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Low work-function tether Deorbit Kit

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ABSTRACT

This works presents a system level analysis of a Deorbit Kit (DK) based on electrodynamic tether technology. The analysis is focused on two relevant scenarios for deorbiting space debris: (i) Earth Observation (EO) satellites with mass in the range of 700 kg -1000 kg and initial orbital altitude of 800 km and 98° inclination, and (ii) Mega Constellation (MC) spacecraft in the order of 200 kg and initial orbit at 1200 km of altitude and 90° of inclination. The scenarios have been selected considering the orbits that are already suffering from the space debris problem or will suffer in the next future. The DK implements a bare electrodynamic tether for capturing electrons passively from the ambient plasma while three different methods are considered for emitting the electrons back to the plasma to reach a steady electrical current on the tether. The three studied options to close the electrical circuit are: (a) a hollow cathode, which has a high technological maturity but needs expellant and a little of power, (b) a thermionic emitter, which does not involve expellant but needs power, and (c) a Low Work-function Tether (LWT) that does not need neither expellant nor power because it has a segment coated with a special material that emits electrons passively through the thermionic and photoelectric effects. In order to provide a fully autonomous operation even in case of critical failure of the mother spacecraft, the DK includes a deployment mechanism, a telemetry and telecommand system, a complete Attitude Determination and Control System with attitude sensors (GNSS, sun sensors, magnetometer) and actuators (magneto torquers), solar panels and batteries. Upon activation, the DK autonomously de-tumbles the satellite, deploys a tether and carries out the satellite's de-orbiting. The study presents DK architectures, mass budgets and simulation results for the two scenarios. It is shown that a complete DK with mass below 6% the mass of the host spacecraft can deorbit EO and MC satellites in about 1.5 years and 10 years, respectively. The importance of the development of the LWT concept to enhance the simplicity and reduce the mass, power and volume budget is highlighted.

1. Introduction

After few decades of controversies and debates, there is currently a broad consensus about the importance of the space debris problem. Thorough analyses have demonstrated that the density of the space debris population has already reached the threshold that triggers an uncontrolled cascade of collision (Kessler syndrome) [1]. Actual trend, i.e. moving from big, few, expensive, and highly reliable spacecraft towards small, very numerous, inexpensive and probably less reliable spacecraft, will deteriorate the environment even more in the future. For instance, Starlink (12,000 satellites), OneWeb (650 satellites), Leosat (108 satellites) and Telesat (292 satellites) constellation are planning to use the 1200 km to 1400 km altitude polar orbit for providing their services and deorbit the satellites using electric propulsion after 5 to 7 years of operations [2,3]. For satellites around 230 kg in 1200 km orbit, it is calculated that a full deorbit below the International Space Station (ISS) will suppose a cost of more than 7 kg and 3 months of operations while partial deorbiting to 500 km followed by a naturally re-entry, in about 10 years would only allow a saving of 1 kg of expellant and 2 weeks of operations. Since a legal international framework for protecting the space environment does not exist other options like graveyard orbits are possible. Besides deorbit operation is only possible for operational satellites.

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Preventing accidental post-mission explosion with passivation means, carrying out active debris removal of the most dangerous objects, and removing spacecraft at the end of life from densely populated orbits are the most effective methods to avoid the population growth [4]. However, since active debris removal and spacecraft deorbiting are both expensive, it is difficult to develop a tough legal framework without affecting the competitiveness of the space sector. Such a vicious circle would be broken by the development of an affordable deorbit technology that would also open a new market. An electrodynamic tether is a promising solution because it is a propellant-less technology that works passively thanks to the Lorentz drag, which is a much more effective mechanism in Low Earth Orbit (LEO) than the aerodynamic drag used by drag augmentation devices [5]. In addition, tethers can be easily tracked from the ground and can perform collision avoidance maneuvers during the deorbiting. Past tether experiments, like Small Expendable Deployer System (SEDS-I) and Plasma Motor Generator (PMG) demonstrated in orbit the tether deployment, the electrodynamic force generation, and the use of tether to produce power [6]. The Tether Physics and Survivability Experiment (TIPS) [7], several hypervelocity impact tests and simulations on tether tapes [8], and theoretical analysis on tether mission design [9], revealed the robustness of tethers in the space environment. The next steps for the technology are passing through an industrialization phase and prove that tethers can be operated in a safe, reliable, and robust manner.

