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SATELLITE DE-ORBITING BY MEANS OF ELECTRODYNAMIC TETHERS PART II: SYSTEM CONFIGURATION AND PERFORMANCE†

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Abstract—This paper aims to assess the efficiency of a de-orbiting system based upon conductive tethers under realistic assumptions for its interaction with the ionospheric environment. We analyze the configuration made up of a 2.5–10 km tether, a passive inflatable collector of 2.5–10 m radius at the positive termination and a hollow cathode at the negative one. Voltages and current in the system are computed from the equation of the equivalent circuit, making use of the IRI-90 ionospheric model. The resulting electromagnetic drag forces have been used to compute the evolution of the orbital elements (especially the semi-major axis) and the re-entry times. Our results indicate that a typical satellite of 500 kg mass at 1300 km altitude can de-orbit in 20–100 days, for a broad range of orbital inclinations and solar activity. The validity of the concept is further strengthened by the comparison with alternative propulsion systems. © 2002 Published by Elsevier Science Ltd.

1. INTRODUCTION

Propulsion and orbital reboost have been considered among the most interesting applications of tethers in space since early studies in the field. This utilization was indeed proposed and analyzed by several authors well before the flight, in July 1992, of the first Tethered Satellite System (TSS-1) by NASA and the Italian Space Agency (ASI)[1,2]. Tether systems provide low thrusts and are therefore particularly effective in counteracting the action of atmospheric drag in a LEO environment. A large structure like the International Space Station with a large area-to-mass ratio could be kept at its operational altitude by a tethered system, greatly reducing propellant consumption.

Recently [3], it has been suggested to use tethers as a way of de-orbiting satellites at the end of their operational life. The estimates performed so far look very promising, as a tether system compares very favorably to conventional chemical propulsion in terms of mass. As a rule of thumb, the mass of the propellant required to de-orbit a LEO satellite ($\Delta v \cong 400$ m/s) is about one-fifth of the total mass, while the same result may be obtained using a simple tether system weighing only 30–50 kg. If this concept will prove to be simple enough and robust from a technological point of view, tethers could become a powerful means of reducing the accumulation and proliferation of man-made space debris. Probably the most important use is related to orbiting upper stages and satellite constellations [4], where the large number of objects and the long lifetime before re-entry by atmospheric drag are reasons of particular concern. The original work by Forward et al. [3] points out how effective tethers can be in de-orbiting LEO satellites. For most constellations the authors obtain a re-entry time of a few weeks and as small as 11 *days* in the case of Orbcomm. Tethers are less efficient for polar

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satellites, whose velocity is nearly parallel to the earth's magnetic field.

The re-entry time or, more appropriately, the product between the re-entry time and the collision cross-section (area–time product) is indeed the critical figure for assessing the effectiveness of tethers as a tool to reduce the threat posed by space debris. This time is inversely proportional to the electrodynamic drag force and, in turn, to the current flowing along the wire. Apart from constraints related to the dynamical stability of the tether, the design of the system must aim at large currents and therefore at a good electrical contact between the conductor and the surrounding charged particle environment. It is precisely the contact impedance of the wire with the ionospheric plasma, related to the onset of sheaths around the high voltage terminations, which may drastically limit the current in the system. In Ref. [3] parasitic impedances were arbitrarily set to a constant value equal to half the ohmic resistance of the tether. In this paper we wish to compute the de-orbiting times by using the appropriate current–voltage characteristics of the terminations and a realistic model of the plasma environment. We will refer to the system configuration described in a companion paper [5] (from now on indicated as Paper I), in which a large passive collector of diameter 5–20 m is deployed from the mother satellite using a tether of 2.5–10 km length. An electron emitter ensures the electrical contact with the ionosphere at the other termination. Our results show that, although the re-entry times turn out to be appreciably larger than in Ref. [3], they still fall in a very interesting range, thus confirming the effectiveness of tether systems in de-orbiting applications.

2. THE TETHER SYSTEM

Propulsion and drag by means of electrodynamic tethers are generated by the interaction of the conductive wire with the Earth's magnetic field. An observer tied to the orbiting tether measures an electric field of magnitude:

$$\mathbf{E} = \mathbf{v} \times \mathbf{B}, \tag{1}$$

where $\mathbf{v}$ is the velocity of the conductor across the magnetic field lines and $\mathbf{B}$ is the local magnetic field, both assumed to be constant along the tether. If no current flows in the system, the potential difference between the two ends is

$$\Phi_{\text{ind}} = \int_L (\mathbf{v} \times \mathbf{B}) \cdot d\mathbf{l} \tag{2}$$

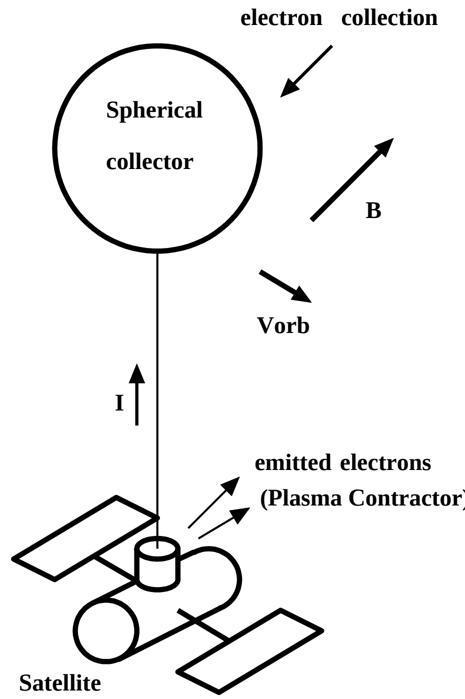

Fig. 1. Electrodynamic tether system configuration.

with the integration extended over the tether (of length L). The system acts therefore as a voltage generator capable of providing a potential difference proportional, for a given geometrical configuration, to the velocity, magnetic field and tether length.

