PERFORMANCE OF PULSED PLASMA THRUSTERS

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Abstract. *Pulsed plasma thrusters (PPTs) generate thrust through a sequence of electrical discharges on the surface of a solid dieletric. This work describes a simplified performance model of a PPT, for coaxial or parallel geometries, considering the effects of the non-ionized gases and the energy losses in the lines, capacitors, electrodes and plasma. Expressions for specific impulses, impulse bits, ablated mass and thrust efficiency are obtained in terms of the mass fraction of the ionized gases. The mass efficiency (payload ratio) of a spacecraft using pulsed plasma thrusters is derived for a given mission with known characteristic velocity.*

Keywords*: pulsed plasma thruster, performance model, energy losses, mass efficiency*

1. INTRODUCTION

A pulsed plasma thruster (PPT) utilizes discharges of electrical energy stored in capacitors to vaporize a solid dielectric and to ionize the vapor. The self-induced magnetic field accelerates the resulting plasma. Nevertheless, not all vapor is ionized and the fraction of non-ionized gas in the PPT can reach 40 % of the ejected mass. Despite that, the plasma and the non-ionized gas are ejected at high velocities, yielding specific impulses higher than 4000 s.

Burton and Turchi (1998) presented a detailed review of the development and applications of PPTs in the previous 35 years, showing some performance data of different PPTs.

Marques and Costa (2004) and Marques (2004) presented a short review, described different PPT configurations and presented initial results obtained for coaxial PPTs developed and tested at the Combustion and Propulsion Laboratory of the Brazilian Space Research Institute (LCP/INPE). They tested the coaxial PPTs with different discharge chambers and surface energy densities, in a vacuum chamber with relatively high pressures (~10E-2 mbar). As a result, the ablation rates were low and the ablated mass per discharge increased exponentially with the discharge energy, in contrast to a linear variation in lower pressures, as observed in other PPTs described in the literature.

In the last 2 years a new high energy PPT prototype (up to 200 J per discharge) with parallel electrodes was designed and tested at the LCP/INPE and at the Astronautics Department of the Southampton University, aiming to obtain higher ablation rates and to improve the discharge control. The preliminary results were encouraging, with ablation rates about 20 % of the literature data. The tests were performed at a lower pressure (10E-5 mbar) than the tests with the coaxial PPTs (Marques et al., 2007).

In general, the utilization of electric propulsion systems allows to obtain high specific impulses, but it is required the transport of power sources which can have a significant mass, such as solar panels and batteries. In order to calculate the payload ratio for a space mission using pulsed plasma thrusters it is necessary first to determine the relation between the energy of a pulse and the resulting ejection velocities, for a given hardware.

Simplified models of ablative pulsed plasma thrusters or gaseous MPD thrusters were presented previously by Andrenucci et al. (1979), Choueiri (1998) and Brito et al. (2004), among others. These models were also based on the literature on plasma guns. Space mission analyses utilizing PPTs were also presented before. Choueiri et al. (1993) compared the performance of MPD thrusters with chemical systems for LEO-GEO orbit transfers, Costa and Carvalho Jr. (1998) presented an analysis the analysis of the performance of electrical and nuclear propulsion systems for space missions in general and Gessini and Paccani (2001) made an optimization study of the utilization of PPT's for a given space mission. Recently, at the International Eletric Propulsion Conference 2005, several papers were presented discussing many aspects of the performance of pulsed plasma thrusters (Uezu et al., 2005, Moller et al., 2005, Kamhawi et al., 2005, and Berkery and Choueiri, 2005).

The objective of this work is to present a simplified theoretical model for the performance of pulsed plasma thrusters and to analyse missions using PPTs. The model can be used for coaxial or parallel PPTs, and takes into account the effects of non ionized gases and the energy losses in the lines, capacitors, electrodes and in the plasma. The model is based on an electric circuit analysis, with a given plasma impedance, and it is based on the model presented by Andrenucci et al. (1979). Expressions for the specific impulse, impulse per discharge, vaporized mass and thrust efficiency are derived. An equation for the mass efficiency (payload ratio) of a satellite or spacecraft utilizing pulsed plasma thusters is obtained for a space mission with a given characteristic velocity. This study represents a first step to derive an improved model to describe a PPT operating with several bursts per discharge (Marques at al., 2007).

