Ablative Pulsed Plasma Thruster System Optimization for Microsatellites [*] [**]

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IEPC-01-182

The Ablative Pulsed Plasma Thruster (APPT) can operate nominally at the very low power levels typical for microsatellites. In this paper we briefly discuss the operating characteristics of an APPT and review the existing data together with the state of the art of the various system parts. In order to characterize the behavior of an APPT and design the propulsion system optimized for a particular mission, it is necessary to have relations connecting the various parameters. Semi-empirical correlations are obtained by interpolating experimental data, available in the literature, regarding thrusters actually used on satellites, flight-qualified or merely laboratory-tested. Their validity and significance is discussed by comparing them with theoretical considerations, similar results and experimental analyses by various authors. Together with expressions for the masses of the various components of the APPT system, they constitute the basis of our model. The model is then applied to the optimization of an APPT system for given satellite mass and mission requirements.

Nomenclature

Acronyms

APPT Ablative pulsed plasma thruster
MPD Magnetoplasmadynamic
PFN Pulse forming network

Symbols

A	Area, cm^2
a, α	Coefficient
b, β	Coefficient
C	Capacitance, C
C_i	Coefficient
E	Energy, J
f	Pulse frequency, t^{-1}
F	Thrust, force, N
I	Impulse, N s
I_b	Impulse bit, N s
I_{sp}	Specific impulse
I_E	Specific thrust, I_b/E, N s/J
i	Current, A
M	Mass, Kg
P	Power, W
t	Time, s
v	Velocity, m/s
w, h	Linear dimensions
η	Efficiency
μ	Magnetic permeability, H/m
σ	Mass proportionality coefficient
τ	Discharge duration, s
ψ	Current parameter, $\int i^2 dt$, A^2 s

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[*] Presented as Paper IEPC-01-182 at the 27th International Electric Propulsion Conference, Pasadena, CA, 15-19 October, 2001.

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Subscripts

B	Bus
CB	Capacitor bank
opt	Optimal
P	Propellant, pulses
PC	Power conditioning
Sys	System
PS	Propulsion system
t	Thrust
th	Theoretical
tot	Total

Introduction

In recent years, the trend toward the deployment of increasingly small satellites in the scientific, commercial and military sector has rekindled the interest in propulsion systems that are particularly advantageous to power and mass-limited satellites. On such satellites, the propulsion subsystem is usually the most massive one. Therefore, the mass savings afforded by high-specific impulse electric thrusters are significant. Electric propulsion is thus emerging as a competitive alternative with respect to chemical systems, allowing the use of smaller launchers for single satellites or the deployment of multiple satellites with a single launcher.

The Ablative Pulsed Plasma Thruster (APPT) can operate nominally at power levels below 300 W, typical for small satellites. APPT systems operating at power levels down to a few tens of Watts have been designed and flown successfully[1]. The characteristics of simplicity and robustness of this type of thruster have been exhaustively described in the literature since the 1960s[2] and have led to its use in spite of an incomplete understanding of some of the physical processes involved. An extensive and excellent review of the theory and applications of this type of thruster by Burton and Turchi[3] and an interesting system optimization study by Choueiri[4] were published, respectively, in 1998 and 1996.

In this paper we examine the optimization of an APPT system for given satellite mass and mission requirements. Our model is based on fundamental definitions and semi-empirical correlations of experimental data, available in the literature, regarding thrusters actually used on satellites, flight-qualified or merely laboratory-tested. The model is then applied to the design of an APPT system optimized for actual, typical missions. The case study considered here is the drag compensation of a microsatellite in a Sun-synchronous orbit.

Physical Background

The APPT utilizes the rapid discharge of energy E from a capacitor or a pulse forming network (PFN) to ablate, ionize and accelerate a small mass of propellant to a high exhaust velocity[5]. A schematic representation of this type of thruster is shown in Figure 1 for a breech-fed configuration and in Figure 2 for a side-fed configuration .

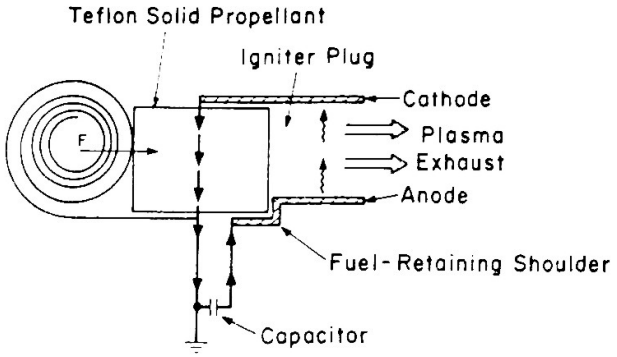

Figure 1 - Breech-fed APPT[6].

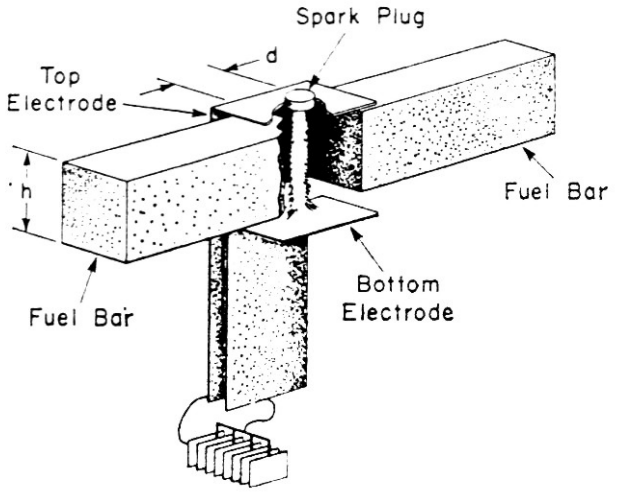

Figure 2 - Side-fed APPT[6].

