Enabling Low-Cost High-Energy Missions with Small Spacecraft by Using Pulsed Plasma Thrusters

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Abstract: Pulsed Plasma Thrusters (PPTs) were the first Electric Propulsion devices ever to be employed in an actual space mission, and continue to be used today when simplicity, robustness and scalability to different power levels are dominant requirements. Therefore, they find a natural niche of application in small-spacecraft 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. While PPTs have drawn renewed attention from the international space community after a long hiatus, this has been generally limited, until now, to low Delta-V, low total impulse missions. In this paper, we investigate the possibility of performing high Delta-v, high total impulse missions, such as orbit raising or even deep-space missions, using PPTs onboard small spacecraft.

Nomenclature

A	=	propellant wetted area
APPT	=	ablative pulsed plasma thruster
$\alpha, \beta, \gamma, \delta$	=	coefficients
Δv	=	velocity increment, Delta-v
E	=	discharge energy
EFF	=	electrostrictive force feeding
EP	=	electric propulsion
FEEP	=	field emission electric propulsion
g	=	acceleration of gravity at sea level
GIE	=	gridded ion engine
I_{bit}	=	impulse bit

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I_{tot}	=	total impulse
I_{sp}	=	specific impulse
LEO	=	low earth orbit
LLO	=	low lunar orbit
LPPT	=	liquid-fed pulsed plasma thruster
LTA	=	late time ablation
m_{bit}	=	mass bit
m_i	=	initial mass
m_p	=	propellant mass
MPDT	=	magnetoplasmadynamic thruster
PFPE	=	perfluoropolyether
PTFE	=	polytetrafluoroethylene
PPT	=	pulsed plasma thruster

I.Introduction

WORKS by many authors have reviewed and analyzed Pulsed Plasma Thrusters (PPTs), yielding in some cases formulas that can be used for preliminary design purposes[1-3]. 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-satellite, mostly low-power applications[2,3], in more recent work special attention was given to larger devices for more ambitious, high-energy (a shorthand for high Delta-v, high total impulse) missions[4-7].

The design of such missions is far from trivial, as the high specific impulse values 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 imply low firing frequencies, due to power limitations onboard a small spacecraft. Consequently, total mission times could become long, of the order of many months, or even a few years. This would still be quite acceptable, though, for a low-cost technology demonstrator/scientific mission.

Until recently, this type of missions with PPT-propelled small spacecraft have been outside of the realm of possibilities, mainly because of the weight of the capacitors that would be needed. With recently developed capacitors, currently in the process of testing and qualification, they now might become possible, due to highly improved energy densities, opening a wealth of applications, including ambitious missions on string budgets.

This paper presents a preliminary study of a potential use of a PPT as a simple and robust primary propulsion system for such low-budget high-energy missions. As a sample mission, a LEO (Low Earth Orbit)-to-LLO (Low Lunar Orbit) transfer is analyzed. In order to assess the feasibility of such a mission, a preliminary design of a PPT system is done analytically, using previously developed scaling laws[2,3].

II.Background

Among the many Electric Propulsion (EP) concepts devised so far, the PPT is one of the most simple, reliable and trusted propulsion systems ever made, especially when using a solid polymer as propellant. This is usually Teflon®, commercial name for Polytetrafluoroethylene (PTFE), but other polymers have been used[8-11]. The basic operation idea of this type of thruster 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[12]. A thruster of this type is generally called ablative PPT, or APPT. 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, over the volume of the plasma[12].

Mainly because of its simplicity, the PPT has had a long history of development and testing, both in the former Soviet Union and 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 APPTs) mission to Mars, only months after a gridded ion engine (GIE) being sent into a suborbital test flight aboard the SERT 1 spacecraft by the United States[12].

Despite their relatively low efficiency, generally below 10%[12], 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[13,14] and has made them even more attractive in recent years, as increasingly smaller satellites, down to CubeSat[15] size (10×10×10 cm), have been built and launched.