2. PROPULSION CHARACTERISTICS OF A PPT

Figure 1a shows a coaxial PPT with an internal electrode of external radius r_i and an external electrode of internal radius r_e. The induced magnetic field B produced by the electrical discharge with a current i between the electrodes has intensity

$$B = \frac{\mu_0}{2\pi}\frac{i}{r} \tag{1}$$

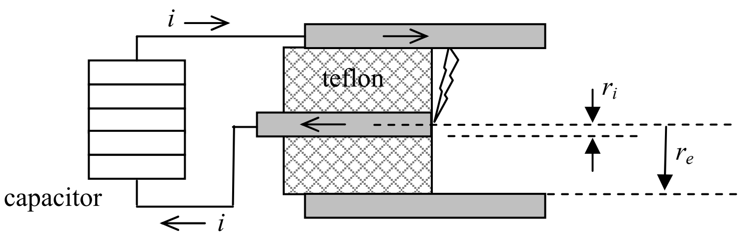
where r is the radial distance between an ion and the cathode.

Therefore, the force applied on an ion, F_i, is given by the integral of $B \times i$ (Lorentz force) between r_i and r_e:

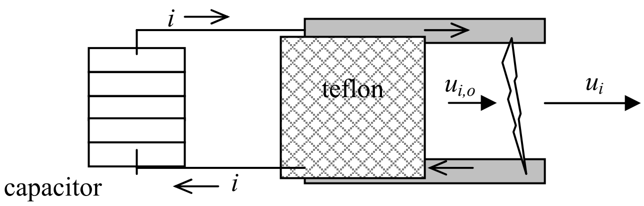
$$F_i = \int_{r_i}^{r_e} B \times i dr = \frac{1}{2} L' i^2 \tag{2}$$

where $L' = \frac{\mu_0}{2\pi} \ln \frac{r_e}{r_i}$ is the inductance per unity length of the discharge chamber.

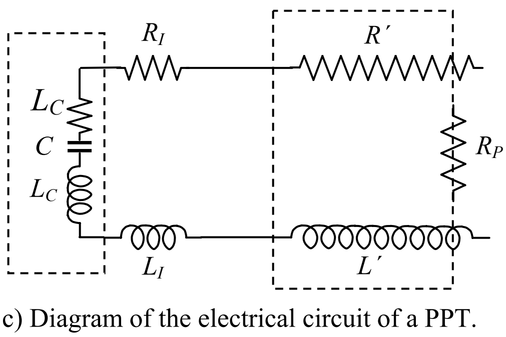
In the case of plasma thrusters with parallel plates, the inductance per unity length is given by $L' = \mu_0 h / d$, where h is the distance between plates and d is the width of the plates. Equation (2) is known as Maecker formula (1955) and its detailed deduction in terms of the current densities is presented by Jahn (1968). A discussion and extension of Maecker formula for gas MPD thrusters is presented by Choueiri (1998).



a) Coaxial PPT



b) PPT of parallel plates (side feed)



c) Diagram of the electrical circuit of a PPT.

Figure 1 – Schemes of PPTs.

Since only a fraction of the vaporized propellant mass is ionized, the total thrust F is calculated from

$$F = \dot{m}_i u_i + \dot{m}_n u_n + P_g A_e \tag{3}$$

where $\dot{m}_i$ is the mass flow rate of ejected ions, $\dot{m}_n$ is the mass flow rate of ejected neutral particles, u_i is the ejection velocity of ions, u_n is the ejection velocity of neutral elements, P_g is the pressure and A_e is the plasma exhaustion area.

According to literature data, mainly from plasma guns, the total mass of consumed propellant is, approximately, proportional to the square of the electrical current, i.e.,

$$\dot{m} = \dot{m}_i + \dot{m}_n = ki^2 \tag{4}$$

and, consequently, the evaporated mass per discharge is given by

$$m = \int_0^{t_p} \dot{m}dt = k\int_0^{t_p} i^2 dt = k\psi \tag{5}$$

where $\psi = \int_0^{t_p} i^2 dt$ and t_p is the discharge period.