A sketch of the proposed electrodynamic tether system is shown in Fig. 1.

If the tether is put in some way in electrical contact with the surrounding ionized medium, charges are collected and an electrical current flows in the system. As already pointed out in the introduction, the effectiveness of tethers for propulsion and de-orbiting depends crucially on the magnitude I of this current. The thrust or drag is indeed generated by the interaction of the current with the local magnetic field according to the Lorentz law:

$$\mathbf{F} = I \int_L d\mathbf{l} \times \mathbf{B}, \tag{3}$$

where the line element $d\mathbf{l}$ along the tether is taken in the direction of the current flow. This force is proportional to the current and the tether length. Several factors limit, in practice, the increasing of the system size, as discussed in [5], the main factors being the increase in weight and complexity of the deploying system and the risk of collisions by debris. Moreover, long tethers develop large voltages across their ends and therefore are more exposed to the possibility of arching and discharges.

For propulsive applications it is essential to create a good electrical contact between the system and the ionosphere by lowering all contact impedances of the plasma sheaths surrounding the high voltage terminations (see eqns (3) and (4) of [5]). The ultimate limit to the current is provided by the impedance of the tether itself, which therefore should also be as low as possible. For a given conductive material, the tether resistance is determined by the length and the mass. For a typical aluminum tether, resistances per unit length of the order of 40–80 Ω/km are achievable with masses of 1–2 kg/km. Our reference configuration consists of an inflatable balloon of 10 m diameter, a 5 km long, 0.4 mm diameter aluminum tether (280 Ω electrical resistance), attached to a 500 kg satellite and equipped with a hollow cathode electron emitter. We will show that for such a configuration the parasitic impedances result in de-orbiting times between 20 and 100 days in a broad range of orbital inclinations and ionospheric conditions.

In this work we do not attempt to optimize the system, but rather aim to demonstrate that the simple configuration proposed above, involving a small fraction of the total mass of the satellite and an inflatable collector of reasonable dimensions, provides an effective mean of de-orbiting satellites. Of course the optimization of the tether length and the collector size depends crucially on the mass of the satellite, the orbit semi-major axis and inclination, and is dictated in the end by the desired re-entry time. The critical system parameters are the tether length (determining the overall available potential drop) and the balloon radius (determining, for given plasma conditions, the contact impedance of the upper termination). In this paper we will examine the effects on de-orbiting times produced by a variation of the system parameters.

The technologies required by the de-orbiting system considered in this paper have already been tested in space and only minor developments are needed. The main issues are (1) the survivability of the tether in the space environment and (2) the deployment of the inflatable after a long storage in orbit.

In spite of its conceptual simplicity, the fabrication of a durable space tether requires great care. The main requirements for the application envisaged here are a good survivability to impacts with micrometeoroids and debris (of size comparable or greater than the tether diameter) and robustness against the onset of arching. Having in mind commercial applications, the risks of tether breaking, coming mainly from large debris (see Paper I [5]), must be kept as low as possible. In order to minimize the probability that the tether itself becomes a debris, the obvious option is to minimize its length. Short tethers would have the additional advantage of producing lower induced voltages, which makes the onset of discharge phenomena less likely. In Paper I it has been shown that a ribbon tether of 5 mm width has a mean time between failure of about 1 year, under probably pessimistic assumptions. However, the TiPS tether [6] (4 km long, 1 mm radius) did not experience any failure since its launch in June 1996, in spite of a predicted lifetime of about 60 days under the same assumptions for the space debris environment. Long duration tethers based on multi-line structures have been proposed and studied [7]. The expected lifetimes of such tethers (of several tens of years) give an excellent margin to ensure a high reliability of the system.

The technology of inflatable structures dates back to the launches of the Echo balloons in the 1960s. Recently, remarkable advances have been made, especially in the fabrication of semi-rigid, accurately shaped structures, such as high gain antennas. In 1996 a 14 m dish, inflatable antenna, connected to a 28 m tripod has been deployed during the STS-77 flight [8]. The total mass of the system, including the control electronics, was only 60 kg. Recent applications of inflatable structures span from the large shielding panels of next generation space telescope (NGST) to the spherical wheels of Mars rovers under development at JPL. It is projected that a spherical inflatable balloon of 10 m diameter, as foreseen in this paper, will weigh about 30 kg. Its fabrication is made easier and cheaper by the very loose shape requirements. The inflatable balloon must have enough surface conductivity to sustain the flow of relatively large currents (about 3 A) and withstand an additional thermal input of about 1 kW, associated to the power dissipation in the plasma sheath. However, the main technological issue is the fabrication of an inflatable structure capable to be deployed correctly after being stored many years in orbit. This issue needs to be addressed carefully in further studies.

