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Electrostatic Plasma Brake for Deorbiting a Satellite

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Space debris in the form of abandoned satellites is a growing concern, especially at the heavily populated 600–1000 km altitude orbits. To prevent new space junk from forming, new satellites should be equipped with a deorbiting mechanism. The problem is especially tricky for the emerging class of very small satellites for which using a braking rocket as a deorbit mechanism may have a prohibitively high relative cost impact. We describe a novel type of deorbiting mechanism that is suitable for small satellites with a mass of up to a few hundred kilograms. The method is a plasma brake device based on coulomb drag interaction between the ionospheric plasma and a negatively charged thin tether. The method resembles the well-known electrodynamic tether deorbit mechanism, but the underlying physical mechanism is different and the new method has an order of magnitude smaller mass and power consumption. The new method uses the same physical principle (coulomb drag) as the recently invented electric solar wind sail propulsion method. Furthermore, the tether required by the plasma brake is so thin that, if accidentally cut, the loose fragments of it pose no threat to other spacecraft and will rapidly descend into the atmosphere. The electrostatic plasma brake could enable an extended use of small satellites by resolving their associated space debris problem.

Nomenclature

A	=	conducting area of spacecraft, m^2
e	=	electron charge, As
F	=	thrust force, N
I	=	tether current, A
I_{el}	=	thermal electron current gathered by spacecraft, A
k_B	=	Boltzmann constant, J/K
L	=	tether length, m
m_e	=	electron mass, kg
m_i	=	ionospheric mean ion mass, kg
n_0	=	ionospheric number density, m^{-3}
P_{dyn}	=	ionospheric dynamic ram pressure, Pa
r_w	=	tether wire radius, m
T_e	=	ionospheric electron temperature, K
V_0	=	tether potential, V
v_0	=	relative velocity of satellite with respect to the ionosphere, m/s
ε_0	=	vacuum permittivity, As/(Vm)

I. Introduction

SPACE debris in the form of abandoned satellites is a growing concern especially at the heavily populated 600–1000 km altitude orbits. To prevent new space junk from forming, new satellites should be equipped with a deorbiting mechanism. The problem is especially tricky for the emerging class of very small satellites (nanosatellites and "CubeSats") for which using a braking rocket as a deorbit mechanism may have a prohibitively high relative cost impact. In such satellites, which are invariably launched as piggyback payloads, any use of pyrotechnic devices such as rockets is usually forbidden for launcher safety reasons. Propellantless propulsion techniques for deorbiting the satellite are therefore called for. Such a technique is provided by the electrodynamic tether (see, for example, [1]), but the mass of such a system is not very small. The purpose of this paper is to present an alternative tether-based plasma brake concept that is more lightweight than the electrodynamic tether.

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II. Negatively Charged Plasma Brake

A positively charged tether can collect momentum from a plasma stream such as the solar wind [2]. A negatively charged tether can also perform a similar action, although to keep it negatively charged generally requires removing the positive charge from the spacecraft by an ion gun [3]. The treatment in [3] is for a tether or wire immersed in the solar wind, but the resulting thrust formula is expected to be valid in the ionosphere as well if and when the tether or wire is thin compared to the plasma Debye length. If operating in the dense ionospheric plasma, however, the ion gun may not be necessary because the conducting body of the spacecraft can act as a collector of thermal electron current from the plasma. This leads to a particularly simple design in which, in addition to the conducting tether, only a voltage source is required that maintains a potential difference between the spacecraft body and the tether. The tether brakes the spacecraft in its motion through the nearly stationary ionospheric plasma by the approximate force [3]

$$F = 1.72 P_{dyn} L [\varepsilon_0 V_0/(e n_0)]^{1/2} \exp[-m_i v_0^2/(2eV_0)] \quad (1)$$

where $P_{dyn} = m_i n_0 v_0^2$ is the flow dynamic pressure, L is the tether length, n_0 the ionospheric electron density, m_i is the ion mass, v_0 is the spacecraft speed relative to the plasma, V_0 is the absolute value of the tether voltage, and the other symbols have their standard meanings. The tether collects an ion current I from the plasma, to

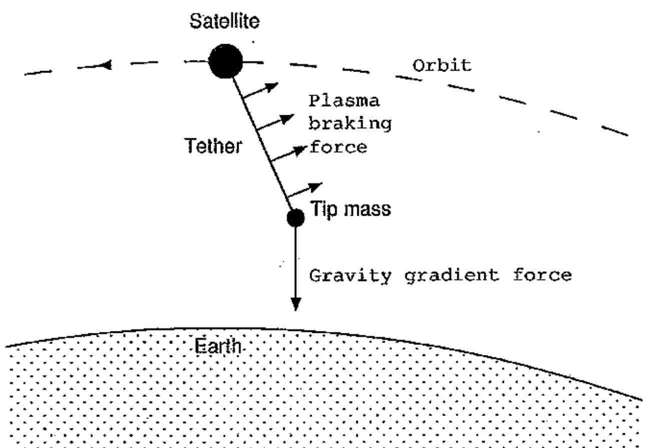

Fig. 1 Plasma brake device conceptual drawing.

