ORBITAL PERFORMANCE OF SMALL SATELLITE DEORBITING KIT BASED ON ELECTRODYNAMIC TAPE/TETHERS

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ABSTRACT

Current plans for large LEO constellations envisage the launch and the transit of thousands of small-size satellites in clustered orbits. As recommended by international guidelines, spacecraft shall implement post mission disposal strategies, to mitigate the hazard they pose on the space debris environment. In particular, all new satellites in LEO are expected to deorbit within 25 years from their end of life.
Among the proposed deorbiting technologies, electrodynamic tethers appear to be a promising and effective option; in this context the European Commission is currently funding the project E.T.PACK – Electrodynamic Tether Technology for Passive Consumable-less Deorbit Kit in the framework of the H2020 Future Emerging Technologies FET Open program. The project focuses on the design of a disposal kit for LEO satellites, that can be activated at spacecraft end of life to perform autonomous re-entry.
In this work we investigated the orbital performance of the proposed disposal kit as a function of host spacecraft mass and for a number of representative orbits, inclusive of worst-case conditions. It is shown that the electrodynamic tether option can be attractive compared to deorbit systems based on traditional (e.g., chemical propulsion) and other alternatives (e.g., neutral-drag sails).

1 INTRODUCTION

The scientific community is growing concerns regarding the increasing number of man-made objects resident in near-Earth orbits. These objects are creating a hazard for the active spacecraft population; in case of large collision events the situation may worsen up to generation of artificial belts of debris in the most crowded orbits [1]. The recent introduction of large constellations of thousands of small satellites (e.g. [2][3]) will further affect the space environment. As reference, the Starlink constellation provider has launched in orbit 2335 satellites to March 30th, 2022, with plans to deploy up to 4100 spacecraft on orbit shells with altitudes between 540 and 570 km, as well as the approval to operate another 2,825 satellites in higher orbits between 1100 km and 1325 km [4]. While current international guidelines recommend a number of actions to mitigate the influence of new satellites on the environment (e.g.: the 25-years rule to deorbit all new satellites within 25 years since orbit injection, if their deployment orbit altitude is below 2000 km [4]), concerns are still arising on their effectiveness on controlling the growth of space debris population [6][7][8].

In this context, all major satellites providers are currently mowing towards a self-regulating approach to respect at least the 25-years recommendation. The most common strategies to ensure their vehicles disposal include the definition of interfaces for servicing and tugging and the implementation of end-of-life autonomous re-entry burns in case a propulsion system is on board. To date, the installation of dedicated deorbit devices is only envisaged in demonstration missions; however, it is expected that in the next decades the employment of such devices will become more common on commercial spacecraft.

Deorbit systems can be classified into passive technologies, including drag augmentation devices (DAD) and electrodynamic tethers, and active propulsion technologies that involve chemical and electric thrusters [9]. With respect to DAD, drag sails have been tested in space as disposal systems for low altitude orbits [10][11], with current studies aiming to develop a small kit to be installed as an independent module on spacecraft (ADEO - [12][13][14]). Propulsion-based disposal can implement chemical or electric thrusters and can be performed with a dedicated kit or with on-board systems (if existing) but shall overcome technological limitations, such as propellant leakage or degradation in case of long storage time and attitude control demands during disposal manoeuvres. Systems based on electrodynamic tethers appear to be a promising option as they overcome the limitations of traditional chemical and electric propulsion and can be implemented in a wider range of altitudes with respect to drag augmentation devices. Electrodynamic tethers collect ionospheric electrons from the plasma environment and re-emit them through a cathode (Bare Electrodynamic Tethers – BET, [15]) or a "Low-Work-Function" segment of the same tether by using thermionic and photoelectric effects (Low Work Function Tether – LWT [16][17]). In both cases, the resulting electric current flowing through the conductive tether generates a Lorentz drag force thanks to the interaction with the Earth's magnetic field that progressively decreases the orbit altitude of the satellite causing its re-entry in atmosphere.