In 2018, the European Commission awarded a H2020 FET-Open project with title "Electrodynamic Tether Technology for Passive Consumable-less Deorbit Kit" (E.T.PACK) and reference number 828,902, which is aimed at the development of a Deorbit Kit (DK) based on electrodynamic tether technology [10]. This work is part of E.T.PACK activities and discusses three different DK configurations to deorbit spacecraft in two critical scenarios for space debris. The first one covers large Earth Observation (EO) satellites (700kg-1000 kg) from orbits catalogued by ESA as hot spot A (82° at 1000 km height), hot spot B (71° at 850 km height) and hot spot C (98° at 800 km height). The second is focused at the mega constellations (MC) case, i.e. spacecraft around 200 kg orbiting at 1200 km and 90° of inclinations. The work is organized as follows. Section 2 presents the basic principles of the three different types of electrodynamic tethers that are currently under development in the framework of E.T.PACK. Section 3 summarizes the most important requirements of the DK. The design at a system-level and numerical simulations of the performance of the DK configurations in the target scenarios are presented in Section 4. The conclusions are summarized in Section 5.

2. Basic principles of electrodynamic tethers

When a tether of length L moves at a relative velocity $\boldsymbol{v}_{rel}$ with respect to the conducting ambient plasma in the presence of a magnetic field $\boldsymbol{B}$ (the geomagnetic field), the following motional electric field $\boldsymbol{E}_m$ appears at the faraway plasma for an observed attached to the tether

$$\boldsymbol{E_m} = \left(\boldsymbol{v}_{rel} \times \boldsymbol{B}\right). \tag{1}$$

If a good electrical contact exists between both conductors, a steady electric current $\boldsymbol{I} = I(s)\boldsymbol{u_t}$ circulates along the tether thanks to the electromotive force provided by $\boldsymbol{E}_m$. A Lorentz force

$$\boldsymbol{F}_L = \int_0^L \boldsymbol{I}\left(\boldsymbol{s}\right) \times \boldsymbol{B} ds \tag{2}$$

appears, where s and $\boldsymbol{u_t}$ are the arclength and a unit vector along the straight tether. Assuming a quasi-circular orbit, Gauss' first planetary equation shows that the semi-major axis of the orbit (a) decreases according to [9]

$$\frac{da}{dt} = \frac{2a^2}{M_s\mu}F_L \cdot v \approx -\frac{2a^2}{M_s\mu}E_t\int_0^L I\left(S\right) ds \tag{3}$$

with $\boldsymbol{v} \approx \boldsymbol{v_{rel}}$ the orbital velocity and M_s the mass of the spacecraft (note that $E_t \equiv u_t \bullet (v \times B) > 0$). Typical values in mid-inclined orbits in LEO are $v_{rel} \sim 7.5$km/s, $B \sim 10^{-5}T$ and $E_m \sim 75V$/km. For a 2km-long tether carrying an average current of 0.5A, the electromotive and Lorentz forces are around 150 V and 10mN (Fig. 1).

The critical point for the operation of electrodynamic tethers is to ensure the good electrical contact, i.e. the electron collection and emission, between the tether and the ambient plasma. Regarding anodic contact, this work considers a bare (without insulation) tether, because it effectively and passively captures the electrons from the ambient plasma [11]. The achievement of a good cathodic contact (electron emission and/or ion collection) is difficult due to the high ion mass. Three different DK configurations are analysed: (a) a bare tether equipped with a Hollow Cathode (HC), (b) bare tether using Thermionic Emitter (TE), and (c) a Low Work function Tether (LWT). In configurations (a) and (b) the electrons are collected by the bare tether passively and they are emitted back to the plasma by the active device, i.e. the HC or the TE. Hollow cathodes can emit significant amount of current, in the order of few ampere, with a power consumption around 20 W. However, they need expellant. Thermionic emitters do not need expellant, but the required power is greater. In an LWT, the full tether or part of it is coated with a material that has a low Work-function (W). Consequently, it can be shown that a tether segment captures electrons passively and the complementary segment emits them back to the plasma through the thermionic [12] and the photoelectric effects [13]. The operation of LWTs is fully passive and it does not require power neither consumable. Unlike HCs and TEs, which are mature technologies, works on manufacturing and testing LWT samples started recently and they are in progress. In the case of E.T.PACK, the consortium is focused on a coating based on the C12A7:e^- electride [14], due to its high electronic conductivity and low W.