The ejection of this mass produces an impulse bit I_b. If the thruster is operated at a fixed pulse frequency f, an equivalent steady state thrust is generated

$$F = f I_b.$$

The corresponding equivalent steady electric power is then

$$P = f E.$$

The relationship between the propellant mass m consumed per thruster discharge and its equivalent steady flow rate is

$$\dot{M} = f m.$$

The specific impulse, in seconds, is therefore

$$I_{sp} = \frac{F}{\dot{m} g} = \frac{I_b}{m g}.$$

The specific thrust is

$$I_E = \frac{F}{P} = \frac{I_b}{E}.$$

If a power conditioning system of efficiency η_{PC} is used to charge the capacitors, the required bus power will be

$$P_B = \frac{P}{\eta_{PC}} = \frac{fE}{\eta_{PC}}.$$

The thrust efficiency η_t is

$$\eta_t = \frac{I_b^2}{2 m E}.$$

If the power conditioning system is included, the system efficiency is then

$$\eta = \eta_{PC} \eta_t = \frac{\eta_{PC} I_b^2}{2 m E}.$$

The APPT presents therefore a peculiar characteristic, because its thrust is varied at constant specific impulse, specific thrust and efficiency. Variable thrust is realized by merely varying the pulse frequency. The electric power requirements are then directly proportional to this latter.

Semi-Empirical Correlations and Theoretical Considerations

In order to characterize the behavior of an APPT and design the propulsion system optimized for a particular mission, it is necessary to have relations connecting the various parameters defined above. In this paper, we will use semi-empirical correlations of experimental data. Their validity and significance will be discussed by comparing them to theoretical considerations.

The impulse bit I_b exhibits roughly linear dependency on the discharge energy E for each propellant-feeding geometry

$$I_E = \frac{I_b}{E} = const \qquad (1)$$

With a least-square interpolation of experimental data from various sources, we obtain, for a breech-fed configuration a characteristic value for the specific thrust[7,8,9,10,11,12,13,14,15] (Figure 3)

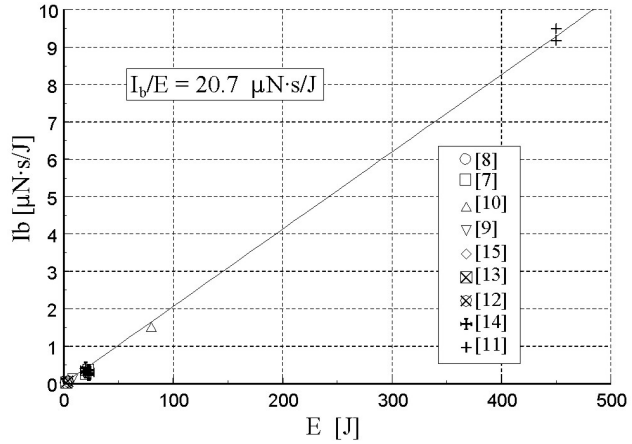

Figure 3 - I_b vs E, breech-fed configuration.

$$I_E = 20.7 \quad \left[\frac{\mu N\, s}{J}\right] \qquad (2)$$

and, for a side-fed configuration a considerably higher characteristic value[6,16,17] (Figure 4)

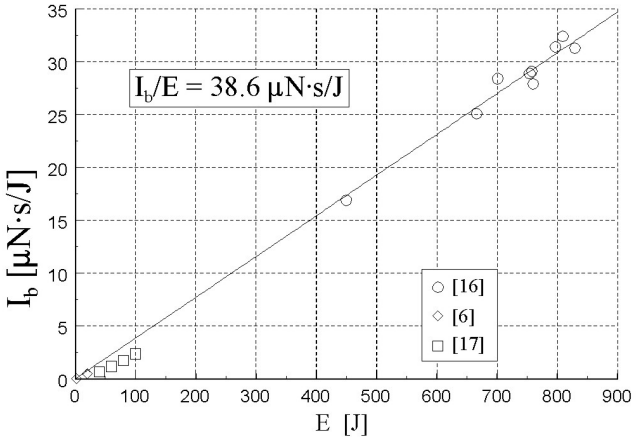

Figure 4 - I_b vs E, side-fed configuration.

$$I_E = 38.6 \left[\frac{\mu N\, s}{J}\right]. \qquad (3)$$

This result is widely confirmed by experimental analyses by various authors. There seems to be a general agreement upon a linear variation of impulse bit with discharge energy[18,19,20,5,21].

Guman, in particular, gives specific thrust values in good agreement with the ones presented in this paper. In the same energy discharge range of several tens to several hundreds of Joules, they are:

$$I_E = 17.8 \left[\frac{\mu N\, s}{J}\right]$$

for a breech-fed configuration[23] and

$$I_E = 40.0 \left[\frac{\mu N\, s}{J}\right]$$

for a side-fed configuration[5]. The results presented in our paper are slightly more conservative than the ones by Guman for side-fed thrusters, while the opposite is true for breech-fed thrusters. It is evident that the specific thrust is primarily a function of the Teflon® propellant geometry. In the side-fed configuration there is a considerable increase in the area of propellant exposed to the electric discharge with respect to the breech-fed one. Recent efforts, aiming at scaling down the APPT for use on nanosatellites (Primex Aerospace Company, AFOSR and DARPA University Nanosatellite Program), indicate that the thrust efficiency could be seriously degraded at low discharge energies. These studies yield a specific thrust

