EDOARD: AN ELECTRODYNAMIC TETHER DEVICE FOR EFFICIENT SPACECRAFT DE-ORBITING

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ABSTRACT

EDOARD (**E**lectrodynamic **D**e-**O**rbiting **A**nd **R**e-entry **D**evice) exploits the basic concepts of electrodynamic drag on conductive tethers to de-boost LEO spacecraft efficiently and reliably. The system is jointly developed by Alenia Spazio and by University "La Sapienza" in view of potential commercial exploitation. An innovative engineering approach has tackled four critical issues: 1) the deployment mechanism, which ensures a purely passive extension of the tether under extremely small gravity gradients; 2) the tether structure and configuration, which guarantees a very high survivability to impacts from artificial and natural debris; 3) the inflatable passive electron collector, which increases the efficiency of the system while reducing the tether length to 4-5 km and 4) the electrodynamic control of the tether librations, which limits the effects of inherent dynamical system instability, while preserving high de-orbiting efficiencies. The configuration ensures de-orbiting times per unit mass of about 0.09 day/kg from a circular, 1500 km altitude, 55 degree inclination.

1. INTRODUCTION

Commercial use of near-earth orbits requires finding innovative solutions to minimize costs and improve access to space. The use of conventional chemical thrusters has limitations due to its low specific impulse. Chemical propellants are not very attractive also since they increase the orbital debris population, as residues of exhaust gases. Since the post-mission disposal of orbiting structures may become a general policy in the near future, efficient de-orbiting devices should be developed. In this field chemical thrusters are marginal or less competitive, since they imply the storage in orbit of propellant (about 10-20% of the total mass to be disposed of) for many years.

The capability of long conductive bodies (tethers) to produce drag and thrust in space via electrodynamic interaction with the Earth magnetic field has been known since more than three decades [1]. Conceptual papers are produced in the area of interplanetary tether missions (i.e., momentum exchange tethers, "Skyhook"(, and others), but their industrial application looks far in the future (i.e., > 20 years from present). However, recent work has shown that this technology is attractive for de-orbiting LEO satellites, with significant advantages over conventional chemical propulsion [2,3,4]. In fact, a tether system of 30-50 Kg is capable of de-orbiting virtually any satellite in LEO (except for very high inclination orbits). Being a true propellantless system, its performance is assessed by the de-orbiting time, scaling as the satellite mass. Chemical propulsion imposes instead a penalty on the launch mass, growing with the mass to be disposed of. Also, re-entry time is less worrisome than launch mass to most space users: hence the interest for electrodynamic tether propulsion. A second advantage of conducting tethers over other propulsion system lies in their ability to work with non-operational/non-controllable objects. This feature is attractive to commercial users, who may be otherwise forced to de-orbit a perfectly functioning satellite at the end of its nominal lifetime. On the contrary, disposal by means of electrodynamic tethers would allow to maximize the use of orbiting satellites, until major internal failures occur.

The purpose of this paper is to describe EDOARD (Electro-Dynamic Orbital Re-entry Device), a state of the art electrodynamic tether system for de-orbiting medium size LEO satellites and upper stages designed in Italy by Alenia Spazio and the University of Rome "La Sapienza". After an evaluation of the market for de-orbiting systems and a summary of the physics of electrodynamic tethers for de-orbiting, we quantify the features of this new type of propulsion, identify the most critical building blocks for a practical tether system and predict overall performance of in terms of de-orbiting time and mass. An extensive numerical simulation of all mission phases has been carried out using proprietary codes, in order to verify the correct dynamical behavior and assess the overall system performances. Finally, we identify critical technologies to select for future research.

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2. MARKET PERSPECTIVES

EDOARD is designed to de-orbit medium and small size LEO satellites or upper stages. As envisaged here, EDOARD is applicable to vehicles in the 600 to 4,000 Kg mass range, orbit altitude between 600 and 2000 Km and orbital inclination up to 65 degrees. These ranges cover most present and future satellite constellations. Market analysis indicates that within the next ten years (2001-2010), for all space applications, 1500-2000 satellites and at least 600 to 700 launchers will be orbited.