In this context, 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), which aims at the development of a Deorbit Kit (DK) prototype based on electrodynamic tether technology [18]. In this paper the orbital performance of the E.T.PACK deorbiting kit is evaluated for three reference configurations (a large satellite in sun-synchronous orbit, a large constellation spacecraft at 1200 km of altitude and an in-orbit demonstrator small-class satellite at 600 km) and is compared with chemical propulsion and drag-sail devices. The paper is organized as follows. Section 2 better depicts the E.T.PACK project, the advantages of the electrodynamic tether disposal, and the proposed in-orbit demonstrator. Section 3 compares the deorbit performances of E.T.PACK kit, chemical propulsion, electric propulsion, and drag sails, showing that the former option is more attractive, in particular for small/medium satellites at high altitudes.

2 THE E.T.PACK PROJECT

The E.T.PACK project was funded in 2018 by the European Commission to develop the technologies for propellant-free tethered satellites deorbiting. The project consortium consists of six institutions among Spain, Germany, and Italy, led by the University Carlos III of Madrid. The members aim to: (1) investigate plasma physics theory and its application in electron emitters; (2) develop and demonstrate novel active materials and coatings for LWTs; and (3) increase the maturity of tether technologies up to the integration in a flight model (the Deorbit Kit Demonstrator shown in Figure 1). Recently, the European Commission funded through the Horizon Europe Program the EIC Transition project with acronym E.T.PACK-F that will continue the development of the prototype up to a flight test planned for 2025.

2.1 Advantages of the proposed approach

Electrodynamic tethers can be used for the deorbiting of satellites at the end of their operational life as they require limited on-board resources and can be deployed once the spacecraft need to be deorbited. The E.T.PACK consortium is aiming in developing a deorbiting kit whose mass would be less than 2% of the host spacecraft (with a minimum kit mass of 15 kg); in addition, the proposed kit will be able to deploy the tether autonomously, regardless of the failure of the host satellite.

The most common concerns regarding the use of space tethers for deorbit are related to: (1) the long time requested to perform the disposal with respect to propulsive manoeuvers, (2) the risk of failure due to the impact with space debris, and (3) the collision hazard they pose to other satellites due to their length. Recent investigations presented to the scientific community seem to mitigate such concerns: the employment of tape tethers helps both in reducing the deorbiting time [19], thus reducing significantly the Area×Time Product and in increasing the survivability to small space debris impact [20][21]. In addition, space tethers are active elements and their rate of descent can be controlled quite simply by switching on and off the current to perform collision avoidance manoeuvres [22]. Based on these considerations, tethers can be considered reliable systems for satellite disposal operations; a comparison with other deorbit solutions should not be affected by preconceived notions, but rather be based on the analysis of their performance.

Figure 1: Artistic representation of the Deorbit Kit Demonstrator

2.2 Deorbit Kit Demonstrator

The Deorbit Kit Demonstrator (DKD) prototype will be delivered in few months by the E.T.PACK consortium and will be the base for the in-orbit flight demonstration. It consists of a 12U CubeSat with a total mass less than 24 kg, to be launched into a circular orbit at an altitude of 600 km and a medium inclination. The DKD will deploy a 500-m-long tape tether (400 m of conductive metallic and 100 m of inert polymeric tether). The natural deorbit time of the system from this orbit is more than 15 years; the DKD is expected to re-enter in less than 100 days thanks to the EDT technology [23]. The DKD consists of 2 connected modules: the Deployment Mechanism Module (DMM), hosting the tether and its deployment mechanism, and the Electron Emitter Module (EEM), that hosts the Electron Emitter. Each module is completely independent with its own power, communication, data handling and attitude control subsystems.