Fig. 2 shows schemes of a tether with active electron emitter, HC or TE (left) and LWT (right) in equatorial orbit. For each of them, the physical model that provides $I(s)$, which is necessary to evaluate the Lorentz force in Eq. (2) and determine the deorbit performance, is different. The models involve tether properties, ambient plasma parame-

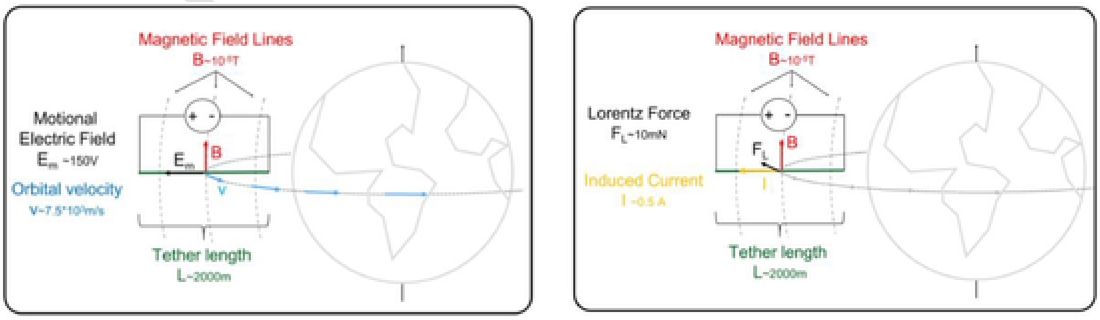

Fig. 1. Typical values of Motional Electric field (left) and Lorentz Force (right) for LEO equatorial orbit tether.

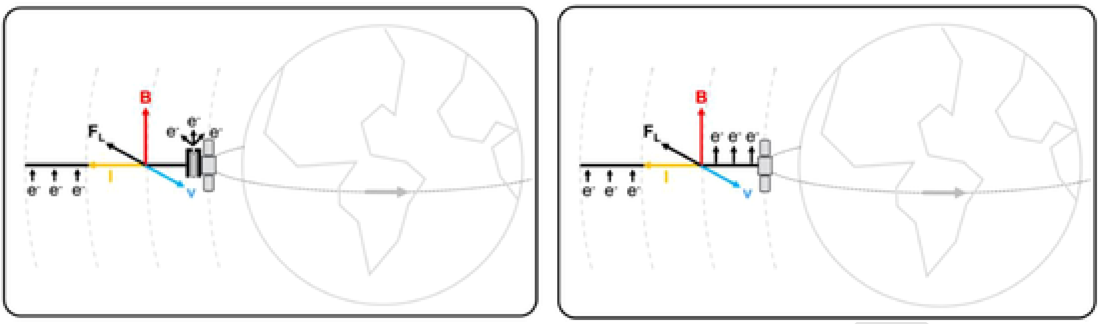

Fig. 2. Schemes of an active electron emitter HC or TE (left) and a LWT (right) in equatorial orbit.

ters, and the characteristics of the HC, the TE or the coating of the LWT. Interested readers can find the details in Ref. [11] for HC and TE, and in Refs. [15] and [13] for LWTs. These models are implemented in BETsMA v.2, a tool for the optimal design and performance determination of electrodynamic tethers [16]. The simulations of Section 4 and the analysis of this work are based on this tool.

Table 1
Deorbit Kit requirements.

Name	Requirement
DK maximum mass	Deorbit Kit mass shall be less than 5% of the host S/C mass. *Rationale. Requirement provided by ESA after consulting with European Spacecraft Manufacturers.*
De-orbit time	Deorbit Kit worst case deorbit time shall be in any case less than 10 years for satellite in orbit up to 1200 km and 25 years for orbit up to 1400 km. *Rationale. Higher orbits selected by telecommunication mega constellation (Starlink, OneWeb, Telesat and LeoSat) range from 1100 km up to 1400 km.*
Host spacecraft mass	Deorbit Kit design shall be scalable to de-orbit satellite of mass from 200 kg up to 1000 kg. *Rationale. Small tether system are of interest for mega constellations spacecraft weighting in the order of 200 kg. Future satellites of 700–1000 kg (Sentinel Class) will be designed for demise. This will allow passive de-orbit by DK.*
Autonomy	Deorbit Kit shall be designed to be fully autonomous and only mechanically connected to the host spacecraft. *Rationale. The kit shall allow de-orbiting also in case of host spacecraft failure.*
Durability	DK shall be designed to be maintained for 10 years in ground stockage and 10 years in space before activation. *Rationale. DK will be stored once manufactured and will only be activated at the end of the satellite operational life. Host satellite maximum operational life is 10 years.*
Fully demisable	The DK shall totally demise when exposed to the thermal flux from an altitude of 78 km. *Note. It is defined totally demised an object when any surviving pieces impact ground with an energy lower than 15 J.*
System reliability	Deorbit Kit reliability shall be higher than 90%. *Rationale. Required by ISO20133:2019. De-Orbit system unreliability shall be lower than 10%.*

