Propulsion for Nanosatellites

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Abstract: Nanosatellites have evolved over the last years to powerful instruments not only for educational purposes but also for test beds for new technology developments. However, the mission range of such small satellites is limited since autonomous propulsion and active attitude control was until now an unachievable goal due to their stringent mass, volume and power limitations. The present paper summarizes the theoretical and experimental efforts to develop a miniaturized Pulsed Plasma Thruster (µPPT) and an advanced Field Emission Electric Propulsion (FEEP) for attitude and orbit control.

The Pulsed Plasma Thruster (PPT) recommends itself for such an application due to its structural simplicity and low power requirements. However, albeit several efforts worldwide were ongoing to develop such a system they generally lack required miniaturization and lifetime. The present paper presents a system consisting of four miniaturized PPTs, installed on a single PCB. The system has a total power requirement of < 2 W, and weights roughly 250 g. The lifetime of an individual µPPT has been experimentally verified to be larger than 600,000 discharge cycles. The system of four µPPTs can deliver an impulse bit of 7µNs on average and can deliver a total Δv of roughly 11 m/s to a CubeSat.

The Field Emission Electric Propulsion system which is under development for satellites with a minimum size of a double CubeSat provides a much larger performance envelope. The power demand of the FEEP system is roughly 4 W and has a thrust range between 0.03 – 0.1 mN. A thruster module was developed and tested. The volume of the system, including the PPU is roughly 700 cm³ and its shape is such that is fits within a standardized CubeSat. The present paper summarizes the development of this system and its experimental investigation.

Nomenclature

I_{bit} = impulse bit
I = discharge current
L' = gradient of inductance
t = time

I. Introduction

Over the duration of the last decade, nanosatellites, also called CubeSats, have evolved from something hardly noticed by the space community to a powerful movement within the community. CubeSat are cubic satellites with a side length of 10 cm and a maximum weight of 1.33 kg[1]. The possibility to attach up to three single CubeSats to a single satellite exists. In general such nanosatellites are launched piggypacked on launchers delivering larger

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spacecraft in orbit. A launch of a single CubeSat comes with a prize tag of roughly 80,000\$[2]. Depending on the mission goals, the hardware costs can be as low as 50,000\$. The low costs and the potentially short development times compared with larger systems makes a nanosatellite very attractive as an educational tool and as a test bed for new technologies. In addition, the CubeSats fit very well in the general trend of miniaturization with is ongoing in particular in the aerospace field. While in the beginning only academia and small research groups got involved in this topic, over time industry such as The Aerospace Cooperation and Boeing[3] and national space agencies such as NASA[4] and ESA realized the benefits and got involved. A number between 3-17 CubeSats (single, double, triple) have been launched each year in the period between 2003 and 2010[5,6] but can this number is expected to rise. A good example of this expectation is the planned project QB50 which is in its magnitude surely the largest CubeSat effort up to date[7]. QB50 will launch 50 double CubeSats such that the individual CubeSats are separated by a few hundred kilometers only. This project has the scientific objective to study in situ the temporal and spatial variations of a number of key constituents and parameters in the lower thermosphere (90-320 km) with a network of 50 double CubeSats, separated by a few hundred kilometers and carrying identical sensors.

Over the years the mission complexity has significantly increased. While the first CubeSats can be considered as "Sputnik" type missions, present missions have real scientific purpose and very often space agencies are involved. Attitude control is achieved with magnetic coils or miniaturized momentum wheels. The performance of those systems is not always to the satisfaction of the CubeSat developer. Furthermore, due to the piggyback nature of the launch, the satellites are released uncontrolled. The final orbit has to be accepted and cannot be changed. Occasionally, the release of the CubeSats from the transportation box in orbit results in a severe rotation of the satellite which sometimes cannot be compensated anymore by the attitude control system. Potentially even more significant is the fact that, depending on the orbit, CubeSats can be in orbit for 25 years. This and the fact that more and more CubeSats are scheduled for launch causes some concern in the space community with regard to space debris. Although a full de-orbiting might be out of scope for presently suggested CubeSat propulsion systems, a partial de-orbiting, leading to a reduction of the in-orbit time, would already be significant.

In summary, propulsion means for CubeSats are urgently needed. However, up to this date only one satellite, the triple CubeSat CanX-2 by the University of Toronto, Canada has been flown successfully with a cold gas propulsion system (SF_6)[8]. Several other efforts (e.g. ION by University of Illionis, USA) were either destroyed during launch or had other problems rendering them unable to show functionality.

