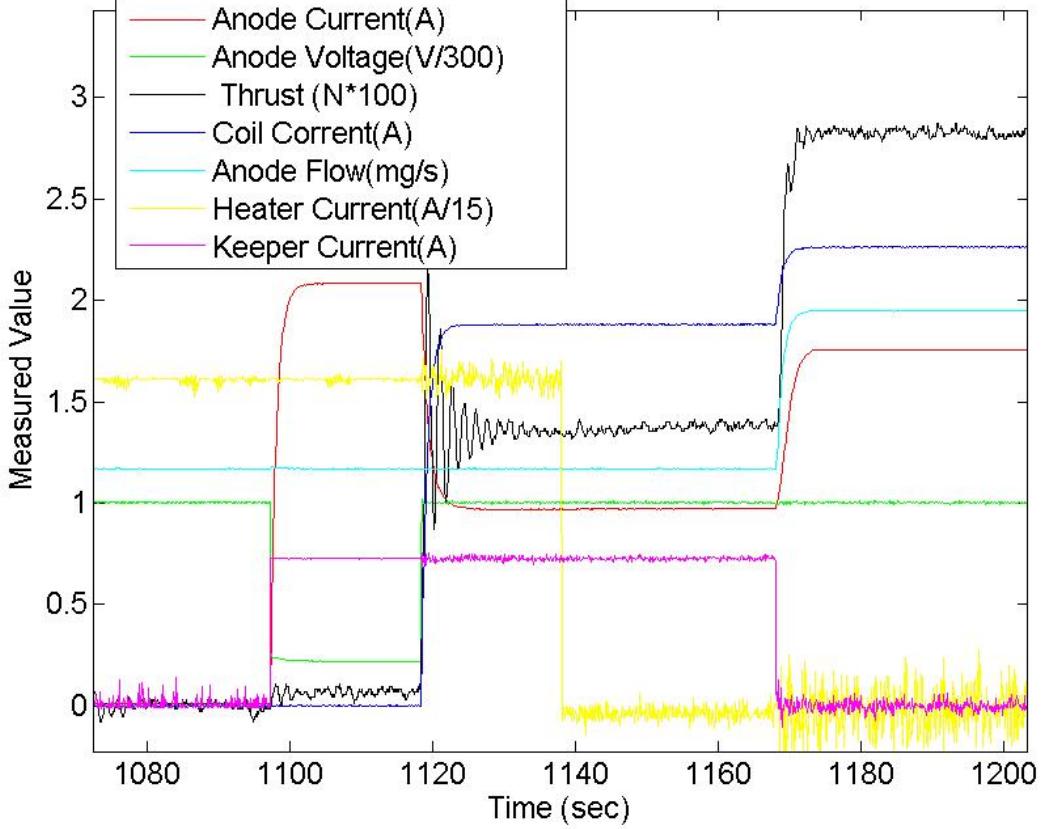


Fig. 4: Thruster ignition procedure: anode voltage, current and propellant flow, magnetic coil current, heater current, keeper current, and thrust vs. time.

4. THE DIGITAL XENON FLOW CONTROLLER

4.1 DXFC control loops

The purpose of the DXFC is to control the Xenon flow rate to the thruster's anode and cathode. Typically, as mentioned in section 3.2, the magnitude of the Hall thruster discharge current, I_d , is linear with the Xenon flow rate. In the VENμS mission the IHET is intended to operate at a fixed discharge voltage, V_d , of 300 Volts. Therefore, the input electric power, $P_e = I_d V_d$, is controlled by the Xenon flow rate. Figure 5 depicts the control loops of the DXFC. For thruster ignition at a given working point the flow controller sets the flow value according to a lookup table. To initiate a new working point while the thruster is at work the flow controller uses the value of I_d for feedback and changes the flow to the required power. In addition, the DXFC controls the flow rate in a closed loop with I_d to compensate for changes of Xenon inlet pressure or inlet temperature.