Since EDOARD has inherent limitation in altitude and orbital inclination, its market over the same ten year period consists of at least 1200 to 1500 LEO satellites (polar missions are estimated to be about 15 to 20 % of total), with approximately an equal number of last stages. Even though some satellites and upper stages will decay naturally due to atmospheric drag in less than 25 years (the present limit past which active de-orbiting is recommended) this potential market is large and electrodynamic tethers look quite attractive to private industry. Such a favourable outlook requires that the design, development, production and launch costs are competitive against the additional launch costs of the more conventional and mature chemical propulsion systems.

The estimated propellant mass necessary to de-orbit LEO satellites from different orbital altitudes is shown in Figure 1, for different values of the mass of the object. For spacecraft of 2000-3000 Kg mass, adopting EDOARD for re-entry would save at least 200-300 Kg of fuel, with a corresponding quite significant cost reduction.

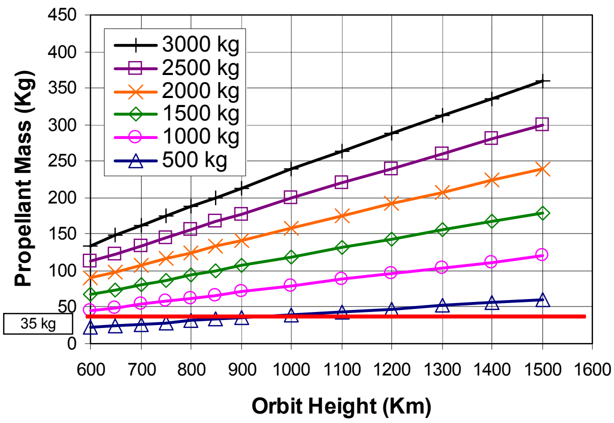

Fig. 1. Propellant mass required for spacecraft de-orbit, as a function of orbital altitude, for different classes of satellites. EDOARD, with a mass of approximately 35 kg, outperforms conventional systems in most cases.

3. PRINCIPLES OF OPERATION

The electrodynamic drag for spacecraft de-orbiting from LEO is generated by the interaction of the conducting wire with the Earth magnetic field and the surrounding charged particle environment. If unperturbed by other external torques, the tether is forced to align along the local vertical by the gravity gradient. Due to the orbital motion an electric field *E* is produced along the conducting wire:

$$\underline{E} = -\underline{v} \times \underline{B}. \tag{1}$$

where *v* is the wire velocity across the magnetic field lines and *B* is the local magnetic field. If no current flows in the system, the potential difference between the two ends is

$$\Phi_{ind} = \int_L (\underline{v} \times \underline{B}) \bullet d\underline{l}. \tag{2}$$

where *dl* is the differential element of tether length. If both ends of the tether (or the tether itself) are in electrical contact with the surrounding ionized medium, a current *I* flows in the wire and a decelerating Lorentz force is applied to the tether

$$\underline{F} = \int_L I(l)\, d\underline{l} \times \underline{B}. \tag{3}$$

This is the simple and powerful principle of electrodynamic drag. The magnitude of the drag force depends in a complex way both on the design parameters of the system, the orbit and the characteristics of the local ionosphere. The induced potential is rather straightforwardly related to the orbital elements (affecting *B* and *v*) and the attitude of the tether. Note that for polar orbits the relative geometry of the three intervening vectors is unfavourable and the resulting voltage is small. The tether current *I* is the other crucial quantity intervening in eq. (3). As it affects directly the re-entry time, the electrical design should aim to achieve the largest current value compatible with the integrity of the system. *I* is in first place affected by the induced potential through the non-linear laws of current collection from highly charged bodies. The attainable current for a given potential is in turn determined by the density and temperature of the ambient electrons, larger values being obviously more favourable. Another critical design element is the electron-emitting device at the negative termination of the wire, which must provide large currents with the lowest possible impedance. As a final remark, note also that the current may in general vary along the tether, as in the case of uninsulated, bare wires.

