LiteEPS – A New Affordable System Developed at Rafael for Large LEO Constellations

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This paper presents a novel and simple design of an Electric Propulsion System (EPS) intended to be integrated and employed in the newly conceived satellite constellations. This EPS, codenamed LiteEPS, is developed by Rafael as a demonstration system to prove and demonstrate the feasibility of such a system. The design follows strict guidelines for reliability, redundancy, tailorability, modularity, manufacturability and affordability. LiteEPS will soon be integrated and tested as a whole module, in an end-to-end test setup. It will provide a proof-of-concept of a system that can be easily manufactured in a propulsion production line and be integrated in a satellite production line.

I. Introduction

IN the past few years, the LEO communication constellations are booming high. Great attention is given to fulfill the need of wideband coverage over remote areas in the world. The evolution, and possibly the revolution, of today's fast communication infrastructures have transformed these features into an obvious need that anyone must have in their daily life. This fact emphasizes the lack of coverage and the urgent need to provide fast and affordable service to these remote areas.

Communication services from space were conceived two decades ago, but only nowadays technologies could afford simple and affordable solutions.

Beside the newest communication technologies of transponders, electronics and sophisticated connectivity algorithms' – a modern spacecraft, which makes one part of a constellation, must have a groundbreaking propulsion system. Among few of its functions is the dispersion of satellites from the launcher orbit to the mission orbit, orbit keeping throughout its lifetime the disposal of the satellite at its end-of-life. Such a propulsion system must be capable of a few hundred meters per second ΔV, it must have a small footprint and low mass – to accommodate small spacecraft, and it must be affordable for the "new space" price range satellites[1]. These factors clearly suggest the use of an Electric Propulsion System.

Such a system, codenamed LiteEPS is currently developed at Rafael. LiteEPS draws its roots from past development activities at Rafael during the past couple of decades, in which Rafael is developing Electric Propulsion Systems (EPS) and components[2]. Notable few are the Venμs satellite EPS, fully developed and qualified by Rafael[3]. This system was launched in 2017 and will provide a sophisticated orbit control scheme using its two Hall effect thrusters operating at a low discharge power level down to 300 W. Another notable achievement is the Micro-satellite Electric Propulsion System (MEPS) project, developed in cooperation with Sitael, under the cooperative umbrella of European and Israeli space agencies (ESA and ISA). MEPS is intended to further reduce the power range needed from a platform, and will operate at discharge power level down to 100 W[4].

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II. General Needs and Requirements

Satellites constellations should generally be quite cost effective, leading to the requirement for a simple and cost effective propulsion system. Table 1 lists the general requirements and guidelines for electric propulsion systems for satellite constellation. Since each launch involves the delivery of multiple satellites, to be positioned in orbit, the immediate need is to allow the satellites to reach their mission orbit independently. The simple maneuvers are dispersing the satellites equally separated in the orbit plane. More complex maneuvers include changing orbit inclination or changing the right ascension. For the mission duration, the satellites need to correct their orbits, especially those in LEO. And finally, each satellite needs to carry the ability to dispose itself, in order to comply with international laws, but more importantly – to vacate its location within the constellation for a new replacement satellite.

All these maneuvers may require a large ΔV that typically ranges from tens to a few hundred meters per second. This need implies on a large capacity of propellant, usually Xenon, but may also be Krypton or alike.

To be cost effective, the satellite must have a small footprint and low mass. Since a part of the launch cost is directly proportional to payload mass, a lightweight and small satellite will be preferred. A typical satellite, making one piece of a constellation, will have a mass of a few hundred kilograms. Traditionally, the propulsion system mass is 10% to 20% of the satellite mass – resulting in a few tens of kilograms for the entire EPS.

A high specific impulse system is preferred, which results in lower propellant mass and aid in keeping the satellite lightweight.

Finally, the cost of the EPS must also be proportional to the cost of the satellite. Traditionally the EPS cost is about 10% to 20% the cost of the satellite. For example, given that one OneWeb satellite is estimated to cost 0.5 M€[1], then the associated costs of its EPS should be around 50 K€ - which is an unprecedented amount for an EPS ever built!

Table 1. Typical requirements and guidelines for the electric propulsion system of a satellite constellation.