Several problems, in particular carbonization and late time ablation (LTA), 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[12].

A. Thruster Scaling

Many authors have reviewed and analyzed APPTs[1-3,12-14,16-20], proposing mechanisms of operation and correlations between geometry, operating parameters and performance characteristics, yielding formulae that can be used for preliminary design purposes. The main of the above-mentioned formulae 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.

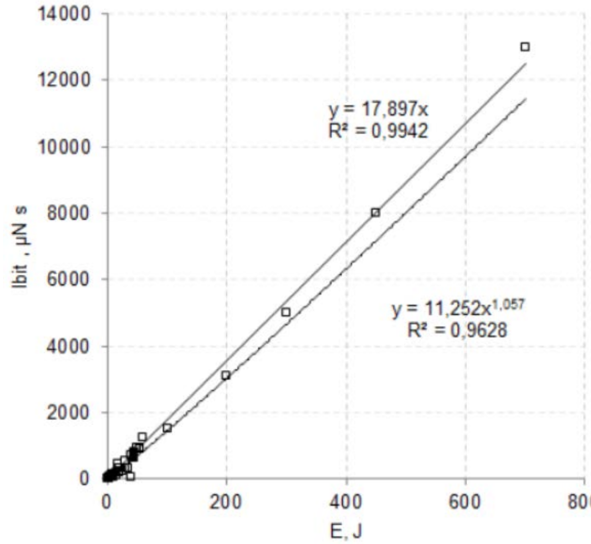

Figure 1. Impulse bit as a function of discharge energy[3].

$$I_{bit} = \alpha E^{\beta} \tag{1}$$

$$I_{sp} = \gamma \left(\frac{E}{A}\right)^{\delta} \tag{2}$$

The coefficients α, β, γ and δ in Eqs. (1) and (2) 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[2,3,18,20]. 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[3,18-21].

Different APPT discharge configurations have been adopted in the past decades[12]. One of the most common is the breech-fed, parallel-electrode configuration, basically consisting of a slab of PTFE propellant sandwiched between two electrodes. Another common configuration is the breech-fed, coaxial-electrode one. Both parallel and coaxial-electrode configurations can also be side-fed, as far as propellant is concerned.

New correlations of experimental data have been proposed in previous papers, in various discharge energy ranges[2,3,18,20]. Examples of such correlations are shown in Figs. 1 and 2 for a breech-fed, parallel-electrode configuration.

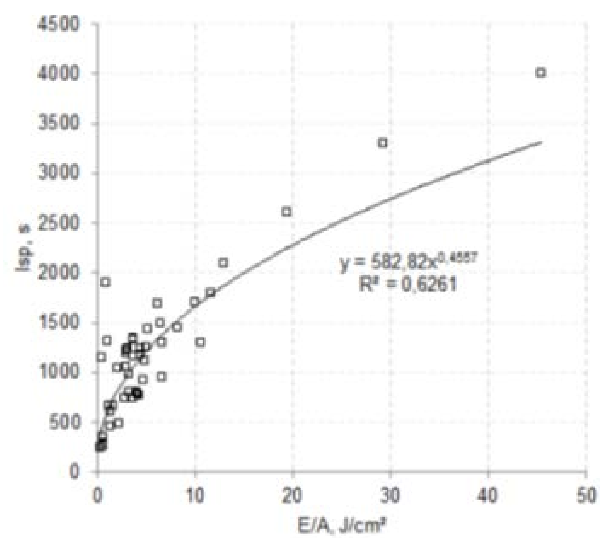

Figure 2. Specific impulse as a function of discharge energy per unit propellant wetted area[3].

From an inspection of the plot in Fig. 1 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. 2, 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.