Table 1 Characteristics of different size plasma brake device; last column is the conducting satellite area needed to gather sufficient electron current

	Voltage	Tether mass	Tip mass	Length	Power	Sat mass	Sat area
Size1	150 V	16 g	30 g	700 m	35 mW	3 kg	0.12 m^2
Size2	500 V	28 g	56 g	1.2 km	360 mW	10 kg	0.38 m^2
Size3	1500 V	160 g	97 g	7 km	11 W	100 kg	3.8 m^2

Table 2 Deorbit times in days from different altitude polar orbits

	500 km	600 km	700 km	800 km	900 km
Size1	45	93	180	300	500
Size2	56	110	200	330	540
Size3	62	120	200	340	540

which the orbital motion limited theory [4] is, as expected, a rather good approximation in this case:

$$I = 4.3en_0L(2eV_0/m_i)^{1/2}2r_w \tag{2}$$

where r_w is the radius of the metal wire the tether has been made of, and the factor 4.3 comes from our assumption that the tether is of the fourfold Hoytether type [5] to withstand the collisions of micrometeoroids and small space debris objects without breaking. Here V_0 is the local potential difference between the tether and the plasma, which we assume to be constant here for simplicity, although in general it would vary along the tether because of ohmic losses in the tether and because of the natural satellite motion-induced electric field. The photoelectron current from the tether due to solar UV is much smaller than the ion current in the ionospheric case, and the electron field emission and secondary emission are not significant because we consider moderate voltages up to 1500 V only. To maintain the negative charging of the tether requires power IV_0 from the voltage source, which keeps the tether negatively charged at voltage $-|V_0|$ with respect to the spacecraft. Under the assumption that the plasma Debye length is much smaller than the spacecraft size, the spacecraft remains only slightly positive and collects the thermal electron current:

$$I_{\mathrm{el}} = en_0[k_BT_e/(2\pi m_e)]^{1/2}A \tag{3}$$

where T_e is the electron temperature (typically 1000–2000 K), and A is the conducting area of the spacecraft. As long as $I_{\mathrm{el}} < I$, the voltage source is enough to keep the tether at potential $-|V_0|$ and no ion gun is needed.

Figure 1 shows the overall concept of the plasma brake device, whose purpose is to deorbit a small-to-moderate size satellite at the end of its operational life to prevent it from forming new space junk. The primary application area is deorbiting from an altitude of up to ~900 km. The plasma brake consists of the gravity-stabilized tether, the tip ballast mass, and the voltage source. Table 1 summarizes the basic properties of plasma brakes of three sizes (1, 2, and 3) intended for 3, 10, and 100 kg satellites, respectively. For simplicity it was assumed that all tethers are made of 50-μm-diam aluminum wire. The tip mass was estimated by requiring that the tether does not incline to more than a 30 deg angle with respect to the vertical direction. In all cases the combined tether plus tip masses are a small fraction of the satellite mass and the power required by the voltage source is easy to obtain from the solar panels of the corresponding satellite.

The total mass of the plasma brake must also include, besides the tether and the tip mass as estimated in Table 1, the masses of the reel, launch lock, dc power source, and controller. We propose to obtain rough estimates for these additional mass items as follows. The ESTCube-1 is an Estonian 1 kg nanosatellite that is currently in Phase B study and whose planned launch is in 2012.† The purpose of ESTCube-1 is to measure the electrostatic plasma braking effects at low Earth orbit in both positive and negative tether modes described by [6] and Eq. (1), respectively, by deploying a 10 m conducting tether and charging it to ±200 V. The total payload mass of ESTCube-1 is about 100 g, which includes the tether; the 6-cm-diam reel; a ±200 V, 1 W power source; and a 1 W, 200 V electron gun for the positive mode. Based on the ongoing engineering development of the ESTCube-1 payload hardware, we estimate that for the Size1 and Size2 plasma brakes (Table 1) the extra mass due to the reel, launch lock, power source, and controller would be in the range of 50–100 g. For the Size3 device the additional mass is harder to estimate at this stage, but would likely fall in the 100–400 g range.

Table 2 summarizes the deorbiting performance. The required electron density and mean ion mass altitude profiles were obtained from the IRI-2007 ionosphere model by averaging over a solar cycle. The deorbit times increase as a function of altitude because the plasma density and the mean ion mass decrease. The satellite masses in Table 1 are indicative only. For a given device size and starting orbital altitude, the deorbit time scales nearly linearly with the satellite mass. Tables 1 and 2 were calculated assuming a polar (90 deg inclination) orbit. For lower inclinations, the deorbit times would be shorter and the power consumptions higher due to the higher ionospheric plasma density in the day-side low-latitude ionosphere. The computed cases are exemplary and, for a specific mission (given the orbit inclination and altitude, solar cycle phase, available power, and contact area), one could tailor the design by optimizing the various parameters.

The electrostatic plasma brake does not require a current to flow in the tether, although a small current develops there as a by-product of charging. For this reason, for small satellites the electrostatic brake has a much smaller mass and power consumption than the electrodynamic tether of corresponding performance. For satellite masses larger than ~100 kg, the electrostatic plasma brake goes gradually to the electrodynamic tether brake because the Lorentz force acting on the tether is proportional to L^2 and the electrostatic force is proportional to the tether length L. By deploying the electrostatic tether both upward and downward from the satellite, the applicable satellite masses could be doubled. If the satellite can be made to spin, one could deploy many tethers to brake a larger satellite still.

III. Conclusions

We have presented and briefly analyzed a novel type of tether-based deorbiting concept that works similar to the well-known electrodynamic tether braking method, but that requires a significantly less massive tether. The presented system could find use especially in small satellites for which pyrotechnics-based braking methods are generally not allowed.

Acknowledgments

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