Accurate modelling of all relevant forces acting on the system, based on widely accepted environmental models and charge collection laws, indicates that the selected configuration for the electrical subsystem is capable to de-orbit in less than 12 months any spacecraft of mass smaller than 3000 kg from an altitude of 1500 km, for any inclination below 55 degrees. Under average solar activity, the de-orbiting time per unit mass is approximately 6 months for 2000 kg spacecraft (0.09 days/kg) for initial altitudes of 1500 km and 55 degrees inclination orbits. At equatorial orbits the de-orbiting times are nearly reduced by a factor three.

4. MAIN EDOARD SYSTEM REQUIREMENTS AND OPERATIONS

The EDOARD system is intended to provide the carrier spacecraft (a LEO satellite or a launcher third stage) with an electrodynamic device capable to de-orbit it within a few months. It is assumed that the in-orbit activation of the EDOARD device, mounted onto a suitable external surface of the spacecraft, be commanded from ground at the end of the operational lifetime of the carrier vehicle or at any time during its in-orbit operations.

EDOARD main mission phases are defined as:

- a Dormant Phase, which covers nearly the full carrier vehicle mission, from its launch to end-of-mission. In this first phase all EDOARD subsystems are in a dormant state, interrupted only by possible health checks from ground control.
- an Initialization Phase, starting with a ground command via TM/TC of the carrier vehicle, initializing all EDOARD subsystems and main components. This will take place at the carrier end-of-life, just before starting the proper sequence of de-orbit operations.
- a Tether Deployment Phase, to be initiated with an impulsive separation of the tethered mass from the carrier vehicle and to be continued with the full deployment of the several-kilometers-long conductive tether
- a Controlled Orbital Decay Phase, when electrical current flows into the wire and the resulting electromagnetic drag causes the artificial orbital decay of the overall tethered system. This phase will also start by a ground command, to be sent after full tether deployment and stabilization operations have been completed. During the de-orbit operation tether system stabilization control is actively performed by the EDOARD electrical subsystem.
- a Re-Entry Phase, representing the last operational phase of the mission and starting when the tethered system orbital decay has reached about 120 km altitude and the atmospheric drag takes control of the trajectory.

EDOARD is capable of deploying the tethered mass and the full tether length autonomously, with minimal requirement of energy and power from the carrier.

In the present configuration, the client vehicle should supply the EDOARD system with power, data and (at the deployment time) with its operational spacecraft attitude control.

The power required by the EDOARD system from the carrier vehicle does not exceed some tens of Watts at any stage of operations. The largest power consumption occurs during the initialization of the electrodynamic de-orbiting subsystem, when the electron emitter is activated. In the Controlled Orbital Decay Phase, when current flows in the wire, the whole tethered system can be made fully autonomous, with no power requirement from the host.

EDOARD also utilizes the carrier TM/TC subsystem to receive commands from ground and to transmit operational data: the required data rates are rather low, not exceeding a few hundred bits per second during transmission. A much more autonomous version is left to future development.

The overall EDOARD mass is envisaged to be less then 30 -35 Kg, even for large LEO satellites, e.g. with dry mass over 3000 Kg. In this way the mass of the EDOARD device will be a very small fraction (between 1% and 5%) of the carrier vehicle mass at launch. As illustrated in the Figure 1, the mass saving with respect to conventional propulsion systems is of the order of hundreds of kilograms for all spacecraft with dry mass larger than 1000 Kg.

5. SYSTEM CONFIGURATION

EDOARD system is based on a short conductive tether (4-5 Km kilometers long), mechanisms capable of deploying the tether from the carrier vehicle after a ground command, electrical/mechanical subsystems ensuring adequate current collection and emission and the associated control electronics. The electron-collecting tether end is equipped with a large passive electron collector (up to 10 m diameter). This configuration ensures effective electron collection and high de-orbiting rates.