These formulas are useful for the design of APPT propulsion systems, which are going to find increasingly wider application in the growing market of small satellites. Unlike most of the previous literature[22-26], in this paper we pay special attention to high-energy devices. It must be stressed that relations of the types (1) and (2), interpolations of experimental data, are only good guidelines for a preliminary, concept design. The development of actual thrusters is a process requiring, in general, an extensive experimental campaign.

Below discharge energies of 1 kJ, a wealth of experimental data from various thrusters is available, and has been reviewed and analized[1-3,17-20]. In the kJ-energy level, actual measurements become much sparser, yet some insight can be gained from experiments on ablative devices operated with a quasi-steady discharge, with values of current approximately constant on timescales of the order of 1 ms[11,27]. A comparison between the results of Refs. 11 and 27 can be made for discharge energies between 0.75 kJ and 1 kJ. The measurements in Ref. 11 yield values of I_{bit}/E and I_{sp} that are rather constant with discharge energy, namely around 20 mNs/J and 3000 s. Results from Ref. 27 yield values of I_{bit}/E consistently around 30 mNs, in that energy range, while I_{sp} varies from 3000 s to 4000 s. These differences, and a certain data spread, can be attributed to different discharge configurations and propellant geometries. Whereas both devices are coaxial, Teflon-ablation thrusters, in Ref. 11 the propellant (polymers other than PTFE were also tested) bars are fed radially into the discharge chamber. The device in Ref. 27 is, more traditionally, breech-fed. Interestingly, the thruster in Ref. 11 is called a quasi-steady, ablative magnetoplasmadynamic thruster (MPDT), while the device in Ref. 27 is called a quasy-steady APPT. Considering all the above, measurements can be taken to be in good agreement, and therefore we will use the data in Refs. 11 and 27 to estimate APPT performance up to several kJ energy levels. At 6 kJ and above, to the authors' best knowledge, only data from Ref. 27 are available, yielding values of I_{bit}/E of about 17 mNs/J and I_{sp} of about 4600 s. Various review papers[1-3,17-20] have pointed out that, in general, coaxial APPTs generate higher values of I_{bit} per unit discharge energy compared to breech-fed, parallel-electrode geometries. These latter are instead advantageous when trying to maximize specific impulse. The reasons for this difference in performance have been identified in the relative importance of electromagnetic and electrothermal acceleration, of the propellant ablation and other physical mechanisms, by various authors. A comprehensive database and an interesting analysis, summarizing many of the above considerations, can be found in Ref. 19. In view of all the above, however, further investigations are necessary before the use of high-energy APPTs can be recommended.

B. Preliminary Mission Design Considerations

The two main parameters used to characterize a given mission are Delta-v and total impulse I_{tot}. From the definition of specific impulse and from the rocket equation[28]

$$\frac{m_p}{m_i} = 1 - e^{-\frac{\Delta v}{gI_{sp}}} \tag{3}$$

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

$$I_{tot} = m_i g I_{sp}\left(1 - e^{-\frac{\Delta v}{gI_{sp}}}\right) = m_p g I_{sp} \tag{4}$$

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

$$I_{tot} = m_i \Delta v \tag{5}$$

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. Delta-v and total impulse become, in such cases, proportional to each other. In order for the difference between the value calculated using Eq. (4) and that using Eq. (5) 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 Ref. 2, where Eq. (5) is used, this will not generally be the case for medium-to-high-energy missions unless very high I_{sp} thrusters, like GIEs or Field Emission Electric Propulsion (FEEP), are employed. The use of Eq. (4) is, therefore, generally preferable when high-energy missions are being analyzed, even using EP, as in Ref. 7 and in the present paper. The feasibility of such 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 Delta-v of about 8 km/s[4,7]. 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 conducted using the relations in Refs. 2 and 3, or analogous relations of the same type, that we cannot respect all the constraints.

The key, in order to conduct high Delta-v 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 20 kg, a micro spacecraft at the high end of the nanosatellite class, and examine different preliminary concept design options. For our capacitors, we will adopt a value of energy density of 2.0 kJ/kg. This is definitely higher than any space-tested technology, but not unreasonable if we consider some types of carbon and polymer-based capacitors[29-31] currently undergoing development and testing.