$$I_E = \frac{I_b}{E} = 14.0 \left[\frac{\mu N\, s}{J}\right] \qquad (4)$$

for a breech-fed configuration in the 5-to-50-J range and substantially confirm much older results obtained in an even lower ($E \leq 5$ *J*) energy range[18,19]. These results could make this kind of thrusters less attractive for low-power missions than previously envisioned. An analogous tendency toward performance degradation at low discharge energies is presented by Ablative Magnetoplasmadynamic (MPD) Thrusters. With Teflon® as a propellant, they show a weak increase of the specific thrust with increasing discharge energy. I_E tends, however, to become constant at higher energies. In particular, for E=3000 J, we have $I_E \sim 10$ $\mu Ns/J$[22]: this is a value which is much lower than the ones for both breech and side-fed APPTs. We have to remember, however, that the thruster these data refer to presents coaxial electrode geometry, radial propellant feeding and operates in a quasisteady mode, with square current pulses of 1 ms duration, at higher energies (some kiloJoules) than APPTs. It is to note that the energy actually flowing through the thruster is much lower than that stored in the capacitor bank at each discharge: for a matched load, the transfer of power is maximum, with a transmission efficiency equal to 50%. This is one of the main factors limiting the performance of this type of thrusters. It is interesting to note that, while the specific thrust calculated with the energy stored in the capacitor bank is approximately constant, showing an increase of only 5% from 1600 J to 3000 J, the same quantity, calculated with the energy actually flowing through the thruster during the discharge, presents a much more substantial increase: 17% in the same energy range. These results show again that the assumption of constant specific thrust is just an approximation, especially accurate at high energies.

The value given by Guman[20] for the specific thrust of all breech-fed thrusters ($I_E = 17.8$ $\mu N\, s/J$) is practically intermediate between those given by (2) up to several hundreds of Joules and by (4) for low energies. This suggests using (4) for APPTs operating up to, say, $E = 50$ *J*, and (2) for those operating at higher discharge energies. At the same time, it sheds some light over the dilemma discussed above. While the specific thrust is not rigorously constant even for a specific thruster, it may to a good approximation be considered as such unless we employ the thruster at the low end of his discharge-energy range. But in order to achieve good performance we will naturally use the thrusters that are most efficient in the particular energy range we are interested in. For a preliminary dimensioning, it is useful to have some general relations connecting the various parameters. A compromise solution, therefore, is to consider I_E as constant in a certain energy range, with values different from range to range. Say, for example, 1 to 10 J, 10 to 50 J, 50 to 250 J and so on. This will entail the compilation of an extensive performance database and new interpolations of data.

From a theoretical point of view, a constant specific thrust corresponds to a purely electromagnetic acceleration of the propellant. An estimate of the electromagnetic force, performed by idealizing the

discharge geometry as a simple flat sheet of width w and length h normal to the thrust axis[21] yields, after integrating over the pulse duration τ,

$$I_b = \mu \frac{h}{2w} \int_0^{\tau} i^2(t)\, dt = \mu \frac{h}{2w} \Psi$$

where we define

$$\Psi = \int_0^{\tau} i^2(t)\, dt$$

as the "current parameter". This agrees well with the notion of proportionality of impulse bit to discharge energy.

The total mass ablated per pulse can scale with the current parameter. In fact, for a discharge surrounding the propellant, we expect some fraction of the resistive heating energy in the plasma to transfer to the propellant surface, thereby ablating propellant mass. The resistive heating rate is proportional to the square of the discharge current. Some authors point out that "the increase in m with ψ is something less than linear"[11]. Many of them, however, agree on a linear variation of m with E[21,19]. Experimental data seem to indicate that the mass ablated is simply proportional to the total energy of the discharge event, especially at high energies[23].

If both these behaviors were rigorously verified, "the specific impulse, obtained by dividing the total impulse by the total mass, could be constant over at least reasonable ranges of systems operation"[21]. Experiments with similar devices (Ablative MPD Thrusters) at higher power levels seem to indicate this kind of behavior[24], with a weak dependence of I_{sp} on E through the discharge resistance and the exposed propellant area. The specific impulse is expected[2] to vary either directly with energy per unit mass or with the square root of energy per unit mass, depending upon whether the thruster behaves only as an MHD device or only as a gasdynamic one, respectively. Some early results carried out with a coaxial electrode geometry showed that I_{sp} varied with E/m as[2]

$$I_{sp} \propto \left(\frac{E}{m}\right)^{0.78}$$

This suggested that the thruster behaves partly as an MHD device and partly as a gasdynamic one, an opinion shared by a great number of authors[2,6,19,11]. Obviously, as long as impulse bit and mass ablated per pulse are taken to be approximately proportional to the discharge energy, the specific impulse will be essentially constant, with at most a weak dependence on E. In some works it was found that the specific impulse of a given thruster geometry correlated quite well simply with the discharge energy[2]. Other experiments showed that I_{sp} vs E did not follow a predictable pattern[18]. This, again, suggests a weak variation of the specific impulse with the discharge energy.