2. DK requirements and target scenarios

The need for the development of a deorbit kit has been formally recognized by the European Commission in 2018 granting to the authors of this paper the E.T.PACK project. In the frame of E.T.PACK, the main requirements of the DK have been established based on need basis while receiving inputs from the European Space Agency and the European spacecraft Prime contractors. Table 1 summarizes the main DK requirements, including DK and mother spacecraft masses, deorbit time, autonomy, durability, demisability, and reliability. According to them, the consortium identified several interesting commercial scenarios to apply the DK. This work focus on the DK design for two different reference missions:

- De-orbit EO satellites in the hotspot C at 800 km altitude and 98° inclination orbit. Typical size of these spacecraft in the sentinel class are between 700 kg and 1000 kg. The maximum DK mass shall therefore be between 35 kg and 50 kg.
- Deorbit small telecommunication satellites from 1200 km altitude at 90° inclination. Typical weight of MC spacecraft is in the order of 200 kg. Maximum DK mass shall therefore be less than 10 kg.

For a perfect and centred dipole geomagnetic field model and a tether aligned with the local vertical flying in polar orbit ($i = 90°$), there is no-deorbiting because $E_t \equiv u_t \bullet (v \times B) > 0$) (see Eq. (3)). Fortunately, the geomagnetic field is tilted with respect to the polar axis and tethers can still operate at high inclinations. In any case, the two selected reference missions correspond to the worst cases for electrodynamic tethers and the performance in orbits with lower inclinations will be significantly better than the one presented in this study.

Table 2 summarizes the main characteristics and the state of maturity of the three DK configurations considered in this work, as well as the proposed application scenario for each of them. The team is making research on the C12A7:e- electride material and trading different tether coating process, thicknesses and substrates. The W and the optical properties of LWT samples will be determined during the project. In parallel, E.T.PACK is working on the HC and TE. HC is the most appropriate emitter for EO satellites because, due to the large mass of the spacecraft, a high level of electrical current is required to meet the de-orbit time requirement. In the case of the MC satellites, which are lighter, a lower current level is needed and the TE can be a better choice to avoid the use of expellant.

Table 2
Electrodynamics tethers considered in this work and their main characteristics.

DK configuration	Expellant Need	Power Need	Electron Emission	Technology Maturity	Host satellite mass
Bare tether + Hollow Cathode (HC)	Yes	Low	Active	High	700 kg to 1000kg
Bare tether + Thermionic Emitter (TE)	No	High	Active	Medium	200 kg
Low Work Function (LWT)	No	None	Passive	Low	200 kg to 1000kg

3. Deorbit Kit design

The DK design build on the consortium tether experience and on the SENER heritage in designing, manufacturing and testing space mechanism and payloads. The design drivers are keeping the design simple and ensure reliability. As already highlighted in the literature [9], one of the key issues in tether mission design is the selection of the tether geometry and tapes tethers have much better performance than round ones. For this reason tape tethers has been selected and length (L), width (w), and thickness (h) have been sized considering a combination of electrodynamic performance, deployment mechanism experience, electron emitters laboratory tests (HC and TE) and dynamic considerations. Such a knowledge was combined with simulations in order to find appropriate values. The long term simulations assumes a tether perfectly aligned with the local vertical and medium solar activity. Short term simulation runs considering tether oscillations indicate this assumption is conservative in terms of deorbit time.

3.1. Active electron emitters DK configurations

During the satellite operations the DK remains in hibernation. HIBERNATION mode is defined to minimize power consumption. DK can receive telecommand and answer with basic telemetry only if required. Upon ground telecommand reception, DK enters in STABILIZATION mode. In this mode DK wake up and turns on the Attitude Determination and Control System (ADCS). Using GNSS, magnetometer and Sun Sensor, DK determines its attitude and angular velocity. DK uses magneto-torquers to slowly removes spacecraft angular velocity and acquire the proper attitude to deploy the tether. The DEPLOYMENT mode can be started either automatically or by ground telecommand. In DEPLOYMENT mode, one part of the DK separates from the other following a controlled trajectory. The initial separation is provided by a cold gas system and helped by a motorised system. The deployment is performed in about 1 h. At deployment completion STANBY mode is automatically entered. In STANDBY mode the DK sensors are active and telemetry is provided to ground. At the end of deployment the tether is stabilized along the local vertical direction and the DK is ready to start deorbit operations. DEORBIT mode is entered from telecommand or automatically after a pre-defined time from the deployment event. In DEORBIT mode, the Electron Emitter (HC or TE) is ignited and a stable and steady current flows along the tether-emitter-plasma circuit and a drag force is generated. DEORBIT mode is the nominal operation mode of the tether. From DEORBIT mode transition to STANDBY is possible via telecommand or in case of failure requiring ground intervention. The transition from DEORBIT to STANDBY stops the active deorbiting force (Lorentz force). This orbital control capability can be used for collision avoidance Fig. 3 depicts the DK operation modes.