II.   Propulsion system review

As mentioned above, Fotec was commissioned in 2006 by the Austrian Space Agency to establish an assessment of the possibility to equip nanosatellites with a propulsion system. The review included 16 types of propulsion systems including chemical as well as electric propulsion systems. Assessment criteria included performance, TRL level, system mass and volume, potential for miniaturization, power requirements, and system complexity to name only some of the most prominent ones.

To summarize the outcome of the review:
- Chemical propulsion systems (mono-, bipropellant, hybrid) were rejected rather early in the study. Major reasons from a technical point of view included the complexity of even a monopropulsion system, the relative low specific impulse, and vthe mass associated with such a system (piping, valves, tank, etc.). Evenly important for the rejection were restrictions based on the CubeSat specification[1]. Those specifications state that no pressure vessels on-board a CubeSat with a pressure exceeding 1.2 bar are allowed and the total stored chemical energy shall not exceed 100 W-hours. Although, to a certain limit, the limits of those specifications might be up to negotiation with the launch provider, a chemical propulsion system in order to be effective might need to bend the existing specification more than any launch provider is ready to risk considering that his main responsibility is the safe delivery of the main payload. Beside of some cold gas systems, the authors are not aware of any systems launched or ready for launch with a chemical propulsion system on-board.

- Cold gas systems have also been rejected for the use on CubeSats mainly due to the low specific impulse, high volume/mass, and the afore mentioned CubeSat specifications. No single or double CubeSat with a gold gas systems has been launched to this date to the best knowledge of the authors. It has to be mentioned

however, that a triple CubeSat (CanX-2) of the University of Toronto, Canada using SF_6 as propellant and the Surrey satellite SNAP-1 with a wet mass of 6.5 kg and using butane as propellant have been launched in the past.

- From the family of electric propulsion system the review included arcjet, hall thruster, gridded ion thruster, field emission electric propulsion (FEEP), pulsed plasma thruster (PPT), vacuum arc thruster (VAT) microdischarge thruster, colloid thruster, magnetoplasmadynamic thruster (MPD) and several other derivatives of the above mentioned systems. It was concluded that in terms of their low power requirement, potential for miniaturization, simplicity and last but not least performance PPTs and VATs are presently the most promising choice.

Facing the choice between PPTs and VATs it was felt that PPTs due to their extensive space history and research experience are the most promising choice.

It has to be noted that in the meantime (post 2006) new propulsion system developments have surfaced some of them featured in the publications of Sanders[9] and Mueller[10]. One of the potentially suitable systems surfacing in the last years is a derivative of the needle FEEP system developed by Fotec. New manufacturing methods have allowed increasing the thrust of the FEEPs from originally ~10μN up to several mN without increasing the volume or mass of the FEEP propulsion system. This and commercially available miniaturized high voltage components have suddenly pushed FEEPs in the focus of potentially suitable candidates as a propulsion system for CubeSats.

In the following, some background of FEEPs and PPTs as well as the development, design, and tests of those systems for use on CubeSats is provided. Furthermore, their potential integration into CubeSats is described.

III. Pulsed Plasma Thruster Development

A: Background:

The Pulsed Plasma Thruster (PPT) was the first electric propulsion device ever flown in space on the Soviet Zond-2 mission. Ever since then, PPTs have been employed in various space missions, with utilization ranging from orbit insertion and drag compensation (TIP/NOVA with a total of 28 thruster) to east-west station keeping (LES-6)[11] and active attitude control (SMS and LES-8/9[12])[13]. Typical discharge energy levels of flight PPTs range from a few joules (LES-6, SMS) to one hundred joules (EO-1, MightySat-II)[14].

The mechanical simplicity not least due to the use of solid propellant, the relative high specific impulse, low-power requirements and the success of previous flight tests[15], make PPTs an ideal candidate for CubeSats. Moreover, due to the quasi neutrality of the expelled plasma, the PPT does not require, as opposed to many other electric propulsion systems, any charge neutralizing devices.

Due to the above and as a result of the outcome of the assessment, the Fotec team (until Dec. 2010 operating within the Austrian Institute of Technology – AIT) initiated in 2006/2007 the development of miniaturized Pulsed Plasma Thrusters (μPPT). Initially the efforts focused on the establishment of an analytical/numerical tool aimed to support the understanding of the dependency of Micro Pulsed Plasma Thruster performance on electrode geometry parameters and on the properties of the circuit elements. This was in particular important as it was felt that many scaling laws which are commonly used in standard PPT designs might not be applicable for a miniaturized PPT geometry. A one dimensional electromechanical model, based on accurate inductance calculation and a detailed description of the inhomogeneous magnetic field distribution accelerating the plasma was established and its suitability to guide the design of a μPPT was experimentally verified[16].

