IL–FEEP: A SIMPLIFIED, LOW COST ELECTRIC THRUSTER FOR MICRO– AND NANO–SATELLITES

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ABSTRACT

The limited mass and volume available make the development of advanced miniaturized propulsion systems a challenge. An advanced version of the Field Emission Electric Propulsion system, namely the Ionic Liquid Field Emission Electric Propulsion (IL-FEEP) thruster, is presented in this paper to equip small platforms with a thruster subsystem able of providing a total impulse of the order of 2 kNs.

The thruster prototype design and the main performance assessed by means of vacuum lab experiments are presented. The complete subsystem is sketched and a strawman 2U Cubesat platform is designed to include such a thruster. From the mass and power budget and the analysis of a number of relevant scenarios it results that, thanks to the IL-FEEP system, such platform can increase its lifetime by years, its LEO altitude by hundreds of km and its inclination by several degrees. The platform may board a set of standard payload equipments, as sensors and cameras. In brief, it is shown that the electric thruster presented is actually a valuable tool to increase the versatility and extend the range of applications of tiny spacecraft.

1 INTRODUCTION

The Ionic Liquid FEEP (IL–FEEP) is a variant of the cesium–fed field emission thruster [1] where the alkali metal propellant is replaced by ionic liquids [2], [3], [4]. Ionic liquids are organic molten salts composed of a mixture of loosely bound cations and anions, with the special property of being liquid at or close to room temperature [5]. Such liquids have been developed by the chemical industry during the last 15 years for their unique properties as process fluids and solvents [6]. The usage of ionic liquids, with respect to alkali metals, leads to considerable simplifications in space and ground operations and the associated supporting equipment. The possibility of contamination to the host spacecraft, as well as the danger of contamination of the thruster from the external environment, is much reduced. In comparison to liquid metals, however, ionic liquid propellants have reduced performance in terms of specific impulse and mass efficiency. In principle, operation in either pure ionic or mixed ion/droplet regime is possible, according to the choice of liquid and the operating conditions. The IL–FEEP thruster design is based on the heritage of the classical Alta cesium FEEP [7] and it is currently funded by ESA and EU programmes [8].

The general trend toward miniaturization of space components and the high standardization allowed by fix specifications and COTS elements, made Cubesat the standard for nano– and pico–satellites [9]. More than 50 Cubesats have been launched in the last decade and more than 60 are going to be launched as piggyback payload within next few years [10].

In general these platforms lack of the possibility of actively modifying and/or maintaining their operative orbit at will. Accordingly, implementing a propulsion system on a Cubesat would be a huge headway towards the definitive success of Cubesat platform. Among the possible applications, the IL–FEEP system might provide autonomous orbital maneuvering and lifetime extension capability through drag compensation (or de–orbiting) to micro- and nano-satellites, such as Cubesats, with limited impact on the onboard resources [11].

This paper summarizes the main performance, together with the experimental campaigns carried out to assess them, of the IL–FEEP thruster. Additionally, the feasibility of a 2U Cubesat equipped with such a device is addressed. The analysis is based on realistic results of experimental tests, on a preliminary mass and power budget and on the mission analysis of a set of potential scenarios taking advantage of the IL–FEEP thruster.

2 IL-FEEP CONFIGURATION

The FEEP system produces a net thrust by extracting and accelerating ions by means of a high electric field applied to the propellant liquid meniscus on the emitter slit [12], [13], [14] Marcuccio, S., Genovese, A., Andrenucci, M., "Experimental Performance of Field Emission Microthrusters", Journal of Propulsion and Power, Vol. 14, No. 5, September-October 1998, pp. 774-781.

[15], [15]. The liquid propellant is required to be electrically conductive and it should have preferably a high atomic mass (to increase the thrust magnitude), a low melting point (to reduce the power consumption), good wetting properties (to flow throughout the ducts), low ionization potential (to facilitate ion extraction) and low vapour pressure (to avoid surface contamination and sparks). Cesium is one of the best options as it copes with many of the previous requirements, but it has an extremely high reactivity with water (including the water vapour present in the air and the water absorbed onto surfaces) which represents a huge drawback for an easy utilization. Accordingly, the usage of cesium as FEEP propellant requires complex and expensive systems to assemble and store the thruster in inert atmosphere and to assure a very high degree of molecular cleanliness for all surfaces.

Ionic liquids, instead, are much more friendly to handle if compared with cesium [16]. This leads to significant simplifications in the thruster design that result in a lightweight and simple configuration. EMI-BF_4, extensively used in literature [17], [18], [19], is the ionic liquid used in the experimental tests and has been chosen as one of the most probable candidates to feed the IL-FEEP system. The main physical parameters of interest of EMI-BF_4 are shown in Table 1.

Density (g/cm^3)	Melting point (°C)	Molecular mass (amu)	Mass anion (amu)	Mass cation (amu)	Conductivity (S/m)	Surface tension (N/m)	Viscosity (Pa s)
1.34	15	197.97	86.805	111.165	1.4	0.05	0.0665

Table 1: Main physical properties of the EMI–BF_4 ionic liquid [20].