The DK based on HC and TE are composed by 3 elements: **fixed Module, tether** and **deployment module**. The tether is packed in coils inside the Deployment Module during the launch and the normal operation of the host spacecraft. This packing allows to optimise the DK volume occupation. The tether connects at one end with the Deployment Module and at the other side with the fixed module. The deployment module is kept attached to the fixed module by means of a commercial Hold Down and Release Mechanism (HDRM). Upon DK activation the deployment module separates from the fixed module and deploys the tether.

The **fixed module** tasks are:

- Emitting the electrons captured by the bare tether in DK operation. Electron emission shall be performed at the host spacecraft side in order to mitigate the instability of electrodynamic tethers [17].
- Keeping the DK connected to the host spacecraft. This is performed by the HDRM.
- Receiving telecommands and sending back housekeeping telemetry.

The **tether** is made of a bare segment and an inert segment and it is required for:

- Collecting electrons from the ambient plasma. One tip of the bare tether segment is connected to the HC or the TE and the other tip is connected to the inert segment.
- Stabilizing the tether. The inert tether segment is required to mitigate the dynamic instability. Since the target orbits have high inclination, a component of the Lorentz Force (Eq. (2)) is perpendicular to the tether velocity and creates a torque that make the tether oscillate out of the orbital plane. The inert segment increases the gravity gradient and helps to keep bounded the amplitude of the oscillations [18].

The **deployment module** is in charge of:

- Removing host satellite angular velocity and providing correct attitude before the tether deployment. Dedicated ADCS is being developed for this scope.
- Deploying the tether along the desired trajectory and extracting it from the storage coil. The initial deployment is performed by a cold gas system aided by a motorised system to overcome the internal friction of the deployer. Motorised system is also used to gently stop the tether deployment maintaining the oscillation limited. The deployment module mass provides a gravity gradient stabilization pull.
- Maintaining contact with ground. Deployment module includes a Communication Subsystem to receive ground telecommand and provide DK telemetry. Deployment module also works as DK main control unit managing the fixed module operations and centralizing the communications with ground.

Tether with Hollow Cathode for Earth Observation Satellites

A bare tether with HC emitter has been selected for the EO scenario. The expellant of the HC is stored in dedicated tanks in the fixed module and its amount determines the maximum operational lifetime of the DK. The ADCS is sized for detumbling 1000 kg host spacecraft rotating at initial angular velocity of 2°/s. This value is provided by ESA according to analysis made on space debris dynamics. In order to minimize the power required by the DK, a heater-less HC based on the C12A7e-electride [19] has been selected. The dimensions of the tether are $L = 3\text{km}$, $w = 2.5\text{cm}$, and $h = 40\mu\text{m}$ and the lengths of the bare and the inert segments were 2 and 1 kms, respectively. Fig. 4 shows a me-

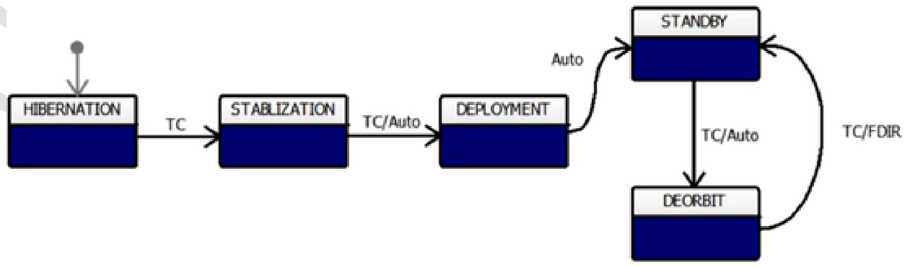

Fig. 3. DK operation modes.

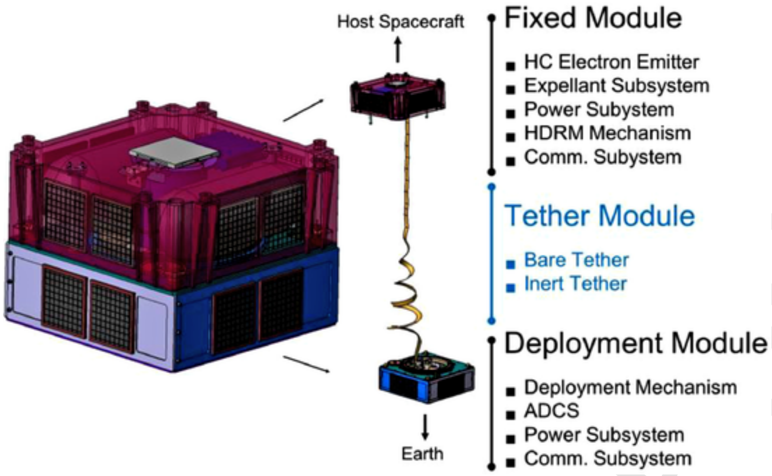

Fig. 4. EO DK design for 700–1000 kg spacecraft.

chanical drawing of the DK. The EO DK mass is 35.5 kg (see mass budget in Table 3).