Using the above mentioned model, several μPPT designs were theoretically investigated and subsequently experimentally assessed. The investigated design variations included standard parallel plate PPT (see left side of Figure 1) with varying discharge channel geometries (aspect ratios as well as electrode length) and electrode designs. The latter includes tongued and flared electrodes as well as variations of electrode materials and electrode thickness[17,18].

Although the above mentioned variations of the parallel plate PPT were operating well and useful to verify the analytic model predictions, persistent lifetime issues mainly linked to the ignition system terminated this line of

investigation. No commercial ignition system sufficiently miniaturized and with suitable performance and lifetime was available. An in-house developed spark-plug type ignition system improved the lifetime but not to an extent making it useful for an extended operation (500,000 – 1,000,000 discharges).

Consequently, the development focus shifted to a coaxial µPPT (see right side of Figure 1) similar to the one developed initially by the AFRL and commercialized by Busek for the Falcon Satellite (launch in 2007). It is important to note that in the present system the ablation face of the Teflon is receding during the operation and not as in a classical breech-fed system kept at the same location (spring fed). The advantage of a rotational symmetric system is the relative ease to implement a third electrode type ignition system (not shown in Figure 1).
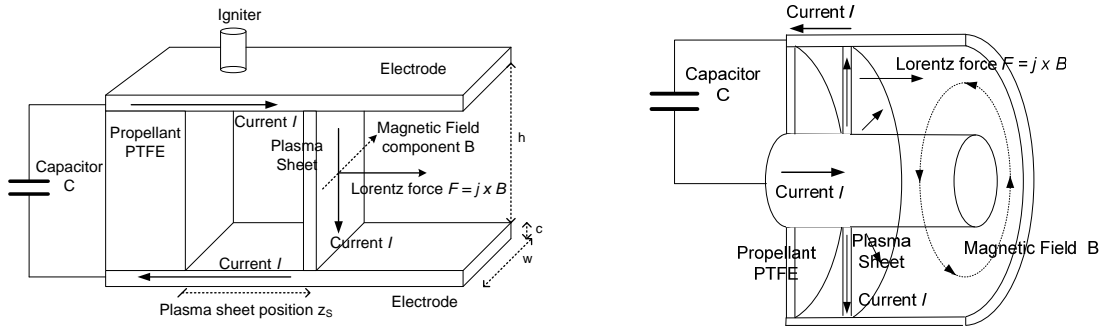

Figure 1: Simplified schematic of a rectangular PPT (left side) and a rotational symmetric PPT (right side)

The subsequent development efforts focused on the design of the main discharge electrodes as well as on the nature of the ignition electrode within the discharge channel, i.e. its geometry, location etc. Already in a very early phase it became clear that this design is by far superior in comparison to the parallel plate design in terms of reliability and in particular in terms of its lifetime. Presently the achieved lifetimes exceed 600,000 discharges (compared with <70,000 for the parallel plate system).

B. Test and Performance evaluation

All developed design variations were extensively tested with regard to their performance, ablation behavior, electrode erosion and in particular their lifetime. In the following only the results of the thruster head which is presently foreseen to be integrated into a CubSat will be presented.
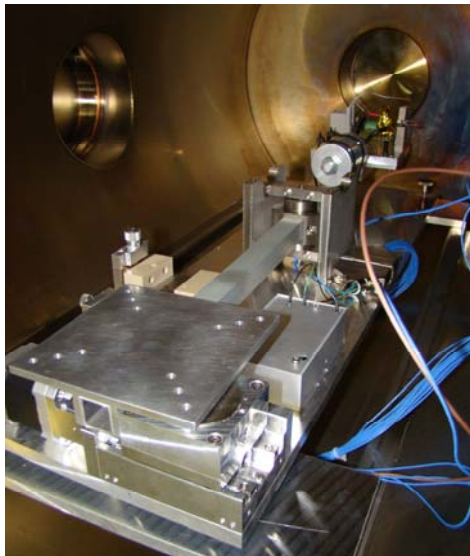