3 ORBITAL PERFORMANCES

To investigate the orbital performance of the E.T.PACK kit, three reference configurations have been selected with different spacecraft orbits and parameters. For each configuration, the deorbit time has been compared considering the utilization of deorbiting kits based on E.T.PACK, on chemical propulsion, electric propulsion, and on drag sails. Results are compared in terms of the deorbiting kit mass fraction and deorbit time.

3.1 Reference configurations

The three reference configurations investigated in this work have been selected considering: (1) a potential customer in sun-synchronous orbit (e.g.: ESA Sentinel 2 and 3 family); (2) a large-constellation spacecraft (e.g. OneWeb, Starlink higher orbital shells); and (3) the proposed orbit for E.T.PACK DKD. In summary, the main data of the three configurations is shown in Table 1.

Table 1: Reference configurations main parameters

Reference Configuration	Description	Apogee altitude, km	Inclination, deg	Eccentricity	S/C mass, g
Case 1	Sun-synchronous S/C	800	98	0	1100
Case 2	Large constellation S/C	1200	55	0	150
Case 3	E.T.PACK DKD	600	51.5	0	24

The disposal operation is considered successful if the spacecraft orbit apogee is lowered under 400 km or if the perigee is lowered to 250 km: in both cases the natural decay due to atmospheric drag would complete the satellite deorbiting in weeks or few months, depending on the spacecraft area to mass ratio and solar activity.

To better compare the results, performances of E.T.PACK and drag sail deorbiting have been evaluated for high and low solar activity. Simulations have been performed considering starting date respectively on January 1st, 2000, and January 1st, 1996.

3.2 Deorbiting performance

The deorbiting performance of E.T.PACK has been evaluated with a campaign of simulations carried out by Universidad Carlos III de Madrid with the software BETsMA [24]. The reference configurations parameters have been used as input for the simulations to evaluate the orbit decay evolution. Six different scenarios have been investigated, considering high and low solar activity, the utilization of a Low Work Function Tether (LWT) or a Bare Electrodynamic Tether (BET) with a cathode, and in the latter case with the implementation of a current limiter. The list of scenarios is shown in Table 2.

Table 2: Simulation scenarios and parameters.

Scenario	Solar activity	Tether	Current limiter
A	High	BET	Active
B	Low	BET	Active
C	High	LWT	/
D	Low	LWT	/
E	High	BET	Inactive
F	Low	BET	Inactive

The simulated tether length is 2 km for the larger spacecraft (Cases 1 and 2) and 400 m for the IOD (Case 3); the main parameters of the tether are listed in Table 3.

Table 3: Simulated tether main parameters.

Thickness, μm	Width, cm	Emissivity	Absorptivity	Work function, eV (scenarios C and D)	Cathode potential drop, V (scenarios A, B, E, and F)
50	2.5	0.06	0.5	1.4	-20

Table 4 shows the deorbit time (in hours) for each scenario for the three reference configurations. Details on the orbit altitude evolution for scenarios A, C, and E (high solar activities) are depicted in Figure 2, Figure 3, and Figure 4, respectively. In general, LWTs deorbit times are longer than those for BETs, and high solar activity helps in further reducing the disposal times. As expected, Case 1 (sun-synchronous S/C, 1100 kg) requires the longest time to lower the orbit to 400 km (from 1.7 years for BET with inactive current limiter at high solar activity to 5.5 years for the LWT in low solar activity). On the contrary, the IOD (Case 3) is expected to reach the altitude of 400 km in a time ranging between 13 days (BET, Scenario E) and 321 days (LWT, Scenario D).

Table 4: Deorbiting time in hours for the selected scenarios and configurations/cases.

