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Plasma brake model for preliminary mission analysis

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ABSTRACT

Plasma brake is an innovative propellantless propulsion system concept that exploits the Coulomb collisions between a charged tether and the ions in the surrounding environment (typically, the ionosphere) to generate an electrostatic force orthogonal to the tether direction. Previous studies on the plasma brake effect have emphasized the existence of a number of different parameters necessary to obtain an accurate description of the propulsive acceleration from a physical viewpoint. The aim of this work is to discuss an analytical model capable of estimating, with the accuracy required by a preliminary mission analysis, the performance of a spacecraft equipped with a plasma brake in a (near-circular) low Earth orbit. The simplified mathematical model is first validated through numerical simulations, and is then used to evaluate the plasma brake performance in some typical mission scenarios, in order to quantify the influence of the system parameters on the mission performance index.

1. Introduction

The plasma brake [1,2] is an innovative technology capable of supplying a propulsive acceleration to a spacecraft on a low Earth orbit (LEO) without any propellant consumption, by exploiting the (electrostatic) Coulomb collisions between a long space tether and the charged particles in a plasma stream. In a typical configuration [2–4] a single charged tether deployed by a spacecraft, see Fig. 1, interacts with the ionized upper stages of Earth's atmosphere (ionosphere), and provides a decelerating thrust (Coulomb drag) orthogonal to the tether line.

The idea of plasma brake is a consequence of the Electric Solar Wind Sail (E-sail) propulsive concept, which can be traced back to 2004 [5]. This propulsion system consists of a spinning grid of tethers, stretched out by centrifugal force, which are kept at a high potential to exchange momentum with the solar wind ions and generate a small (but continuous) propulsive acceleration without any propellant consumption [6,7]. An E-sail-based spacecraft is a good candidate for some special heliocentric mission scenarios that would be difficult or impossible to achieve with a conventional propulsion system, including displaced non-Keplerian orbits [8,9], outer Solar System exploration [10], asteroids deflection [11], near-Earth asteroid flyby [12] or sample return mission [13].

The hypothesized structure of a plasma brake tether was originally based on the Hoytether [14] concept, in which the tether has some primary lines, with multiple interconnections made of smaller wires. This design provides a significant safety improvement against impacts with micrometeorids, thus increasing the system lifetime with respect to a single line tether. A more recent innovation in the field of space tether manufacturing is the Heytether [15], which is made of a single primary line with multiple secondary connections. Its structure guarantees a sufficient reliability, with a reduction of total mass and design complexity when compared to the Hoytether. In this context, a 1km-long Heytether has been obtained as one of the final outcomes of the European project EU FP7 [15–17], in view of future space tests of the E-sail technology.

A first validation test for a plasma brake (and E-sail) system was tried in 2013 by the Estonian satellite ESTCube-1 [18,19], but a failure occurred to the tether reel mechanism, probably due to the vibrational loads during the launch phase. A second technology demonstration-spacecraft, the Finnish satellite Aalto-1 [20], should be launched during 2017. The Aalto-1 is a 3U CubeSat equipped with a 100m-long electrostatically charged tether, stretched out with the centrifugal force generated by spinning the satellite [21]. A plasma brake in-situ experiment will be conducted, both for positive and negative polarity of the tether, to obtain experimental evidence of the propulsion system concept. Another plasma brake experiment is planned by the more advanced ESTCube-2 satellite [22], which will carry a 300m-long tether. The spacecraft systems are currently under development and the launch readiness tests should be completed within 2018.

Some preliminary numerical simulations have been conducted to investigate the potential performance of a plasma brake system [4,23], with encouraging results in a classical deorbiting mission scenario. In this context, simulations give decaying times of few years for near or

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Nomenclature

A	spacecraft frontal area, [m^2]
a	semimajor axis of the spacecraft osculating orbit, [km]
b_t	tether width, [cm]
C_D	drag coefficient
c_b	ballistic coefficient, [kg/m^2]
e	elementary charge, [C]
F	magnitude of plasma brake-induced drag, [mN]
f_i	auxiliary functions, see Eqs. (6)–(8)
g_0	standard gravity, [m/s^2]
h	spacecraft altitude, [km]
h_0	spacecraft initial altitude, [km]
K	plasma brake constant, see Eq. (1)
k_B	Boltzmann constant, [J/K]
L_t	tether length, [m]
m_0	spacecraft initial mass, [kg]
m_i	mean molecular mass of the incoming flow, [u]
m_L	payload mass, [kg]
m_{rm}	tether reel mechanism mass, [kg]
m_{sav}	fraction of mass saving
n_0	plasma bulk number density, [m^{-3}]
$R_\oplus$	Earth's radius, [km]
r_w	wire radius, [μm]
T	temperature of ions, [K]
V_r	modified voltage, see Eq. (2) [V]
V_t	tether voltage, [V]
v_0	plasma bulk relative velocity, [km/s]
ε_0	vacuum dielectric constant, [F/m]
η_r	packaging factor of the reeled tether
λ	payload ratio
ρ_r	density of the reel structure with respect to the internal volume, [kg/m^3]
ρ_w	density of the wires material, [kg/m^3]
σ_S	thruster structural coefficient, [kg/m]
σ_t	tether structural coefficient, [mg/m]