Although there are hundreds of ionic liquids suitable as FEEP propellants, the melting point is one of the preferred figures of merit for the choice of propellant. Ionic liquids with lower melting point than EMI–BF_4 exist, but these have worse values of other physical properties, resulting in limited propulsive performance.

As regards the thruster design, the core of the device is represented by two electrodes, the emitter and the accelerator, required to apply the high electric field on the propellant liquid meniscus. The linear slit emitter, one of the more massive components, is made in stainless steel and its geometry is derived from the classical cesium-fed emitter. The accelerator is designed as a simple titanium plate (15x60 mm and 2 mm thickness) with a proper opening for the ion beam exhaust. In the thruster assembly, the accelerator is grounded to the spacecraft bus (or to the vacuum chamber during laboratory tests) while the emitter is biased to the positive or negative potential.

As the thruster is not required to be protected in inert atmosphere during ground operations, a sealing container and an opening mechanism are not required. Besides, the overall configuration is not required to be axisymmetric and the insulation between electrodes is achieved by using two cylindrical spacers made in PolyEtherEther–Ketone (PEEK, [21]), a polymeric material lighter, easier to machine and less expensive than alumina (the material used for the insulator in the cesium-fed FEEP).

The propellant is stored in a non–pressurized tank made also in PEEK. The baseline thruster design includes an embedded tank with an internal volume of about 50 cm^3 that, with the EMI-BF_4 liquid, means 67g of propellant. By considering a thruster specific impulse of 3000 s (a conservative value, considering the experimental tests), these correspond to a total impulse of about 2000 Ns, sufficient for a wide range of applications ranging from orbit maintenance, deorbiting and repositioning (see Sec. 4). This tank, however, is a constitutive part of the thruster and can be modelled according with the specific mission envisaged.

The thruster design includes a ground shield connected with the accelerator electrode encompassing the emitter and the electrode insulator. The shield has the same shape of the tank and is realized with a thin aluminium foil. Preliminary analyses lead to assume a working temperature between –20 and +60 °C and a survival temperature between –30°C and +70°C. These values are intended as temperature ranges at the thruster interface.

3 IL-FEEP EXPERIMENTAL CHARACTERIZATION

The experimental apparatus used for the thruster characterization includes the thruster unit (i.e. the slit emitter, the accelerator, the electrode PEEK insulators, the emitter heaters and the ground shields), the propellant feeding system and a set of probes for ion beam characterization.

The divergence half angles of the ion beam in the vertical and horizontal plane are measured via a couple of electrostatic probes that move in front of the thruster by means of two stepper motors. The beam composition analysis is carried out by using a Time–Of–Flight (TOF) technique. Two probes are used to perform these measurements: a metallic wire (TOF1) placed in vertical position at the centre of the vacuum chamber and a metallic collector (TOF2) placed at the end of the chamber. The distances between the emitter slit and TOF1 and TOF2 are 600 mm and 1300 mm respectively.

The whole apparatus was integrated in one of the Alta's FEEP laboratory vacuum facilities with an ultimate vacuum pressure of 10^{-8} mbar. Two views of the setup showing the feeding system, the ground shield and the two probes are presented in Figure 1.

