REFERÊNCIA
GESSINI, Paolo et al. Smallsat high-energy missions using ablative pulsed plasma thrusters. In: IAA LATIN AMERICAN CUBESAT WORKSHOP, 3., 2018, Ubatuba, SP, Brasil.

SMALLSAT HIGH-ENERGY MISSIONS USING ABLATIVE PULSED PLASMA THRUSTERS

Paolo Gessini[(1)], Rodrigo I. Marques[(2)], Pedro L. K. Da Cás[(3)], Cristian Vendittozzi[(4)], Giancarlo Santilli[(5)], Gabriela C. Possa[(6)], José L. Ferreira[(7)], Lui T. C. Habl[(8)], Stephen B. Gabriel[(9)]

[(1)] Universidade de Brasília (UnB), Gama Campus, 72444-240 Brasília, DF, Brazil, +5561982091145, paolo.gessini@aerospace.unb.br

[(2)] Instituto Nacional de Pesquisas Espaciais (INPE), Laboratório de Combustão e Propulsão, 12630-970 Cachoeira Paulista, SP, Brazil, +5561997440477, rodrigo.intini@inpe.br

[(3)] Agência Espacial Brasileira (AEB), Setor Policial, 70610 200 Brasília, DF, Brazil, +5561981300086, pedrokdc@gmail.com

[(4)] Universidade de Brasília (UnB), Gama Campus, 72444-240 Brasília, DF, Brazil, +5561983624277, vendittozzi@aerospace.unb.br

[(5)] Universidade de Brasília (UnB), Gama Campus, 72444-240 Brasília, DF, Brazil, +5561983550174, santilli@aerospace.unb.br

[(6)] Universidade de Brasília (UnB), Gama Campus, 72444-240 Brasília, DF, Brazil, +5561998551505, gabriela.possa@gmail.com

[(7)] Universidade de Brasília (UnB), Darcy Ribeiro Campus, 70919-970 Brasília, DF, Brazil, +5561992964962, leo@fis.unb.br

[(8)] Universidade de Brasília (UnB), Darcy Ribeiro Campus, 70919-970 Brasília, DF, Brazil, +5561981205515, txaicalvoso@gmail.com

[(9)] University of Southampton (UoS), Electronics and Computer Science, SO17 1BJ Southampton, Hants, UK, +447956676298, sbg2@ecs.soton.ac.uk

Keywords*: Electric Propulsion (EP), Ablative Pulsed Plasma Thruster (APPT), Small Satellite (SmallSat), CubeSat, High-Energy Missions*

Ablative Pulsed Plasma Thrusters (APPTs) were the first Electric Propulsion (EP) devices ever to be flown onboard an actual spacecraft, and continue to be used today in missions were simplicity, robustness and scalability to different power levels are dominant requirements. Therefore, they find a natural niche of application in SmallSat missions, where mass, volume and onboard power are at a premium, in spite of their low overall efficiency and not fully understood physical operating principles. Works by many authors have reviewed and analyzed APPTs, yielding formulas that can be used for preliminary design purposes. Such relations take the general form of power laws with coefficients depending, mainly, on the thruster configuration and, to some extent, on the range of discharge energy values. Whereas in previous papers new correlations of experimental data were proposed with focus on small, low-power applications, now special attention is given to high-energy devices. These will be useful for the design of high-efficiency APPT propulsion systems, which are going to find wide application in the growing market of micro and nanosatellites for increasingly ambitious missions. In fact, while APPTs have drawn renewed attention from the international space community after a long hiatus, this has been generally limited, until now, to low ΔV, low total impulse missions. In this paper, we investigate the possibility of performing high-energy (a shorthand for high Δv, high total impulse) missions, such as orbit raising or even deep-space missions, for example LEO (Low Earth Orbit)-to-LLO (Low Lunar Orbit) transfer, using APPTs onboard small spacecraft. The design of such missions is far from trivial, as the high specific impulse values that seem desirable to obtain a high payload ratio are generally obtained at the

expense of impulse bit (the impulse produced at each pulse) vs discharge energy. This implies a high number of shots, which could strain the capacitor capabilities, or high values of discharge energy, which would increase capacitor weight and, due to power limitations onboard a small spacecraft, imply low firing frequencies and consequently increased mission times. This type of missions with APPT-propelled small spacecraft have been, until recently, outside of the realm of possibilities, because of the weight of the capacitors that would be needed. With recently developed supercapacitors, currently in the process of space qualification, they now become possible, due to highly improved energy densities. This opens a wealth of applications, including ambitious missions on string budgets, as those generally available in academic institutions and developing countries. This paper presents a preliminary study of a potential use of an APPT as a simple and robust primary propulsion system for such low-budget high-energy missions. Scaling laws based on data available in the literature were employed. As APPTs in the kJ-energy level, as the one being proposed here, were never actually tested, though, a deeper analysis will be necessary before their use can be recommended.

1. Introduction

Pulsed Plasma Thrusters (PPTs), and in particular Ablative Pulsed Plasma Thrusters (APPTs) were the first Electric Propulsion (EP) devices ever to be flown onboard an actual spacecraft [1], and continue to be used today in missions were simplicity, robustness and scalability to different power levels are dominant requirements. Therefore, they find a natural niche of application in small satellite (SmallSat – a satellite with a mass below 500 kg) missions, where mass, volume and onboard power are at a premium, in spite of their low overall efficiency and not fully understood physical operating principles [1].

Whereas APPTs have drawn renewed attention from the international space community after a long hiatus, this has been generally limited to low-energy, a common shorthand for low Δv, low total impulse (I_{tot}) missions. In recent papers, we investigated instead the possibility of performing high-energy (high Δv, high total impulse) missions, such as orbit raising or even deep space missions, using APPTs onboard small spacecraft [2-4]. The design of such missions is far from trivial, as the high specific impulse (I_{sp}) values that seem desirable to obtain a high payload ratio are generally obtained at the expense of the impulse produced at each pulse, or impulse bit (I_{bit}), vs discharge energy (E) ratio (thrust to power ratio). This implies a high number of shots, which could strain the capacitor capabilities, or high values of discharge energy, which would increase capacitor weight and, due to power limitations onboard a small spacecraft, imply low firing frequencies and consequently increased mission times. Historically, low power availability onboard spacecraft has limited the scope and applicability of electric propulsion, and low energy density of capacitors has seriously hindered the use of APPTs on many ambitious missions.