III.Case Study

For our preliminary thruster design, we will adopt a breech-fed, parallel-electrode configuration, as it is the most widely used, and the one where it is easier to define important parameters of the geometry, such as the propellant wetted area A. The design relations of type (1) and, even more, type (2), present a much higher data spread in the case of coaxial geometries, more sensitive to parameter variations between different thruster models[3]. From the considerations in Section II.A, we can confidently extrapolate the value of 18 mNs/kJ, shown in Fig. 1, to kJ-energy levels, and expect to be able to achieve values of I_{sp} of the order of 3000 s.

With an I_{sp} of 3000 s, we see from Eq. (3) that the propellant mass will be 24% of the initial mass, that is, 4.8 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. (4) 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 (1). From the discharge energy, we then calculate the capacitor mass, using appropriate values of energy density. From Eq. (4), we calculate a total impulse of about 130 kNs. By assuming, conservatively, that our capacitors can withstand only one million pulses, we thus have for I_{bit} a value of 130 mNs. Then, by using for I_{bit}/E the value of 18 mNs/kJ, we calculate a value for E of 7.2 kJ, not much higher than that of the device tested in Ref. 27. With a value for energy density of 2.0 kJ/kg, we obtain a capacitor mass of 3.6 kg. The combined mass of propellant and capacitor bank, therefore, is about 8.4 kg, corresponding to 42% of the total mass of our spacecraft.

For what concerns the design of the APPT itself, from the relation type (2) reported in Fig. 2, we have that a discharge energy of 7.2 kJ and a specific impulse of 3000 s yield a wetted area of about 200 cm^2. Our APPT would thus be a big device, for example with 20 cm wide electrodes distanced by 10 cm. With a density of 2.2 g/cm^3 for PTFE, we then obtain for the propellant bar a volume of about 2200 cm^3 and a length of only 11 cm, to provide the total impulse needed for the mission. This may not be a practical design as such dimensions could be quite awkward to fit in a small platform. In addition to this, a 10-cm gap would probably make discharge breakdown rather difficult to initiate.

From the example 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[2]. As for the relation shown in Fig. 2, 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 could 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. Also, we have to remember that, in general, this type of expression shows significant data spreads, all the more so for coaxial configurations[3].

As for the total mission time, if we assume an onboard power available for propulsion of 150 W, unusual in very small spacecraft, but achievable with deployable solar panels, by operating at 7.2-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[2], are nowadays very conservative, so we can safely assume that this subsystem will have a mass of less than 1.5 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 Ref. 2, which ought to give us significant safety margins, and multiply the total mass of propellant, capacitors and power conditioner by a factor 1.5, thus estimating our total system mass at less than 15 kg. This would leave more than 5 kg for the other subsystems, like the ADCS, and the payload. This is probably a pessimistic estimate, as we have been quite conservative in our assumptions about the mass of power conditioner, electrodes, structure and so on. Provided we can achieve the values of specific impulse and energy density assumed in our case study, this type of mission would definitely become feasible. Notice also that capacitors may be available capable of withstanding much more than one million charge-discharge cycles. This would give us greater flexibility in the design, allowing us to use lower values of energy per pulse.

A deployable solar array configuration with a total area of 1-1.5 m^2 could provide an average power of about 150 W, even just using commercial, off-the-shelf components[32]. It is important to note that, in case such a configuration proved difficult to fit in a particular 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. This is determined, ultimately, by the mission total impulse and the available onboard power.

IV.Discussion

Electrode gap size is evidently fundamental in 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 some advanced capacitors, like for example carbon-based 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[29,30] would definitely require complex circuitry to make breakdown possible, even connecting many cells in a series. The use of polymer-based capacitors, with lower energy densities but much higher voltage ratings[31] could therefore be preferable.