3. TETHER SYSTEM PERFORMANCE

The secular changes of the orbital elements are calculated by averaging the Lagrange equation along the orbit of the satellite, a computation which requires the knowledge of the in-plane and out-of-plane components of the acceleration. The most important quantity to keep track of is, of course, the semi-major axis, but also important

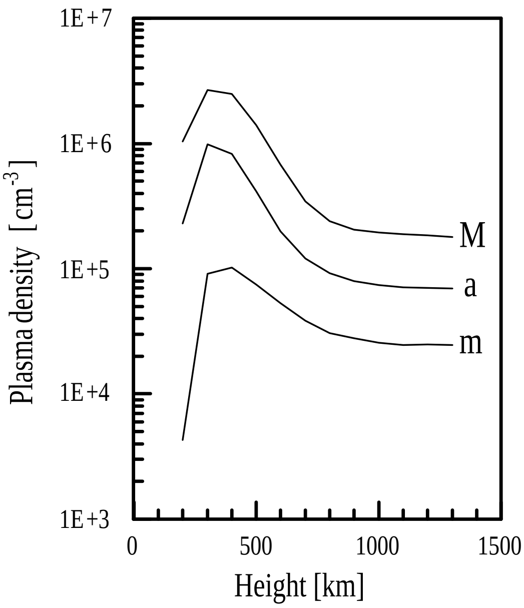

Fig. 2. Profile of the plasma density estimated by IRI90 model between 200 and 1300 km altitude, for mean sunspot ($F10.7$ cm = 150) and 55° orbit inclination. The three lines refer to the maximum (M), average (a) and minimum (m) densities determined by the model along the orbit.

are the variations of inclination and eccentricity. Changes in the inclination are especially relevant for nearly polar orbits, where even moderate variations produce significant change in the (small) induced voltages and thus in the re-entry times. The eccentricity is especially relevant to the stability and control of the system, as it is well known that tethers in highly eccentric orbits are unstable.

From eqn (3), the current is the critical quantity determining the Lorentz acceleration. Its value depends not only on the induced potential, but also on the local ionospheric environment, especially the plasma density. This quantity, which varies strongly with altitude, local time and solar activity (see Fig. 2), influences crucially the resistivity of the sheath around the upper termination. It is precisely the complex dependence on the plasma environment that makes impossible to compute analytically the acceleration. Another reason for a numerical treatment is the non-linear behavior of the current–voltage characteristics of the sheaths. In general, the current I must simultaneously satisfy the equation of the equivalent circuit and the current–voltage characteristics of the sheaths:

$$\begin{aligned} &\Phi_{\mathrm{ind}} - R_{\mathrm{w}} I = V_{\mathrm{sh}+} + V_{\mathrm{sh}-}, \\ &I = I(V_{\mathrm{sh}+}), \\ &I = I(V_{\mathrm{sh}-}). \end{aligned} \tag{4}$$

Assuming a rigid tether aligned with the local vertical (a good assumption a posteriori), the induced voltage Φ_{ind}, determined by eqn (4), depends only on the orbital elements and anomaly. In the computations we have assumed circular orbits and an eccentric dipole model for the Earth magnetic field (accurate to about 2%). Electric fields up to 240 V/km are obtained along equatorial orbits (see Fig. 3b).

The potential drops $V_{\mathrm{sh}+}$ and $V_{\mathrm{sh}-}$ at the positive and negative terminations are determined by the characteristic curves of each element (balloon and hollow cathode, see Paper I, Figs. 3 and 4). Both characteristics depend on the density and temperature of the plasma. This dependence is particularly strong for the positive (upper) termination, where electrons are collected from the ionosphere, while the very dense plasma produced by the hollow cathode makes the characteristic curve much less sensitive to the local ionospheric environment. The I–V characteristic of the hollow cathode can be adequately approximated by an exponential function, which fits the experimental results obtained in plasma chambers [9,10]. The curves shown in Fig. 4 of Paper I were measured by polarizing the hollow cathode with respect to a preexistent ambient plasma of ionospheric parameters. In particular, here we consider only the part of the characteristics relevant to the electron emission regime, that is the one obtained when the hollow cathode is negatively biased. The fit procedure does not require a great accuracy as long as the characteristic is steep, because the device acts essentially as a plasma contactor clamping the lower termination to a potential close to the plasma potential. The function $I = I(V_{\mathrm{sh}+})$ for the balloon is less accurately known. The available measurements obtained during the TSS-1R flight indicated good agreement with isotropic charge collection models [11] and have shown larger than expected currents. The main unknown is the dependence of the characteristics on the radius of the collector. However, there is good theoretical ground to believe that isotropic models are also accurate for dimensions much larger than the TSS-1R (1.6 m diameter), provided that the potential is sufficiently high. As the characteristics entailed by these models are universal in terms of dimensionless combinations of parameters (including the radius of the body), the extrapolation is straightforward.