Requirement / Guideline	Value
Maneuvers	Orbit positioning Orbit inclination change De-orbiting
ΔV	O(10-100) m/s
Propellant	Xenon, Krypton
Mass	O(10) kg
Isp	High
Cost	O(10) k€

III. Design Considerations

A. Reliability and Redundancy

Traditional space systems and satellites are designed to be highly reliable, with contingency and redundancies[5]. Consequently, this paradigm, of traditional space, drives to conducting many expensive activities and hardware tests; leading to the large cost of concurrent space system. But all this results in an expensive system, which may be heavy; therefore making the satellite heavier and imposing an expensive launch. This “positive feedback” loop results in large and expensive satellites.

The “new space” paradigm, tries to reverse the traditional trend and to impose a “negative feedback”, which restrains costs by using cost effective COTS components, performing only essential tests and involves careful considerations of reliability and redundancies. This approach would make the propulsion system small, lightweight and cost effective.

But how can this “space reliability” be achieved while ensuring the required traditional mission lifetime performance?

The answer is in the concept of constellation. We can move the reliability and redundancy from the satellite system level – to the constellation level, if it is composed of small, non-expensive satellites. Thus, in case of a system malfunction, it is worthwhile to replace the satellite in orbit with a new one, instead of attempting to repair it on-orbit.

This new paradigm is often heard recently, when satellite constellation integrators, starving to comply with the very low cost of a satellite – request simple propulsion systems, with minimum components and redundancy. For an EPS, a common requirement is a single branch or single thruster, as shown also in Ref.6.

B. Modularity and Interfaces

Integration time and complexity drive the satellite price high. To deploy an entire constellation, many satellites are needed to be manufactured and tested in a short period of time. For such a quick campaign, seldom seen before in the satellite industry, there is a real need of high level integration and modularity, a "Plug & Play" approach as presented in ref.[7]. Traditionally, satellite integrators are spreading the propulsion system components wherever possible in the satellite. The result is a complex and expensive design, integration and testing process, with mixed discipline teams.

To comply with the new paradigm, a Module Approach (MA) must be adopted. A modular EPS is a system that is designed, built and tested by the EPS manufacturer. It is supplied to the satellite and integrated as a whole. Of course, the integration effort is minimal because of the shallow interfaces between the satellite platform and the EPS. Traditional interfaces to EPS may be complicated, because the structure, thermal, electrical and communication interfaces complexity. By adopting the MA – all these responsibilities are shifted to the EPS provider and the satellite integration is shortened.

The module approach can be used on several levels:

- Full independent module: This approach treats the propulsion module as an independent part of the satellite.
- Module as part of the satellite: This approach is similar to the full independent module but treats the propulsion module as part of the satellite.
- Basic module approach: In this approach the propulsion system is supplied as a mechanically assembled and tested subsystem. It does not include the wiring and the thermal elements.
- Subassembly approach: This approach is not a "system module" approach. The propulsion supplier delivers component assemblies ready to be attached to the satellite.

C. Tailorability

Since different satellites fulfill different missions the EPS solution must be tailored to the particular mission needs, without compromising on important factors such as component heritage or component qualification; which are important factors when selecting an EPS for a satellite. Tailorability generally refers to the ability of an EPS to be adapted for a given mission or performance requirement.

One common need that drives tailorability is mission performance. For example, the EPS propellant tank may need to be adapted for the right quantity of fuel and/or thrusters may need to be added for the propulsion system to achieve the required total impulse or ΔV.

D. Design for Manufacturability

EPS manufacturing and testing must be carefully designed, to gain all the essential characteristics, as listed above. Most of all, the system needs to be modular and to have minimum number of components with common interconnections. System integration must be done in separate stations on batches of subsystems and components, in order to optimize the time and cost spent on each system integration. All components and subsystems need to be batch-tested, to save resources.

A possible integration concept could be to assemble all subsystems and components on the propulsion baseplate and perform an electrical test, a leakage test and a functional interface test. Subsequently, the entire baseplate should be inserted into a testing vacuum chamber for an end-to-end test. All thrusters should be fired in a sequence, to verify proper operation, while only essential parameters should be measured and logged. This type of test will assure that each delivered thruster, as part of the EPS, is operational. The End-to-End test will conclude EPS testing and will provide a system level COC.

At the satellite level, the satellite integrator need only test the interfaces to the EPS. Acceptance environmental tests should be performed at satellite level.

IV. LiteEPS Demonstrator

LiteEPS is RAFAEL's interpretation to a possible EPS solution for a communications satellite constellation. It tries to comply with the above mentioned principles. The system is described herein.