The specific impulse of a thruster, *Isp*, is defined by

$$Isp = F/\dot{m}g_o \tag{6}$$

where g_o is the gravity acceleration at sea level. Therefore, the specific impulse of a PPT is given by

$$Isp = \frac{\dot{m}_i u_i + \dot{m}_n u_n + P_g A_e}{\dot{m}g_o} = \frac{1}{g_o}\left(f_i u_i + (1-f_i)u_n + \frac{P_g A_e t_p}{k\psi} \right) \tag{7}$$

where $f_i = \dot{m}_i/\dot{m}$ *is* the mass fraction of ions in the plasma.

The applied magnetic force can be related to the variation of the ion flow momentum, i.e.,

$$\frac{1}{2}L' i^2 = \dot{m}_i\left(u_i - u_{i,o}\right) = f_i\dot{m}\left(u_i - u_{i,o}\right) = f_i k i^2\left(u_i - u_{i,o}\right)$$

where $u_{i,o}$ is the initial gas velocity before ionization. Therefore,

$$u_i = u_{i,o} + L'/\left(2 f_i k\right) \tag{8}$$

and the specific impulse becomes:

$$Isp = \frac{1}{g_o}\left(\frac{L'}{2k} + f_i u_{i,o} + (1-f_i)u_n + \frac{P_g A_e t_p}{k\psi} \right) \tag{9}$$

Assuming that $u_{i,o} \approx u_n$, Eq. (9) simplifies to:

$$Isp = \frac{1}{g_o}\left(\frac{L'}{2k} + u_{i,o} + \frac{P_g A_e t_p}{k\psi} \right) \tag{10}$$

The total impulse per discharge is given by

$$I = mg_o Isp \tag{11}$$

and the thrust efficiency is calculated from

$$\eta = I^2 / 2mE \tag{12}$$

where $E = CV_o^2 / 2$ is the energy initially stored within the capacitors. Combining Eqs. (11) and (12) with Eqs. (5) and (9), results:

$$I = \psi\left((L'/2) + kf_i u_{i,o} + k(1-f_i)u_n + P_g A_e t_p / \psi\right) \tag{13}$$

$$\eta = \frac{\psi}{2kE}\left((L'/2) + kf_i u_{i,o} + k(1-f_i)u_n + P_g A_e t_p / \psi\right)^2 \tag{14}$$

According to the scheme shown in Fig. 1c, the power balance for the PPT circuit is:

$$Vi = Ri^2 + Li\, di/dt + E_{k,i} + E_{k,n} + \dot{q}_l \tag{15}$$

where R and L are the total resistance and the total inductance, respectively, including the wiring (R_l and L_l), capacitor (R_C and L_C), electrodes (R' and L') and plasma (R_P). The resistance and the inductance of the electrodes can vary with propellant consumption, since the active area of the electrodes is reduced. $E_{k,i}$ and $E_{k,n}$ are the kinetic energies of ions and neutral elements, given, respectively, by:

$$E_{k,i} = \dot{m}_i u_i^2 / 2 \tag{16}$$

$$E_{k,n} = \dot{m}_n u_n^2 / 2 \tag{17}$$

e $\dot{q}_l$ represents the energy losses, calculated from:

$$\dot{q}_l = \dot{q}_s + \dot{q}_{rd} + \dot{q}_{cv} + \dot{q}_{ab} \tag{18}$$

where $\dot{q}_s$ is the energy loss in the plasma sheath, $\dot{q}_{rd}$ is the energy loss by radiation, $\dot{q}_{cv}$, is the energy loss by heat convection and $\dot{q}_{ab}$ is energy for ablation and sublimation of the propellant surface.

Defining $f_q = \dot{q}_l / Vi$ (~ 15%) as the fraction of energy losses, from Eq. (15), it follows that:

$$V = Ri + L\, di/dt + (1/2i)\left(\dot{m}_i u_i^2 + \dot{m}_n u_n^2\right) + f_q V \tag{19}$$

or

$$\left(1 - f_q\right)V = Ri + L\left(di/dt\right) + \left(ki/2\right)\left(f_i\left(u_{i,o} + L'/\left(2f_i k\right)\right)^2 + \left(1 - f_i\right)u_n^2\right) \tag{20}$$

Adopting the parameters

$$R^* = \left[R + \left(k/2\right)\left(f_i\left(u_{i,o} + L'/\left(2f_i k\right)\right)^2 + \left(1 - f_i\right)u_n^2\right)\right] / \left(1 - f_q\right) \tag{21}$$

and

$$L^* = L / \left(1 - f_q\right) \tag{22}$$

allows to simplify Eq. (20) to:

$$V = R^* i + L^*\, di/dt \tag{23}$$

Substituting the relation $i = -C\left(dV/dt\right)$ into Eq. (23) and rearranging, it results the classical equation for a RLC circuit:

$$L^{*}C\, d^{2}V/dt^{2} + R^{*}C\, dV/dt + V = 0 \tag{24}$$

with the initial conditions: $t = 0$, $V = V_0$ and $i = dV/dt = 0$.