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 Refs. 33-35, as it can impart an arbitrary amount of energy in the second stage to increase the specific impulse. Preliminary work has been able to achieve values of I_{sp} up to 4000 s with regular capacitors[36]. In a possible TS-PPT high-energy configuration, the first stage would act as a trigger to the second stage, where high-energy density capacitors could be discharged. The use of high-energy density capacitors 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}. The preliminary ideas presented in this and previous papers[4-7] represent just seeds towards developing a candidate configuration for a high-energy APPT.

Anticipating possible difficulties with achieving repeatable ignition and discharge conditions using large 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. 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, although at the cost of increasing system complexity.

As seen in Section II-A, although PPTs have found applicability as microthrusters, they can be scaled up to operate at discharge energies of several kilojoules[11,27]. However, an APPT operating at such energies would require a mass bit (m_{bit}) in the milligram range, where the uncontrolled ablation would significantly limit the propellant utilisation efficiency and increase LTA effects, resulting in performance downgrading. Evidently, discretising the mass bit would increase the performance. This is possible when using gas-fed of liquid-fed PPTs. These devices decouple propellant feeding from acceleration, which are intrinsically coupled in APPTs. Ablative devices tend to operate at values of specific impulse approximately constant, for constant propellant wetted area A, as both I_{bit} and m_{bit} increase with E[10,11]. We have to keep in mind, though, that, especially in the case of certain configurations, the propellant area actually exposed to the discharge may vary with E[27]. In spite of their advantages, the development of both gas and liquid fed PPTs has been hindered by the lack of feeding systems that maintain the simplicity, robustness and low-cost manufacturing typical of APPTs.

An electrostatic pump, termed Electrostrictive Force Feeding (EFF) unit, has been developed at the University of Southampton that employs an electric pressure gradient to deliver the m_{bit} in a coaxial Liquid-fed PPT (LPPT). This promising feeding device offers active mass bit delivery and requires no moving parts. Its simplicity does not entail a significant increase in manufacturing costs, compared to traditional PPTs. The ability to control liquids like Perfluoropolyether (PFPE) via strong and non-uniform electric fields has been demonstrated previously[37]. The EEF pump is a dedicated feeding sub-system for LPPTs. The pump consists of an open-end capillary connecting the discharge chamber and propellant tank. An electrode is placed inside the fluidic line, which, together with the anode, generates a diverging electric field inside the line. The electric field creates the pressure, which then pumps the liquid. Thus far, the EFF unit has proven reliable operation with PFPE as propellant, achieving mass bits in the range of 0.08 – 0.96 mg[38]. The experimental results prove that the mass bit increases with the applied electric field, hence, the feeding sub-unit is voltage-controlled. Moreover, the electrostatic pump has been operated over a period of 5.7 seconds, where the strong linearity of delivered mass bit and operational time of the pump indicates that continuous mass flow capabilities, necessary for large mass bit thrusters, will be achievable.

Scaling from the literature[39] suggests that a gas-fed PPT, operating with a mass bit of 0.2 mg at a discharge energy of 5 kJ, can be expected to generate an impulse bit of 32 mNs, with a specific impulse of 16000 s. The use of a EFF unit would increase the PPT versatility, as mass bit to energy ratio could be actively controlled, therefore allowing thrust modulation without compromising the specific impulse and overall efficiency. This constitutes a significant advantage to traditional, solid-propellant PPTs, while preserving a simplicity and robustness of the overall system comparable to APPTs.

V.Conclusion

High-energy missions with PPT-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, high-energy density capacitors, they now become possible. If they can be reliably operated for millions of pulses, this will open 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 and performance degradation due to LTA, in the case of APPTs, while trying to preserve overall system simplicity and robustness, which are the most attractive characteristics of these devices. It appears that the use of two-stage and/or liquid-fed configurations may be an avenue for improving performance while maintaining a level of system complexity not much higher than that of APPTs.

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