Simulations with BETsMA v2.0 show that the EO DK can deorbit from 800 km orbit in 551 days a 700 kg spacecraft, in 684 days a 850 kg spacecraft and a in 804 days a 1000 kg spacecraft (see Fig. 5). These times are well below the deorbit time requirement and are compatible with the expellant mass allocated in Table 3.

Table 3
EO DK mass budget.

Module	Mass	%	Comment
Fixed	16.2 kg	45.6%	C12A7:e- heatherless hollow cathode emitting a maximum current of around 0.5A with 5 kg of expellant mass.
Tether	6.9 kg	19.4%	Bare aluminium tether (2 km) plus inert PEEK tether (1 km). Width 2.5 cm and thickness 40microns.
Deployment	11.3 kg	31.8%	Including magneto-torquers for detumbling the host spacecraft rotating at 2°/s.
Total	**35.5 kg**	**100%**	

Tether with Thermionic Cathode for Mega Constellation Satellites

The mega constellation scenario introduces 3 challenges for the design of a DK: (i) the altitude of the initial orbit is 1200 km, (ii) the orbital inclination is the worst for the tether (90°), and (iii) the DK mass should be less than 10 kg. amongst the challenges, the mass constraint is the more demanding. From Table 3 it can be observed that 45.6% of the EO mass is located in the fixed part. In particular, the electron emitter expellant subsystem weigh is 11.5 kg. The advantage of using HCs is that they require very limited power and can emit high current, resulting in limited deorbit time for large spacecraft. However, current levels required for deorbiting 200 kg host satellites are small enough to use a different electron emitter technology: the Thermionic Emitter (TE). In TE, a low-W material is heated at high temperature, in the order of 1000 °C, and the electrons are extracted into free space using a biased extraction grid. TEs have reduced mass and volume, have proven lifetimes longer than 10,000 h, and are commercially available. DK based on TE is not limited by the expellant mass, therefore the deorbit time can be longer. As drawback, TEs require high power to heat the cathode and in particular to electrostatically accelerate the electrons. Required power consumption is in the order of 200 W for emitting currents of 0.1A. Since the DK cannot generate this power, it should be provided by the host spacecraft during the deorbit. The autonomy requirement would be violated but lightweight DK of less than 12 kg can be designed (Table 4).

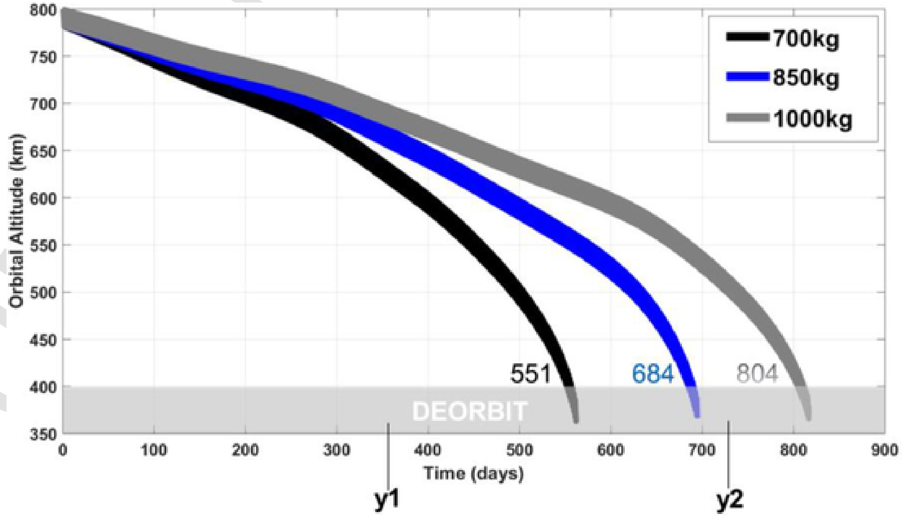

Fig. 5. Performances of EO DK for 700 kg, 850 kg and to 1000 kg host satellites.

Table 4
MC DK mass budget.