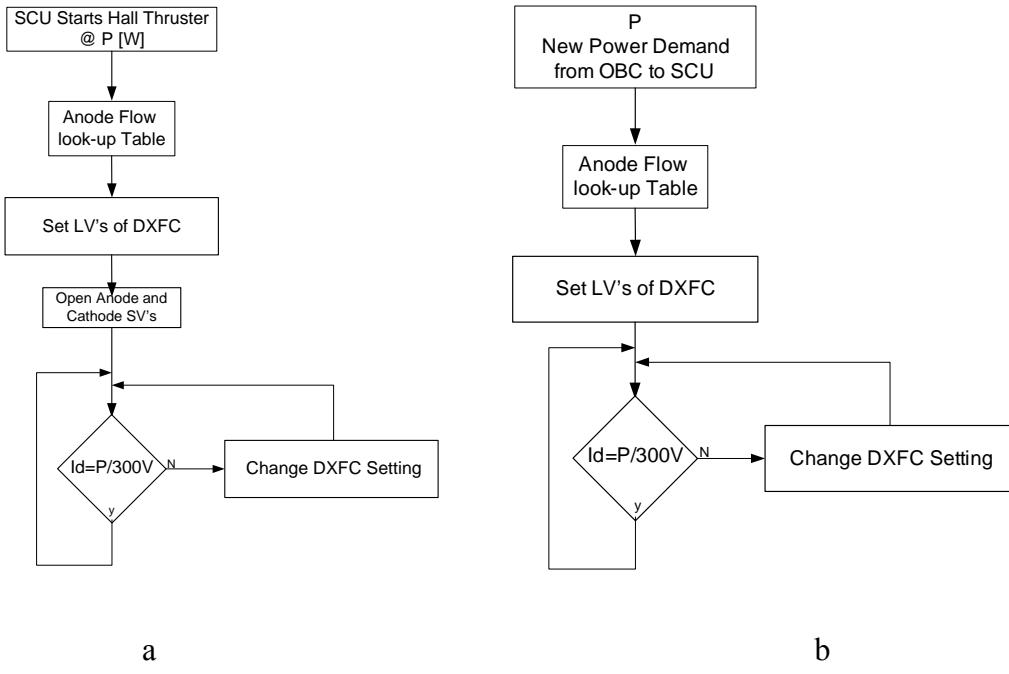


Fig. 5: DXFC control loops; a) thruster ignition stage; b) change of working point.

4.2 DXFC description

Figure 6 shows a schematic drawing of the DXFC which is under development at Rafael. It consists of an anode branch and a cathode branch. The Anode branch contains six flow restrictors, where the flow through five of them is controlled by upstream latch valves. A specific flow rate value is determined by a specific combination of open and closed valves. A diagram of the discrete flow values through these restrictors is shown in Fig. 7. This arrangement enables 32 Xenon flow rate options from 0.5 mg/s to 2.05 mg/s with a resolution of 0.05 mg/s. In the cathode branch there are only two flow restrictors. One of them is for the constant rate at which the cathode operates. The second is a backup used when additional flow is required in case of cathode ignition difficulties.

On both branches, there are additional solenoid valves which are located near the thruster. This arrangement enables fast hydraulic response of filling the thruster chamber and the cathode with the right amount of Xenon in order to save propellant at cycled operation.