Figure 2: Torsion balance shown in the µPPT vacuum test chamber

In general, the impulse bit of a PPT can be calculated by the following expression:

$$I_{Bit} = \frac{1}{2} L' \int I^2(t)\, dt$$

In order to measure the discharge current I a Rogowski coil has been implemented into the circuitry. The gradient of the inductance L' is in general not known but can be estimated by analytical expression. Several expression for the inductance have been suggested in the past by various authors[19,20,21,22,23]. In particular for miniaturized PPTs (fringe effects have larger influence) care has to been taken when choosing the appropriate expression. Using equation 1 to calculate the impulse bit is very convenient for identifying relative performance changes (e.g. for two (slightly) different PPT designs) but one should not rely on it for absolute values. For this reason, Fotec has manufactured a thrust balance. Details about the µN thrust balance, balance calibration, and verification can be found in reference 24. An example of a typical thrust measurement sequence, consisting of thrust-on and thrust-off periods is shown in Figure 3.
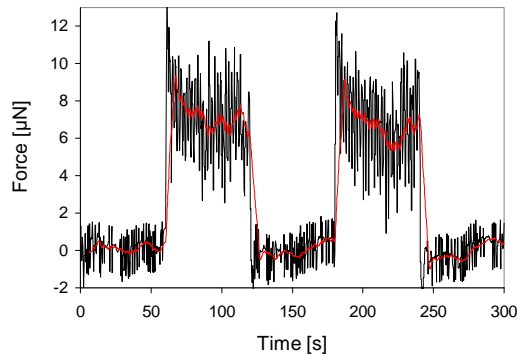
Figure 3: Thrust measurement sequence

The coaxial design which was used to obtain the following results has a total propellant area of 13.6mm². The circuitry used for those tests feature a 2µF capacitor for energy storage which was charged to 1200V resulting in a discharge energy of 1.45 J.

The left side of Figure 4 shows the direct BOL thrust measurements for three different firing frequencies and the mean impulse bit thereof measured as a function of discharge energy. Interestingly, based on the values shown in Figure 4, the ratio of impulse bit to discharge energy is with 10-14 µN-s/J rather similar to a parallel plate PPT when compared with historical values[19] although one could expect a higher value due to the coaxial design. Unfortunately, nothing can be said about the specific impulse of the system. The herefore necessary weight measurements of the Teflon bar following a test is made difficult since it is not possible to remove the Teflon bar without destroying the thruster and part of the Teflon itself. The error in weight measurement is therefore presumably large. Weight measurements of the complete thruster head indicated a mass loss of 0.05 – 0.1 µg/J which is nearly two magnitudes lower than historical values[19] and cannot be trusted for specific impulse calculation at this point in time.
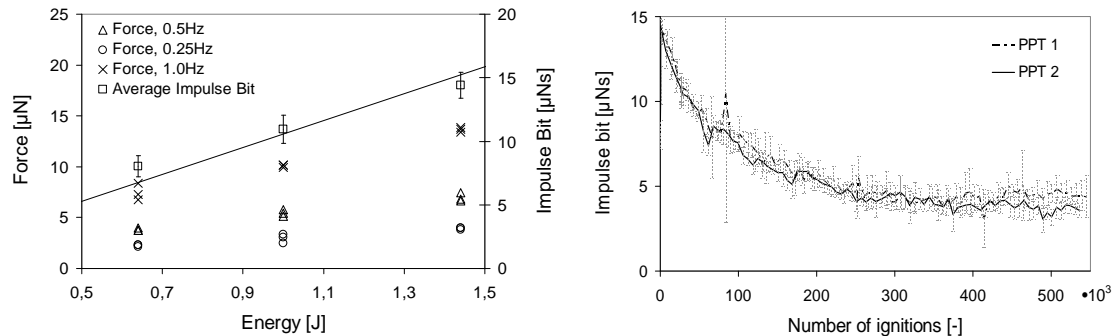

Figure 4: PPT performance evaluation: thrust/impulse bit (left side), impulse bit as a function of operational time (right side)

For the system above also lifetime tests have been performed. For this purpose, the thruster was mounted on the balance and fired at 1Hz continuously to simulate CubeSat operational conditions with power consumption below 2W. Thrust measurements using the test procedure described afore were conducted every 2 hours. The µPPT was removed when reaching a total number of discharges of ~5.5x10^5 shots. This test was repeated twice with identical thrusters. The thrust measurements as function of the number of ignitions are shown on the right side in Figure 4 for the two lifetime tests.

It is interesting to observe that the impulse bit drops from a relative high level (~14 µN-s) down to 4-5 µN-s in a rather short time (200,000 discharges). At this point the performance seems to level off. The reason(s) for this behavior is not yet fully understood. Environmental effects, e.g. initial outgasing of the Teflon was investigated but found not to be the reason for the decrease. Presently it is assumed that the receding ablation face of the Teflon bar might be the a factor although the speed the Teflon face recedes does not correlate with the speed with which the impulse bit decreases.