2. The Ablative Pulsed Plasma Thruster

Between the many EP concepts devised so far, the APPT is one of the most simple, reliable and trusted propulsion systems ever made, using a solid polymer as propellant. This is usually Teflon®, commercial name for Polytetrafluoroethylene (PTFE), but other polymers have been used [5-8]. The basic operation idea of the APPT is based on an electronic circuitry that stores energy in a capacitor bank and cyclically discharges it producing pulsed high voltage arcs (some thousands of volts) on the surface of the propellant bar, causing its vaporization, dissociation (known as the ablation process) and ionization. The resulting gas is accelerated partly by the Lorentz force and partly thermally, resulting in the generation of thrust [1]. An APPT schematic and an idealized representation of its discharge configuration, showing the electromagnetic thrust generation mechanism, are shown in Fig. 1 [9].

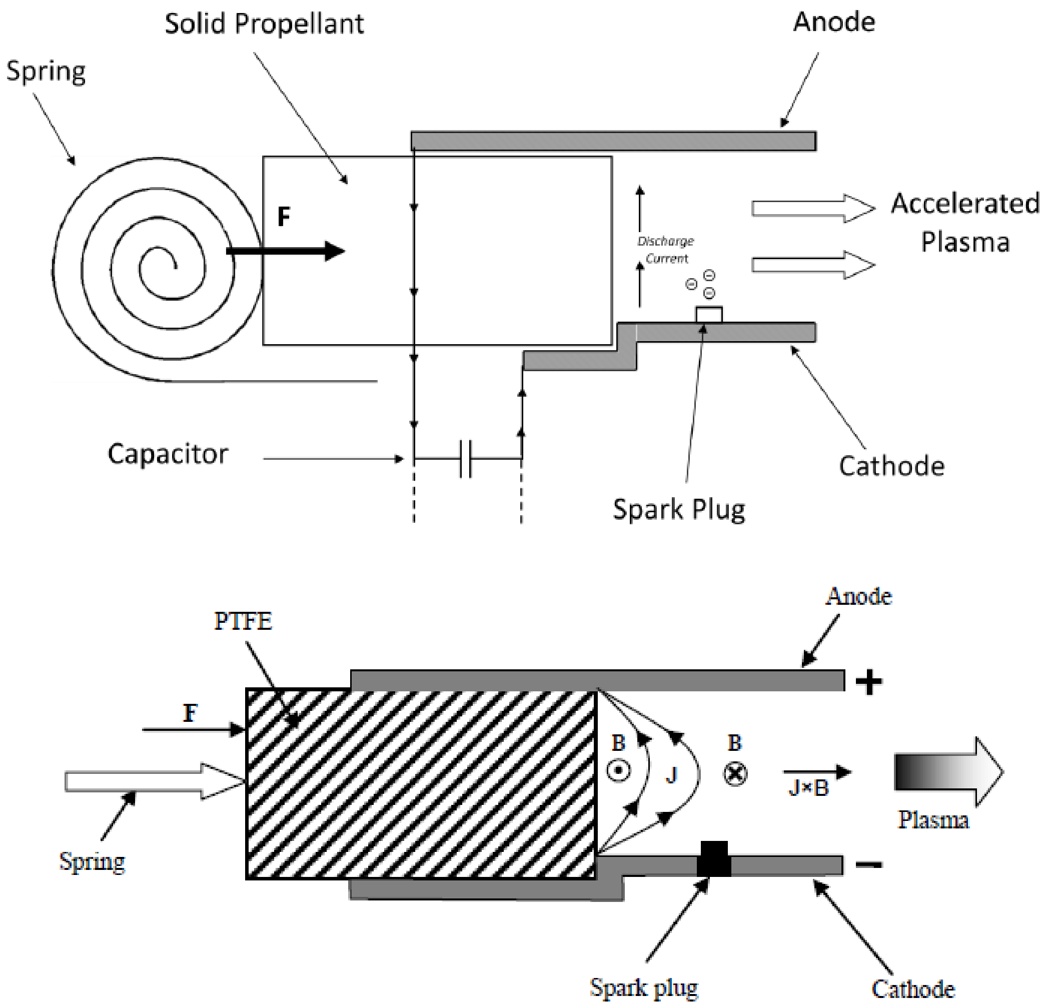

Figure 1: APPT schematic and idealized representation [9].

The electromagnetic thrust is given by the integration of the axial component of the Lorentz force density, which is the vector product of the current density j and the self-induced magnetic field B, according to the formula

$$f_L = \vec{j} \times \vec{B} \tag{1}$$

over the volume of the plasma [1], as shown in Fig. 1 [9], where the current is the actual electron flux, as opposed to the conventional one.

Because of its simplicity, this was among the first types of electric thrusters to be developed and tested, both in the former Soviet Union and, later, in the United States. In 1964 it became also the first EP system to actually be flown on a spacecraft, the Soviet probe Zond 2, on an unsuccessful (for reasons unrelated to the AP-

PTs) mission to Mars, shortly before an ion thruster was sent into a suborbital flight aboard the SERT 1 spacecraft by the United States [1].

Despite their relatively low efficiency, generally below 10% [1], APPTs have been employed because of their outstanding reliability. The absence of tanks, piping and moving parts in general makes them very little prone to malfunctioning and failure, while at the same time easy to scale down to low power levels. This caused a resurgence of interest in APPTs in the 1990s [10, 11] and has made them even more attractive in recent years, as increasingly smaller satellites, down to CubeSat [12] size (10×10×10 cm), have been built and launched.

Several problems, in particular carbonization and late time ablation, with a large fraction of the mass being exhausted at essentially thermal speeds, thus lowering specific impulse and efficiency, remain unresolved, notwithstanding decades of experimental research and numerical/analytical modeling [1].

3. Scaling Laws from Correlations of Experimental Data

Works by many authors have reviewed and analyzed APPTs [1, 10, 11, 13-20], proposing mechanisms of operation and correlations between geometry, operating parameters and performance characteristics, yielding formulas that can be used for preliminary design purposes. The main of the above-mentioned formulas relate I_{bit} with E and I_{sp} with the ratio E/A, discharge energy per unit propellant area (A), intended as the *wetted area*, the area of propellant that is exposed to the discharge. Such relations take the general form of power laws.

$$I_{bit} = \alpha E^{\beta} \tag{2}$$

$$I_{sp} = \gamma \left(\frac{E}{A} \right)^{\delta} \tag{3}$$

The coefficients α, β, γ and δ in Eqs. (2) and (3) depend, mainly, on the thruster configuration and, to some extent, on the range of discharge energy values. Impulse bit, in particular, has been commonly assumed as proportional to discharge energy for a long time. The validity of this assumption has, however, been challenged in more recent papers, especially over wide energy ranges and at the very low end of the discharge energy spectrum, as data reviews were updated [16-18, 20]. Whereas for such low energies a general degradation of performance is observed, with values of the thrust to power ratio sensibly lower than those observed at higher energy levels and a considerable data spread, at high discharge energies the proportionality of I_{bit} to E is well verified, and high values of I_{sp} are observed. This suggests a thrust production mechanism predominantly electromagnetic at high values of E, as expected, with an electrothermal component becoming increasingly important at lower energies. Some experimental investigations have suggested that the data spread could be caused by a strong dependence of performance (impulse bit and specific impulse) on thruster design, and in particular on electrode geometry [17-21].