Module	Mass	Comment
Fixed	2.7 kg	TE to emit a maximum current around 0.1A. Power from the host spacecraft is required
Tether	3 to 4kg	Bare aluminium tether $L = 1.5$ km and $L = 2$ km. Inert tether not required due to limited current. Width 2.5 cm and thickness 30microns.
Deployment	6.2 kg	Module sized to host 2 km of tether.
Total	**11.9 –12.9 kg**	

The resulting fixed module (powered from the host satellite) includes a compact thermionic cathode, the communication subsystem and the HDRM. Given the reduced currents, the inert tether is removed and the tether length and thickness are reduced to 1.5 km and 30 µm. The deployment system structure mass is also reduced as direct consequence of the tether volume reduction. MC DK mechanical design is shown in Fig. 6.

Simulations with the mega constellation configuration has been run with tether length of 1.5 km and 2 km. Results shows that the bare tether DK with TE can deorbit from 1200 km polar orbits to 400 km orbit a 200 kg spacecraft in 3776 days ($L = 1.5$ km) and 2342 days ($L = 2.0$ km).

4.2. DK design based on LWT

LWT is of great interest since requires no active electron emitter. LWT would lead to a lighter EO DK and a fully autonomous MC DK (not requiring power from the host spacecraft). However, LWT represents a challenge in material science due to the strong requirement for the coating. In order to understand them, please note that, ignoring ohmic effects, tether equilibrium temperature is given by a radiative cooling and solar absorption balance

$$T = \left(\alpha_{abs} S_{Sun}/\pi\sigma_B\varepsilon_{em}\right)^{1/4}$$

with $S_{Sun} \approx 1.37\text{kW/m}^2$, $\sigma_B \approx 5.68 \times 10^{-8}\text{W/m}^2\text{K}^4$. Therefore, tether temperature is controlled by the ratio solar absorptance-to-thermal emittance ratio $\alpha_{abs}/\varepsilon_{em}$. Having a hot tether is crucial for triggering photoelectron and thermionic emissions in an LWT because they are mainly controlled by the ratio W/T (find details in Ref. [13]). For instance, the Richardson-Dushman law for the thermionic current is

$$J_{th} = -AT^2 exp\left(-W/k_B T\right) \tag{3}$$

with $A \approx 1.20 \times 10^6\text{A/m}^2\text{K}^2$. Therefore the main challenge of E.T.PACK is the development of a coating with large $\alpha_{abs}/\varepsilon_{em}$ and a low $W/k_B T$. For $J_{th} \approx 2.3\text{mA/m}^2$, a coated tether segment of dimensions 1km × 2.5cm could emit around 0.06A in a fully passive manner. Replacing the HC in the DK for EO by an LWT tether segment could save 10 kg (see Table 4). The deorbit time would be longer, anyway it will not be an issue because the LWT will not be limited by the expellant mass. In the case of the MC, the LWT would make the DK fully autonomous since no power from the host spacecraft is needed (see Table 5).

Fig. 7.

Fig. 8 reports the performances of LWT DK configuration for deorbiting EO satellites of mass 700 kg (black), 850 kg (blue) and 1000 kg (grey) and 200 kg MC satellites from 1200 km orbit (orange and green).

5. Conclusions

This work presents the design and performances of a Deorbit Kit for satellites of up to 1000 kg in orbit of up to 1200 km altitude. DK is bolt-on to the customer satellite before launch and activated at the end of the satellite operational life or in case of critical failure. Upon activation, DK autonomously de-tumbles the satellite, deploys a tether and carries out the satellite's de-orbiting. Two reference mission has been selected: Earth Observation satellites in the range of 700–1000 kg and telecommunication spacecraft of 200 kg. Three different configurations of DKs are presented based on bare tether and modern electron emission devices: heater-less hollow cathode (HC), thermionic emitter (TE) and low work-function tether (LWT). In all cases DK mass is maintained below 6% of the mass of the host satellite. DK with HC and TE, which are perfectly feasible with state-of-the-art components, present competitive performance and highlight the suitability of electrodynamic tethers for deorbiting satellites in an affordable manner. The analysis highlights the importance of the development of the LWT concept to have access to an even simpler and lighter solution.

Declaration of Competing Interest

The authors declare that they have no known competing financial interests or personal relationships that could have appeared to influence the work reported in this paper.