The total impulse delivered by the two thrusters based on the data from Figure 4 (right side) is equal to 2.94 Ns and 3.17 Ns. Assuming a single CubeSat, a propulsion module consisting of 4 µPPT units (see Figure 5) will provide a minimum Δv change of 11.8 m/s and 12.7 m/s respectively.

C. Integration of µPPTs into CubeSat structure

The above described µPPT thruster is obviously only one part of a complete propulsion system which requires in addition a Power Processing Unit (PPU) consisting of the main energy storage capacitor, the ignition circuit, microcontroller, telemetry (if possible) and last but not least the interfaces to the spacecraft. The present goal is to

assemble all this in a single unit providing space for four thruster heads such that sufficient volume of the spacecraft is left for the payload.

Major difficulty is the identification of suitable capacitors. The electrical characteristics (capacitance, voltage) of the capacitor are derived from the mission and spacecraft requirements. At the same time the capacitor should be lightweight and occupy a small volume. In particular the latter makes it difficult to find appropriate capacitors. Presently, rather large capacitors (40x30x32 mm) are used which increase the thruster unit size. Figure 5 shows an example of a thruster unit foreseen for the use on a CubeSat. It contains all the elements mentioned afore and four µPPTs mounted on a printed circuit board (PCB). The four µPPTs share largely the same circuitry meaning that only one thruster at a time can be fired (controlled by the microcontroller). The side length of the PCB is chosen such that it fits into a CubeSat. The height of the unit is 30 mm which is driven by the height of the capacitors. The length of the µPPTs is simply fitted to the height of the capacitor. The total weight of the present unit (shown in Figure 5) is 294 g and, again, the weight of the capacitors with 130 g is the dominant contributor.

Figure 5 show on the left side how such a unit can be integrated into a CubeSat. Test with the unit shown in Figure 5 indicated occasional failures of the microprocessor. In the present set-up, the location of the microprocessor is rather close (~1mm) to the main discharge lead and therefore exposed to relative large EM noise which is most probably the reason for its failure. However, it is expected that this problem can be solved by e.g. rearranging the elements on the PCB such that the microcontroller is removed from the vicinity of the main sources of the EM noise.
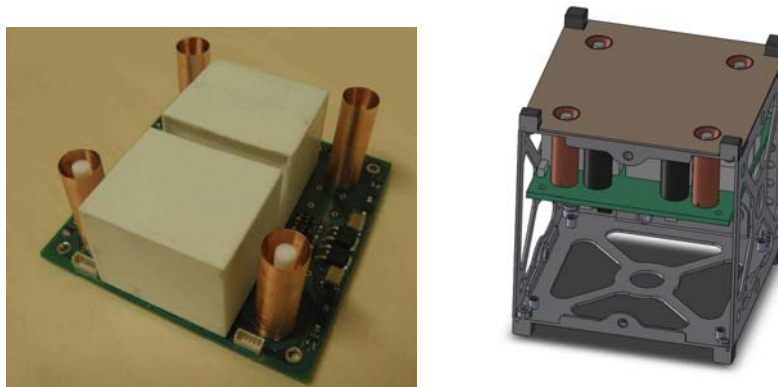

Figure 5: Design Study of a complete thruster unit providing space for 4 µPPTs and its implementation into a CubeSat (left side)

IV.   Field Emission Electric Propulsion Development

D: Background

Field emission electric propulsion (FEEP) uses a highly energetic ion beam to generate thrust in a very efficient and controllable manner. This class of thrusters is therefore very suitable for application where ultra-precision maneuvering is required. The key component consists of a sharp structure, typically a needle, which is coated with a liquid metal. Upon application of several kV to opposing electrodes, ions are field emitted from the liquid metal and accelerated to speeds in the range of 80 km/s. This high level of acceleration of the ions is responsible for the exceptionally high level of specific impulse (Isp). Typically, a FEEP thruster is capable of an Isp from 4000s to 8000s, which is one of the highest values available with current technology[25,26].