Different APPT discharge configurations have been adopted in the past decades [1]. One of the most common is the breech-fed configuration, basically consisting of a slab of PTFE propellant sandwiched between two parallel electrodes. It is shown in Fig. 2 [18], together with other configurations, either used or proposed. In the current analysis, we will employ a breech-fed APPT.

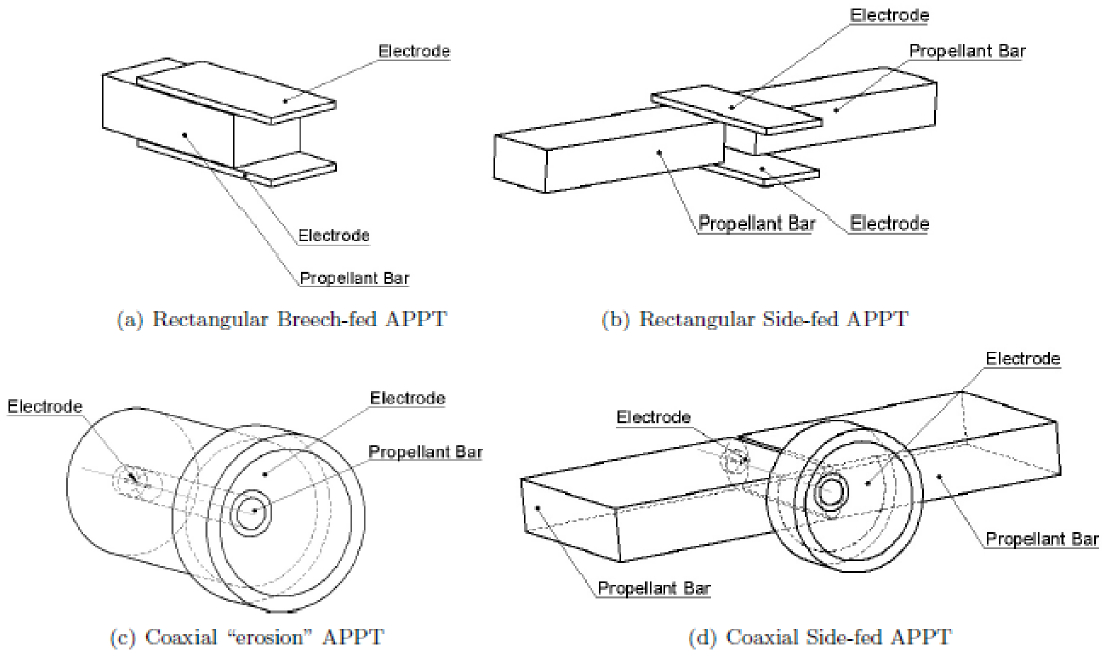

Figure 2: Various APPT discharge configurations [18].

New correlations of experimental data have been proposed in previous papers, in various discharge energy ranges [16-18, 20]. Examples of such correlations are shown in Figs. 3 and 4 for a breech-fed configuration.

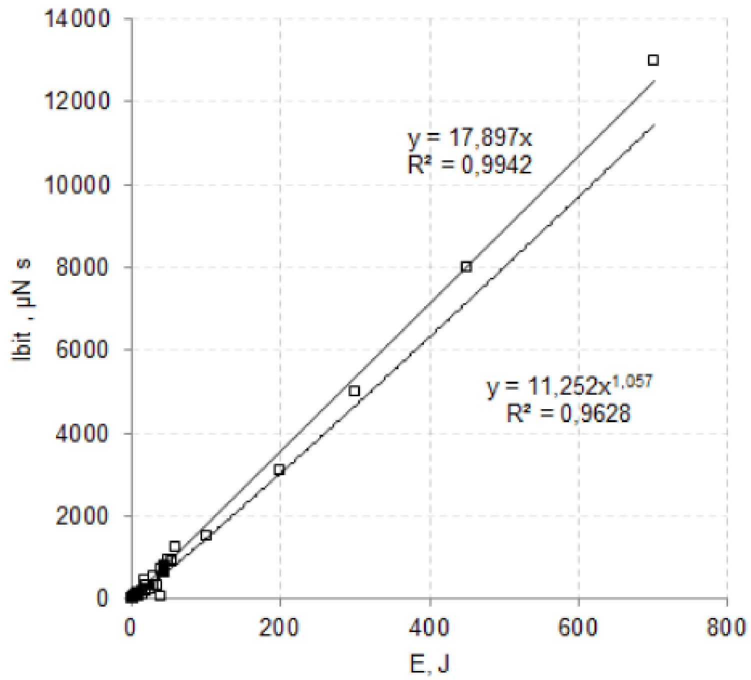

Figure 3: Impulse bit as a function of discharge energy [18].

From an inspection of the plot in Fig. 3 appears clearly that the impulse bit increases with increasing discharge energy and that this relation is approximately linear, especially at high energies. From the plot in Fig. 4, in turn, it can be seen that the specific impulse increases with increasing discharge energy per unit propellant *wetted* area, with a relation that may be roughly approximated by a square-root function.