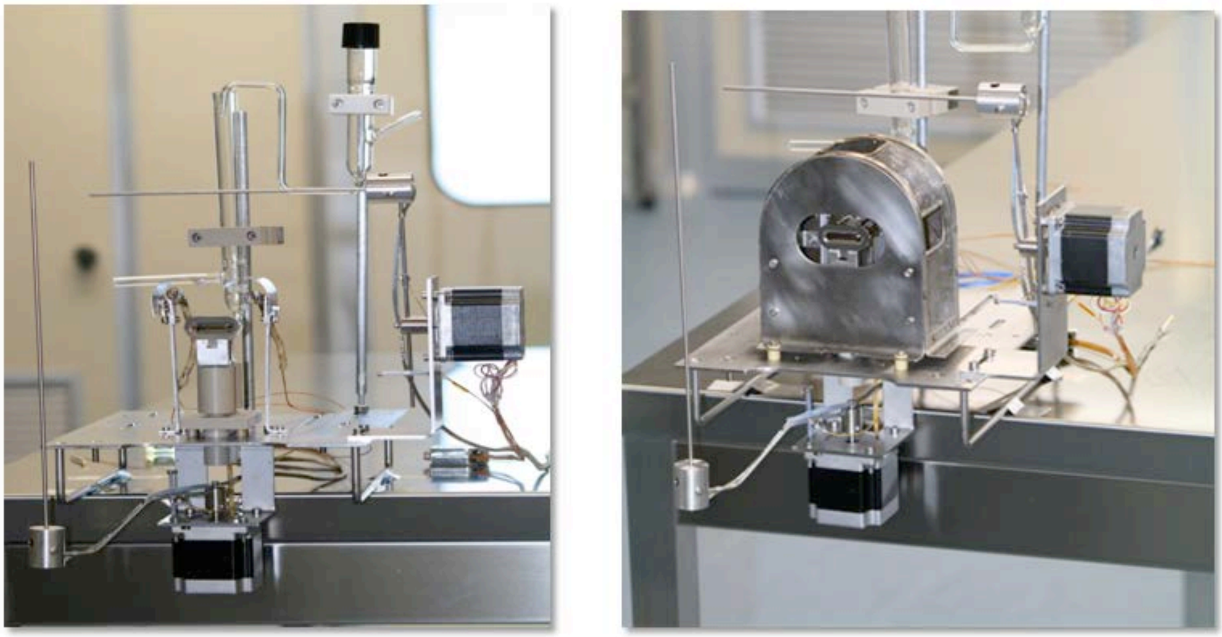

Figure 1: Two views of the experimental setup showing the glass feeding system, the two orthogonal probes and the ground shield around the emitter–accelerator assembly.

The IL-FEEP thruster prototype was operated both in positive and negative polarity. The total firing time was more than 500 hours; for 300 hours the thruster fired in alternate polarity with a switch frequency of the order of 0.1-1 Hz; for the remaining 200 hours the thruster fired in constant positive polarity.

Current–Tension (I–V) characteristics were recorded for different emitter temperatures in the range 20–100°C. The voltage threshold to start emission resulted to be about 7 kV in positive polarity and about -5 kV in negative polarity (Figure 2). A rather strong dependency of the electrical parameters (especially the emitter current) from the emitter temperature was detected.

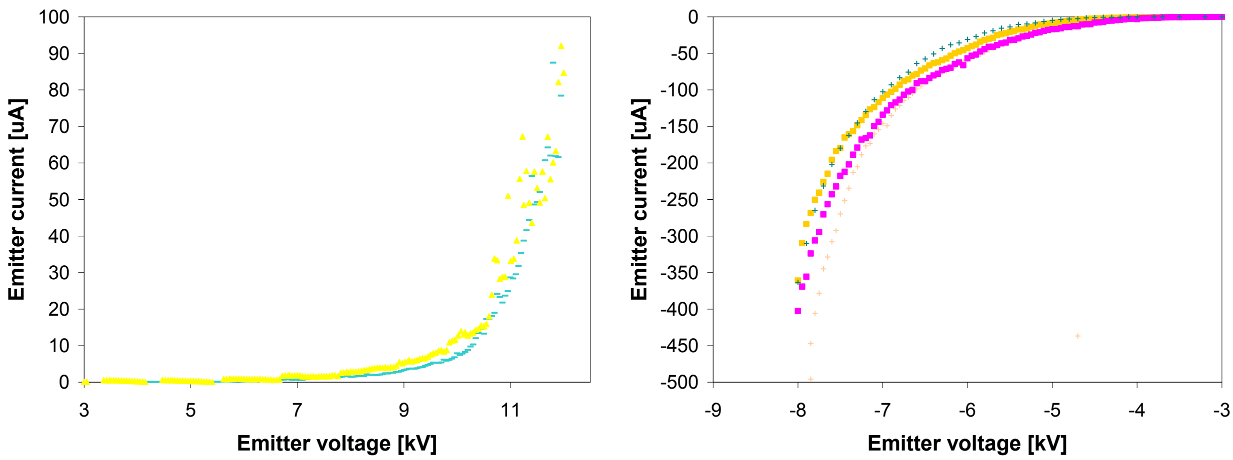

Figure 2: Current-tension characteristic in positive (left) and negative (right) polarity, performed with an emitter temperature of 40°C.

The divergences half angles obtained with the electrostatic probes resulted to be less than 10 deg in the vertical plane and 30-40 deg in the horizontal plane. The beam divergence half angles resulted to be not very influenced by the emitter temperature, while the currents drained by the probe resulted to be highly affected (the probe currents increased of 500% by varying the emitter temperature from 20°C to 100°C, Figure 3).