In order to extend the thrust range of FEEP emitters from the µN to the mN level, a novel multi-emitter was developed. In this manner, multiple ion beams are emitted from a single thruster, thus distributing the emission current across a larger area. This was made possible through use of an appropriately shaped porous tungsten matrix through which the liquid metal indium can flow internally. Because of its shape, it is termed a 'porous tungsten crown emitter' as displayed in Figure 6. It consists of a single piece of porous tungsten and has 28 needles, each with a tip radius in the range of 1-3 µm for high efficiency.
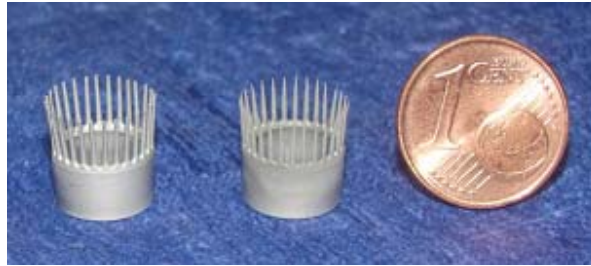
Figure 6: Porous tungsten crown emitter.

A core concept during the development of this thruster was its scalability to generate a wide range of thrusts. While it was designed to operate at a nominal thrust of 0.35 mN, it was shown, that it is capable of producing more than 1 mN for more than 100 hours of operation[27]. Since nanosatellites usually are not capable of producing large amounts of power, a specialized version of this crown emitter was manufactured with only 14 needles on the structure. This procedure increases the distance between two adjacent needles and thus increases the electric field at each remaining needle tip, thus facilitating operation at lower voltages and lower power.

Figure 7 shows typical operating voltages for such a 14-needle porous tungsten crown emitter as function of the emission current. Using a distance of 4 mm between the needles and the oppositely biased extraction electrodes, the crown emitter commences ion emission at a voltage of 4.25 kV and requires around 8 kV for an emission current of 1 mA. The power consumption typically lies in the range of 80 W / mN, but can be adjusted to lower or higher values by changing the potential of the extraction electrode. Lowering its potential to more negative values (i.e. -4 kV) while maintaining a constant emission current will lead to less energetic ions and a correspondingly lower power consumption. In this manner, true variable-Isp maneuvers are possible, such that phases of high-thrust and low specific impulse (apo/perigee) can alternate with low-thrust but high specific impulse (i.e. interplanetary).
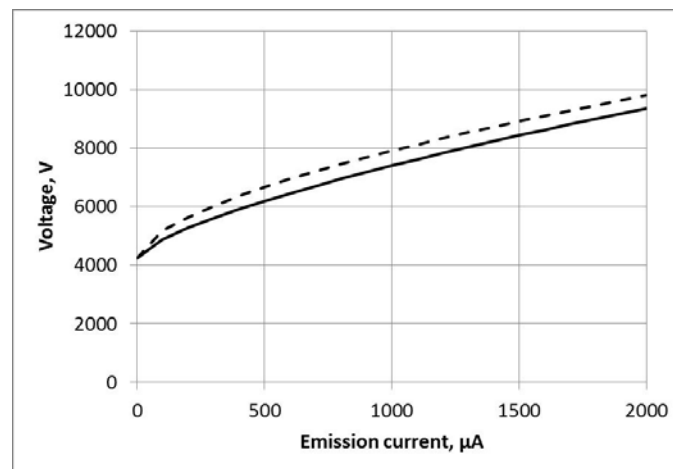

Figure 7: Current voltage characteristic of a porous tungsten crown emitter using indium as propellant (dashed line – before thermal cycle, solid line – after thermal cycle)

A key feature of this new class of FEEP thrusters is its internal flow of liquid metal. This sets it apart from classical approaches using solid needles covered by the liquid metal on their outside. Because the liquid metal can flow through the porous matrix, it is protected from potential contaminations on the outside; an important feature when outgassing of satellites is considered. Furthermore, it has been observed that an internal flow has a beneficial effect on the resilience of the emitter when a thermal cycle is performed. This entails the solidification and re-melting of the liquid metal, which may be initiated to save power during a long period of time in which no thrust is required. Thermal contraction and expansion of the liquid leads to a considerably larger movement of the liquid metal when it is arranged in a thin film on the surface of a solid needle than if mass is distributed across the entire internal volume of the porous needle. In this, the porous needle exhibits the beneficial properties of a capillary emitter, but without sacrificing the high mass efficiency of needle emitters. The two curves in Figure 7 were recorded 40 days apart, and

the emitter was exposed to air during this time. It shows that there is no significant change to the characteristics of the emitter after performing a thermal cycle.

E. Test and Performance evaluation

The FEEP unit for CubeSat was tested for roughly 10 hrs, including 6 thermal cycles. Specific impulse and mass efficiency correlates strongly with the operational conditions, i.e. thrust and extractor voltage. However, on average a mass efficiency of 30-45% and a specific impulse between 3000 s to 6000 s was evaluated.

