Electric Propulsion Thruster Assembly
for Small GEO: End-to-End Testing and Final Delivery

IEPC-2013-222

Presented at the 33rd International Electric Propulsion Conference,
The George Washington University • Washington, D.C. • USA
October 6 – 10, 2013

Olivier Duchemin¹, Vaitua Leroi², Michael Öberg³ and David Le Méhauté⁴
Snecma, Safran Group, Space Engines Division, 27208 Vernon, France

Ramón Pérez Vara⁵
IberEspacio, Magallanes 3, 4a Planta, 28015 Madrid, Spain

Alain Demairié⁶, Mikael Björklund⁷ and Staffán Persson⁸
OHB-Sweden AB, SE-17122 Sölna, Sweden

Marco De Tata⁹ and Stefan Beekmans¹⁰
OHB-System AG, 28359 Bremen, Germany

and

Niccola Kutufa¹¹
ESA/ESTEC, Keplerlaan 1, Postbus 299, 2200 AG Noordwijk, The Netherlands

Abstract: The first satellite of the new European Small GEO platform is currently undergoing assembly, integration and testing ahead of a launch scheduled in late 2014. In the frame of this ESA development program, Snecma was awarded a contract for the development and procurement of an Electric Propulsion Thruster Assembly based on Hall-effect plasma thrusters. Meanwhile, the Propellant Supply Assembly was developed at IberEspacio. This paper provides a brief description of the two assemblies which form part of the Electric Propulsion Subsystem, and reports on the End-to-End test campaign combining both the PSA and a complete flight EPTA branch, which constituted a final and important stage of system verification before delivery.

Nomenclature

\[ F = \text{thrust, N} \]
\[ I_d, U_d, P = \text{discharge current, A (resp. voltage, V; and power, W)} \]

¹ Senior EP Engineer and EPTA Project Manager, Plasma Propulsion Section, olivier.duchemin@snecma.fr
² Propulsion Engineer, Plasma Propulsion Section, vaitua.leroi@snecma.fr
³ Electrical Engineer, Plasma Propulsion Section, michael.oberg@snecma.fr
⁴ Electric Propulsion Test Engineer, Plasma Propulsion Section, david.lemehaute@snecma.fr
⁵ Thermo-Fluid Engineer, rpv@iberespacio.es
⁶ Head of Propulsion, alain.demaire@ohb-sweden.se
⁷ Propulsion Engineer, mikael.bjorklund@ohb-sweden.se
⁸ EPPS Project Manager, staffan.persson@ohb-sweden.se
⁹ Electric Propulsion System Engineer, marco.detata@ohb-system.de
¹⁰ Electric Propulsion System Engineer, stefan.beekmans@ohb-system.de
¹¹ Propulsion System Engineer, niccola.kutufa@esa.int
Idosc = discharge current oscillations amplitude, A
Im = magnet trim current, A
Isp = specific impulse, s
Itt = thermothrottle current, A
m[dot] = total mass flow rate, mg/s
U[CRP] = cathode reference potential, V
η = thrust efficiency, %

I. Introduction

In the summer of 2009, Snecma (Safran) was awarded a contract to develop, manufacture and deliver an Electric Propulsion Thruster Assembly (EPTA) for the Small GEO program. Small GEO is a general-purpose, small geostationary satellite platform developed within the framework of an ESA ARTES (Advanced Research in Telecommunications Systems) program. ARTES 11 is the specific element dedicated to the development and implementation of the Small GEO system – a geostationary communications satellite that, in partnership with Hispasat, will fly the Luxor platform developed by a consortium led by the German company OHB System AG, and including OHB-Sweden AB, RUAG Space AG of Switzerland, and LuxSpace of Luxembourg. Small GEO (Figure 1) accommodates payloads up to 400 kg in mass and 3.5 kW in power, for a launch mass into Geostationary Transfer Orbit (GTO) of up to 3 tons depending on satellite version. One version relies on direct injection into GEO by the launch vehicle, whereas another (hybrid) version features an optional MON-MMH chemical Apogee Engine Module (AEM) to transfer the satellite from GTO to GEO. Another version of the platform uses solely chemical propulsion, whereas future versions are planned where electric propulsion will be used both during transfer and on station.