Another important correlation is the one that links impulse bit, specific impulse and energy. The product $I_b I_{sp}$, when plotted on a logarithmic scale as a function of E, is well approximated by a straight line. With a least-square interpolation of experimental data we obtain, for each different thruster geometry, a semiempirical correlation of the type

$$\ln(I_b I_{sp}) = a + b \ln E.$$

This may be rearranged into the form

$$I_b I_{sp} = \alpha E^{\beta} \qquad (5)$$

with $\alpha = e^a$, $\beta = b$. In particular, from data available in the literature, we have:

$$I_b I_{sp} = 1.70 E^{1.65} \qquad (6)$$

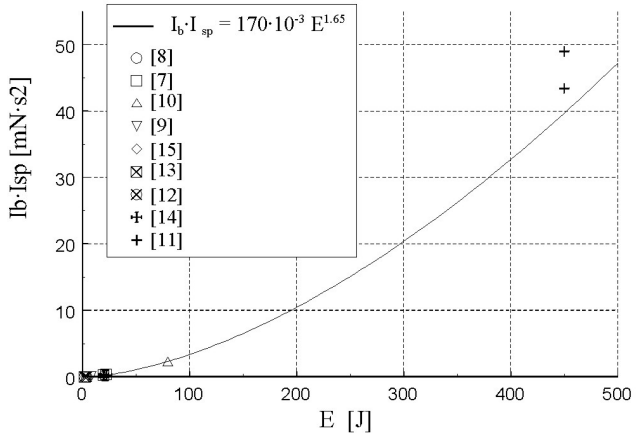

Figure 5 - $I_b I_{sp}$ vs E, breech-fed configuration.

for breech-fed thrusters (Figure 5) and

$$I_b I_{sp} = 3.97 E^{1.41} \qquad (7)$$

for side-fed thrusters (Figure 6).

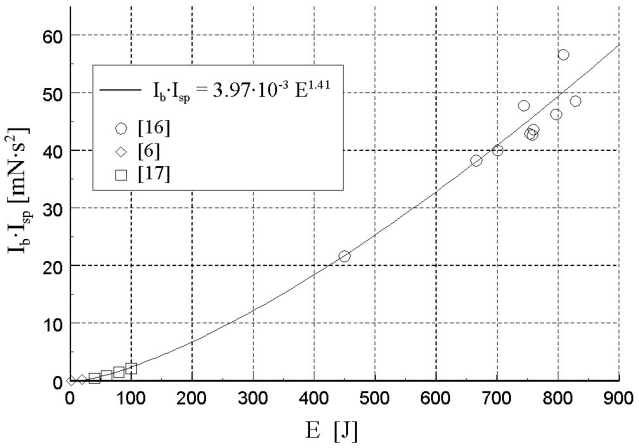

Figure 6 - $I_b I_{sp}$ vs E, side-fed configuration.

In these relations the impulse bit is expressed in millinewton-seconds, the specific impulse in seconds and the energy in joules.

Guman, again, gives analogous expressions[20]:

$$I_b\, I_{sp} = 2.45\, E^{1.60}$$

for breech-fed thrusters and

$$I_b\, I_{sp} = 1.12\, E^{1.60}$$

for side-fed thrusters. For breech-fed thrusters, the curve obtained in the present paper is only slightly more conservative than the one suggested by Guman. The opposite is true for side-fed thrusters (Figure 7). The situation is reversed with respect to the formulas for the specific thrust.

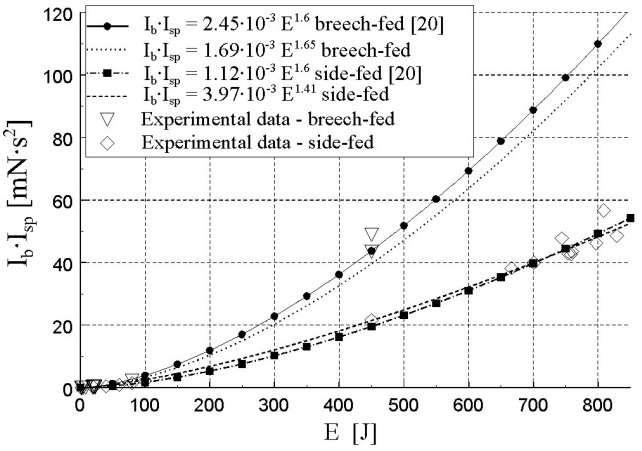

Figure 7 - $I_b I_{sp}$ vs E, comparison between different configurations and previously obtained relations.

The existence of these semi-empirical relations for the product $I_b I_{sp}$, and a linear, at least approximately, variation of I_b with E, both seem to indicate that I_{sp} may not be actually constant, at least over large ranges of operation, but exhibit an appreciable variation with E. This, in turn, would point out that the variation of m with E is, in reality, much less than linear. The dependence of mass ablated per pulse on discharge energy seems actually to be closer to a square-root law.

Another very useful correlation is the one between I_{sp} and the ratio E/A_P, where A_P is the area of propellant exposed to the discharge ("wetted" propellant area). The specific impulse, when plotted on a logarithmic scale as a function of E/A_P, is well approximated by a straight line, i.e.:

$$I_{sp} = constant \left(\frac{E}{A_P}\right)^{constant} \qquad (8)$$

With a least-square interpolation of experimental data we obtain, proceeding as before:

$$I_{sp} = 247 \left(\frac{E}{A_P}\right)^{0.87} \qquad (9)$$

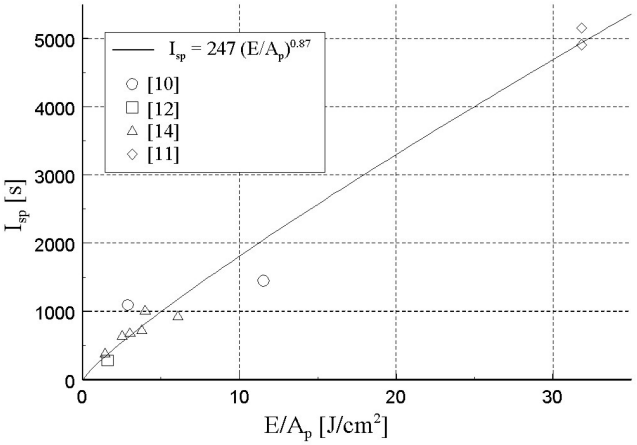

Figure 8 - I_{sp} vs E/A_p, breech-fed configuration.

for breech-fed thrusters (Figure 8) and

$$I_{sp} = 504 \left(\frac{E}{A_P}\right)^{0.40} \qquad (10)$$

for side-fed thrusters (Figure 9).