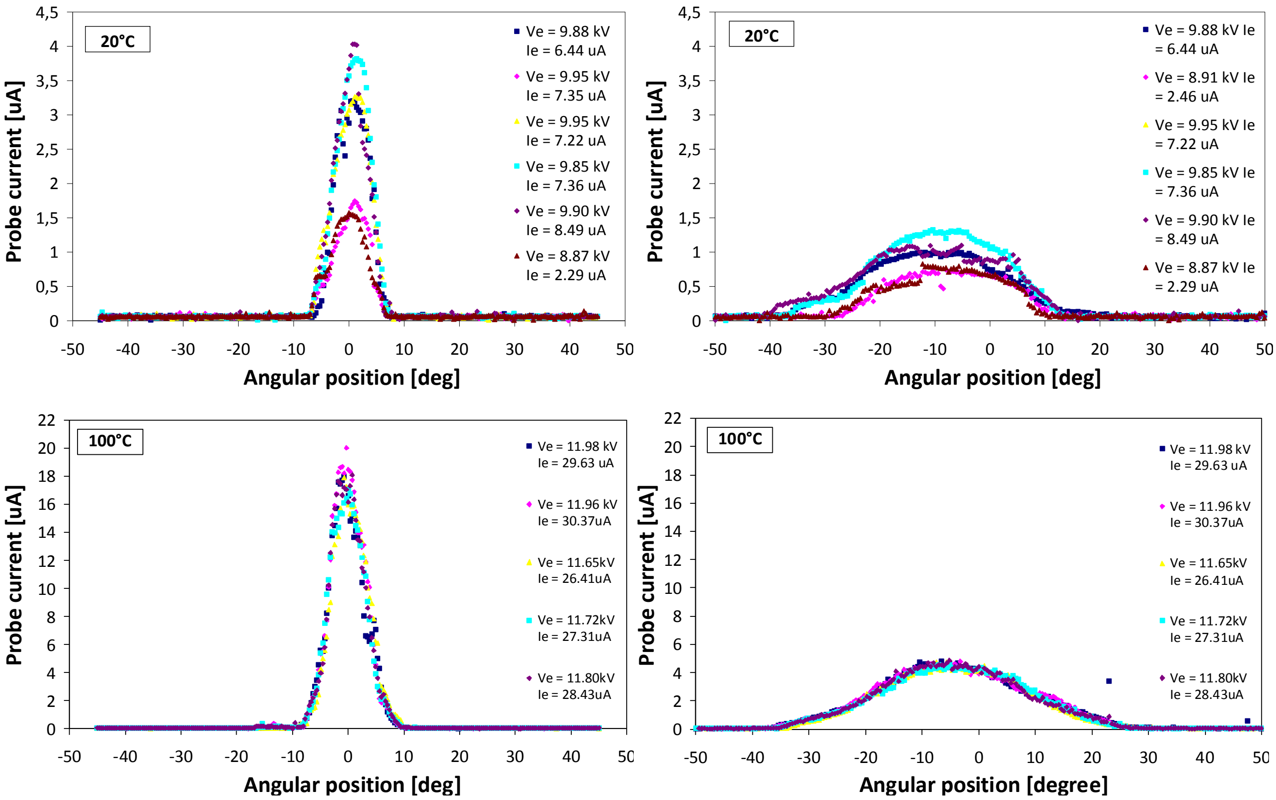

Figure 3: Vertical (left) and horizontal (right) beam current scans at different emitter voltage levels and different temperatures (20 °C up, 100 °C bottom).

Typical output of the oscilloscope for the TOF measurements is like the one presented in Figure 4, obtained with a temperature of 100°C and an emitter voltage of 12.9 kV.

The TOF technique determines the mass-to-charge ratio of the species present in the beam by measuring the different times they spent to reach the target. In the measurements performed, the fastest species are not detected due to the noise of the setup. In Figure 4 the time spent by the slowest species to reach TOF1 and TOF2 is identified by the slope variation of the curve. Accordingly to Figure 4, the velocity of the slowest species in the beam is:

- $v = 0.6\ m / 6.8 \cdot 10^{-6}\ s = 88235\ m/s$ (TOF1),
- $v = 1.3\ m / 15.4 \cdot 10^{-6}\ s = 84416 \cdot 10^{3}\ m/s$ (TOF2).

The mass-to-charge ratio is:

- $m/q = 2\ V_e/v^2 = 12.9 \cdot 10^3\ V / (88235\ m/s)^2 = 3.29 \cdot 10^{-6}\ kg/C$ (TOF1),
- $m/q = 2\ V_e/v^2 = 12.9 \cdot 10^3\ V / (84416\ m/s)^2 = 3.59 \cdot 10^{-6}\ kg/C$ (TOF2).

Assuming only single charge species, the atomic number is:

- $M = 3.2 \cdot 10^{-6}\ kg/C \cdot 1.6 \cdot 10^{-19}\ C / 1.66 \cdot 10^{-27}\ kg = 317.3$ (TOF1),

- $M = 3.6 \cdot 10^{-6}$ kg/C · $1.6 \cdot 10^{-19}$ C / $1.66 \cdot 10^{-27}$ kg = 346.6 (TOF2).