A. Architecture

LiteEPS is a simple EPS based on Hall Effect Thruster (HET) technology. It comprises of a dozen components, as shown in Figure 1 (Left). It has a single branch with no redundancies. It can be tailored according to required performance by sizing the fuel tank and/or adding additional thrusters. The interfaces to the satellite are simple and shallow. LiteEPS is enclosed in a dedicated compartment, as illustrated in Figure 1 (Right). This compartment may represent the propulsion base of the satellite. It is modular and self-contained.

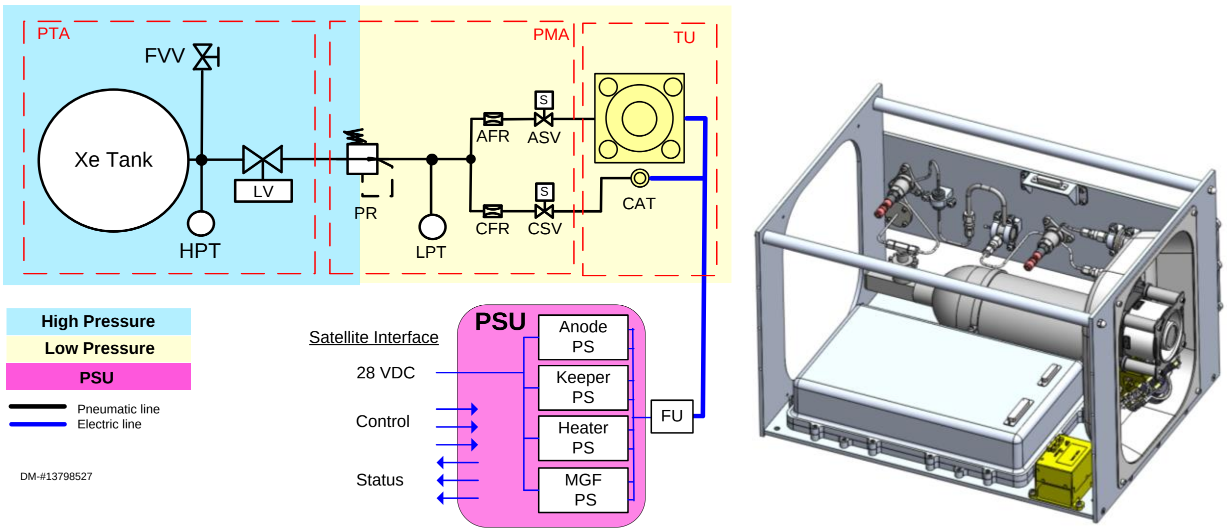

Figure 1. Left: System architecture of the LiteEPS. Right: CAD model of the LiteEPS

The LiteEPS is based on the IHET-300 Hall thruster (see Figure 2), which was developed by RAFAEL, and is now gaining space heritage on the Venµs satellite[8]. In the LiteEPS demonstrator, IHET-300 will be operated at constant discharge power of 300 W. The thruster's discharge power may be adjusted between 250W to 600W, and it was developed for the Venµs mission. As such, the TU is fully qualified for LEO missions. At discharge power of 300 W, IHET-300 produces thrust of 14 mN and anode Isp greater than 1,200 s, at EOL. The total impulse of IHET-300 is greater than 135 kN-s.

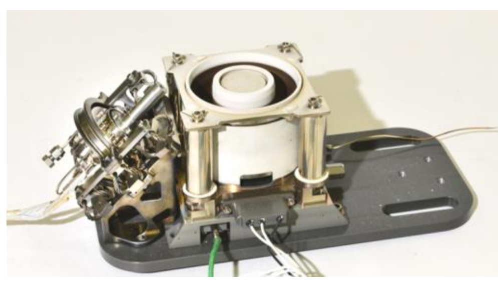

Figure 2. Picture of the IHET-300 Hall thruster.

In order to save cost, the LiteEPS demonstrator is using an open-loop operation of the thruster in terms of operating power - instead of a closed-loop control using an expensive flow controller and electronics for accurate discharge power setting. The architecture is based on fixed flow restrictors for both the anode and cathode feed lines, resulting in discharge power that may vary approximately ±10% due to pressure and temperature variations, and may have an additional ~10% degradation over lifetime. Usually, this might be a reasonable compromise for the satellite integrator. It should not affect the mission, as most of the lifetime the EPS is dedicated to disposal orbit transfer and the operational orbit corrections can be cleverly managed thanks to the low and delicate thrusts.