The possible solutions for Eq. (24), with given initial conditions, are an under damped wave, a critically damped pulse or an over damped pulse, if R^{*} is larger, equal or smaller than $2\sqrt{L^{*}/C}$, respectively. Figure 2 shows the voltage curve from the high energy PPT prototype tested at LCP/INPE, and it can be verified that it corresponds to an under damped wave.

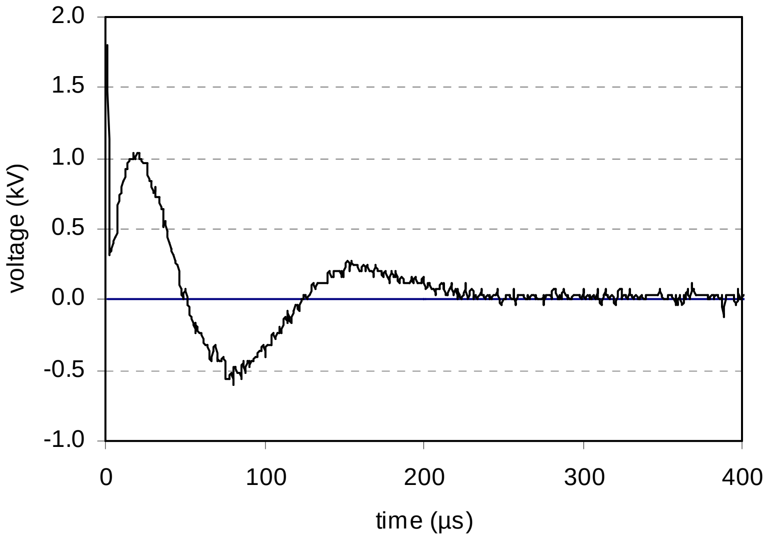


Figure 2 –PPT discharge voltage obtained in a PPT prototype with parallel electrodes.

It is verified that current reversion causes considerable losses in the PPT performance (Burton and Turchi, 1998). Due to this current reversion the first PPTs attained an efficiency of about 10 %, but use of discharge control allowed efficiencies of up to 30 % (Kamhawi et al., 2005).

Independently of the solution of Eq. (24), all dissipation in the circuit is included in the total resistance R^*, therefore, it can be written, directly:

$$E = \int_{0}^{\infty} R^{*} i^{2}\, dt \tag{25}$$

from where it follows that $\psi = E / R^{*}$. The PPT performance parameters can be rewritten in terms of the discharge energy:

$$m = kE / R^{*} \tag{26}$$

$$I = \left((L'/2) + kf_{i}u_{i,o} + k(1-f_{i})u_{n}\right)\left(E / R^{*}\right) + P_{g}A_{e}t_{p} \tag{27}$$

$$Isp = \frac{1}{g_{o}}\left(\frac{L'}{2k} + f_{i}u_{i,o} + (1-f_{i})u_{n} + \frac{P_{g}A_{e}t_{p}R^{*}}{kE}\right) \tag{28}$$

$$\eta = \frac{1}{2kR^*}\left((L'/2) + kf_i u_{i,o} + k(1-f_i)u_n + P_g A_e t_p R^* \Big/ E \right)^2 \tag{29}$$

It can be observed in the previous equations that the vaporized mass per discharge and the impulse per discharge are directly proportional to the stored energy in the capacitors and inversely proportional to the total equivalent resistance. The total equivalent resistance increases with increasing energy losses and total efficiency decreases with equivalent resistance. If the discharge energy is below the dielectric breakdown voltage the gas pressure will be zero and the pressure term becomes zero.