The engineering design is driven by low mass (35 kg), low volume (a few tens of liters) and low complexity as well as by high reliability (0.95 minimum) requirements. The innovative approach has tackled four critical issues: 1) the deployment mechanism, which ensures a purely passive extension of the tether under extremely small gravity gradients; 2) the tether structure and configuration, which must guarantee a very high survivability to impacts from artificial and natural debris; 3) the inflatable passive electron collector, which increases the efficiency of the system while reducing the tether length and 4) the electrodynamic control of the tether librations, which limits the potentially destructive effects of the inherent dynamical instability of the system under the forcing magnetic torque, while preserving high de-orbiting efficiencies.

5.1 Electrical Subsystem

The EDOARD electrical sub-system consists of the conductive tether, an electron collector at the positive end and an electron emitter at the negative end. A switch also is needed, in order to control current flow in the tether. The typical de-orbiting system configuration can be electrically represented by the equivalent circuit of Fig. 2.

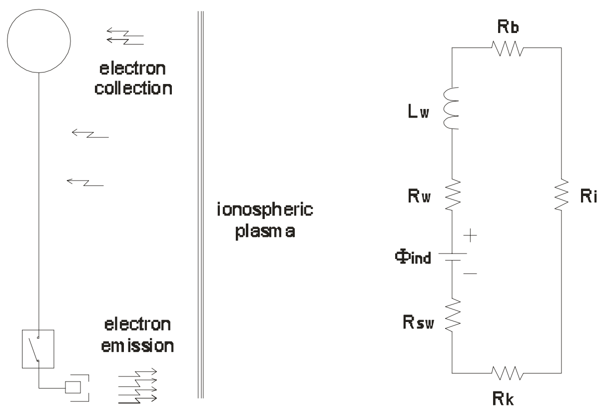

Fig. 2. EDOARD equivalent electric circuit

The wire current mainly depends on the potential drop associated with electrical field in the orbiting frame (see eq. 1 and 2). For a given orbit and attitude configuration, the current can be maximised by reducing the total electric resistance of the circuit, given by:

$$R_{tot} = R_w + R_b + R_k + R_i + R_{sw}. \tag{4}$$

where R_w is the wire resistance, R_b and R_k are the contact resistances associated respectively with the potential gap across the plasma sheaths between ionosphere and electron collector and emitter; R_i is associated with the ionospheric closure loop (usually a negligible quantity) and R_{sw} is the ON-state resistance of the switch. As the de-orbiting time is inversely proportional to the current (neglecting in a first approximation the effects of the attitude dynamics), a major requirement on the design of the electrical sub-system is the minimization of the overall electrical resistance. This requirement has to be traded-off against other design specifications on mass, complexity, reliability etc. Using the functional and performance requirements associated with cable design discussed in the previous section, a trade-off study provides an electrical resistance R_w of a few hundreds Ohms.

R_b, the contact resistance between the positive electrode and the ionospheric plasma, can be reduced by exposing large collecting surfaces. An appreciable flux of electrons can be directly collected by the positively biased wire, which is directly exposed to the surrounding plasma. In order to increase the drag force and reduce the tether length, the positive termination of the cable will be connected to an inflatable structure, ensuring a large collecting area, with adequate surface conductivity and power dissipation.

A low sheath resistance R_k at the negative end is provided by a plasma contactor. The most mature and tested technology is based upon hollow cathodes, in which the emitted electrons are produced by a local plasma generated from a Xe gas flow. On the other hand, the emerging technology of cold cathodes is quite appealing. Electrons are emitted by field effect, requiring small electrical power, and no gas supply. The favourable mass and power budgets are counterbalanced by several risk items: first of all, the real production capability of present suppliers is unknown. Moreover, so far no fly test has been reported for these devices, and doubts are still unsolved about emitting surface stability as well as about correct operation of cold cathodes in the wide range of environmental conditions encountered during controlled orbital decay.

The switch is an electronic device allowing us to start and stop current flow in the circuit by transmission of proper TLC signals. It can also be autonomously triggered by on-board computer and by attitude control logic.

5.2. Mechanical Subsystem and interfaces with carrier vehicle

The EDOARD system is attached to the carrier vehicle by a simple mechanical interface plate mounted on a suitable external surface of the carrier satellite.