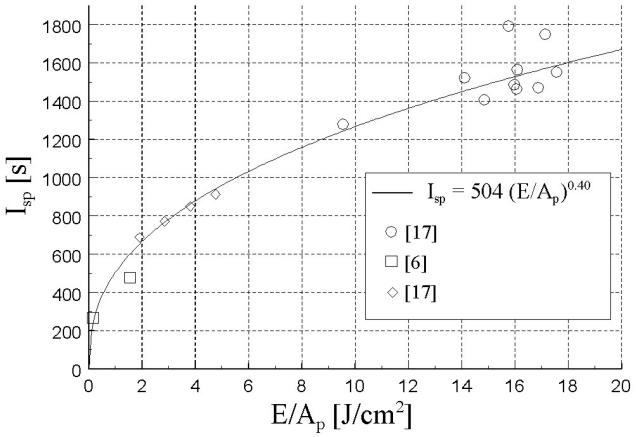

Figure 9 - I_{sp} vs E/A_p, side-fed configuration.

In these relations I_{sp} is measured in seconds, E in joules and A_P in square centimeters.

Guman suggests a single correlation of the same type:

$$I_{sp} = 317 \left(\frac{E}{A_P} \right)^{0.585}$$

valid for all different propellant-feeding geometries[20]. By comparing it with the correlations found in our paper, we see that it approximates reasonably well experimental data for side-fed thrusters, but not for breech-fed ones (Figure 9).

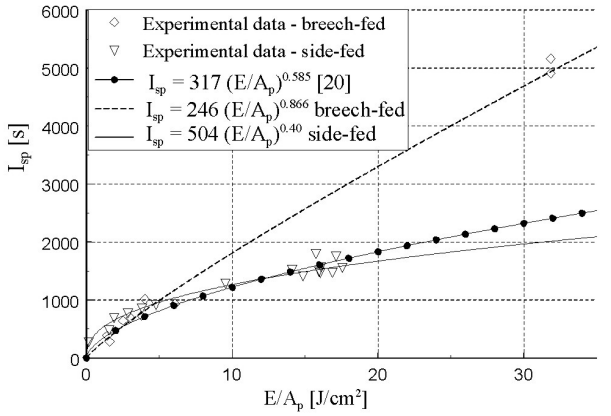

Figure 10 - I_{sp} vs E/A_p, comparison between different configurations and previously obtained relations.

These latter present a higher specific impulse than the side–fed ones, where a larger area of propellant exposed to the discharge ensures, as we have showed above, a higher impulse bit for a given discharge energy. The greater amount of mass ablated, though, degrades the specific impulse accordingly.

System Mass Breakdown

The total mass of the propulsion system, M_{PS}, can be expressed as the sum of several components:

$$M_{PS} = M_P + M_{CB} + M_{PC} + M_{Sys}$$

The propellant mass, M_P, can be immediately calculated from the value of the total impulse I_{tot} required to perform the mission as:

$$M_P = \frac{I_{tot}}{g\, I_{sp}}$$

The capacitor bank mass, M_{CB}, can be written in terms of the discharge energy E and the specific mass m_{CB}, in kilograms per Joule, of the capacitors:

$$M_{CB} = m_{CB}\, E.$$

Notwithstanding some advances, especially in the capacitor field, most of the technology of the 1970s is still suited for today's applications. Even if new types of energy storage capacitors are presently being developed, with energy densities m_{CB}^{-1} as high as 10^5 J/kg, ceramic capacitors are still the only off-the-shelf technology deemed reliable for long-term space applications[4]. The ceramic Z5U capacitor is benchmarked at 50.3 J/kg[25]. For our current study, we take an energy density of 50 J/kg (m_{CB}=0.02 kg/J) and a maximum number of pulses of 3×10^7, limited mostly by capacitor failure[4]. Aluminum electrolytic capacitors with m_{CB}^{-1} as high as 99 J/kg are presently available[26], but they are still deemed unreliable[27] and are damaged by current inversions. It is interesting to note that 35 J/kg capacitors had already been life-tested in the 1970s and that 88 J/kg capacitors were then expected to be available soon[5].

The power conditioner mass, M_{PC}, can be written as the sum of a fixed mass M_{PCf} and a mass dependent on the spacecraft bus power P_B through the coefficient m_{PC}, specific mass of the power conditioning system, in kilograms per watt:

$$M_{PC} = M_{PCf} + m_{PC}\, P_B$$

The power conditioner technology of the mid-1970s[5] offers a high value of efficiency (η_{pc}=0.8), a fixed mass of 0.5 kg and a specific mass $m_{PC}=11\times10^{-3}$ kg/W for P_B<200 W and $m_{PC}=5\times10^{-3}$ kg/W for P_B>200 W. For

values of P_B around 200 W, an average value of $m_{PC}=8\times10^{-3}$ kg/W can be used[4].