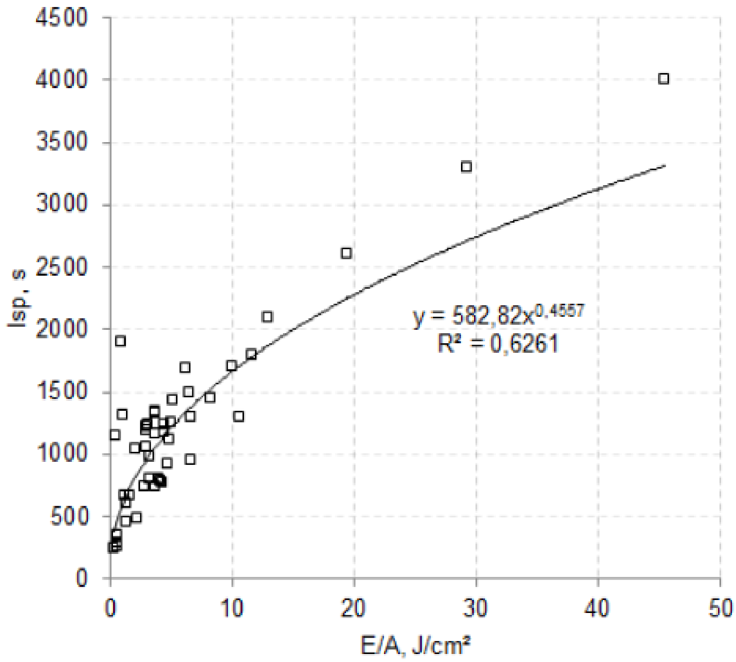

Figure 4: Specific impulse as a function of discharge energy per unit propellant *wetted* area [18].

These formulas will be useful for the design of higher-efficiency APPT propulsion systems, which are going to find increasingly wider application in the growing market of micro, nano and picosatellites [22-26]. In this paper, we pay special attention to high-energy devices. It must be stressed that relations of the types (2) and (3), interpolations of experimental data, are only good guidelines for a preliminary, concept design. The development of actual thrusters will require an extensive experimental campaign. At the beginning, scaled-down models will be used to explore fundamental issues at a lower cost and without requiring large facilities. A high-vacuum chamber (0.75 m diameter, 1.25 m length) has been recently installed in the Advanced Space Propulsion Laboratory (ASPLab) at the University of Brasília (UnB) Gama Campus, Aerospace Engineering [27]. Equipped with a 2000 l/s turbomolecular pump, it is well suited to such experiments.

4. Preliminary APPT Concept Design Procedure for a Sample Mission

The two main parameters used to characterize a given mission are Δv and total impulse. From the definition of specific impulse

$$I_{sp} = \frac{I_{tot}}{g m_p} \qquad (4)$$

and from the rocket equation [28]

$$\frac{m_p}{m_i} = 1 - e^{-\frac{\Delta v}{g I_{sp}}} \qquad (5)$$

it is possible to derive the following relation between these two parameters:

$$I_{tot} = m_i g I_{sp}\left(1 - e^{-\frac{\Delta v}{g I_{sp}}}\right) = m_p g I_{sp} \tag{6}$$

If the specific impulse is sufficiently high, compared to the Δv, such relation can be simplified, using a first-order Taylor series expansion, as:

$$I_{tot} = m_i \Delta v \tag{7}$$

In other words, high values of I_{sp} imply low values of propellant mass, negligible compared to the total spacecraft mass, which can then be assumed constant. Δv and total impulse become, in such cases, proportional to each other. In order for the difference between the value calculated using Eq. (6) and that using Eq. (7) to be negligible, say less than 5%, the value of the ratio $\Delta v/gI_{sp}$ has to be less than 0.1. While such values could be easily achieved for low-energy missions using electric propulsion as in [16], where Eq. (7) is used, this will not generally be the case for medium-to-high-energy missions unless very high I_{sp} thrusters, like Gridded Ion Engines (GIEs) or Field Emission Electric Propulsion (FEEP), are employed. The use of Eq. (6) is, therefore, generally preferable when high-energy missions are being analyzed, even using EP, as in the present case. Here, the feasibility of certain classes of high-energy missions, within the constraints of a small, and especially of a micro (<100 kg mass) spacecraft, is explored in the light of the latest developments in APPT and capacitor technology.

Let us consider, for example, a low-thrust transfer from LEO to LLO, with a Δv of about 8 km/s [2-4]. Such a mission will severely strain the capabilities of a small spacecraft using APPTs. If we adopt conservative values for the various parameters, using state-of-the-art technology, it is easily seen, with an analysis of the type conducted in [16, 18], that we cannot respect all the constraints.

The key, in order to conduct high Δv missions with this kind of spacecraft, is to use extrapolation values to near-term technology, especially as concerns capacitors, both for energy density and maximum number of discharges. This will allow us to limit the capacitor mass to an acceptable fraction of the total mass.

Another very important term, in the total mass budget, is the propellant mass. In order to keep this down, we have to work with high values of specific impulse, higher, in general, than those commonly used in APPTs.

Let us consider a spacecraft with a total initial mass of about 16 kg, as in the case of a 12U CubeSat platform at the upper limit of its allowed mass [12], and examine different preliminary concept design options. It is to be noted that this is actually a realistic scenario, as 12U CubeSat deployers are commercially available [29].

Case 1

With an I_{sp} of 2000 s, we see from Eq. (5) that the propellant mass will be 33% of the initial mass, 5.3 kg. While this value may seem somewhat high, it is still acceptable for an EP technology demonstrator, and much lower than what we would get by using chemical propulsion.

By calculating the total impulse with Eq. (6) and dividing by the total number of pulses, we obtain the impulse bit needed and, from this, the discharge energy, using an empirical relation of the form (2). From the discharge energy, we then calculate the capacitor mass, using appropriate values of energy density. From Eq. (6), we calculate a total impulse of 105 kNs. By assuming our capacitors can withstand one million pulses, we thus have for I_{bit} a value of 105 mNs. Then, by using the value for I_{bit}/E of Fig. 3 (breech-fed configuration), which we can approximate, for high energies, as 18 μNs/J, we calculate a value for E of 5.8 kJ. If we adopt a value of energy density of 10 kJ/kg, we obtain a capacitor mass of just about 0.6 kg. Such a value for energy density is much higher than space-tested technology, but quite conservative if we consider some types of recently developed supercapacitors [30, 31]. These devices are currently undergoing space qualification, and have already been tested for over a million cycles.

For what concerns the design of the APPT itself, from the relation type (3) reported in Fig. 4, we have that a discharge energy of 5.8 kJ and a specific impulse of 2000 s yield a wetted area of about 390 cm^2. Our APPT would then be a big device, for example with 26 cm wide electrodes distanced by 15 cm. With a density of 2.2 g/cm^3 for PTFE, we then obtain for the propellant bar a volume of about 2400 cm^3 and a length of only 6.5 cm, to provide the total impulse needed for the mission with a good safety margin. This may not be a practical design as such dimensions would be quite awkward to fit in a 12U CubeSat platform. In addition to this, a 15-cm gap would probably make discharge breakdown rather difficult to initiate.