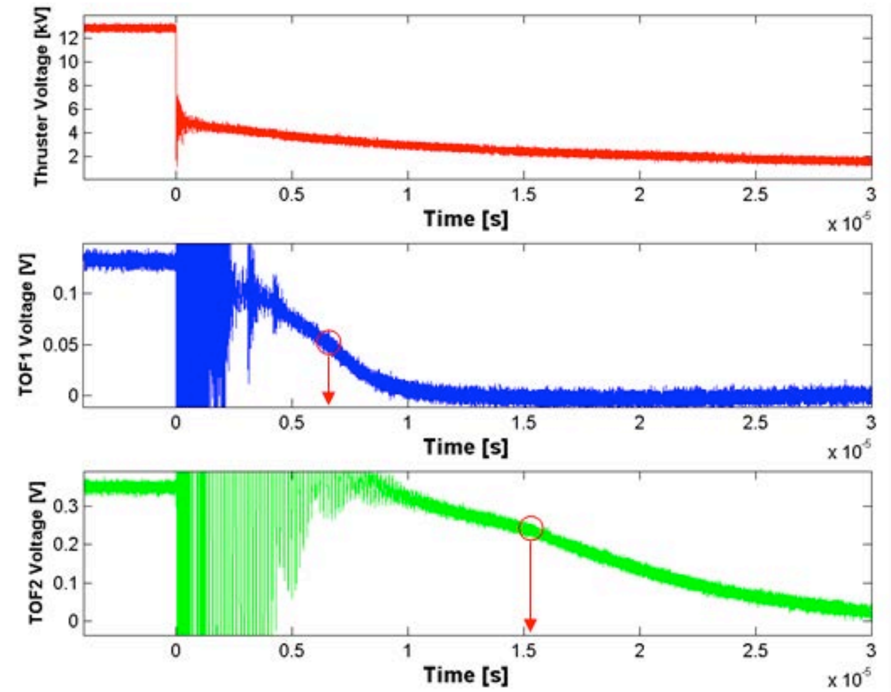

Figure 4: Time-of-Flight measurements. From top to bottom: emitter voltage, TOF1 voltage and TOF2 voltage (T_e=100°C, V_e = 12.9 kV). The slope changes are indicated with the red arrows.

Table 2 summarizes the TOF measurements for the complete range of operating conditions.

Temperature [°C]	Applied voltage [kV]	Distance [m]	Time of flight [µs]	Velocity [m/s]	Mass to charge ratio [kg/C]	Molecular mass [amu]
60°C	9.2	0.6	8.0	75,000	$3.27 \cdot 10^{-6}$	315.6
		1.3	18.5	70,270	$3.73 \cdot 10^{-6}$	359.5
80°C	12.4	0.6	6.9	86,956	$3.28 \cdot 10^{-6}$	316.5
		1.3	15.7	82,802	$3.62 \cdot 10^{-6}$	349.0
100°C	12.9	0.6	6.8	88,235	$3.29 \cdot 10^{-6}$	317.3
		1.3	15.4	84,416	$3.59 \cdot 10^{-6}$	346.6
120°C	11.5	0.6	7.5	80,000	$3.59 \cdot 10^{-6}$	346.7
		1.3	16.6	78,313	$3.75 \cdot 10^{-6}$	361.8

Table 2: Time-of-Flight data summary for 9–13 kV emitter voltage and 60–100 °C operating temperature.

From Table 2, it results that the IL-FEEP prototype, considering a conservative mass efficiency of 50%, has a specific impulse in excess of 3000 s and the beam is mainly composed of EMI^+ (monomer), $(EMI–BF_4)EMI^+$ (dimer), $(EMI–BF^4)_2EMI^+$ (trimer). The small differences between the values obtained from the two probes can be explained by the large differences in probe geometry: the wire probe (TOF1) achieves a local measurement on a restricted portion of the beam, while the target (TOF2) collects the current over a very large beam portion.

4 NANO–SATELLITE MISSION SCENARIO

Assuming the current thruster development status and standard figures for the propellant tank and harness, the total propulsion system dry mass turns out to be about 700 g, including the control and power units required to operate the thruster.

Form the above considerations about the thruster specific impulse and by assuming 100 μN thrust, the 67 g propellant mass considered in the thruster baseline configuration ensures that a total impulse of ~2000 Ns that can be delivered during over 230 days of operations. This results in a maximum ΔV of more than 0.5 km/s for a 4 kg, 3U Cubesat. As a comparison, orbit raising or plane change in LEO require approximately 135 m/s per degree of plane change and 0.6 m/s per km of altitude change. The estimated power consumption of the device is of the order of 4 W, including also the power needed for the control electronics and the thermal conditioning subsystem.

There exist a broad number of possible scenarios where a thruster with such characteristics can be adopted. In particular, the combination of Cubesat and IL–FEEP turns out to be particularly advantageous in performing orbit maintenance operations (e.g. on a 300 km, non-sunsynchronous circular orbit), in a orbit transfer from LEO to MEO (e.g. form a 300 km to a 1000 km SSO orbit), or to perform a groundtrack repositioning of a LEO/MEO orbit.

4.1 IL–FEEP Cubesat configuration

As an example, the design of a 2U Cubesat equipped with such a thruster is outlined in this section. Both the main performance in terms of orbital maneuvers and the spacecraft subsystem design are described. Standard off–the–shelf components are assumed for the main Cubesat systems and the state of the art of the IL–FEEP laboratory prototype is considered as baseline for the thruster performance, mass and power.