All the remaining parts of the propulsion system can be included in a so-called system mass, M_{Sys}. The discharge initiating circuit mass is about 0.2 kg, independent of thruster size and power[5]. The mass of the electrodes and of the structural assembly, including a lightweight fuel track to guide the solid propellant, varies from a little below 0.5 kg to a little over 3 kg[5], depending on thruster size and power. The packaging mass can be considered a percentage of the total mass of the propulsion system. For APPT flight-ready prototypes this percentage can be as low as 20%[28]. The system mass, M_{Sys}, can therefore be expressed as:

$$M_{Sys} = \sigma M_{PS}$$

The mass proportionality coefficient σ is not easy to estimate a priori, because it can vary considerably. A value of 0.35 is felt to be a quite conservative assumption[29].

The total mass of the propulsion system, M_{PS}, can thus be written as:

$$M_{PS} = M_P + M_{CB} + M_{PC} + M_{Sys}$$
$$= M_P + M_{CB} + M_{PC} + \sigma M_{PS}$$

This yields, for the total mass of one APPT module:

$$M_{PS} = \frac{M_P + M_{CB} + M_{PC}}{1 - \sigma}$$

Mission Analysis

Assuming that, from an analysis of our mission, the thrust F needed to execute certain maneuvers and the total impulse I_{tot} are known, choosing the pulse frequency f allows us to calculate the impulse bit I_b and the total number of pulses

$$N_p = \frac{I_{tot} f}{F}$$

After selecting a propellant-feeding configuration and therefore a value for I_E, we use (1) to calculate E and the bus power

$$P_B = \frac{F}{\eta_{PC} I_E}.$$

Then by using (5) we see that the specific impulse

$$I_{sp} = \alpha I_e^{-\beta} F^{\beta-1} f^{1-\beta} = C_1 f^{1-\beta}$$

decreases with increasing f ($\beta>1$ for both propellant-feeding configurations). The propellant mass

$$M_P = \frac{I_{tot}}{g I_{sp}} = g^{-1} I_{tot} C_1^{-1} f^{\beta-1}$$

instead increases with increasing f. The "wetted" area A_P is calculated from (8). M_P and A_P determine the size of our propellant bar, and therefore the dimensions of the electrodes. From the system mass breakdown we finally calculate the total mass of our APPT module

$$M_{PS} = (1-\sigma)^{-1} [g^{-1} I_{tot} f^{\beta-1} C_2 f^{-1} + C_3].$$

where

$$C_2 = m_{CB} F I_E^{-1}, \quad C_3 = M_{PCf} + m_{PC} F I_E^{-1} \eta_{PC}^{-1}$$

In a preliminary dimensioning, the two main constraints are P_B and M_{PS}. P_B is determined by F and I_E, and therefore the propellant feeding configuration, and defines a "family" of thrusters. Within each "family", different values of f directly imply different values of E and, through the procedure outlined above, M_{PS}. The optimal value of f, f_{opt}, is the one corresponding to the minimum value of M_{PS}. We can thus determine our optimal design configuration.

In the case of a microsatellite, the constraints on M_{PS} and P_B are particularly stringent, given the limited total size of the spacecraft. We can then start our preliminary design procedure from some imposed values for P_B and M_{PS} and see for which couples of values (M_{PS}, P_B) we have a configuration capable of satisfying our mission requirements. This type of procedure is also useful in case we want to fit an APPT system on a preexistent platform. We can evaluate F from our (1) after selecting a propellant-feeding configuration and therefore a value for I_E. We can thus assess the kind of maneuvers our spacecraft will be able to perform. Selecting a pulse frequency f allows us to calculate E. Now for every chosen value of P_B the thrust F is a constant and the impulse bit

$$I_b = \frac{\eta_{PC} P_B I_E}{f} \quad [\mu N\ s]$$

decreases with increasing f. From (5) we can then calculate the specific impulse

$$I_{sp} = \alpha I_E^{-1} (\eta_{PC} P_B)^{\beta-1} f^{1-\beta} = C_1 f^{1-\beta}\times10^3 \quad [s].$$

The various masses can be evaluated from the system mass breakdown. From the value of the propellant mass

$$M_p = (1-\sigma)M_{PS} - M_{PCf} - M_{PC}P_B - M_{CB}\eta_{PC}P_B f^{-1} \quad [kg]$$

we can then calculate the total impulse

$$I_{tot} = M_p I_{sp} g$$

$$= g C_1 \{[(1-\sigma)M_{PS} - C_3]f^{1-\beta} - C_2 f^{-\beta}\} \times 10^3 \quad [N\ s]$$

and finally, knowing I_b, the total number of pulses

$$N_p = \frac{I_{tot}}{I_b}.$$

From the knowledge of I_{sp} and E we can calculate A_P using (8) and therefore the size of the propellant bar. This is particularly important in the case of a microsatellite.