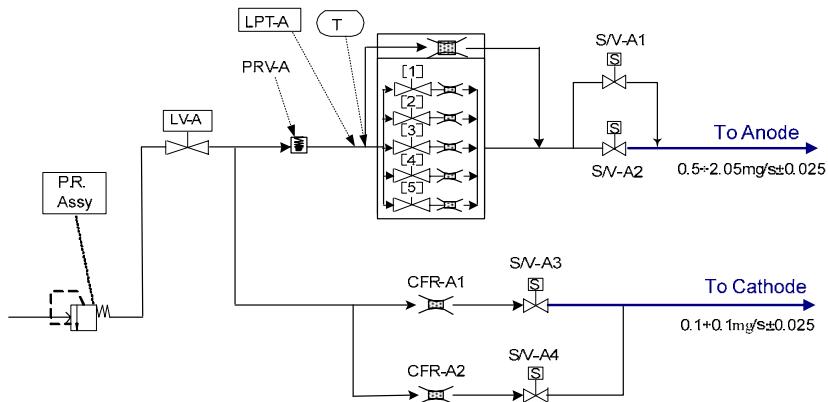


Fig. 6: Schematic drawing of the DXFC

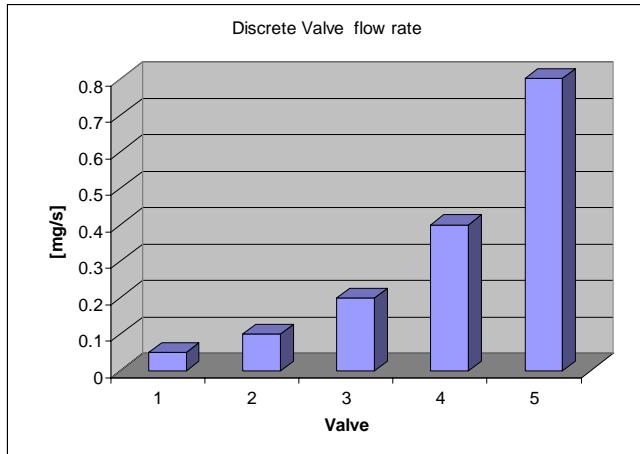


Fig. 7: A diagram of the discrete flow rate values through the restrictors in the anode branch.

The assembly of the DXFC parts is implemented with screwed connectors. The assembled DXFC weighs less than 350 grams including a pressure transducer and a thermocouple. It consumes less than 2.25 Watts which are required to hold the solenoid valves open. The DXFC is now undergoing space qualification tests which include dynamic environment, endurance, and material fitness to space environment and use with Xenon.

5. PROPELLANT MANAGEMENT ASSEMBLY

The PMA enables to load Xenon propellant and store it during the mission onboard the spacecraft and to regulate the pressure up to the low level required for the control of the flow rate to the thruster. The PMA, developed by Rafael and now at different stages of engineering model design and approval, includes the following subassemblies and components: fill and vent valve (FVV), Xenon tank and manifold assembly, particles filter unit, pressure regulator assembly (PRA), and high pressure latch valves.

The FVV that weighs 70 grams provides the interface between the Xenon loading ground system and the Xenon tank. The Xenon tank assembly weighs 3.7 kg with its manifold. It is a titanium tank designed for maximum operating pressure of 140 bars. Figure 8 shows a drawing of the tank attached to the satellite base. Figure 9 shows the pressure vs. Xenon mass in the tank at 325°K and 290°K. The expected pressure values at the various stages of the VENμS mission are also indicated. The xenon pressure will be monitored by a light weight pressure transducer embedded in the tank manifold. In addition, the tank will be thermally controlled to keep Xenon temperature inside between 20°C and 40°C.

The pressure regulator assembly (PRA) consists of two mechanical pressure regulators (spring loaded) which are connected between them to create a reliable and accurate two-stage pressure regulator. The PRA was already qualified for space applications. Within the PMA it will be thermally controlled by thermostats and heaters. The PRA weighs 550 grams. Two High pressure Latch valves each weighing 50 grams connect or disconnect the Xenon flow to the active thruster. The LV working pressure is 138 bars and proved to MEOP of 210 bars. They are located at the low pressure side of the system so actually the factor of safety with regard to pressure is very high.

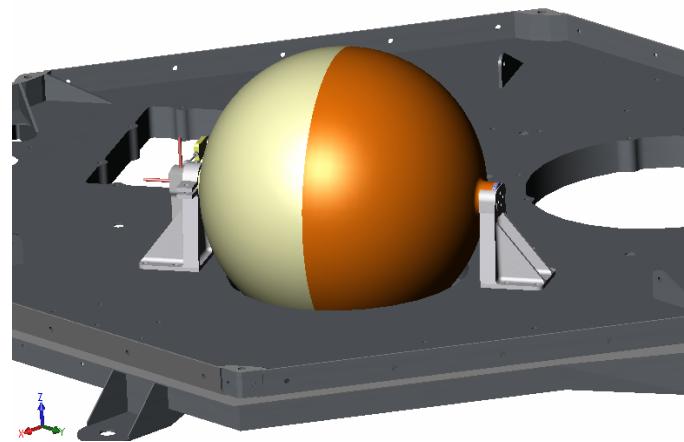


Fig. 8: Xenon Tank polar attachment to satellite base.