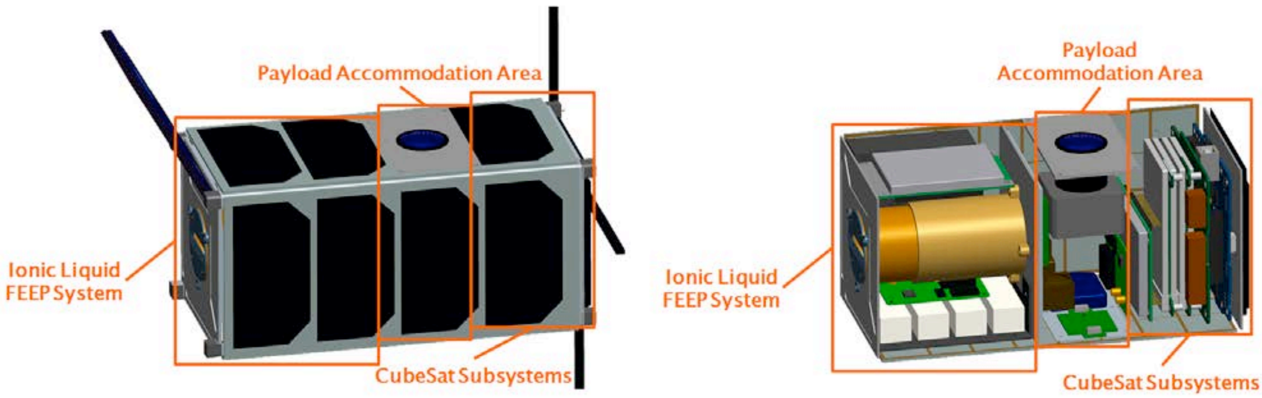

Figure 5: External (left) and internal (right) view of a 2U Cubesat equipped with the IL–FEEP system. The propulsion bay, the Cubesat subsystem bay and the payload sections are highlighted.

The maximum power consumption for the thruster subsystem is estimated to be around 4 W; 1.5 W are reserved for the Cubesat bus (telecommunication, control, main computer and so on) and 2 W for the payload. The total power consumption is of the order of 7.5 W, while assuming a power generation system including 6 GaAs triple–junction solar panels, body mounted on the Cubesat surfaces and on the thruster lid, with an efficiency of 16.9%. The total incoming power turns out to be between 4.3 and 6 W (see Sec. 4.2 for a more refined power budget).

The IL–FEEP subsystem is conceived as composed of the thruster unit, control unit, LV–HV converter and associated electronics and switches. The maximum dimensions are 90x100x100 mm (~0.9 U). A standard COTS LV–HV converter is considered, namely a proportional transformer EMCO F101 series [22] able to reach up to 10 kV in output with an input between 0 and 15 V. The power processed by such a device is up to 10 W with a total efficiency >70%.

The mass budget for the 2U Cubesat equipped with the IL–FEEP system is shown in Figure 6.

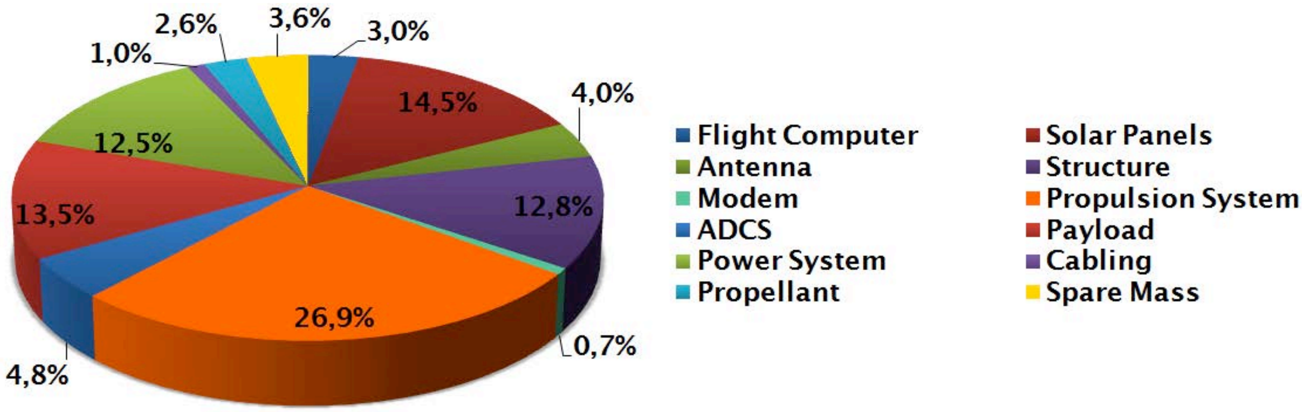

Figure 6: Mass budget of a 2U Cubesat equipped with the IL–FEEP system.