The nonlinear set of eqn (4) is solved numerically for the current at each point of the orbit, using the plasma densities and temperatures predicted by the IRI-90 ionospheric model [12]. Currents and induced voltages as a function of altitude

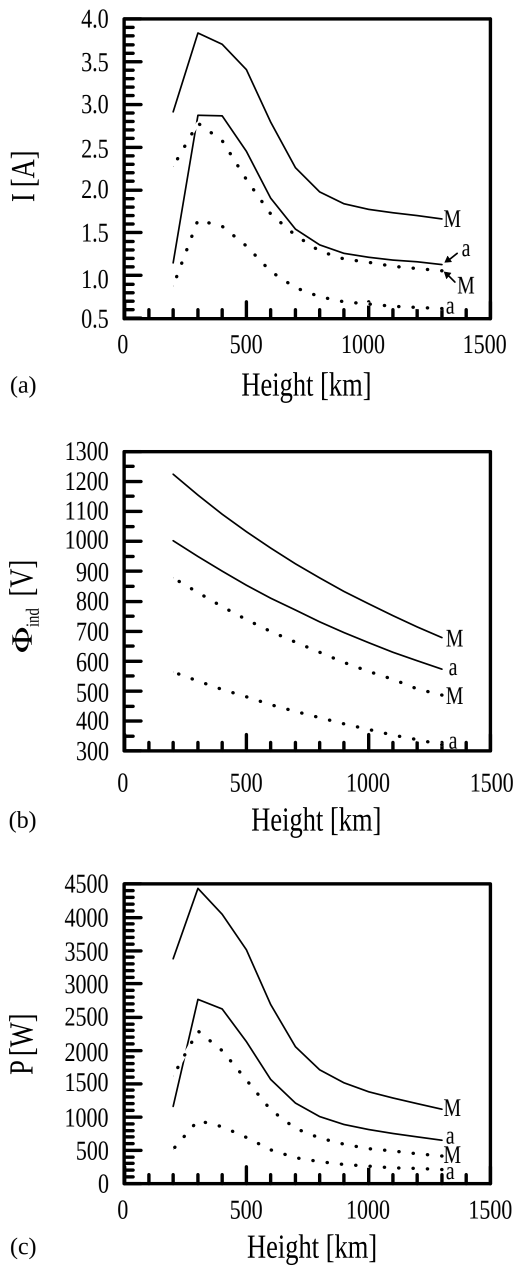

Fig. 3. Current I, induced voltage $\Phi_{\rm ind}$ and total dissipated power P, computed at mean sunspot ($F10.7\ {\rm cm} = 150$) for a tether system 5 km long equipped with a balloon 10 m diameter. Solid lines refer to $i = 0^\circ$ and dashed lines to $i = 55^\circ$. The lines labeled a and M refer to average and maximum values, respectively.

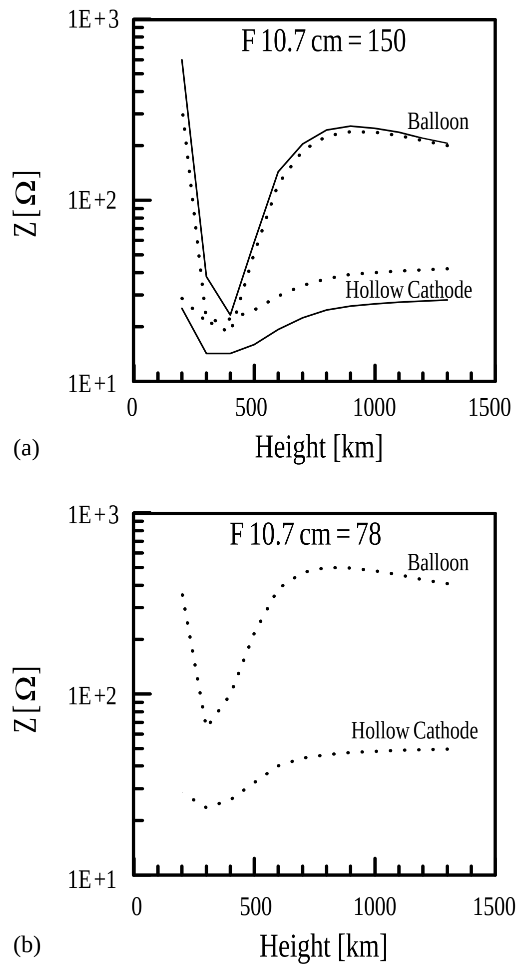

Fig. 4. Contact impedances of the hollow cathode and the balloon (10 m diameter) computed for average (panel (a)) and minimum (panel (b)) sunspot. Solid and dashed lines refer to 0° and 55° orbit inclinations, respectively.

are shown in Figs. 3a and 3b, for a 5 km length tether and a 5 m radius balloon, under average solar activity conditions ($F10.7\ {\rm cm} = 150$). The largest power dissipation (nearly 4.5 kW) is attained in daytime for equatorial orbits near the peak of the $F2$ ionospheric layer (about 300 km altitude, see Fig. 3c). Under such conditions a large fraction of the total power is dissipated along the wire, as the impedance of the sheaths is very low due to the large plasma density.