B. Pneumatics

The LiteEPS pneumatics is simple and straightforward. The high pressure section is shown on the left part of Figure 1 (Left). It comprises of a Pressure Tank Assembly (PTA) to store the propellant which is filled through a Fill and Vent Valve (FVV). A High Pressure Transducer (HPT) is used to monitor propellant tank pressure and a Latch Valve (LV) acts as a flow isolator to the rest of the system. The Pressure Regulator (PR) reduces the pressure, at the propellant tank outlet, to a constant 2 bar. In the low pressure section a Low Pressure Transducer (LPT) measures the pressure in this section. Two valves and two flow restrictors are used to set a constant propellant flow rate to the anode and cathode that compose the Thruster Unit (TU).

The Xenon tank in the LiteEPS demonstrator has a volume of 2 liters and can hold approximately 4 kg of Xe. Depending on the specific mission and fuel requirements, various other propellant tanks can be integrated on LiteEPS.

C. Electronics

The Power Supply Unit (PSU) is the major electronics component. The PSU is a single string of power supplies with no redundancies. The PSU comprises of four power supplies: (1) anode PS that produces 300 VDC to operate the main thruster discharge; (2) magnetic field PS that delivers current to the magnetic coils; (3) cathode keeper PS for cathode discharge generation; and (4) cathode heater PS for cathode heating prior to ignition.

The PSU conducts the dissipated heat, through the bottom section, to the baseplate that radiates it out to space. However, other thermal solutions may be specifically adopted given the specific design of the satellite platform.

Electrical interfaces to the satellite are mainly the operating power, of 28 VDC and a few discrete control lines, as shown in Table 2. A more sophisticated communication line and network may be implemented, but not without additional cost in the form of a dedicated PPU controller. Thus, the satellite on-board computer controls the PSU and the whole LiteEPS operation. The PSU is designed and implemented using COTS for most of its components.

Table 2. LiteEPS electrical interface to the satellite.

In / Out	Name	State	Comment
In	ASV	on/off	
In	CSV	on/off	
In	LV	on/off	
In	ANPS	on/off	
In	MGFPS	on/off	
In	CAK	on/off	
In	CAH	on/off	
In	CAH_C1	0/1	“00” = 5A, “01” = 8A, “10” = 9A, “11” = 11A
In	CAH_C2	0/1	
In	MGFPS_C	0/1	“0” = 1.3A, “1” = 1.9A
Out	ANPS_C	analog	
Out	ANPS_V	analog	
Out	HPT	analog	
Out	LPT	analog	

D. General

The LiteEPS mass budget is shown in Table 3. The presented design can accommodate a propellant tank of maximum 12 kg Xe, assuming the satellite will use the total impulse of the thruster and an extra 10% for the cathode. For shorter missions, a smaller propellant tank may be used.

LiteEPS can be upgraded to two thrusters, in case the mission justifies it. In this case, a Thruster Selection Unit (TSU) will be added, to switch between the operational thrusters.

Table 3. LiteEPS mass budget.

Component/Assembly	Unit Mass [kg]	Qty	Total Mass [kg]
TU (IHET-300 & Cathode)	1.50	1	1.5
Xe Tank 2L (COPV)	1.30	1	1.3
PSU (estimation)	6.00	1	6.0
ASV, AFR, CSV, CFR	0.1	1	0.1
LV	0.10	1	0.1
PR	0.1	1	0.1
FVV	0.07	1	0.1
HPT, LPT	0.1	2	0.2
Manifolds (estimation)	0.40	1	0.4
Harness (estimation)	0.40	1	0.4
Fasteners (estimation)	0.20	1	0.2
Brackets (estimation)	0.50	1	0.5
Total			**10.9**

V. Implementation of a LiteEPS Demonstrator

E. Integration

Currently, the LiteEPS demonstrator is being integrated. Since there is no designated satellite to integrate it on, we designed a propulsion frame to house the EPS, as shown in Figure 1 (Left). Its dimensions are 530x400x350 mm. It follows the modularity and "all-in-one" concepts[1]. These dimensions emulate a real microsatellite of 100 kg to 200 kg class. The system is encapsulated in one module, using only three facets: the TU is mounted on the outer right face. The PPU and propellant tank, which are the largest components of the system, are mounted on the bottom. One important factor is the thermal control and heat dissipation mechanisms of the PSU. The structure takes part in the heat dissipation process and heat radiation out to space. The third facet is hosting the PMA and its components. The tubing is interconnecting the components with threaded fasteners, to enable a quick-to-assemble and affordable system.