The first Small SGEO satellite, Hispasat Advanced Generation 1 (HAG1), is due for launch in late 2014 on board an Ariane 5 ECA. It is a hybrid version where the spacecraft will rely on the Electric Propulsion Subsystem (EPPS) for all routine propulsive tasks once in Geostationary Orbit (GEO) after orbit raising with chemical propulsion. The orbital maneuvers include station acquisition; station keeping for a service life of 15 years; momentum management during GEO mission using reaction wheels; intermediate repositioning, if required; and transfer to graveyard orbit at end of life. For this purpose, the EPPS baselined on the SGEO platform comprises two independent (redundant) branches of four thruster units each, where the eight thrusters are hard-mounted onto the satellite walls, with no thrust pointing mechanism. This is in similar fashion, for example, to the Russian GALS, Express, Sesat, Kazsat, and Yamal family of telecommunications satellites.

This paper provides a brief overview of the EPPS, with particular emphasis on the EPTA and on the Propellant Supply Assembly (PSA), and discusses the final stage of verification of the EPTA and PSA compatibility before the EPTA was delivered in 2012: the end-to-end test.

Figure 1. Artist rendering of the Small GEO platform.

Figure 2. The Electric Propulsion Subsystem for HAG1.
II. Overview of the Electric Propulsion Subsystem

A. Electric Propulsion Thruster Assembly

The Electric Propulsion Thruster Assembly (EPTA) is one of the four assemblies of the EPPS (Figure 2). The other assemblies are the Xenon Tank Assembly (XETA); the Propellant Supply Assembly (PSA), which electronically regulates the xenon feed pressure to the required 2.2 bar upstream of the Electric Propulsion Thrusters (EPTs); and the Cold Gas Thruster Assembly (CGTA), which comprises eight cold-gas thrusters. The CGTA can provide reaction control torques on specific instances where electric propulsion is not available, i.e., during detumbling after spacecraft separation from the launch vehicle; and when the spacecraft enters safe mode.

The EPTA comprises the two redundant branches (EPTA1 and EPTA2), which depending on platform configuration may be based on two different thruster technologies. A block diagram is shown in Figure 3 and the complete EPTA design accommodation is represented in Figure 4. Both Figures represent the configuration in which the two EPTA branches are based on Hall-effect plasma thrusters (HETs). In this configuration, each branch can independently operate one Thruster Unit at a time using a single Power Processing Unit (PPU). Thruster selection is done via a 1-to-2 Thruster Selection Unit (TSU), which is internal to the existing PPU, and a 2-to-4 External TSU (ETSU). Each HET branch thus comprises:

- 1 × PPU including internal TSU;
- 1 × ETSU;
- 4 × Thruster Units including Xenon Flow Control (XFC) modules and associated tubing;
- 4 × electrical Filtering Units (FU) with electrical harness and connectors.

Both the PPU and the ETSU are independently powered by the spacecraft Power Distribution Unit (PDU) via a 50-V regulated bus. The PPU communicates with the on-board Satellite Management Unit (SMU) through a MIL-STD-1553B serial data bus. The ETSU, a much simpler unit, communicates using direct telemetry and direct telecommands.

Apart from the ETSU, all hardware units were preexisting and flight proven. The ETSU, although it is a new unit, did not require any new technology as it is based on the TSU hardware. The PPU and ETSU were provided by Thales Belgium.

The procurement contract awarded to Snecma included the development and delivery of a complete EPTA, featuring SPT-100U Thruster Units from EDB Fakel of Russia. It also included delivery of Structural and Thermal Models (STMs) as well as Engineering Models (EMs) of all hardware units, along with Ground Support Equipment (GSE) and simulators in software to support Assembly, Integration and Test (AIT) operations at spacecraft level. All development models were delivered and successfully tested at spacecraft level by OHB System. The flight models are constituted by a first (PFM) branch, which is the nominal EPTA branch of the EPPS, and a redundant (FM) branch.