As expected, the current flow decreases significantly at higher altitude, simply because the resistivity of the system is larger. A breakdown of each contribution to the total impedance (see Figs. 4a and 4b) shows indeed that the tether resistance (280 Ω in our reference configuration) is, for average solar activity, comparable to the impedance of the upper termination. As expected, the contribution from the hollow cathode is negligible. At low solar activity levels the resistivity associated with the balloon strongly limits the current over a broad range of altitudes. It is worth noticing that the sheath impedance reaches a maximum at about 700–800 km. The decreasing trend of the sheath impedance observed at the higher altitudes is due

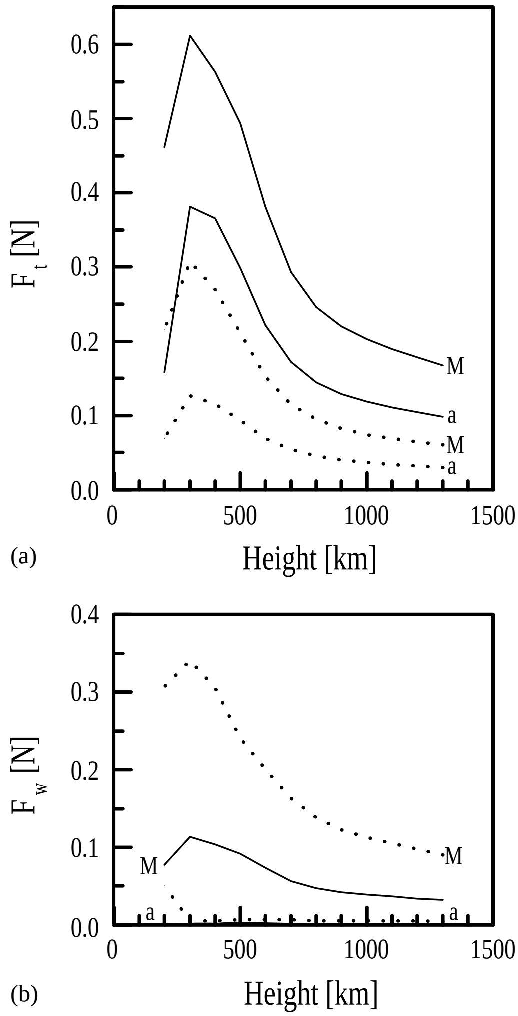

Fig. 5. In-plane (panel (a)) and out-of-plane (panel (b)) forces computed at mean sunspot ($F10.7\ \text{cm} = 150$) for a tether system $L = 5$ km and balloon $D = 10$ m. Solid lines refer to $i = 0^\circ$ and dashed lines to $i = 55^\circ$. The lines labeled a and M refer to average and maximum values, respectively.

to a combination of two effects: the increase of the electron temperature (while the density stays approximately constant) and the decrease of the dipolar magnetic field.

The knowledge of the current, combined with the assumption of a vertically aligned tether, allows the computation of the in-plane and out-of-plane components of the force given by eqn (3) (see Figs. 5a and 5b). For the reference configuration considered above ($L = 5$ km, $D = 10$ m, $R_w = 280\ \Omega$, $i = 55^\circ$, $F10.7\ \text{cm} = 150$), along-track forces F_t up to 0.3 N are predicted at altitudes of about 300 km. The average values along the orbit are in the range 0.03–0.13 N, depending on the altitude. For high inclination orbits the out-of-plane component F_w has a comparable value. Only for equatorial orbits, when the magnetic field is orthogonal to the orbital plane, the along-track component becomes dominant. In this case the magnitude of the force increases by about a factor of two.

The effect of the electrodynamic torque on the stability of the system is negligible. For the configuration considered above, the gravity gradient force is equal to the largest electrodynamic force experienced by the system in equatorial orbit. Under such conditions the angle between the tether and the local vertical is determined by the equilibrium between the electrodynamic and the gravity gradient torques acting upon the tether (assumed to be rigid). Due to the disparity between the mass of the satellite and that of the balloon (500 kg versus 30–50 kg), the center of mass of the system is very close to the lower termination (satellite) and the maximum deviation angle is about 20°. For most of the time (always for high inclination orbits) this angle is much smaller, thus confirming that the assumption of a vertically aligned tether is a good first approximation.

The variations of the orbital elements can be conveniently computed using the Lagrange planetary equations. We have traced the evolution of the semi-major axis a, inclination i and eccentricity e, under the assumption (verified a posteriori) that the variation of i and e are small during the whole orbital decay. Indeed, indicating with T_r the re-entry time, n the mean motion and a the semi-major axis, the change of inclination, of order $T_r F_w / m_{tot} n a$, is limited to a few degrees even for medium inclination orbits, where the effect is maximum. The same remark applies to the orbital eccentricity, which remains always small. It is worth noticing that the drag force, being larger at perigee due to the larger magnetic field and ionospheric density, effectively circularizes the orbit. A significant build-up of the eccentricity from an initial circular orbit is therefore impossible.