F. Testing

Testing the LiteEPS demonstrator is planned in three phases: (1) clean-room/laboratory tests, (2) PMA and TU tests in a vacuum chamber and (3) full end-to-end system test in a vacuum chamber.

The clean-room laboratory tests will verify the workmanship and the assembly correctness of all components and sub-assemblies. On the system level, the pressure and flow values will be tested and calibrated. Also, electrical tests will be performed on each PSU power source to verify the correct output voltages and currents.

The second phase of testing will be in a vacuum chamber. In this test, the entire LiteEPS module, less the PSU, will be mounted in the vacuum chamber. The PSU will externally power the EPS, as shown in Figure 3. The whole system will be controlled by the Satellite Simulator Control Unit (SSCU), which will represent the satellite's on-board computer. The satellite's power supply will be simulated and provided by a laboratory 28 VDC PS. The aim of this phase is to verify and validate the PMA and the TU, both operating in space (vacuum) environment, while controlling and adjusting the PSU operation outside the chamber.

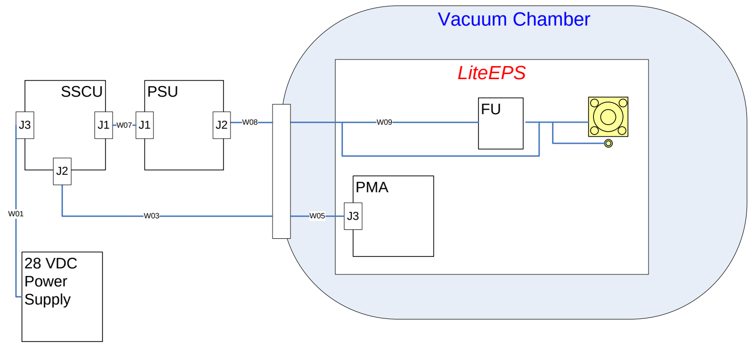

Figure 3. Schematic of the experimental setup of phase II of the LiteEPS tests.

On the third and final phase of the testing, an end-to-end test setup will be operated. The complete LiteEPS will be operated in the vacuum chamber, as shown in Figure 4. The SSCU and PS are the only satellite's representatives, operating the LiteEPS from outside the chamber. In this setup, the full functionality of LiteEPS may be tested, along with the thermal dissipation and heat disposal mechanisms. As the name implies, the end-to-end test is following our concept[1] to factory test and supply a fully tested module, which will be easily and flawlessly integrated with the satellite platform.

The gradual and stepwise test phases are all aimed to proceed with full verification in a controlled manner, where no more than one major component is introduced and tested. Since the TU is already qualified and with space heritage, it is being tested along with the PMA, within the same phase.

The parameters that will be measured in this phase are the power, efficiency, specific impulse and other functional characteristics.

As for the regular, mass production – only phase I and III are planned. After integration in the clean room, the system will be pressurized and tested in an end-to-end fire test. This scheme is intended to be the most cost-effective and functional.

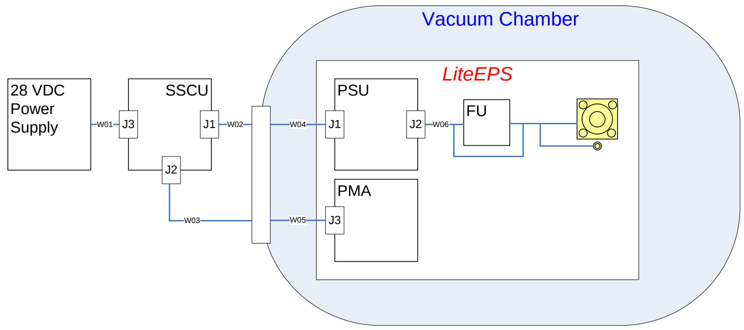

Figure 4. Schematic of the experimental setup of phase III of the LiteEPS tests.

VI. Conclusion and Future Work

This paper presents a novel and simple design of LiteEPS, an EPS designed for new satellite constellations. The concept is using a simple architecture, one branch and no redundancies. Implementation is done with COTS and heritage-based components. The design is modular and tailorable.

Future work will involve a proof-of-concept test of the propulsion system. It will be achieved by end-to-end tests in a vacuum chamber.

An alternative PMA design will also be tested. This design will employ an Electric Pressure Regulator (EPR) that will replace some of the mechanical components. Study results will be published as soon as they are achieved.

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