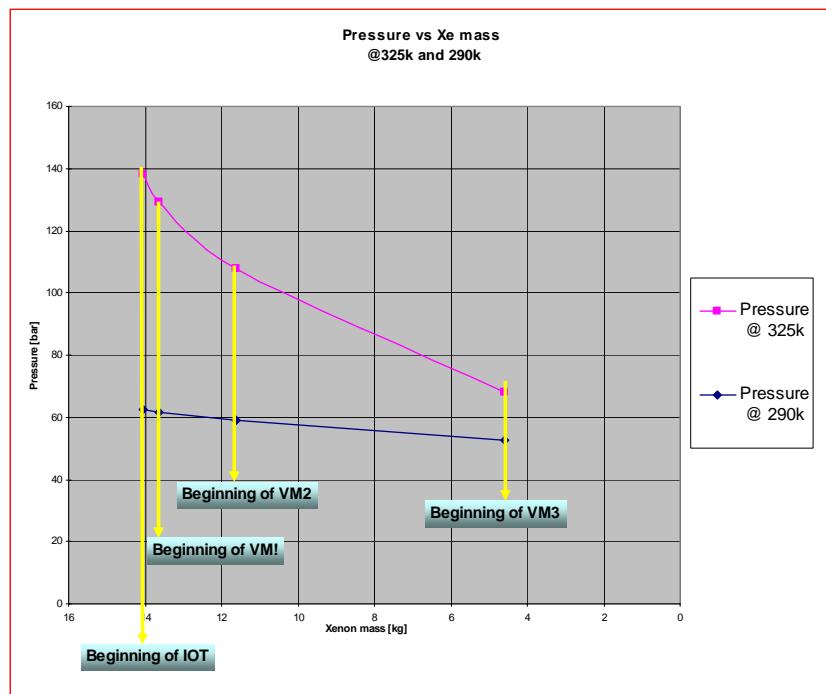


Fig. 9: Xenon tank calculated pressure vs. Xenon mass for the maximum and minimum allowed temperature values. The expected pressure values at the various VEN μ S mission stages are indicated.

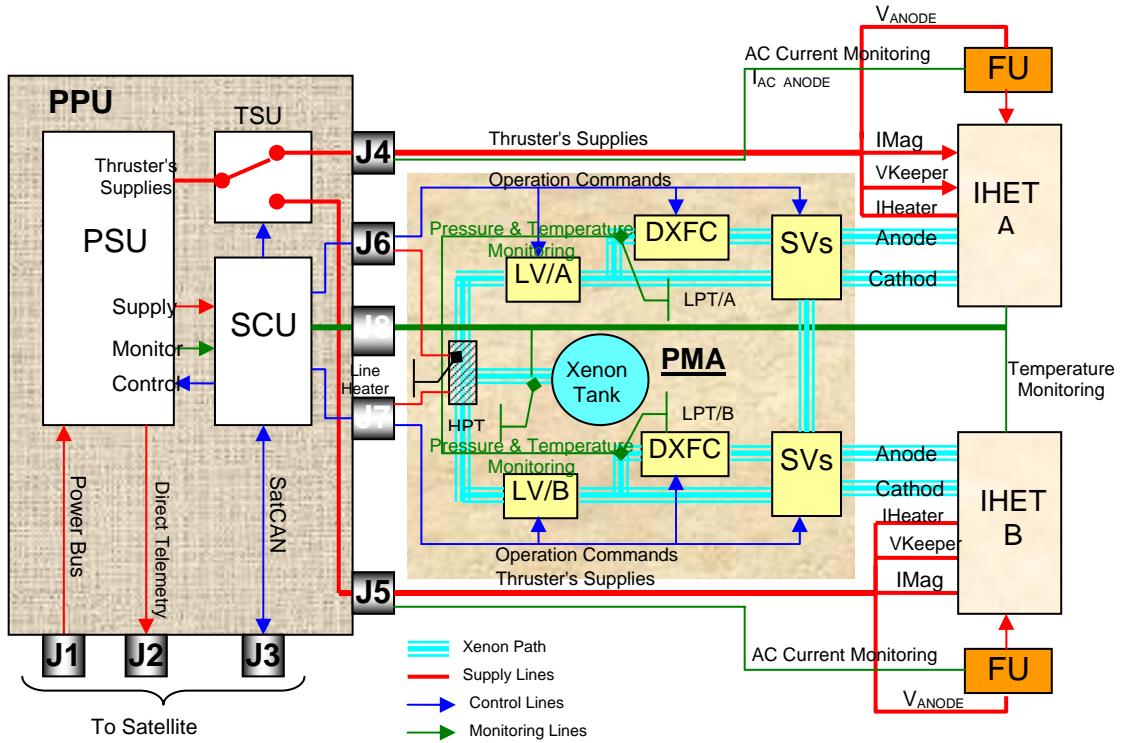
6. THE POWER PROCESSING UNIT

The PPU is responsible for the ignition and regular operation of the IHET. For this purpose, the PPU has to control the Xenon flow and simultaneously to generate the required power and supply the various voltages and currents to the different IHET elements in defined levels and sequence. The PPU serves also as the interface between the EPS and the spacecraft. As such, it has to condition the electrical power from the spacecraft power distribution unit (PDU) to meet the thruster requirements, and to function as an interface between the spacecraft onboard computer (OBC) and EPS. It has also to provide status indications of selected components and to monitor system performance and health signals.