The secular changes of the semi-major axis are obtained by integrating the equation, valid for $e = 0$,

$$\dot{a} = 2\langle F_t \rangle / m_{tot} n \tag{5}$$

where the brackets indicate average along the orbit. The re-entry time may then be computed using eqn (8) of Paper I [5]:

$$[\Delta t]_{a1}^{a2} = \int_{a1}^{a2} \frac{n m_{tot}}{2\langle F_t \rangle}\,\mathrm{d}a = \int_{a1}^{a2} \frac{G M m_{tot}}{2a^2 \Phi_{ind} I}\,\mathrm{d}a. \tag{6}$$

The profile of the orbital decay for two inclinations ($i = 0^\circ$ and 55°) and maximum ($F10.7\ \text{cm} = 200$), average ($F10.7\ \text{cm} = 150$) and

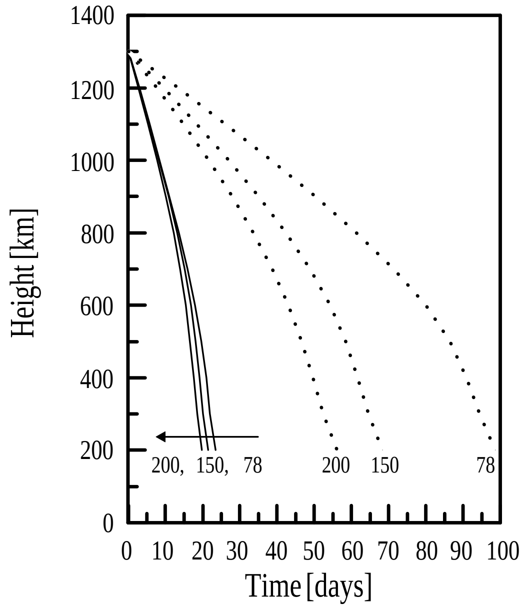

Fig. 6. Orbital decay for $i = 0^\circ$ (solid line) and 55° (dashed line) at maximum, average and minimum sunspot (see text), for a tether system $L = 5$ km and balloon $D = 10$ m.

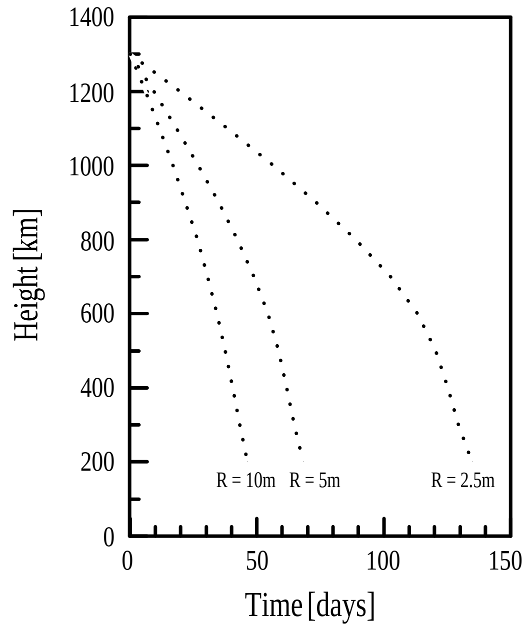

Fig. 7. Orbital decay for $i = 55^\circ$ at mean sunspot ($F10.7$ cm $= 150$) vs. balloon radius (tether $L = 5$ km).

minimum ($F10.7$ cm $= 78$) solar activity level is plotted in Fig. 6. We have assumed an initial semi-major axis of 7700 km, while the other system parameters are the same as in the reference configuration ($L = 5$ km, $D = 10$ m, $R_w = 280\ \Omega$, $m_{sat} = 500$ kg). Even under the less favorable conditions ($i = 55^\circ$ and $F10.7$ cm $= 78$) the re-entry time is about 100 days, a figure which we consider still very interesting. As expected, the rate of the decay increases below 500 km, where the lower impedance of the system produces larger drag forces.

Figure 6 points out the importance of the orbital inclination on the re-entry time. The maximum efficiency is obtained for equatorial orbits, due to a combination of larger induced voltages and ionospheric densities. For nearly polar orbits the component of the electric field along the tether is small and the system becomes very inefficient. It is worth pointing out, however, that the Earth rotation always produces a non-zero relative velocity between the tether and the magnetic field lines. Therefore, a small induced voltage ($\omega_{\text{Earth}} a L B \approx 75$ V) is always present across the two terminations, at least near the equator. At high inclinations also the effect of the solar activity becomes quite significant, as the main contribution to the total impedance comes from the plasma sheath around the balloon.

The re-entry time is also affected by a variation of the other system parameters, notably the tether length and the balloon radius. What matters, in the end, is always the capability of the system, under different configurations, to drive large currents along the tether. It is therefore not surprising that smaller collectors significantly increase the de-orbiting time (Fig. 7). Under average solar activity conditions, a 2.5 m radius balloon develops a sheath with a contact impedance much larger than the tether resistance, at least for altitudes above 600 km, leading to re-entry times a factor of two larger than for the reference configuration. Increasing the size of the balloon to 10 m has a less significant effect. Again, this is due the fact that now the dominant contribution to the total impedance comes from the wire, while the sheath impedance becomes negligible. Figure Fig. 7 shows also that the vertical velocity ($\dot{a}$) is affected by the size of the balloon only at altitudes above 500 km, where the sheath impedance plays the dominant role. Below 500 km the ionospheric plasma is so dense that the resistivity of the sheath decreases to very low values.