Now a "family" is defined by values for M_{PS}, P_B, F and I_E. Again, within each family different values of f imply different values of E. Now f_{opt} will correspond to a maximum value of I_{tot}. In order to determine this, we differentiate the expression obtained for I_{tot} with respect to f and equate the result to zero, obtaining:

$$\frac{dI_{tot}}{df} = g C_1 f^{-\beta} \{[(1-\sigma)M_{PS} - C_3](1-\beta) + C_2 \beta f^{-1}\}$$

$$= 0.$$

The evident solution $f \to \infty$ is, of course, devoid of any physical interest. By studying the second derivative, we see that it corresponds to a horizontal asymptote, as $I_{tot} \to 0$ when $f \to \infty$ of and that the other solution

$$f = \frac{C_2}{(1-\sigma)M_{PS} - C_3} \frac{\beta}{\beta - 1} = f_{opt}$$

actually corresponds to a maximum. We find this by substituting back the value calculated for f_{opt} into the expression for I_{tot}:

$$I_{totMax} = g C_1 C_2^{1-\beta} [(1-\sigma)M_{PS} - C_3]^{\beta} \frac{(\beta-1)^{\beta-1}}{\beta^{\beta}}.$$

If this value of I_{tot} does not allow us to satisfy our initial mission requirements, we are left with the choice of either modifying them, planning for instance a shorter mission, or increasing the mass of the spacecraft or the power available on board. If, on the other hand, we can achieve a value of I_{tot} that exceeds our mission requirements, this means that we will be able to use a smaller spacecraft, with a smaller solar panel area, or even increase the duration of our mission. Of course, we may also choose not to operate at f_{opt}, but in this way we would not exploit our configuration to the fullest of its potential. Values of f less than

$$f = \frac{C_2}{(1-\sigma)M_{PS} - C_3} = f_0$$

for which $I_{tot}=0$ correspond to negative values for I_{tot} and have obviously no physical relevance.

Case Study

As a mission-analysis example, we choose the drag compensation of a small satellite in a "near noon-near midnight" Sun-synchronous orbit at an altitude of 500 km and with an orbital period of 94.62 minutes. This 80-Kg mass satellite falls into the microsatellite (10-to-100-kg mass) category, according to the standard proposed by the Air Force Research Laboratory (AFRL) in 1998[30]. We will use a single propulsion system. This mission requires a velocity increment per year

$$\Delta v = 17.5 \quad \left[\frac{m}{s}\right]$$

with a corresponding total impulse [31,32,33]

$$I_{tot} = 1400 \quad [N\ s].$$

The initial design mission duration is two years.

If we choose a drag compensation strategy, by selecting the thrusting time during one orbit, we then have the thrust required of our APPT system.

Instead of following the first of the two procedures that we have outlined, in this case it is more appropriate, as we said above, to start from our constraints on bus power and total mass of the propulsion system. Both these are in general extremely limited on microsatellites. Let us consider in particular $M_{PS} = 2.5\ kg$ and three possible values for the available bus power, $P_B = 10, 30, 50\ W$.

By choosing a propellant-feeding configuration, and therefore a value for I_E, we can calculate the thrust F. Given the low values of available bus power, we might consider employing the more conservative value of I_E for breech-fed configurations at low discharge energies as given by (4). The thrust values are shown below, in Table 1.

Table 1. Thrust values for different configurations

P_B [W]	F [μN]		
	Low-Energy	Breech-fed	Side-fed
10	112	166	309
30	336	497	926
50	560	828	1540

In a drag-compensation type of mission, our spacecraft is not going to be expected to perform any maneuver that may require a certain minimum level of thrust. We can therefore consider very low values for the available bus power, having chosen a suitable drag-compensation strategy.

Let us now examine a breech-fed configuration, and therefore a specific thrust of 20.7 μN s/J and $\alpha = 1.70$, $\beta = 1.65$, as given by (2) and (6). By following the optimization procedure that we have outlined above we obtain the results summarized in Table 2.

Table 2. Breech-fed thruster optimization

P_B [W]	10	30	50
f_{opt} [Hz]	0.400	1.53	3.53
$I_{tot\,Max}$ [N s]	3470	2320	1360
E [J]	20.0	15.7	11.3
I_B [μN s]	415	325	235
I_{sp} [s]	575	491	397
N_p	8.36×10^6	7.14×10^6	5.79×10^6
M_{CB} [kg]	0.400	0.314	0.226
M_{PC} [kg]	0.610	0.830	1.05
M_P [kg]	0.615	0.481	0.348

We can see that, with $P_B = 10\ W$, we obtain a value of total impulse sufficient to perform a two-year mission with a safety margin of nearly six months. At higher values of the available bus power, the mass of the power conditioning system begins to dominate the mass budget, for a fixed total mass. Therefore, the mass of the propellant we can carry decreases and with it the total impulse attainable. The total number of pulses is, in all cases, well below the maximum safe limit of 3×10^7.

It is interesting to compare these results with those we would obtain by using the more conservative estimate of specific thrust, valid for low discharge energies, as given by (4). At $E = 20\ J$, in fact, it seems more appropriate to use the correlation $I_E = 14.0\ \mu N\ s/J$. We use the same values as before for α and β, from (6). The results are summarized in Table 3.

Table 3. Breech-fed thruster optimization

P_B [W]	10	30	50
f_{opt} [Hz]	0.400	1.53	3.53
$I_{tot\,Max}$ [N s]	5120	3420	2000
E [J]	20.0	15.7	11.3
I_b [μN s]	280	220	159
I_{sp} [s]	848	724	586
N_p	1.83×10^7	1.55×10^7	1.26×10^7
M_{CB} [kg]	0.400	0.314	0.226
M_{PC} [kg]	0.610	0.830	1.05
M_P [kg]	0.615	0.481	0.348

Now, the values for thrust (Table 1) and impulse bit are smaller, while those for the specific impulse are higher. Optimum pulse frequencies, discharge energies and masses of power conditioning system and propellant are the same. The values of the maximum achievable total impulse are thus higher. This means that if we still use a 10-W bus power we can nearly double the duration of our mission by extending it to three years and eight months. If we are not interested in doing this, our spacecraft can still perform a two-year mission with nearly the same margin as in the previous case with a 30-W bus power. This means that the thrust available will be three times as big, enabling our spacecraft to perform a wider range of maneuvers or a different drag compensation strategy. This might not very important, in general, for this type of mission, so we might even be interested in cutting down the propellant mass, therefore carrying enough only for a two-year mission and further reducing the total mass of our microsatellite. We could save 0.2 kg and still maintain the same margin as before. With the continuous progress being made in miniaturization, this could allow us to carry a small extra payload. Finally, the total number of pulses, even if higher than before, is still below 3×10^7. For $P_B = 50$ W, the optimum pulse frequency is close to the highest values (~4 Hz) reported in the literature[29]. Higher values of f might imply time intervals between pulses not long enough to charge the capacitors.