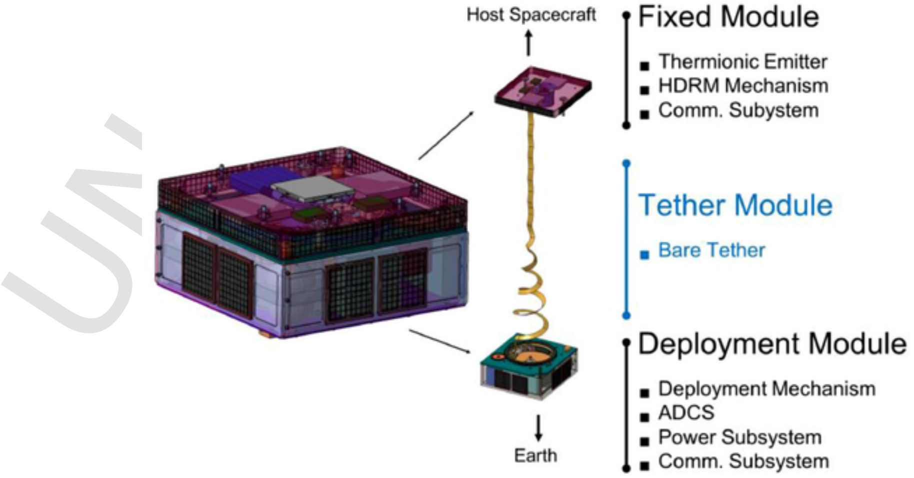

Fig. 6. MC DK design.

Table 5
LWT tether DK for EO satellites (left) and MC satellites (right).

Module	Mass	Comment	Module	Mass	Comment
Fixed	0.5 kg	Support structure.	Fixed	0.5 kg	Support structure.
Tether	12.1 kg	LWT with $L = 3km$, $w = 2.5cm$, and $h = 40\mu m$.	Tether	4.4- 5.9 kg	LWT with $L = 1.5 - 2km$, $w = 2.5cm$, and $h = 30\mu m$.
Deployment	13.5 kg	Including HDRM and magneto-torquers.	Deployment	7.2 kg	Including HDRM and magneto-torquers.
Total	**26.1 kg**		**Total**	**12.1–13.6 kg**	

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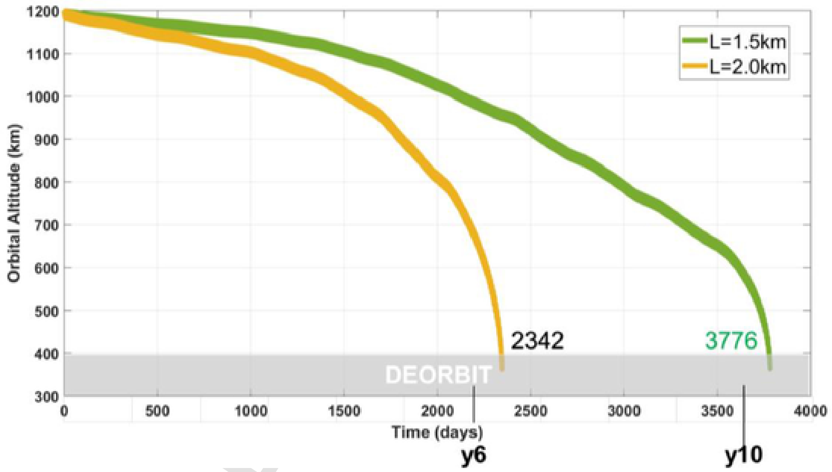

Fig. 7. MC DK performances.

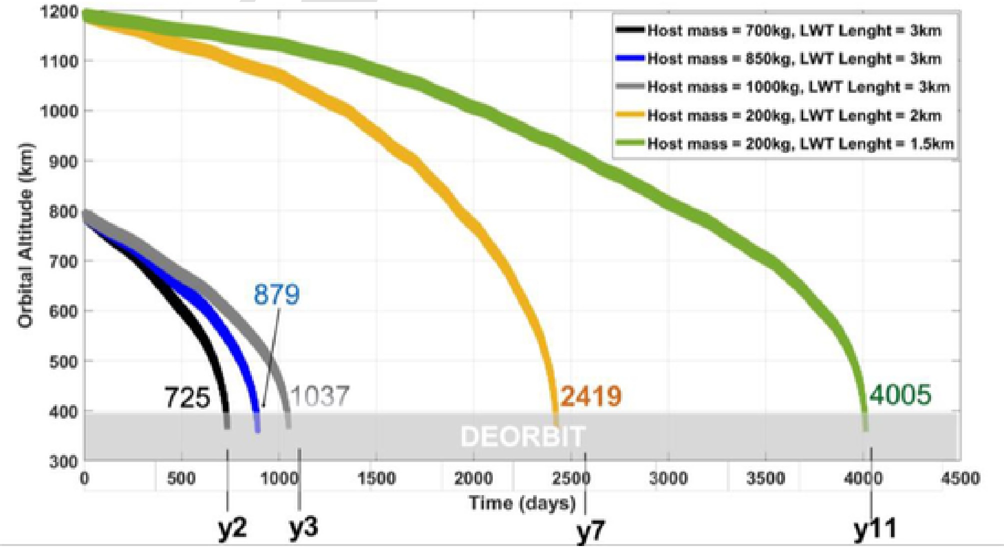

Fig. 8. Performances of LWT DK configuration for EO and MC.

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