Subscripts

C	chemical or electric thruster
i	generic part of the spacecraft
PB	plasma brake
SC	spacecraft
t	tether
w	wire

Superscripts

$\sim$	reference value
$-$	mean value

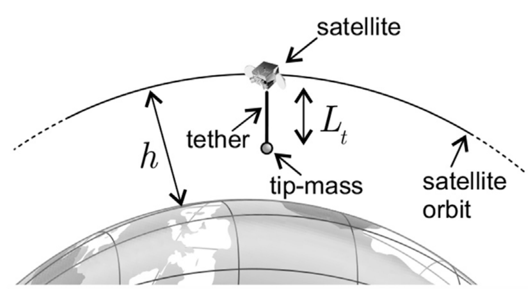

Fig. 1. Plasma brake conceptual scheme, adapted from Ref. [2].

mid-term tether lengths, and significant mass savings with respect to active deorbiting strategies with chemical or electric thrusters. For example, according to Ref. [4], a 5km-long tether produces a decelerating drag of 0.43mN at an altitude of 800km. This braking force is able to reduce the altitude of a 260kg spacecraft of about 100km in 1year. Based on previous technology developments [15–17], a tether length of 5km is estimated to be ready within the next 5 years.

The available mathematical models for plasma brake are however quite complex and depend on many different parameters, and a thorough analysis of its actual potentialities is not yet available. Therefore, the aim of this paper is to illustrate an analytical model capable of predicting the mission performance of a gravity gradient-stabilized spacecraft equipped with a plasma brake tether, with small computational costs compared to a simulative approach.

This paper is organized as follows. Section 2 presents an approximate mathematical model, based on some simplifying assumptions, which can be used to estimate the plasma brake-induced force by means of analytical formulas. The approximate model is then validated by simulation with a more general model in which the equations of motions are integrated numerically. Section 3 analyzes the plasma brake performance in a typical mission scenario as, for example, a deorbiting strategy. Finally, Section 4 contains some concluding remarks.

2. Approximate thrust model

Previous works [1,3,24] on plasma brake system analysis have shown that a negatively-charged tether is more convenient compared to a positively-charged one in terms of design simplicity, for a LEO mission scenario. In fact, in the positive polarity case, the plasma brake system requires a voltage source and an electron gun to maintain the necessary voltage by expelling the accumulated electrons. On the contrary, when the tether is negatively-charged, although the voltage source is still required, the ion gun (which is theoretically needed to keep the tether at the design voltage) is not essential for a proper system operation, since the spacecraft itself acts as an electron collector due to the high thermal mobility of electrons. Hence, only the negatively-charged tether case will be addressed in this paper. As the spacecraft is used as an electron collector, a part of the vehicle is electrically connected to the negatively biased tether. This arrangement poses additional constraints to the spacecraft system design as, for example, the fact that some components must be properly shielded with electric insulators.

Consider a spacecraft orbiting a LEO that at a certain time instant releases a single charged tether (with a tip mass) to generate a Coulomb drag, see Fig. 1. The spacecraft-tether-mass system is stabilized by gravity gradient, with the tether axis pointing towards the Earth's center-of-mass. According to Refs. [3,22], the thrust per unit length dl of a single negatively-charged tether can be expressed as

$$\frac{\mathrm{d}F}{\mathrm{d}l} = K\, m_i\, n_0\, v_0^2 \sqrt{\frac{\varepsilon_0\, V_r}{e\, n_0}}\, \exp\left(-\frac{m_i\, v_0^2}{2\, e\, V_r}\right) \tag{1}$$

where F is the magnitude of the plasma brake-induced drag force, m_i is the mean molecular mass of the incoming flow, n_0 and v_0 are the plasma bulk number density and the component of the plasma relative velocity (with respect to the tether) orthogonal to the tether axis, e is the elementary charge, ε_0 is the vacuum dielectric constant, and $K = 3.864$ is a dimensionless constant [3]. The term V_r in Eq. (1) is a sort of modified tether voltage [4] given by