The exhaustion velocity of the neutral particles can are of the order of the sound speed (Brito et al., 2004), since there is an expansion process of a compressible fluid along a constant area duct (A_e). Therefore, $u_n \cong \sqrt{\gamma R_g T_g}$ and the propellant vapor temperature T_g can be obtained from the Clapeyron equation and from the perfect gas equation:

$$P_g = P_c \exp(-T_c / T_s) \tag{30}$$

$$P_g = \rho_g R_g T_g \tag{31}$$

where P_c is the characteristic pressure, T_c the characteristic temperature, T_s is the gas temperature at the propellant surface, γ is the gas specific heat ratio and R_g is the gas constant. For Teflon, P_c = 1,84×10^{15} N/m^2 and T_c = 20815 K (Burton and Turchi, 1998). The gas constant depends on the molar fractions of gases C and F, $M = X_C M_C + X_F M_F$, however according to Brito et al. (2004) it can be considered that, in general, $X_F \approx$ 10-11X_C and, therefore, $M \approx$ 18,43. Then, $R_g \approx$ 451 J/kg/K and the specific heat ratio for monoatomic gases is 5/3. The constant k is about 5×10^{-11} kg/A^2/s, according to Andrenucci et al. (1979).

The gas density is calculated by $\rho_g = \dot{m} / u_n = m / (u_n t_p)$ and, consequently,

$$T_g = \frac{\gamma}{R_g}\left[\frac{t_p R^*}{kE} P_c \exp(-T_c / T_s) \right]^2 \tag{32}$$

From Eq. (32), once T_s is known, T_g is calculated and then P_g and u_n can be calculated. The initial velocity of the ions can be approximated as $u_{i,o} \cong u_n$.

3. COMPARISON OF RESULTS

Gessini and Paccani (2001) made fits of various experimental data for different PPTs in the literature and verified that the impulse per discharge varies linearly with the discharge energy, in agreement with the present model, assuming that the contribution of pressure is also linear with the discharge energy. They also determined that the impulse per discharge is given approximately by I = 20,7.E μNs/J for PPTs with parallel plates with back feed and by I = 38,6.E μNs/J for PPTs with parallel plates with side feed. In the case of coaxial PPTs they verified that $Isp \sim (E/m)^{0,78}$, implying that the specific impulse is independent of E, since m is proportional to E.

4. MASS EFFICIENCY

Electric propulsion systems can provide high specific impulses, nevertheless the power source and the power conditioning unity can also have a large mass making the use of electric thrusters not viable. To determine the mass efficiency (payload ratio) of a propulsion system using PPTs, the mass distribution of a satellite or spacecraft is written:

$$m_o = m_{prop} + m_p + m_u + m_s \tag{33}$$

where m_o is the total initial mass, m_u is the payload mass, m_{prop} is the propellant mass (teflon), m_s is the structural mass and m_p is the mass of the power source and the thruster.

The power source and thruster mass, without the propellant, can be estimated by:

$$m_p = m_{po} + m_{bc} + m_{fe} \tag{34}$$

where m_{bc} *is* the capacitor mass, m_{fe} is the power source mass of and m_{po} is the residual mass including electrodes, wiring, packaging, triggering circuits of discharge and thruster supports.

The mass of capacitors is proportional to the stored energy:

$$m_{bc} = \rho_{bc} E \tag{35}$$

where ρ_{BC} is the specific mass per unity energy of the capacitor bench. The mass of the power source is proportional to the power P supplied by the source:

$$m_{fe} = \alpha_{fe} P \tag{36}$$

where α_{fe} is the specific mass of the power source per unity power supplied. The mass m_{po} can be taken as the fraction σ of the mass of the power source and thruster:

$$m_{po} \cong \sigma m_p \tag{37}$$

For solar panels $\alpha_{FE} \sim 0,01$ kg/W, for capacitors, in general, $\rho_{BC} \sim 0,02$ kg/J and $\sigma \cong 0,35$, according to Gessini and Paccani (2001). The effective power to the capacitors is $P_{ef} = \eta_p P$, where η_p is the conversion efficiency. Considering that $P_{ef} = fE$, where f is the operation frequency, then $P = fE/\eta_p$. Substituting now Eqs. (35,36,37) into Eq. (34), it is obtained

$$m_p = \frac{\rho_{bc} + \alpha_{fe} f/\eta_p}{(1-\sigma)} E \tag{38}$$

indicating that the mass of the power supply and thruster is proportional to the energy stored in the capacitors and to the specific masses of the capacitors and of the power source.