Another important system parameter to consider is the tether length. Its value determines the induced voltage, through eqn (2), and therefore the current, through eqn (4). Of course the length determines also the ohmic resistance of the tether, but here we wish to assess only how the re-entry time is affected by the available potential drop. For a given mass and material, the tether resistance is

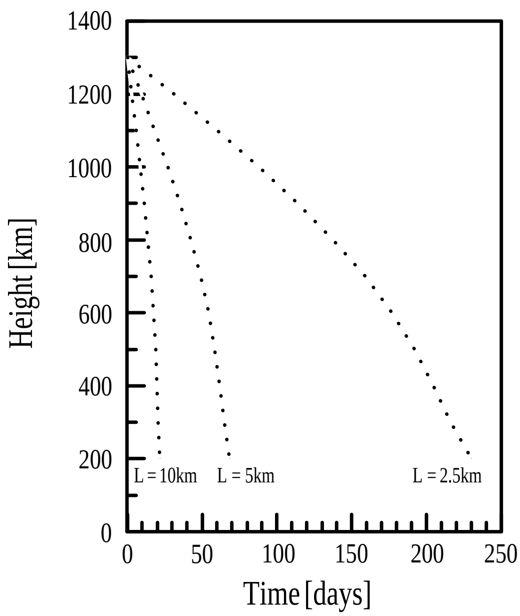

Fig. 8. Orbital decay for $i = 55^\circ$ at mean sunspot ($F10.7\ \text{cm} = 150$) vs. tether length (balloon $R = 5$ m).

proportional to L^2, quite a strong dependence. The mass of the tether has the same dependence on the length if one requires a constant electrical resistance, indicating that the optimization of the parameters is crucial. Figure 8 shows how the effectiveness of the system changes as a function of L, while keeping the tether resistance to its reference value of 280 Ω. Short tethers imply significantly longer decay times, due to the combined effect of smaller currents and induced voltages (see eqn (6)). Although the performance of long tethers are attractive, the price to pay in terms of mass, risk of arching and debris impact could be too high for reliable operations.

4. COMPARISON WITH ALTERNATIVE PROPULSION SYSTEMS

A comparison between tethers and other types of propulsion should address two issues:

1. performance (mainly de-orbiting time), and
2. penalties (mainly weight) associated with the propulsion device.

From all the estimates done so far, a reasonably compact tether system will weigh about 30–50 kg and (with a thrust up to about 0.5 N) de-orbit a 500 kg "debris" (dead satellite or third stage) in 1–2 months. Alternatives to this novel thruster system are conventional chemical and electric thrusters. These conventional thrusters must be able to reliably operate for mission times much longer than the usual applications they were designed for.

Chemical thrusters must remain in orbit for many years ready to fire. This restricts the choice basically to hydrazine devices: their monopropellant is space-storable, decomposes easily on the Shell X-405 catalyst, and are space-proven. The main performance drawback is the low specific impulse, I_{sp}, of order 200–300 s for continuous firing and 110 s for bit impulse firing. For the same thrust level (≈ 0.2 N), the mass of hydrazine required would be close to 200 kg. The thrust of typical hydrazine motors, however, is much higher, suggesting a kick-type de-orbiting maneuver and a total impulse appropriate to the negative Δv needed. Even so, because I_{sp} is so low, the mass ratio needed still turns out to be high, and the final figure for the propellant mass is still in the hundreds of kg. To this, one must add the weight of the thruster itself and of its ancillary subsystems (electrical power to preheat the catalyst bed, and so on). This by itself can add up to 10 kg. Besides, for third stages de-orbiting, it appears more difficult to engineer a chemical thruster than a tether.

Electric thrusters possess, in principle, much higher I_{sp}, with thrust levels that are comparable to those of a tether or, at most, a factor 10 higher (for MPD devices). Resistojets can be ruled out as competitors on the same ground of hydrazine thrusters, since the I_{sp} is roughly of the order of 200–300 s.

Arcjets can yield $I_{sp} = 500$ s, with a corresponding reduction of propellant mass. The propellant can be hydrazine, and then the system works as a combined chemical-arc system (augmented arcojets). The power conditioning unit is relatively light: for instance, a 0.2 N arcjet thruster weighs of the order of 6 kg. Most of the mass of an arc de-orbiter consists then of propellant. The propellant weight would still amount to 100–200 kg, with a de-orbiting time comparable, or even longer, than using a tether de-orbiter.

Ion thrusters, especially the latest generation already being installed on commercial satellites (e.g., XIPS by Hughes Aircraft Company), are capable of much higher I_{sp}, reaching up (potentially) to 3000–5000 s. In practice, 2600 s are standard on the Hughes XIPS. Therefore, the propellant mass needed for a typical de-orbiting device would be substantially reduced with respect to other propulsion systems: a realistic estimate is of order of 100 kg. The thrust available from ion thrusters, however, is of order 0.02–0.08 N, which would stretch the de-orbiting time to many months rather than 1–2 months. This would pose problems; higher thrust ion thrusters would need to be developed. The mass of an ion thruster is mainly affected by the high voltage and power required, and scales more than linearly with thrust. An estimate of the mass of a 0.6 N thruster yields about 40 kg. To this the mass of the propellant must be added.