Case 2

Let us now try a higher value for I_{sp} in order to improve our system design. With an I_{sp} of 3000 s, from Eq. (5) we calculate a propellant mass fraction of 0.24, that is, just 3.8 kg of PTFE. From Eq. (6), we then calculate a total impulse of 112 kNs. Still assuming a conservative one million-pulse operation, we have for I_{bit} a value of 112 mNs and, from the relation in Fig. 3, a value of 6.2 kJ for E. Now the total capacitor mass is still about 0.6 kg, and from Fig. 4 we can estimate, for an I_{sp} of 3000 s, a wetted area of about 180 cm^2. This would correspond, for example, to 18 cm wide electrodes distanced by 10 cm. The length of our PTFE propellant bar would be just 10 cm, with a safety margin. Such dimensions could be accommodated more easily in our small spacecraft envelope, even if the gap width would still be quite high.

From the examples above it is quite clear that, due to large increases in energy densities, the capacitor mass is not such a driving factor in the overall mass budget any longer, as it still used to be only a couple of decades ago [16]. Also, the values of E that we have been using lie outside of the range reported in Fig. 3. Yet, the good linearity of the relation, especially at high energies, allows us to confidently extrapolate and use a constant value of 18 μNs/J for I_{bit}/E, significant deviation from this value having been noticed, instead, at low energy levels [18]. As for the relation shown in Fig. 4, we notice a high spread of experimental data points at low values of E/A, with the actual data points at high values of E/A significantly higher than the interpolation curve. We can thus make a point for using the actual experimental data in that region, instead of the interpolation, which is weighted down by many data points at low energies per unit area.

Case 3

Let us consider, therefore, the case of an essentially electromagnetic thruster, with a high I_{sp} of 4000 s, and assume that such an I_{sp} would correspond more closely to a value of E/A of 45, from the data points in Fig. 4. In this case, from Eq. (5), we see that the propellant mass is further reduced, to a mere 18% of the initial mass, corresponding to just 2.9 kg of PTFE. Now, from Eq. (6) we calculate a total impulse, for our mission, of 115 kNs. Still assuming a conservative one million-pulse operation, we have for I_{bit} a value of 115 mNs and, from the relation in Fig. 3, a value of 6.4 kJ for E. The total capacitor mass is still just about 0.65 kg, and from Fig. 4 we can estimate, for an I_{sp} of 4000 s, a *wetted* area of about 140 cm^2. Remembering that we have been quite conservative in this case, by just using our data points instead of the interpolating curve, and that our relations are just useful tools for preliminary design, we could try a value of E/A between 50 and 55 and see if we still obtain an I_{sp} close to 4000 s. By reducing the propellant area exposed to the discharge (*wetted*), for the same energy, we should increase electromagnetic acceleration and I_{sp}, arguably. With this type of assumption, we can estimate a *wetted* propellant area of just over 120 cm^2. We now have a flat slab of PTFE, with 19-cm wide electrodes, separated by a 6.5-cm gap, as a propellant bar. With a PTFE density of 2.2 g/cm^3, our slab will have to be 11 cm long, with a mission safety margin. The smaller gap would facilitate breakdown, and the APPT would easily fit in our platform envelope, on the smaller (20×20 cm) side.

As for the total mission time, if we assume an onboard power available for propulsion of 120 W, unusual in very small spacecraft, but achievable with deployable solar panels, by operating at 6-kJ discharge energies we are limited to pulse frequencies of about 0.02 Hz. With one million pulses, this yields a minimum total mission time of about 580 days, a little more than 19 months, assuming continuous, uninterrupted thrusting. This may seem long, but is quite acceptable for a technology demonstrator with a very low cost, as in this case.

The mass of the power conditioner can be assumed, as a first approximation, proportional to the power. Values of 20 years ago for this proportionality coefficient, of about 0.01 kg/W [16], are nowadays very conservative, so we can safely assume that this subsystem will have a mass of no more than 1 kg. In order to assess the total mass of our propulsion system, we have to add the mass of the electrodes, of the discharge initiating circuit and of various structural and packaging components. These can also be estimated, as a first approximation, using proportionality coefficients. Again, progress has been made in the last two decades, but we will be conservative, by using the values reported in [16], which ought to give us significant safety margins, thus estimating our total system mass at about 7 kg. This would leave approximately 9 kg for the other subsystems, like the ADCS, and the payload. The whole propulsion subsystem could therefore nearly fit in a 4U CubeSat envelope, occupying just over a third of our 12U CubeSat. An artist rendition of the spacecraft, with the solar panels fully unfolded, is shown in Fig. 5, with deployable antennas included for completeness.

The deployable array configuration shown in Fig. 5 has a total area just below one m^2. Even just using commercial, off-the-shelf components, it could easily provide an average power above 120 W [32]. It is important to note that, in case such a configuration proved difficult to fit in our concept design, reducing the array total area and hence the available power would only entail decreasing the pulse frequency, correspondingly increasing the total mission time.

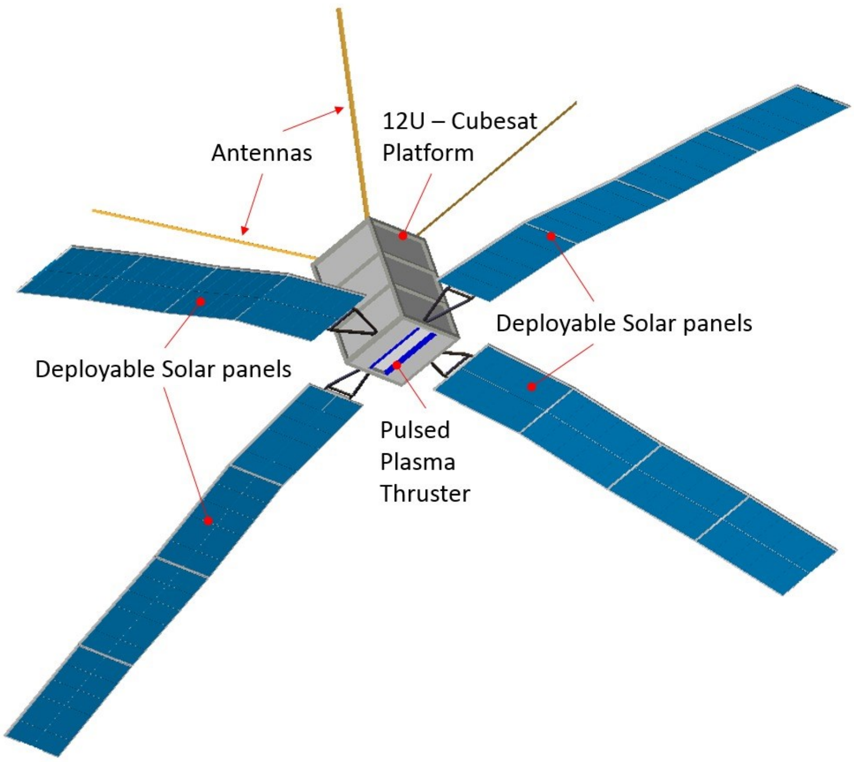

Figure 5: Artist rendition of the spacecraft.