Dividing Eq. (33) by m_o and substituting Eq. (38), it follows that:

$$\eta_m = 1 - \frac{m_{prop}}{m_o} - \frac{\rho_{bc} + \alpha_{fe} f/\eta_p}{(1-\sigma)} \frac{E}{m_o} - f_s \tag{39}$$

where $\eta_m = m_u/m_o$ is the mass efficiency and $f_s = m_s/m_o$ is the structure fraction of the space vehicle.

The ratio m_{prop}/m_o is obtained from the momentum conservation applied to the space vehicle:

$$\frac{m_{prop}}{m_0} = 1 - \exp\left(-\Delta V/g_0 Isp\right) \tag{40}$$

where ΔV is the characteristic increment of velocity (m/s) for the specified space mission. Substituting Eq. (40) into Eq. (39), it gives:

$$\eta_m = \exp\left(-\frac{k\Delta V}{L'/2 + kf_i u_{i,o} + k(1-f_i)u_n + P_g A_e t_p R^*/E}\right) - f_s - \frac{\rho_{bc} + \alpha_{fe} f/\eta_p}{(1-\sigma)} \frac{E}{m_o} \tag{41}$$

or, simplifying the velocities of exhaustion of ions and neutral particles:

$$\eta_m = \exp\left(-\frac{k\Delta V}{L'/2 + ku_n + P_g A_e t_p R^*/E}\right) - f_s - \frac{\rho_{bc} + \alpha_{fe} f/\eta_p}{(1-\sigma)} \frac{E}{m_o} \tag{42}$$

Equation (42) indicates that large inductances per unity length and large ejection velocities of neutral gases increase the mass efficiency of a propulsion system using PPTs. Low specific masses of the power source and capacitor bench and low structural fractions make the propulsion system more efficient in terms of payload ratio.

Next it is presented an example for a space mission utilizing a coaxial PPT, performing a sequence of maneuvres for orbit correction and attitude control with $\Delta V = 1000$ m/s. The following data are used: $u_n = u_{i,\text{the}} = 2000$ m/s, $P_g = 0$, $f_s = 0{,}15$, $\eta_p = 0{,}9$, $\alpha_{FE} = 0{,}01$ kg/W, $\rho_{BC} = 0{,}02$ kJ/J, $\sigma = 0{,}35$ and $\mu_{\text{the}} = 4\pi\times10^{-7}$ N·A^{-2}.

Then, the following expression for the mass efficiency is obtained:

$$\eta_m = \exp\left(-\left(2 + \frac{10^{-10}}{k} \times \ln\frac{r_e}{r_i}\right)^{-1}\right) - 0.15 - \frac{0{,}02 + 0{,}0111 f}{0{,}65}\frac{E}{m_o} \tag{43}$$

Figures 3, 4 and 5 show the curves of mass efficiency for the specified mission versus E/m_o, for several values of k, r_e/r_i and f. There is a linear dependence of the mass efficiency with the E/m_o. The increase of the power source and thruster mass reduces the mass efficiency of the spacecraft, decreasing the mass of payload to values that eventually can make the mission not viable. In order to optimize the propulsion system, k should be decreased, r_e/r_i increased and f decreased.

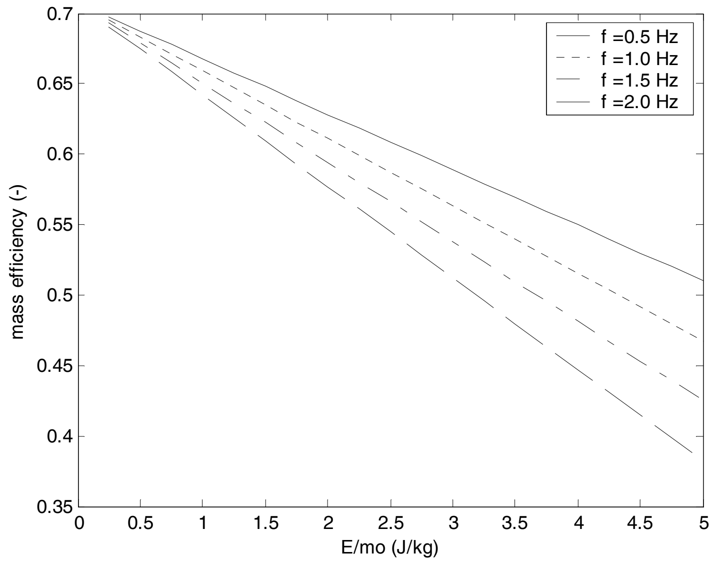


Figure 3 – Effects of E/m_o on mass efficiency for several values of the ablation constant k, with $r_e/r_i = 6$ and $f = 1$ Hz.