Figure 10 depicts a schematic drawing of the PPU and its interfaces with the EPS subassemblies. The PPU itself comprises of 3 major subassemblies: Power Supplies Unit (PSU), Sequencing and Control Unit (SCU) and Thruster Selection Unit (TSU). The PSU consists of 4 power supplies required for the IHET operation: main (anode) power supply (V_{ANODE}), magnetic coil's current supply (IMag), cathode heater power supply (IHeater), and cathode keeper power supply (VKeeper). In regular operation, the cathode heater and keeper power supplies are required only during thruster ignition. The PSU contains also some auxiliary power supplies used for the powering of the logics and the PMA elements. The SCU controls the operation of the power sources, the Xenon flow path, the monitoring process and the communication management in a strict sequence and schedule. The TSU is responsible to select and electrically connect the active thruster to the power sources according to SCU command received from the ground. Shown also in Fig. 10 are the filter units (FU) which are used to protect the PPU and the rest of spacecraft circuitry from interference due to current oscillations, generated by the plasma during IHET operation. The PSU, FU and TSU are being developed by Enercon Ltd, Israel, while the SCU is being developed by dsIT, Givat Shmuel, Israel. Both groups are guided by Rafael's engineering staff. A breadboard PSU was already tested successfully with an EM IHET. Currently, the main effort is devoted to the development of an engineering model PPU.

The main PPU characteristics are as follows:

- ◆ Main (anode) operating power: may vary between 200W to 600W. Power is supplied into the IHET nonlinear and fluctuating load.
- ◆ Anode voltage: 250, 270 or 300V \pm 1% (2 bits digitally selectable)
- ◆ Anode power supply efficiency: above 87% (forced by satellite's PDU constrains and the demand for floating supplies architecture).
- ◆ Cathode heating power: up to 48W into a variable load (0.4Ω max.).
- ◆ Cathode heating currents: 5,8,9 and 11 \pm 0.5 Ampere (2 bits digitally selectable)
- ◆ Keeper open circuit voltage: 60 \pm 5V (0 to 1mA); Keeper Current: 0.8 \pm 0.05A @ 12-21V
- ◆ Magnet coils current: 1.6 \div 2.6A \pm 0.05% (continuously adjustable)
- ◆ Total weight: less than 11 kg.
- ◆ The PPU design should be protected against total ionization dose (TID) of 10KRad, should be latch-up free and protected from single event gate rupture (SEGR).
- ◆ The design eliminates single point of failure using full redundancy and prevents propagation among circuits.
- ◆ The PPU design should demonstrate reliability of higher than 0.99.
- ◆ The unit has to demonstrate 20000 cycles of operation without failures. Should withstand harsh environmental conditions including EMI/ RFI immunity.



DPS- Digital Processing System

PSU- Power Supply Unit

SCU- Sequencing & Control Unit

TSU- Thruster Selection Unit

FU- Filter Unit

LV- Latch Valve

SV- Solenoid Valve

DXFC- Digital Xenon Flow Controller

HPT- High Pressure Transducer

LPT- Low Pressure Transducer

Fig. 10: Schematic drawing of the PPU and its interfaces with the EPS subassemblies.

7. CONCLUSION

The VENμS electric propulsion system was overviewed. The principle of operation of the Hall thruster, its performance and characteristics, the development of the Israeli Hall effect thruster, and thruster ignition procedure were briefly reviewed. The other main elements of the electric propulsion system: the Xenon flow controller, the propellant management assembly, and the power processing unit were also described. This paper demonstrates the complex and interdisciplinary nature of the propulsion system required for a successful operation of Hall thrusters in space. Moreover, the VENμS electric propulsion system has to cope with some additional challenges: the tight volume and mass constraints, the low and variable available power, and the very large number of operating cycles. The different elements of the VENμS electric propulsion system are now in various stages of development and qualification.

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