5. Discussion

As already said, relations of the types (2) and (3) are only good guidelines for preliminary design, and the development of actual thrusters will require experimental testing. At the beginning, scaled-down models will be used to explore fundamental issues at a lower cost and without requiring large facilities. A device operating at energies around 100 J is expected to produce an I_{bit} of about 1.8 mNs. If the wetted area is reduced in proportion, to about 2 cm^2, such a device ought to operate at values of I_{sp} close to 4000 s, similar to those of the large devices we have been proposing in this paper, and could be easily tested inside the high-vacuum facility installed in the Advanced Space Propulsion Laboratory [27].

Different electrode gap widths will be explored, in order to study the discharge breakdown phenomenon. A spark plug mechanism has been routinely employed to initiate APPT discharges, as generally voltages much higher than those used to charge the capacitors are required for breakdown. If such a mechanism could be eliminated, in favor of simpler discharge initiation methods, APPTs would become even more attractive, from a system-engineering point of view. A major issue with the use of supercapacitors with APPTs is that, while their high energy density makes then attractive, enabling mission that would be impossible with more traditional capacitors, their low voltage rating [30, 31] would require complex circuitry to make breakdown possible. Connecting many cells in a series to get a higher voltage would, of course, put a lower constraint on the capacitor mass, connected with the size of each cell, and limit the concept usability on very small spacecraft. Therefore, much further analysis is required.

Another possible candidate for the concept design of a high-energy, high I_{sp} APPT is the Two-Stage Pulsed Plasma Thruster (TS-PPT), described in [9, 33, 34], as it can impart an arbitrary amount of energy in the second stage to increase the specific impulse. A schematic of this device is shown in Fig. 6 [34].

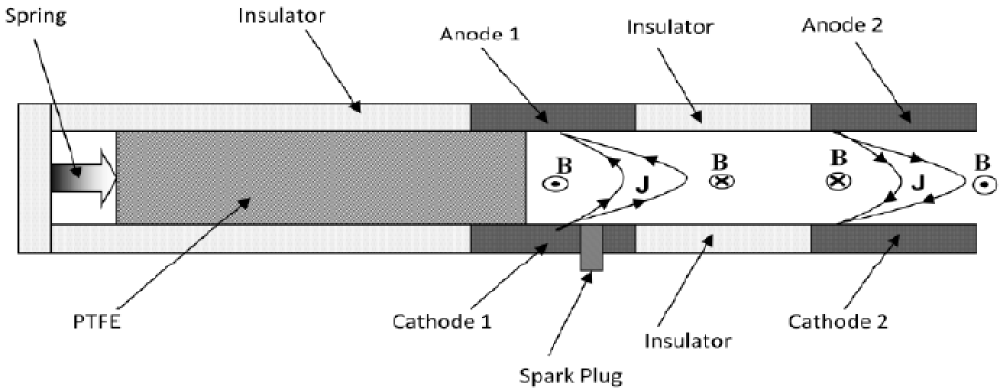

Figure 6: Schematic of the Two-Stage Pulsed Plasma Thruster (TS-PPT) [34].

Preliminary work has been able to achieve values of I_{sp} up to 4000 s with regular capacitors [35]. A photo of a second-generation TS-PPT, testing energy distribution amongst the two stages in vacuum, is shown in Fig. 7 [35].

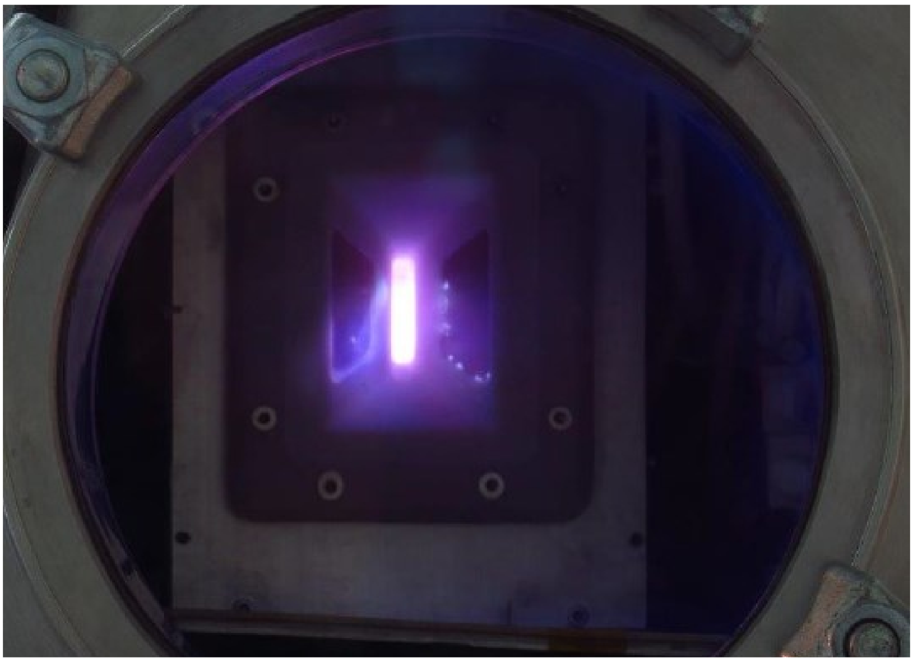

Figure 7: Testing of the Two-Stage Pulsed Plasma Thruster (TS-PPT) [35].

In a possible TS-PPT high-energy configuration, the first stage would act as a trigger to the second stage, where supercapacitors could be discharged. Currently the TS-PPT is scheduled for characterization testing in a larger high-vacuum facility at the Electric Space Propulsion Laboratory, part of the Combustion and Propulsion Laboratory of the National Institute for Space Research (LABCP/INPE) in 2019. The use of supercapacitors with the TS-PPT could greatly simplify the circuitry and possibly make its use viable. More importantly, it could even further increase its I_{sp}. As of now, research and development is underway to reach a candidate configuration for a high-energy APPT. The preliminary work described in this paper represents a seed in that direction.