In the mass budget and Cubesat configuration proposed, 350 g and 2 W are allocated for the payload. These figures are sufficient to include: an optical camera (GomSpace NanoCam C1U, 3MP 10 bit CMOS Sensor [23]), a GPS receiver (SSBV GPS Receiver [24]), a Sun sensor (AllSpace DSS–01, 1 deg accuracy [25]) and additional temperature, pressure and acceleration sensors.

4.2 Mission scenario

The IL–FEEP system can provide up to 1 km/s of ΔV on a 2U Cubesat and 0.5 km/s on a 3U Cubesat. Starting from a 300 km circular LEO, the IL–FEEP thruster can increase the altitude of a 2–3 kg Cubesat by 900–2000 km. The inclination can be modified up to 4.6 deg and the considered propellant mass can maintain 300 km altitude for 1.5–10 years, depending on the solar activity and spacecraft mass, see Figure 7.

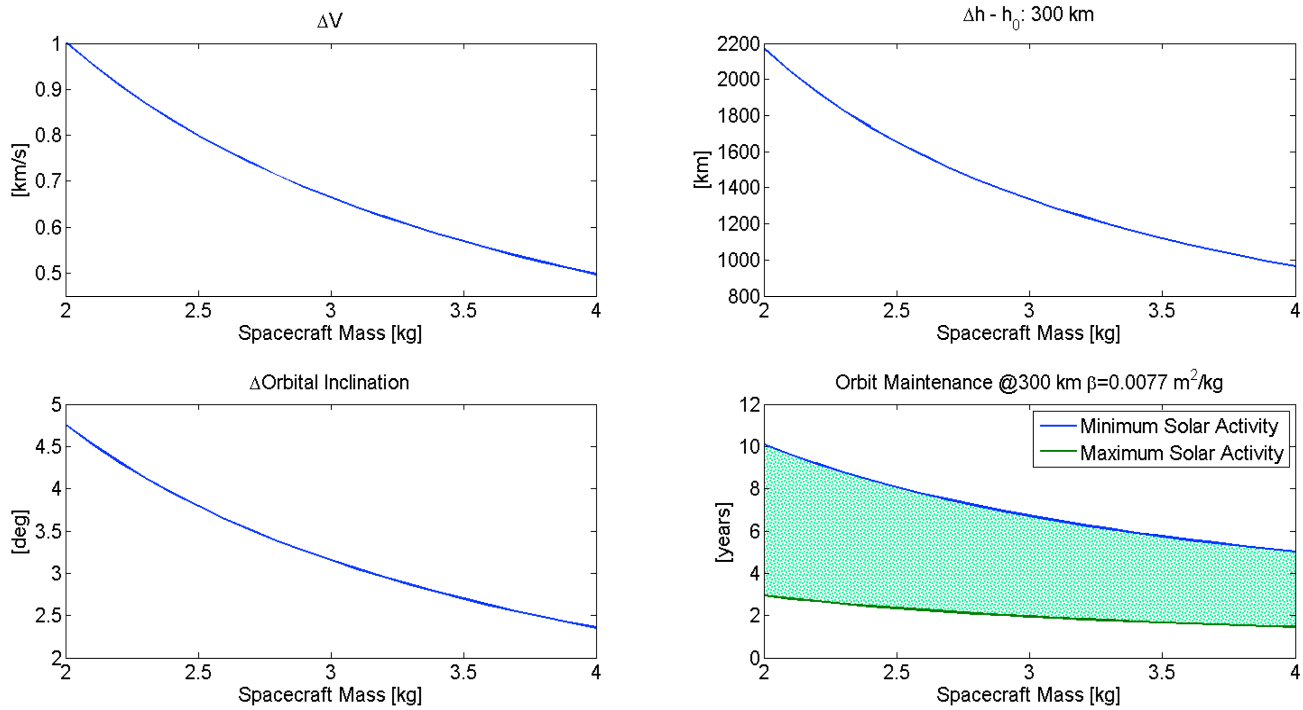

Figure 7: Left to right, top to bottom: the total velocity increment (ΔV), the altitude change, the inclination change and the orbit lifetime for a nanospacecraft (mass from 2 to 4 kg) equipped with the IL–FEEP thruster.

The total mission ΔV can be estimated by means of the Tsiolkovsky rocket equations. The achievable semi-major axis and inclination change were assessed via the Edelbaum analytic approximation [26]. The orbital lifetime increase, instead, was computed by comparing the total impulse exerted by the atmospheric drag at 300 km and the one delivered by the thruster. In this way, the yearly ΔV required for the orbit maintenance of a Cubesat can be obtained.

About 8 g of propellant (ΔV= 90 m/s) are sufficient to maintain the 2U Cubesat in a 200 m box at 300 km for 6 months by considering a static medium density atmospheric model. This corresponds to about 3.8 years of expected lifetime with 4 thruster operations per day, 1 hour each.