Detailed descriptions of the main EPTA units and heritage have been given in previous papers. As shown in Figure 4, the eight Thruster Units of the two EPTA branches are mounted onto thruster subassemblies to be integrated onto the spacecraft walls. Each thruster subassembly, in turn, includes the thruster; the XFC and associated xenon tubing; an electrical interface called Hot Interconnecting Box (HIB) and associated electrical harness; and ancillary mechanical supports. All elements are integrated onto a HET Spacer Assembly. A photograph
of one of the eight flight thruster subassemblies is shown in Figure 5, along with PPU PFM1, ETSU PFM1, and an electrical Filtering Unit (FU PFM3).

Figure 4. The Small GEO Electric Propulsion Thruster Assembly for HAG1.

Figure 5. EPTA flight hardware units for HAG1, clockwise from left: thruster subassembly FM5; PPU PFM1; ETSU PFM1; and FU PFM3.
B. Propellant Supply Assembly

The PSA, manufactured by IberEspacio of Spain, is an electronic pressure regulator based on the bang-bang control logic. The SGEO PSA (Figure 6) is composed of two physically independent units: the Pressure Regulator Panel, or PRP (Figure 7), where the fluid control hardware is mounted onto a dedicated base-plate; and the Support and Control Electronics (SCE), the purpose of which is to power, drive, and monitor the health status of the PRP.

The PSA functions are to reduce and regulate the xenon pressure from tank conditions (Maximum Expected operating Pressure, or MEOP, of 186 bar at 50ºC) to an in-flight programmable set point within the range 0 – 4.5 bar, while accommodating the downstream usage conditions. The PSA features an internal closed-loop thermal management which guarantees that the xenon propellant remains in gaseous state. The PSA can manage a flow rate of over 200 mg/s for an upstream pressure greater than 40 bar, down to 25 mg/s for an upstream pressure greater than 5 bar, and down to 10 mg/s with a pressure drop of 0.6 bar on a downstream set-point of 2.2 bar. The estimated total xenon mass throughput capability of the PSA is greater than 650 kg.

The PSA architecture is based on a fully cold redundant scheme, both in the PRP and in the SCE, as highlighted in Figure 6. There is no cross strapping between the two branches. When the PRP is powered off or in standby mode, three solenoid valves isolate the high pressure side from the low pressure side. During PRP operation, one fully closed valve is always present, thus avoiding the exposure of the downstream part to the high pressure side. This is an important safety feature which is incorporated into the design. The PSA provides a fill and drain port (on the high pressure side); a test port (on the low pressure side); and internal filtration units to protect from in/out-coming particle contamination.

The PSA is equipped, for each branch, with one high pressure transducer and two low pressure transducers, allowing fault detection in the low pressure node, and fault identification when using the information from the redundant branch. The internal thermal control loop is also fully redundant. The PSA can be equipped optionally with isolation valves on two of the three outlets to match isolation requirements towards downstream users. The large range of xenon propellant flow rates of the PSA accommodates, in particular, the differing flow rate requirements between the EPTA1 and EPTA2 electric thruster branches, and the Cold Gas Thruster Assembly (CGTA) represented in Figure 2.

The rest of this paper provides a description of the EPTA and EPTA+PSA End-to-End test campaign. This test was carried out in June 2012 at Snecma, Vernon operations, and constituted the final verification stage before EPTA delivery.

III. EPTA+PSA End-to-End Test

A. Test Objectives

The main purpose of the End-to-End (E2E) test campaign was twofold: 1) provide final verification of the complete EPTA PFM branch including power processing and control, xenon flow control, data handling and electrical interface control; and 2) verify by testing the full compatibility of the EPTA and PSA. Prior to the E2E test, all hardware units had been fully acceptance-tested separately, therefore, full performance characterization was
not required. In particular, the specific test arrangement whereby all four HETs were placed under vacuum simultaneously (see Section B) prevented the use of a thrust balance, but this was in no way detrimental to the test objectives.