Figure 8 shows the result of a representative test with the thrust capability and response over time of a 14-needle crown. The maximum thrust during this test exceeded 0.5 mN. The small triangles serve to characterize the current-voltage response, as shown above. The response time of FEEP emitters is measured in µs, so that rapid changes in thrust are possible. In the test program, a wide range of thrust levels was investigated, including low thrust levels of 20 µN, shown from 750 s to 1000 s. From 2000 seconds onward, a constant thrust of 80 µN was commanded, showing the constancy of the performance over time.
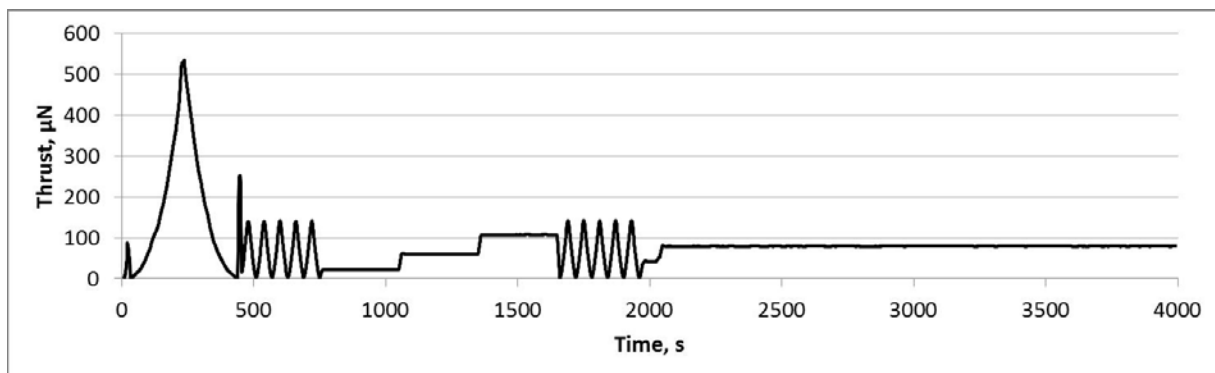

Figure 8: Thrust capability and response to triangle profiles and step profiles. A maximum thrust of 0.5 mN was recorded during this test.

F. Integration of the FEEP system into a CubeSat structure

The module developed for the operation of the FEEP emitter is constituted by three main components: the heating system, the electrode and high voltage system and the main frame (see Figure 9). The heating system has the function to keep the emitter at a temperature higher than the indium melting point, and is composed of an aluminum oxide "shell" which holds the emitter in place, and a nickel-chromium wire heater. A specially designed thermal shielding allows reducing the heating power required to keep the operational temperature (3.5 W are sufficient to keep the temperature of the emitter at 200°C).

The electrode system is comprised of a plate extractor and a central cylindrical extractor, both having the built-in additional capability of shielding the interiors of the module against the droplets emitted by the FEEP emitter. This is a very important feature, as an uncontrolled contamination coming from the indium droplets would compromise the electrical insulation between the emitter (which operates at high voltage) and the parts of the module which are at ground potential, thus limiting the operational life of the module itself.

The high voltage connection system provides a safe interface of the cables coming from the HV power supplies with the electrode system and the emitter. The maximum voltage ratings are +20 kV for the emitter and -10 kV for the extractors, while the nominal ratings are +15 kV for the emitter and -2 kV for the extractors.

The housing of the module provides the interface with the CubeSat frame and offers added thermal shielding. The total weight of the unit (wet mass) shown in Figure 10 is 415 g.

During FEEP operation, because of positive charge leaving the thruster, the satellite ground will be biased negative. This is compensated by the use of a neutralizer, which, emitting electrons, allows to keep the potential of the ground at values relatively close to zero volt. In the case of a double CubeSat the neutralizer is located in the

intersection between the two CubeSat units. Preliminary testing with this kind of neutralizer showed that it is possible to keep the ground potential down to about 250V with a power of 2.5 W, in the case of emitter currents of the order of 1mA.

The module occupies about 2/3 of the length of a single element CubeSat (see right side of Figure 9). The remaining 1/3 can be used to accommodate the DC-DC converters, which transform the low voltage provided by the power control unit to the high voltage required for the thruster operation. Obviously, in order to have a useful mission in terms of payload, the FEEP solution is only possible for at least a double CubeSat sized satellite. Although there is still some margin for reducing the weight and volume of the FEEP unit, the need for a double CubeSat (or larger) is driven more by the power requirements than the volume.