Anticipating possible difficulties with achieving repeatable ignition and discharge conditions using the proposed electrode gaps, the propulsion system could be modified from a monolithic APPT to a set of independent discharge chambers of a more typical size. We could actually have separated APPT modules, each with its own capacitor bank, only sharing, for maximum integration and system optimization, the same power conditioning system. By inspecting Eqs. (2) and (3) we see that, for example, the same performance could be attained by splitting the discharge energy in half, into two discharge chambers with the same propellant wetted area. The total capacitor mass would be unchanged, as it is proportional to the total energy. This multi-discharge chamber configuration could be advantageous from the development effort point of view, as smaller, more traditional devices would be needed, and from the system point of view, as multiple off-centered APPTs could be used for attitude control and reaction wheel desaturation. In this way, a certain level of redundancy could also be added. An artist rendition of our 12U CubeSat using two APPTs with halved electrode gaps, which could facilitate breakdown, is shown in Fig. 8.

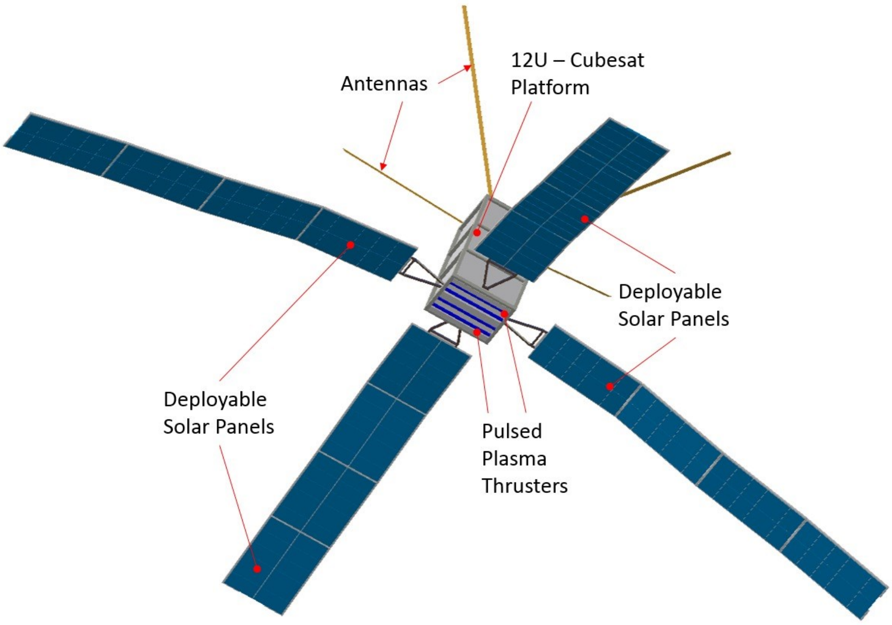

Figure 8: Artist rendition of the spacecraft using two identical APPTs.

As this scenario would increase the complexity of the system, a detailed tradeoff analysis becomes necessary to assess the cost/benefit of this design choice.

It is important to stress again that what is presented here is not actually a preliminary mission design, and that Figs. 5 and 8 are just, strictly speaking, artist renditions of a concept, a merely initial study for the application and possible use of prospective high-energy APPTs employing supercapacitors, assuming that all technical challenges can be overcome.

6. Conclusions

High-energy missions with APPT-propelled small spacecraft have been, until recently, outside of the realm of possibilities, because of the weight of the capacitors that would be needed. With recently developed supercapacitors, currently in the process of space qualification, they now become possible, due to highly improved (orders of magnitude) energy densities. This opens a wealth of applications, like orbit raising and deep-space missions on a string budget, as generally available in academic institutions and developing countries.

Interpolations of experimental data from the literature can be used as effective guidelines for preliminary APPT design, but extensive experimental testing is needed in order to improve efficiency and confirm the validity of scaling these devices up to higher energy levels. Much research and development effort will also be needed to address the issue of discharge initiation, especially with low-voltage supercapacitors, while trying to preserve overall system simplicity and robustness, which are the most attractive characteristics of these devices.

Acknowledgments

The Authors wish to thank the Brazilian Space Agency (AEB) for supporting Electric Propulsion research at the University of Brasília (UnB), through its program Uniespaço, and the Deanery for Research and Innovation of the University of Brasília (DPI/UnB) for funding their participation in the 3rd IAA Latin American CubeSat Workshop (IAA-LACW 2018). The second author, in addition to the above, acknowledges the support of FAPESP – the São Paulo Research Foundation Grant #2016/05927-3, scholarships from CAPES and financial aid from INPE – The Brazilian National Institute for Space Research.

References

[1] R. L. Burton, P. J. Turchi, Pulsed Plasma Thruster, Journal of Propulsion and Power, Vol. 14, No. 5, 1998, pp. 716-735.

[2] P. Gessini, P, L. T. C. Habl, Ablative Pulsed Plasma Thrusters for High Delta-v, High Total Impulse Missions with Small Spacecraft, in: Proceedings of the 2nd IAA Latin American CubeSat Workshop (II LACW), Florianopolis, SC, Brazil, February 28-March 2, 2016. IAA-BR-16-0P-71.

[3] P. Gessini, L. T. C. Habl, G. C. Possa, S. B. Gabriel, Ablative Pulsed Plasma Thrusters for Microspacecraft High Energy Missions, in: Proceedings of the 10th International Conference on Plasma Assisted Technologies (ICPAT-10), Cancun, Mexico, March 7-9, 2016.

[4] P. Gessini, L. T. C. Habl, G. C. Possa, J. L. Ferreira, G. Santilli, C. Vendittozzi, S. B. Gabriel, Ablative Pulsed Plasma Thrusters for SmallSat High-Energy Missions, in: Proceedings of the 1º Congresso Aeroespacial Brasileiro (I CAB), Foz do Iguaçu, PR, Brazil, November 1-3, 2018.

[5] D. J. Palumbo, W. J. Guman, Effects of Propellant and Electrode Geometry on Pulsed Ablative Plasma Thruster Performance, Journal of Spacecraft and Rockets, Vol. 13, No. 3, 1976, pp. 163-167.

[6] E. J. Pencil, H. Kamhawi, L. A. Arrington, W. B. Warren, Evaluation of Alternate Propellants for Pulsed Plasma Thrusters, in: Proceedings of the 27th International Electric Propulsion Conference, Pasadena, CA, October 2001. IEPC 2001-147.