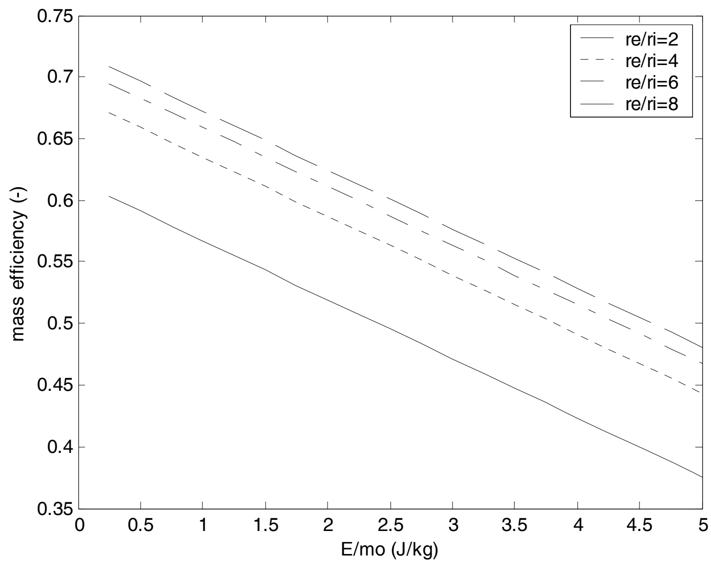


Figure 4 – Effects of E/m_o on mass efficiency for several values of r_e/r_i , with $k = 4.10^{-11}$ kg/A^{2}s and $f = 1$ Hz.

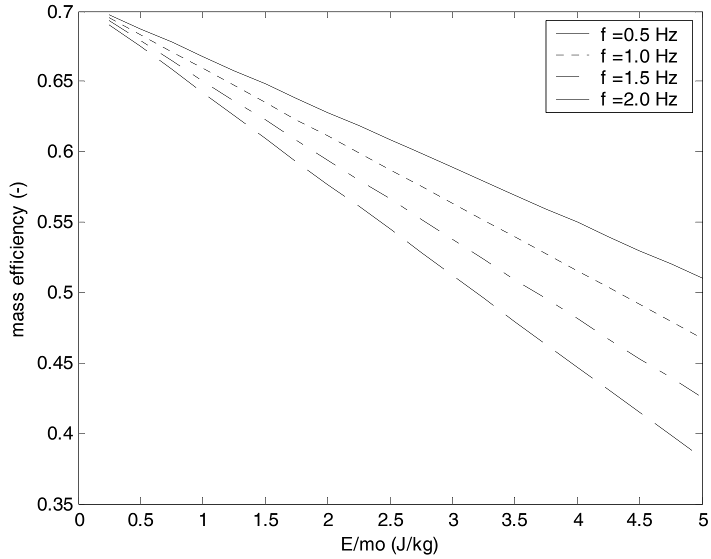


Figure 5 – Effects of E/m_o on mass efficiency for several values of f, with $r_e/r_i = 6$ and $k = 4.10^{-11}$ kg/A^{2}s.

4. CONCLUSIONS

A simplified theoretical model of coaxial or parallel pulsed plasma thrusters (PPTs) was developed, considering the effects of the non-ionized gases and of the energy losses in lines, capacitors, electrodes and plasma, based on an electric circuit analysis. Expressions for specific impulses, impulse bits, ablated mass and thrust efficiency were obtained in terms of the mass fraction of the ionized gases. An expression for the mass efficiency (payload ratio) of a spacecraft using pulsed plasma thrusters was derived for a given mission with known characteristic velocity. The theoretical results were in accordance with experimental fits obtained from several PPTs.

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5. RESPONSIBILITY NOTICE

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