The units of the EDOARD system which remain onboard the carrier vehicle after the separation (in particular, the on-board micro-processor unit for attitude control, the plasma contactor, the tether attachment point and part of the separation mechanism) are mounted on the outside of this interface plate. The same plate hosts also the power supply and the data interfaces with the carrier (standard pre-selected connectors).

During the launch and the carrier vehicle mission EDOARD is in the so-called Dormant phase; the tether and the related mechanisms are protected from space environment agents and impact risks with micrometeorites or debris by a cylindrical shield, housing the spool and the released tether end.

5.3. Tether deployment subsystem

In order to achieve minimum mass, volume and complexity, the design of EDOARD has adopted a passive tether deployment strategy, initiated by an impulsive actuation imparted by the separation mechanism and commanded from ground. In the early stage of the deployment, the acting dynamic forces (differential gravity gradient and Coriolis') are rather small; in order to guarantee the successful completion of a passive deployment, the friction forces acting on the tether during unwinding must be kept very small (of the order of few grams). This is possible thanks to the particular tether winding and deployer design, using a fixed spool. It is currently under international patenting.
According to the adopted deployment strategy, the tether end mass, after a small deceleration lasting up to few hundred meters, will keep accelerating, unless an increase in tether deployment friction or deployment breaking force is purposely applied. In order to achieve a pre-selected deceleration at the end of the deployment and avoid dangerous shocks on the tether, the EDOARD system makes use of a passive device increasing the deployment friction in the final phases of the extension. The device ensures that the terminal deployment rate of the tether is kept to below 0.5 m/s.
Figure 3 illustrates the history of the tether deployment rate using the adopted passive control strategy. With an initial separation velocity of about 2 m/s, the maximum deployment rate is about 4.5 m/s. By starting the deceleration phase in the last kilometer, the terminal velocity attains a null value when the tether is fully deployed. In this simulation the total deployment time is about 3,000 seconds.
An extensive series of ground tests will complement numerical simulations, in order to assess the friction forces during deployment. A computer-controlled TMM&M test stand has been already implemented, equipped with a fine tension measurement unit, a motor drive unit with tether drive rollers, clutch and flywheel to properly follow a given tether deployment profile.

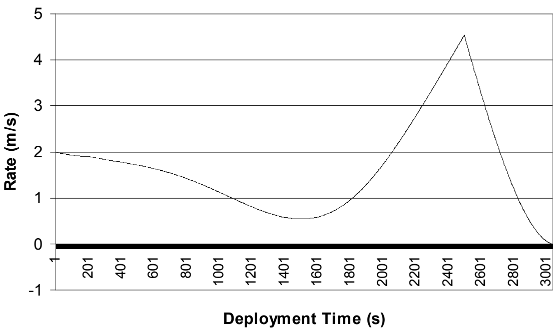

Fig. 3. Deployment rate history of the EDOARD tether.

5.4. Attitude control subsystem

The electrodynamic force acting on the tether generates a torque affecting very significantly the attitude dynamics. The actual location of the center of application of the electrodynamic forces depends on the current profile along the tether. For EDOARD, which uses both a spherical and a bare wire anode, this point is always located close to the tether's geometrical center. Due to the large difference in the mass of the two end bodies, the system's center of mass is very close to the satellite to be de-orbited and a large lever arm has to be expected
Large ED-torque-induced librations are expected under dynamic conditions [5], unless a proper control strategy is adopted. In the EDOARD system tether attitude estimations, combined with the knowledge of the local magnetic field and wire current, allow to control both in-plane and out-of-plane libration by proper phasing of the current flow. The duty cycle ranges between 0.65 and 0.95, depending on the orbital altitude and inclination.
An example of the residual attitude motion is shown in Figure 4, which refers to an altitude of 500 km and an inclination of 55 degrees. If uncontrolled, the tether would start tumbling within a few tens of orbits.