[7] G. Paccani, U. Chiarotti, Behavior of Quasi-steady Ablative MPD Thrusters with Different Propellants, in: Proceedings of the 24th International Electric Propulsion Conference, Moscow, Russia, September 1995. IEPC 1995-118.

[8] G. Paccani, U. Chiarotti, W. D. Deininger, Quasisteady Ablative Magnetoplasmadynamic Thruster Performance with Different Propellants, Journal of Propulsion and Power, Vol. 14, No. 4, 1998, pp. 254-260.

[9] R. I. Marques, A Mechanism to Accelerate the Late Ablation in Pulsed Plasma Thrusters, PhD Thesis, School of Engineering Sciences, University of Southampton, 2009.

[10] E. Y. Choueiri, System Optimization of Ablative Pulsed Plasma Thruster for Stationkeeping, Journal of Spacecraft and Rockets, Vol. 33, No. 1, 1996, pp. 96-100.

[11] J. K. Ziemer, E. A. Cubbin, E. Y. Choueiri, Pulsed Plasma Propulsion for a Small Satellite: Mission COMPASS P^{3}OINT, in: Proceedings of the 32nd AIAA/ASME/SAE/ASEE Joint Propulsion Conference and Exhibit, Orlando, FL, July 1996. AIAA-96-3292.

[12] A. Mehrparvar, CubeSat Design Specification Rev. 13, (February 2014), The CubeSat Program, Cal Poly SLO. California Polytechnic State University, 2014.

[13] R. J. Vondra, K. I. Thomassen, Flight Qualified Pulsed Electric Thruster for Satellite Control, Journal of Spacecraft and Rockets, Vol. 11, No. 9, 1974, pp. 613-617.

[14] W. J. Guman, Designing Solid Propellant Pulsed Plasma Thrusters, in: Proceedings of the AIAA 11th Electric Propulsion Conference, March 19-21, 1975. AIAA-75-410.

[15] W. J. Guman, Solid Propellant Pulsed Plasma Propulsion System Design, Journal of Spacecraft and Rockets, Vol. 13, No. 1, 1976, pp. 51-53.
[16] P. Gessini, G. Paccani, Ablative Pulsed Plasma Thruster System Optimization for Microsatellites, in: Proceedings of the 27th International Electric Propulsion Conference, Pasadena, CA, October 2001. IEPC 2001-182.
[17] P. Gessini, R. I. Marques, S. B. Gabriel, S. M. Veres, Propulsion System Optimisation for Satellite Formation Flying, in: Proceedings of the 4th International Workshop on Satellite Constellations and Formation Flying, INPE, São José dos Campos, Brazil, February 2005.
[18] P. Gessini, et al., Low Power Ablative Pulsed Plasma Thrusters, in: Proceedings of the 33rd International Electric Propulsion Conference, Washington, DC, October 2013. IEPC 2013-344.
[19] P. Molina-Cabrera, et al., Pulsed Plasma Thrusters: a worldwide review and long yearned classification, in: Proceedings of the 32nd International Electric Propulsion Conference, Wiesbaden, Germany, September 2011. IEPC-2011-340.
[20] P. Gessini, et al., Very Low Power Pulsed Plasma Propulsion, in: Proceedings of the 4th IAA International Conference on Space Technologies: Present and Future, Dnepropetrovsk, Ukraine, April 2013.
[21] H. Kamhawi, E. Pencil, T. Haag, High Thrust-to-Power Rectangular Pulsed Plasma Thruster, in: Proceedings of the 38th AIAA/ASME/SAE/ASEE Joint Propulsion Conference and Exhibit, Indianapolis, IN, July 2002. AIAA-2002-3975.
[22] S. J. Pottinger, C. A. Scharlemann, Micro Pulsed Plasma Thruster Development, in: Proceedings of the 30th International Electric Propulsion Conference, Florence, Italy, September 2007. IEPC-2007-125.
[23] M. Coletti, F. Guarducci, S. B. Gabriel, A micro PPT for Cubesat application: Design and preliminary experimental results, Acta Astronautica, Vol. 69, No. 3, 2011, pp. 200-208.
[24] F. Guarducci, M. Coletti, S. B. Gabriel, Design and Testing of a Micro Pulsed Plasma Thruster for Cubesat Application, in: Proceedings of the 32nd International Electric Propulsion Conference, Wiesbaden, Germany, September 2011. IEPC-2011-239.
[25] A. Mingo Perez, M. Coletti, S. B. Gabriel, Development of a Microthruster Module for Nanosatellite Applications, in: Proceedings of the 32nd International Electric Propulsion Conference, Wiesbaden, Germany, September 2011. IEPC-2011-144.
[26] M. Lau, G. Herdrich, S. Fasoulas, H. P. Rser, A Thermal Pulsed Plasma Thruster for Microsatellite Propulsion, in: Proceedings of the 32nd International Electric Propulsion Conference, Wiesbaden, Germany, September 2011. IEPC-2011-140.
[27] P. Gessini, et al., Electric Propulsion Activities in Brazil, in: Proceedings of the 35th International Electric Propulsion Conference, Atlanta, GA, USA, October 2017. IEPC-2017-522.
[28] R. G. Jahn, Physics of Electric Propulsion, 2nd ed., Dover Books on Physics, New York, Chap. 1, 2006.
[29] https://www.isispace.nl/product/quadpack-cubesat-deployer/-(retr. November 2018).
[30] Y. Zhu, et al., Carbon-Based Supercapacitors Produced by Activation of Graphene, Science, Vol. 332, 2011, pp. 1537-1541.
[31] Skeleton Technologies, Miscellaneous Technical Material and Private Communication, 2018.
[32] https://www.isispace.nl/product/custom-solar-panels-cubesats-nanosats/-retr. November 2018.
[33] M. Coletti, R. I. Marques, S. B. Gabriel, Design of a Two-Stage PPT for Cubesat, in: Proceedings of the 31st International Electric Propulsion Conference, Ann Arbor, MI, USA, September 2009. IEPC-2009-244.
[34] R. I. Marques, S. B. Gabriel, F. d. S. Costa, The Two-Stage Pulsed Plasma Thruster, in: Proceedings of the 31st International Electric Propulsion Conference, Ann Arbor, MI, USA, September 2009. IEPC-2009-250.
[35] L. F. C. Marin, R. I. Marques, Performance Analysis of a Double Discharge Pulsed Plasma Thruster by Varying the Energy Distribution amongst its Two Stages, in: Proceedings of the 1st IAA Latin American CubeSat Workshop (I LACW), Brasília, DF, Brazil, December 8-11, 2014. IAA-BR-14-16-05.