For this purpose, all the EPTA PFM branch hardware was used in this test, i.e., the four electric thruster subassemblies including their XFC, HIB, flight xenon tubing and flight electrical harness; the electrical FUs (SGEO PFM1 through 4); and the SGE all PPU and ETSU PFM1 units. For the second sequence of the test, which was to verify the compatibility of the PSA and EPTA, the facility xenon supply was disconnected and replaced by the PSA EM unit outside the vacuum chamber.

The fluid interface compatibility between the PSA and the EPTA was verified by analysis and simulations at earlier stages of the development, but the requirement for verification by testing in an appropriate facility remained fully justified. A similar test was performed on the EPPS provided by Snecma for the Smart-1 ESA lunar mission in December 2002.

For the Small GEO application, two operating points for the HETs were defined in the early stages of the program, to fit within the power constraints of the small platform, corresponding to thrust levels of 40 mN and 75 mN. Although the 40-mN operating point has become obsolete from a mission standpoint, full verification of the applicable requirements mandated that the EPTA closed-loop control, and EPTA and PSA compatibility, be tested at this operating point as well.

Table 1 provides a summary of the thruster control and performance parameters at Beginning of Life (BOL) for the two nominal thrust levels required for SGE. In this Table, $I_{th}$ is the thermothrottle current controlling xenon flow rate; $I_m$ is the magnet trim current; $I_d$, $U_d$ and $P$ are the discharge current, voltage and power, respectively; $m_{tot}$ is the total xenon mass flow rate; $F$ is the thrust; $Isp$ is the (total) specific impulse; and $\eta$ is the total thrust efficiency. The need for magnet trim current $I_m$ at low discharge current comes from the fact that the electro-magnet coils are electrically connected in series with the main discharge, so that lowering the discharge current to throttle the engine to low power also means that the magnetic induction field strength is lowered. The magnet trim supply in the PPU provides the possibility of adding the current $I_m$ to the discharge current $I_d$ into the electromagnet coils.

In order to keep the startup power in-rush below the acceptable limit transient on the 50-V spacecraft bus, a low-power startup condition was also defined, drawing from the experience of operating the PPS®1350-G from Snecma on the small Smart-1 spacecraft. On SGE, thruster discharge is ignited near 900 W and 250 V of discharge power and voltage, respectively. The PPU then automatically ramps up the discharge

<table>
<thead>
<tr>
<th>$I_{th}$</th>
<th>$I_d$</th>
<th>$U_d$</th>
<th>$m_{tot}$</th>
<th>$F$</th>
<th>$I_m$</th>
<th>$Isp$</th>
<th>$P$</th>
<th>$\eta$</th>
</tr>
</thead>
<tbody>
<tr>
<td>A</td>
<td>A</td>
<td>V</td>
<td>mg/s</td>
<td>mN</td>
<td>A</td>
<td>sec</td>
<td>W</td>
<td>%</td>
</tr>
<tr>
<td>1.79</td>
<td>4.22</td>
<td>300</td>
<td>5.04</td>
<td>75.2</td>
<td>0.00</td>
<td>1521</td>
<td>1265</td>
<td>44.3</td>
</tr>
<tr>
<td>3.65</td>
<td>2.31</td>
<td>300</td>
<td>3.09</td>
<td>40.1</td>
<td>1.00</td>
<td>1321</td>
<td>693</td>
<td>37.5</td>
</tr>
</tbody>
</table>

Figure 8. Low-power startup, and thruster operating points for HAG1.
voltage to the nominal 300 V within less than 3.2 seconds, at which point the SMU takes over to throttle the engine to the desired thrust level by adjusting the (regulated) discharge current set point. The low-power startup throttle condition, as well as the two SGEo nominal operating points, are represented in Figure 8.

Once the thruster is in operation (at 75 mN for the HAG1 mission), the PPU regulates the flow rate by means of a feedback control loop retroacting on the thermothrottle current \( I_t \) in order to maintain a constant discharge current, i.e., a constant power at the fixed voltage level of 300 V. The PPU output discharge power limitation is set at 1430 W (Figure 8) to ensure that, even accounting for the PPU efficiency, the EPTA input power meets the maximum steady-state power allocation of 1560 W with margins.