Power generated by the solar panels during the orbit changes as a consequence of the instantaneous Cubesat attitude and the relative Sun position. Therefore, the power available to the thruster, payload and other subsystems also changes periodically. Numerical simulations accounting also for the major orbital perturbations (thruster acceleration, Earth oblateness, third body, atmospheric drag, solar radiation pressure) were carried out by means of the Alta's orbital propagator SATSLab [27].

The nat power flux, i.e. the difference between power generated and required, ranges from -7.5 to 2.5 W. In particular, this latter situation with 2.5 W of power flux occurs when the thruster is not firing and the power generated is only used to operate the payload and the spacecraft bus (1.5 W). In the worst case, the 7.5 W power flux peak is caused by the thruster, the payload and the other subsystems operating at the same time while the spacecraft is in eclipse and no power is produced by the solar arrays. Nevertheless, the numerical propagations of the Cubesat trajectory and of the system power status show that, considering a 20 Wh battery (ClydeSpace CS-iBAT2-20, 230 g mass) [28], the thruster can be operated also during the eclipse phases. Figure 8 summarizes the power flux (left) and the battery Depth Of Discharge (DOD) for one day of propagation.

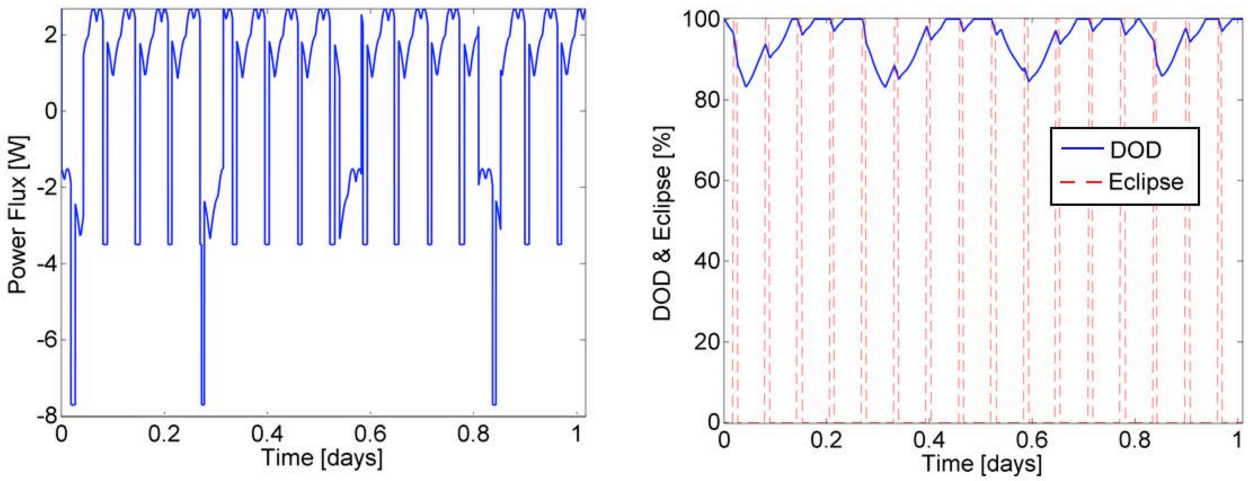

Figure 8: Power flux (left) and battery DOD (right) for one day of propagation of a 300 km orbit maintenance scenario.

As a comparison for the other scenarios sketched in Sec. 4, about 7 months are required to perform the orbit transfer from the initial Sun–synchronous orbit (a=6678 km, i=96.68 deg) to the target one (a=7378 km, i=99.48 deg). The orbit transfer manoeuvre requires about 60 g of propellant.

Once that target SSO Orbit has been achieved, 7 g of propellant are still available for possible orbit phasing or correction manoeuvres. Groundtrack repositioning manoeuvres can be performed by increasing orbital altitude of few tens km in few days. Considering a 1000 km SSO orbit, the groundtrack at medium latitude can be shifted by about 100 km in 2 days with less than 1 g of propellant.

5 CONCLUSIONS

The Ionic Liquid FEEP is a compact and versatile thruster enabling a wide range of orbital transfers and repositioning capabilities. The lightweight thruster under development may represent a very attractive option to increase small platform capabilities by means of autonomous orbital manoeuvres. The experimental test undergoing at Alta's premises have demonstrated the IL-FEEP performance and the TOF measurements confirm the pure ionic working regime and the very high specific impulse.

The thruster configuration developed for Cubesat applications allows for a total impulse of 2000 Ns corresponding to a total ΔV of 1 km/s for a 2U configuration and 0.5 km/s for a 3U. Almost 1U of volume and 0.55 U of mass has to be allocated for the IL–FEEP subsystem. It represents the main power load of the platform, while the impact on the other subsystems is marginal. Significant orbital manoeuvres (e.g. Δh and ΔSSO) and responsive groundtrack changing manoeuvres (50 km/day @ 1000 km) can be realized by an IL–FEEP equipped 2U Cubesat.

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