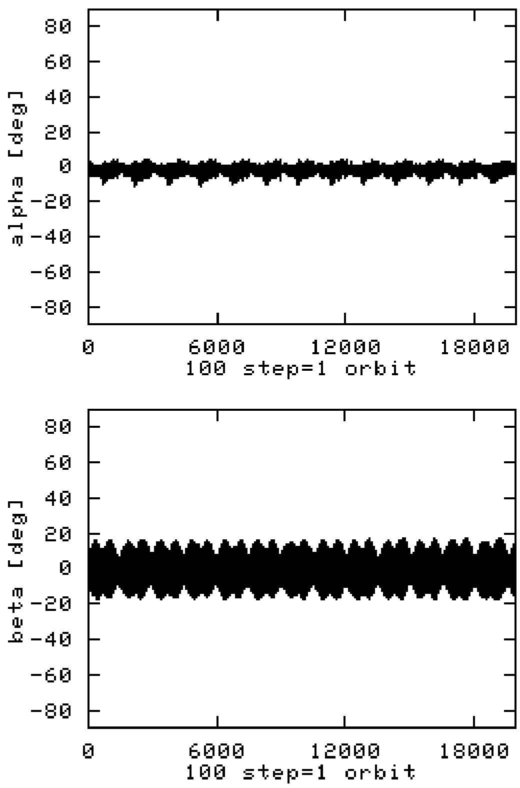

Fig. 4. Estimated in-plane and out-of-plane libration angles, under electrodynamic attitude control.

6. TETHER SURVIVABILITY

The issue of survivability (in particular for the longest de-orbit missions) is of paramount importance for the choice of the tether structure and therefore deeply affects the design of the whole EDOARD tether mechanism. The tether must be a conductive, unprotected wire with a suitable diameter and length. This adopted design must be compatible with other technological and environmental requirements. In particular, the selected material and structure, must offer good flexibility and ensure environmental survivability when fully extended. As rather long de-orbit times (up to 12 months) are expected in the case of large satellites at highly inclined orbits, several potential threats have been considered, including UV radiation damage and atomic oxygen erosion and especially the risk of severing by impact with micro-meteorites and/or space debris. The design of the tether must ensure a survival probability in the longest missions of at least 95 - 98%, including the risk of severing impact with micrometeorites and artificial debris.

In order to appropriately design the tether against the risk of severing by impact, differential flux models of space debris and micro-meteorites have been considered as functions of several parameters, including orbital altitude and inclination, calendar date and solar flux and activity. Unfortunately single strand tethers achieve the required survival probability only if the diameter is unacceptably large. As an example, for a high altitude (1,500 Km), large satellite requiring about 12 months to de-orbit, the survivability of a single strand 5-Km long tether with a diameter of less than 1 mm is as small as 10-15%. Adequate survivability (around 95 % or larger) is possible by adopting a double strand tether structure. The EDOARD system makes use of an appropriately designed, double strand tether which meets all requirements as for mass, electrical resistance, flexibility and survivability.

7. CONCLUSIONS

A potentially large demand for de-orbiting propulsion systems is likely to grow in the near future, motivated by the need to limit the population of artificial debris in the near-Earth space. Due to the large mass penalty associated with the use of conventional chemical propellants, the development of more efficient, alternative systems is highly desirable. Electrodynamic tethers offer a valid and attractive solution. Imposing a penalty in terms of de-orbiting time rather than mass and being able to operate with dead satellites, they offer cost savings and capabilities unmatched by any other propulsion system. EDOARD, jointly designed by Alenia Spazio SpA and the Rome University La Sapienza, is a tether device capable to generate a low, nearly continuous, electrodynamic drag force which ultimately leads to the re-entry in the atmosphere of the carrier vehicle. With a total mass of about 35 kg, the system would allow a saving in the range of hundreds of kilograms of propellant. The attainable de-orbiting times are below 12 months even for large satellites orbiting at 1500 km altitude. The design ensures the correct deployment of the tether and its dynamical stability against the perturbing electrodynamic torque. The present, initial version still requires power and telemetry from the host and therefore does not provide the capability to de-orbit non-operational satellites. A fully autonomous device can be largely based upon the existing design and is being considered after the successful evaluation and testing of the present configuration.

8. ACKNOWLEDGMENTS

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