The PPU closed-loop control has two main time constants, one (shorter) associated with xenon flow, and the other (longer) with thermal accommodation within the thermothrottle. Because the resulting overall time constant is on the order of 10 seconds, short disturbances such as the instantaneous pressure spikes generated by the bang-bang pressure regulation of the electronic PSA upstream of the XFC are not filtered and will affect flow rate. This, in turn, will generate transient discharge current spikes. If this transient exceeds the limit current at the knee of the PPU V-I characteristic curve, this will further generate a transient voltage drop, as depicted in Figure 9 (detail of Figure 8).

Such a transient behavior does not constitute a problem at EPTA level, but may in principle affect average performance at mission level since a periodic drop in discharge voltage will result in a corresponding drop in specific impulse.

To illustrate this point, we may recall that for the Smart-1 EPPS this (nominal) behavior was expected and observed, both on the ground during the E2E test,\(^\text{9}\) and in flight:\(^\text{12}\) analysis of the EPPS telemetries for various amplitudes of the pressure

---

**Figure 9.** Possible EPTA transient behavior induced by PSA pressure spikes.

---

**Figure 10.** Example of effect of bang-bang-type pressure regulation on discharge current, voltage and power, Smart-1 flight data. In this example at beginning of life, the pressure regulation range was similar to the one planned for SGEo, i.e., 2.0 bar with pressure steps of +0.2 bar.
spikes confirmed the linear dependence of the transient discharge current spikes and voltage drops (Figure 10). The thermostrottle current response was also found to vary linearly with pressure spike amplitude for small or medium perturbations, and to saturate (as designed) for high-amplitude pressure perturbations. Overall, however, the effect of pressure regulation on the discharge power was minor (slightly above one percent). For the Small GEO HAG1 application, the EPTA design featured sufficient margin between the regulated current at 75 mN and the knee current in the V-I characteristic to avoid even experiencing any voltage drop in the nominal case. On the hardware itself, the actual margin is affected by a number of such parameters as dispersion on knee current, DC current setting accuracy, current telemetry digitization error, temperature, or component ageing. A worst case analysis for Small GEO, including maximum expected amplitude of the pressure spikes, showed that a small voltage drop could in theory be experienced when combining all worst situations. This would impact the Isp by less than 0.3%, which is within the mission propellant budget. Naturally, one of the main purposes of the EPTA and PSA compatibility testing was to verify this behavior on the PFM hardware.

B. Test Setup and Commissioning

The EPTA and PSA E2E test was carried out in June 2012 in the LI-C test facility of Snecma, Vernon operations located in Normandy, France. The requirement was for each individual thruster to be tested sequentially, but in order to avoid multiple ventings and openings of the vacuum chamber, the entire EPTA PFM branch, including the four thruster subassemblies, were placed inside the vacuum chamber. This configuration rendered possible simple thruster switching by means of the PPU and ETSU during the testing sequence, in similar fashion to flight operations. In fact, for each E2E test sequence, all four thrusters could be operated, each on both cathodes in turn, within one day of testing.

The LI-C vacuum chamber is a 2.2-m diameter, 5-m long cylindrical test chamber which is cryogenically pumped down to a base pressure lower than 1×10⁻⁶ mbar. In order to fit the EPTA PFM branch under vacuum, the thrust balance and beam probe arm, which are normally mounted within the vacuum chamber, were removed and a large mounting structure was manufactured and integrated within the chamber to hold the EPTA hardware and the thermal regulation interface. Thermal control was implemented in order to ensure that all EPTA hardware temperatures remained well within the qualified limits at all times. Because the flight internal EPTA electrical harness was used, the mounting arrangement was to account for a relative placement of the PPU and ETSU representative of the accommodation within the spacecraft in order to meet the harness length, connector backshell orientation, and radius of curvature constraints (Figure 11).