MPD thrusters flown on satellites can produce much higher thrust than other electrical devices, i.e. 10–20 N. Their specific impulse is in the 2000–4000 s range, although 2000 s is more typical of actual applications. The mass of propellant would be still higher than in ion devices, typically 30% higher, while the thruster mass is moderate (for a 20 N thruster, about 22 kg). The main appeal of this propulsion system for de-orbiting purposes is the relatively high thrust, which can substantially shorten the orbital decay time with respect to a tether. Considering the shorter mission time due to the higher thrust level an estimate of propellant mass yields about 50 kg.

Drawing preliminary conclusions, a tether de-orbiting device shows interesting potential when compared to more conventional propulsion. This is inherently due to the fact that no propellant mass is ejected and thus needs not to be carried onboard. Roughly speaking, a tether de-orbiter will, on paper, have the upper hand when pitted against other thrusters, unless their I_{sp} can be increased by a factor of 10, which does not appear realistically possible. However, other factors (not considered in this short analysis), such as the need to space-qualify any new system, additional complexity related to the balloon, dynamic effects and other probably still unknown, will need to be investigated prior to draw final conclusions about the worthiness of a tether de-orbiting system.

5. CONCLUSIONS

The main goal of this paper has been to demonstrate that an electrodynamic tether system using an inflatable passive collector and a hollow cathode at its terminations is an effective means of de-orbiting large, man-made space debris (non-operational satellites or spent upper stages). The re-entry times have been computed under realistic assumptions for the interaction between the system and the ionospheric environment, an aspect not considered in previous works [3]. The currents and the drag forces have been obtained by solving along the orbit the equations of the equivalent circuit, with the data supplemented by the IRI-90 ionospheric model. A relatively light (30–50 kg) and technologically simple system (as compared to existing satellite thrusters), made up by a tether of 5 km length and a balloon of 10 m diameter can de-orbit a 500 kg satellite at 1300 km altitude in 20–60 days, depending on the orbital inclination. The drag forces produced by the system (up to 0.6 N) do not impair the stability of the tether.

The space tether considered here compares very favorably in terms of mass with respect to many alternative de-orbiting systems. One of its advantages lays in the fact that the same system, with no or little change, can be used for satellites with very different mass. Indeed, re-entry is anyway ensured even for very large satellites, at the expenses of longer de-orbiting times. On the contrary, chemical thrusters require an amount of propellant roughly proportional to the mass of the satellite. We feel that the penalty of longer re-entry times, paid when tethers are used, is much more acceptable than the mass penalty required by chemical propulsion. To produce re-entry, a tether system exploits the energy already acquired by the satellite during launch and orbit insertion, while for chemical propulsion the required energy must be brought, stored and expended in orbit. The inherent inefficiency of the chemical propulsion could only be justified by its well-proven reliability and controllability. However, the need for propellant increases significantly, if not prohibitively, the mass at launch, not mentioning the problems associated with a long storage in orbit.

In this paper we have not attempted to optimize the tether system. By a careful exploration of the parameter space one could find a better compromise among mass, complexity and decay times. However, a few remarks are in order. First, short tethers ($L < 5$ km) are more appealing because of the smaller risk of breaking by impact or arching and, for a given mass, because of their lower electrical resistance. Second, the total mass of the system, which should stay as small as possible, poses strong limitations to the combination of balloon size and tether length. We have computed the effect on the performance of a variation of the main system parameters (i.e. tether length and balloon radius) and solar activity levels. Longer tethers (10 km) and larger balloon radii (10 m) lead to shorter re-entry times, at the expense of the mass and complexity of the system, while shorter tethers (2.5 km) and smaller balloon radii (2.5 m) make the decay phase significantly longer, especially when the solar activity is low.

The technology of the inflatable structures is rapidly improving, but of course work needs to be done to ensure a reliable deployment after a long storage in orbit and the sustainability of the large currents required by tether propulsion. A demonstration of the system, maybe with a spent upper stage, would be very appropriate. To increase the reliability and ensure re-entry even in case of failure of the inflatable, one could consider leaving part of the metallic tether directly exposed to the ionospheric plasma. The results of the Pro-SEDS experiment [13], planned for the end of 2002, will be particularly interesting to assess the performance and the reliability of bare wires.

Other important issues are the probability of failure of the tether and the controllability of the decay. The first aspect has been treated in [5]. Although the predicted mean time between failure is relatively short (1 year) for typical system considered here, a suitably manufactured, multi-strand tether can reach much longer lifetimes [7]. A limited controllability of the system is highly desirable because of the risk that a decaying satellite may force large structures, like the International Space Station, to expensive collision avoidance maneuvers. The velocity of the re-entry may be very easily limited by means of series resistors, which increase the total impedance and hence control the current. If the orbital propagation shows an unacceptable risk of collision, the decay may be temporarily suspended. A variable thrust may also be used to increase the stability of the tether and control its librations.

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