The sum of heater and neutralizer power requirement is already 6W. Depending on the requested thrust level, additional beam power between 5 (0.1 mN) and 8 W (0.35 mN) have to be included. On average a CubeSat can provide 3-5 W of solar power. The sweet spot obviously is a triple CubeSat, but in the case of a double CubeSat, batteries have to be included.
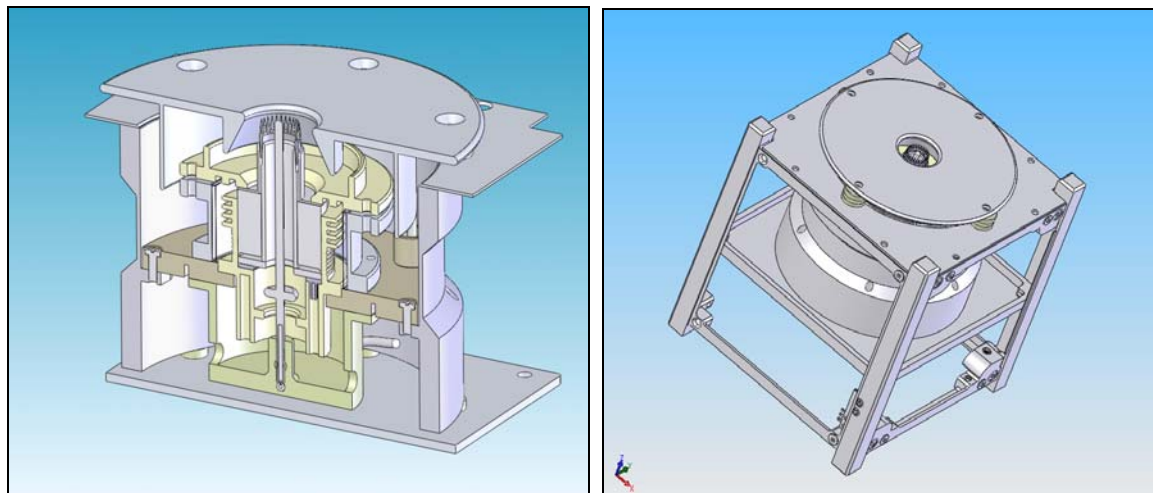

Figure 9: Engineering drawings of the FEEP unit (left) and its integration into a CubeSat structure (PPU and neutralizer not shown)
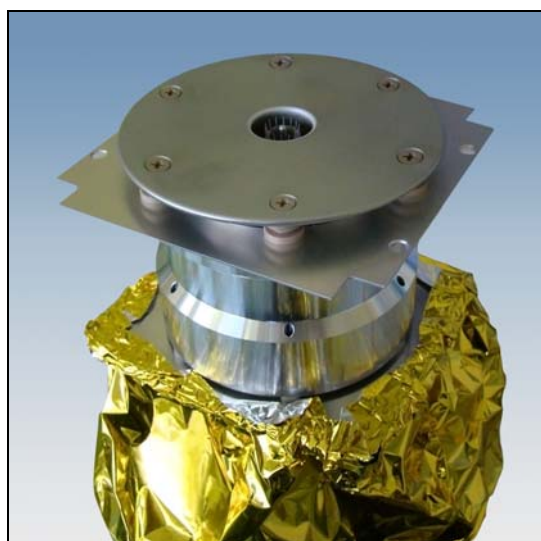

Figure 10: FEEP unit for a CubeSat

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V.   Conclusion

The mission range of CubeSats is presently limited by the fact that no propulsion system exists which can provide attitude control and Δv capability. The development, of a miniaturized PPT and an advanced version of the needle FEEP might close this gap. The developed µPPT propulsion unit, consisting of 4 independent µPPTs can provide a Δv of around 12 m/s to a CubeSat which is sufficient for limited orbit change and/or formation flight task. The system requires only 1.5 W and its total mass is less than 300 g (including all necessary circuity), taking up roughly 1/3 of the available volume in a CubeSat. While the thruster in its present stage can be considered as fully developed issues with the reliability of the circuitry (EM noise sensitivity) are not yet solved.

The FEEP system proposed for use on double CubeSats (or larger) can operate in thrust levels ranging from 50µN up to 1 mN, depending on the availability of power. For a thrust level of 0.1 mN, the required power is 11 W. Under such operational conditions the total impulse capability of the system is 1260Ns.

Due to the high level of integration and high density of propellant, the concept of using a porous tungsten crown emitter as a highly efficient and accurate thruster in a nanosatellite was found to be feasible. Depending on the available power of the satellite, currently achievable thrust levels of up to 1 mN may enable the use of this concept as main thruster, or as ultra-precision attitude control. The scalable power and Isp furthermore permit operation in a wide range of conditions.

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