A principle schematic of the test setup is represented in Figure 12. In this Figure, the EPTA hardware, including flight harness and xenon tubing between the XFCs and thrusters, is placed under vacuum and mounted onto thermally-regulated mechanical interfaces. The PPU and ETSU are driven by Electrical Ground Support Equipments (EGSEs) through a Line Impedance Stabilization Network (LISN) in order to simulate the spacecraft data (TC/TM) and power interface, including dynamic behavior.
For the first E2E test sequence, where only the EPTA was implemented, the xenon propellant was provided directly to the XFCs by means of a (mechanically) pressure-regulated facility distribution system. After complete EPTA E2E testing, however, the PSA was implemented as shown in Figure 12, and both the EPTA1 and EPTA2 PSA outlets were tested. This is because, as per the possibility noted in Section II.B, the EPTA1 on the HAG1 EPPS may be fitted with an optional isolation valve and it was necessary to check the EPTA and PSA compatibility also in the case where the extra pressure drop of that valve is present. To ensure proper test representativeness, the volume of the xenon pipe work upstream of the XFC was adjusted to match that of the flight accommodation between the PSA and the EPTA.

The last stages of commissioning of the test setup and EGSE test software were performed using thruster simulators as well as qualification models for the PPU and ETSU. A picture of the setup external to the vacuum chamber during the EPTA and PSA E2E test is shown in Figure 13.

C. Test Sequence

The test was subdivided into two main sequences: the first one involved E2E testing at EPTA level only, i.e., with the test facility xenon supply; and the second one during which the facility xenon supply was disconnected and the PSA, outside the vacuum chamber, was connected instead. For each sequence (EPTA only, then EPTA+PSA), all thrusters were operated on each cathode and at both SGEO nominal operating points, i.e., 75 mN then 40 mN. During each operating cycle, the thruster was first throttled by the PPU EGSE to 75 mN for a minimum duration of 20 min., and then throttled to 40 mN, again for a minimum of 20 min. Altogether, therefore, each thruster was started a total of four times, twice with the lab xenon supply, and twice with the PSA, and operated for a minimum total duration of 160 min (4 × 20 min. on each cathode). The start sequence was
driven by the PPU using the automatic execution mode, as per Figure 8.

D. End-to-End Test Results

After pump down of the chamber and cold outgassing of the EPTA under high vacuum, the E2E test began by
the xenon line venting sequence. The sequence which was followed was the one programmed into a dedicated PPU
mode for flight operation, where all XFC valves for a given thruster subassembly are open simultaneously. This
permitted verification of this mode as well.

Following successful venting, the firing sequence was then initiated and each thruster was first operated one after
the other, with the facility xenon supply (EPTA only). Thruster selection was operated using the PPU and ETSU.
Figure 14 shows two thrusters of the EPTA PFM branch during operation in the LI-C test chamber. On the
photograph, the plasma beam generated by the thrusters appears saturated because the time exposure was set to
render the non-firing thrusters visible as well.

One of the test objectives, as discussed in Section III.A, was to provide verification by test of the EPTA
interfaces, including the spacecraft power bus interface. To reach this objective, the primary (bus) current was
measured in three different ways: using the PPU telemetry (TM) and using the bus current measured by the PPU
EGSE, which provided slow (1 Hz) data; and using active current probes placed upstream of the PPU, which
provided fast data (10 MHz). The agreement between the PPU TM and EGSE primary current measurement was
found to be better than 2.2% (worst case), which is within the TM accuracy reported in the PPU Interface Control
Document (ICD). No current probe could be placed under vacuum, e.g., between a FU and associated thruster,
because of the (shielded) flight harness used. The thruster discharge current oscillation is nevertheless provided by a
probe located within each FU and transmitter via the PPU telemetry.

To illustrate the EPTA behavior at both 75-mN and 40-mN thrust levels, the primary current and bus power
consumption as measured on thruster PFM1 (cathode B) are shown in Figure 15 during ignition and for both thrust
levels. The current and power drawn on the primary bus prior to ignition ($T_0$) are dominated by the cathode heater
consumption during the pre-heat phase, with the XFC valves and thermosthrottle also contributing. Figure 16 shows
the thruster discharge parameters during the same operating cycle: discharge current $I_d$, and voltage $U_d$, discharge
current oscillations $I_{dosc}$; and Cathode Reference Potential (CRP). The discharge current set point is indicated by
“IDconsigne”.