Let us now repeat our optimization for the case of a side-fed configuration. Now we have $I_E = 38.6$ μ*N s/J*, $A = 3.97$ and $B = 1.41$, as given by (3) and (7). The results are summarized in Table 4.

Table 4. Side-fed thruster optimization

P_B [W]	10	30	50
f_{opt} [Hz]	0.542	2.08	4.78
$I_{tot\,Max}$ [N s]	2190	1550	984
E [J]	14.8	11.5	8.37
I_b [μN s]	570	445	322
I_{sp} [s]	311	281	246
N_p	3.84×10^6	3.48×10^6	3.06×10^6
M_{CB} [kg]	0.296	0.230	0.167
M_{PC} [kg]	0.610	0.830	1.05
M_P [kg]	0.720	0.564	0.408

In this case our thruster is operating at high thrust and impulse bit and low specific impulse. The propellant consumption is, therefore, much higher. If we wish to keep the total mass of our propulsion system fixed, this greatly reduces the total impulse. Now, even at 10 W bus power we cannot meet the initial mission requirements, but fall short by nearly six months. We could operate at lower bus power. With $P_B = 5$ W, we would still have a thrust of 154 μN. Now the optimum pulse frequency is much lower, 0.257 Hz, therefore $I_b =$ 600 μN s. The specific impulse is now 318 s. We do not gain much in I_b and I_{sp}. The propellant mass available does not increase much either: $M_P = 0.759$ kg. The reduction in the mass of the power conditioning system is partially offset by the increase in the mass of the capacitor bank, as now the discharge energy has gone up to 15.7 J. M_{PC} is now close to its fixed minimum value, therefore there is no appreciable gain to be expected from a further reduction in bus power. The maximum total impulse attainable is 2370 Ns, still far short of our initial mission requirement of a two-year drag compensation. A side-fed configuration does not appear, therefore, as a viable choice for our mission.

In cases where both configurations could be employed, a factor that may be determinant is the volume occupied by the propulsion system. This is, to a great extent, determined by the size of the propellant bar. Teflon® has a density of about 2.17 kg/m^3. From M_P we calculate the volume of the propellant bar. Then, using the relation (8), we can determine the "wetted" propellant area A_P and thus the length of the bar. A breech-fed configuration is simpler, but in a side-fed one the propellant mass is divided in two bars. Teflon® can be stored in the form of a straight rod, a circular segment or a helical coil. Various components, like for instance the capacitor bank, can be placed in the volume surrounded by the propellant bars[20,34]. For applications with a high total impulse requirement, therefore, where the size of the propellant bar becomes a primary factor, a side-fed configuration may be the preferred approach. For lower-I_{tot} missions, instead, a simpler breech-fed thruster is generally deemed adequate.

Conclusions

The operation of the Ablative Pulsed Plasma Thruster (APPT) has been analyzed. Correlations of experimental data have been obtained. These have then been compared with results in the literature, obtained by interpolating different data with the same type of formulas, and evaluated in the light of theoretical considerations. Together with expressions for the masses of the various components of the APPT system, they constitute the basis of our model.

The model is then applied to the design of an APPT system optimized for actual, typical missions. Given the various mission requirements and constraints, it is possible to proceed to the dimensioning of our propulsion system along different lines. From the knowledge of the Δv characteristic of our mission, and therefore of the I_{tot} required, taking into account the limitations on the total mass M_{ps} of the propulsion system and on the bus power P_b, it is possible to determine the mass of the various components and the exposed propellant area A_p. The number of propulsion modules, their thrust F and the discharge frequency f are parameters that can be varied by the designer, as well as the choice between a breech-fed or side-fed configuration, which determines I_E. Our model explores the interdependencies of all these quantities and endeavors to illustrate their significance from a physical standpoint. From the example discussed in the present paper it is made clear how to determine an optimum preliminary design for a given mission.

Simple relations, e.g., constant specific thrust, are rigorously valid only for each single thruster in its

optimum performance medium-to-high-energy range. More general correlations are useful for the preliminary design of an APPT propulsion system. In this paper, a linear relation between discharge energy and impulse bit is still adopted as best working hypothesis, but two different values are proposed for breech-fed thrusters, for energies above an below 50 Joules. This is still far from satisfactory. The compilation of an extensive database and new interpolations will be necessary in order to obtain relations between operating parameters valid for a certain number of basic configurations in different energy ranges. This work will be the subject of future publications.

Acknowledgments

The authors wish to thank the School of Engineering and Applied Sciences, University of Southampton, and in particular Dr. Stephen B. Gabriel, for allowing Mr. Paolo Gessini the use of the University facilities and for supporting him during the writing of this paper. The authors also wish to thank Mr. Massimiliano Agnesi, whose thesis work has directly inspired the writing of this paper and constitutes its foundation.

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