The fast measurement of the ignition transient on the primary bus was acquired by the current probes and
captured by an oscilloscope. The peak current transient measured during each one of the 16 thruster ignitions was
found to be significantly lower in amplitude than that which was predicted by analysis, and measured at the PPU
input on a thruster simulator during PPU acceptance. In all ignition cases observed during the E2E test campaign,
and taking into account a maximum 3-sigma upper bound, this transient was demonstrated to remain within
acceptable limits at the spacecraft bus interface.

Figure 14. EPTA thruster operation during E2E tests. Left: PFM1; right: PFM4.
Figure 15. Primary current and power consumption during EPTA operation at 75 mN and 40 mN. Thruster PFM1, cathode B.

Figure 16. Discharge parameters during EPTA operation at 75 mN and 40 mN. Thruster PFM1, cathode B.
Figure 17. Discharge parameters during PSA+EPTA operation at 75 mN and 40 mN. Thruster PFM1, cathode B.

Figure 18. Flow parameters during PSA+EPTA operation at 75 mN and 40 mN. Thruster PFM1, cathode B.
The thruster discharge and flow parameters when connecting the PSA to the EPTA for the same operating sequence are shown respectively in Figure 17 and Figure 18. The effect of the pressure spikes generated by the bang-bang cycles of the PSA are readily visible and remained both consistent with expectations and below the authorized limits. In order to highlight this effect, Figure 17 (PSA+EPTA) can be directly compared to Figure 16 (EPTA only). As expected from analysis, the voltage remains constant is not affected at all by the PSA bang-bang cycles. As discussed in Section III.A (Figure 10), this is in contrast with observations made on other systems which were deliberately operated closer to the knee current of the PPU V-I characteristic. Again, a voltage drop would have no detrimental effect on the EPTA, and would have a negligible impact on average EPTA performance; however, the demonstration by test was useful for the purpose of system verification and for accurately predicting flight behavior.

Regarding steady-state performance, it can be concluded that the power consumption never exceeded the required values on the bus line during EPTA operation at both 75 mN and 40 mN thrust levels, whether the EPTA was operating off of the laboratory xenon supply or off of the PSA.

The maximum allowable peak level of 1600 W during operation with the PSA, including the effect of the pressure spikes, is never exceeded. The bang-bang pressure spikes generated corresponding discharge current spikes of approximately 0.3 A (7%) at 75 mN, and peaks of 0.8 A on the thermosthrottle current. At 40 mN, these values decrease to 0.2 A (8%) and 0.5 A, respectively. The duration of the transients, before the thermosthrottle closed-loop flow control implemented in the PPU could effectively regulate back the discharge current to its set point, was less than 5 seconds.

IV. Conclusion

A comprehensive End-to-End test campaign was carried out in June 2012, implementing the EM model of the Propellant Supply Assembly and the PFM branch of the Electric Propulsion Thruster Assembly. Both assemblies form part of the Electric Propulsion Subsystem for Hispasat Advanced Generation 1, the first satellite based on the new Small GEO platform. In this test campaign, all EPTA PFM branch hardware was placed under vacuum, including flight harness and flight xenon tubing between the XFCs and thrusters. The test setup permitted operation of all four thrusters on the branch in sequence, by simple switching of the PPU and ETSU.

The E2E test campaign was entirely successful and provided final verification by testing of the EPTA power, fluid, and data interfaces before delivery to the spacecraft prime contractor (Figure 19). In particular, the startup current and power transients were measured at the spacecraft power bus, and proved lower than worst-case expectations and simulations at unit level. The effectiveness of the PPU closed-loop control of the discharge current was also demonstrated for the specific (low-end) discharge current and xenon feed pressure conditions required on Small GEO, including in the presence of the xenon pressure spikes generated by the PSA.

The test also demonstrated the excellent compatibility between the PSA and EPTA. It validated fully the analyses which predicted sufficient margin between the discharge current set point and the knee current on the PPU V-I characteristic to avoid any effect of the bang-bang cycles on the discharge voltage and system performance.

![Figure 19. Preparation for delivery of the eight thruster subassemblies as part of the EPTA for HAG1.](image-url)
References