\	Tether length	A *BET*	B *BET*	C *LWT*	D *LWT*	E *BET*	F *BET*
Case 1	2 km	19926 (2.3 y)	32973 (3.8 y)	35939 (4.1 y)	48068 (5.5 y)	15035 (1.7 y)	27783 (3.1 y)
Case 2	2 km	2324 (96 d)	4243 (177 d)	7837 (327 d)	13816 (576 d)	1337 (56 d)	3496 (146 d)
Case 3	400 m	406 (17 d)	895 (37 d)	1446 (60 d)	5787 (241 d)	321 (13 d)	831 (35 d)

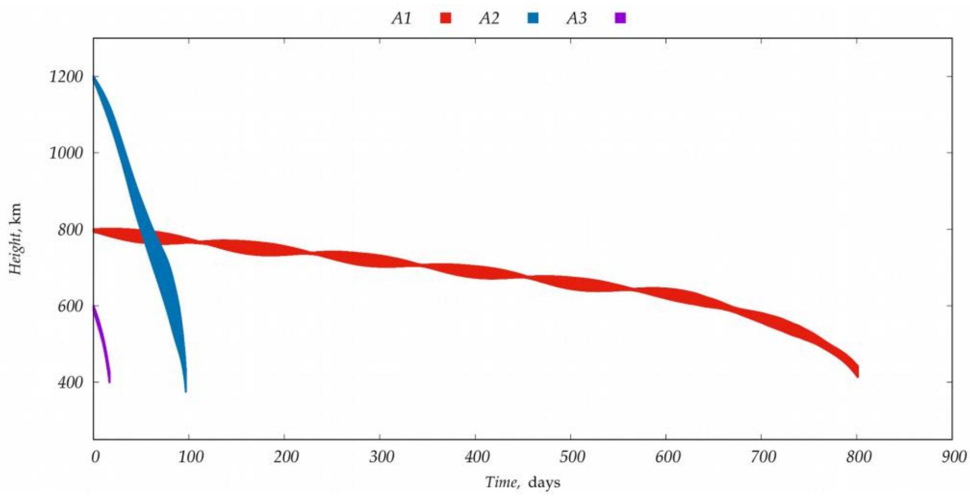

Figure 2: Apogee altitude evolution for the three configurations/cases with E.T.PACK deorbiting kit: Scenario A (High solar activity / Bare Electrodynamic Tether / Current Limiter Active).

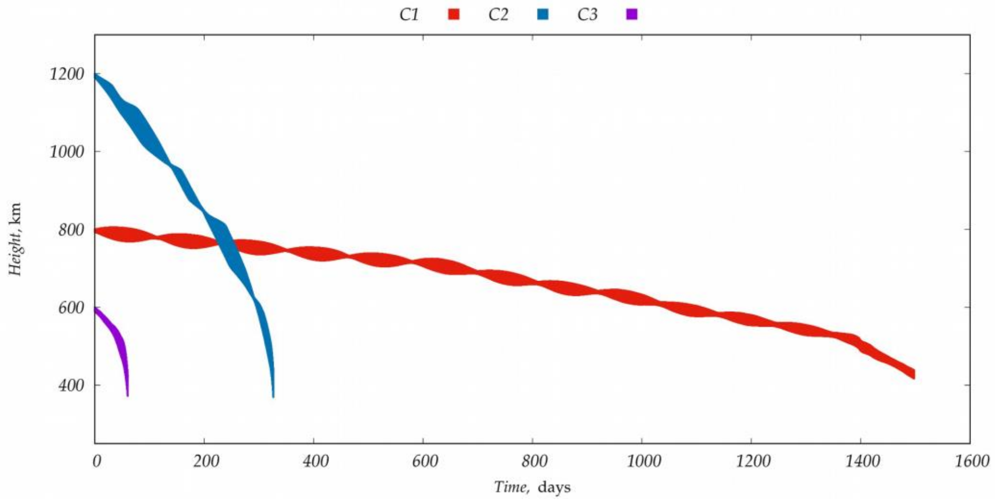

Figure 3: Apogee altitude evolution for the three configurations/cases with E.T.PACK deorbiting kit: Scenario C (High solar activity / Low Work Function Tether).

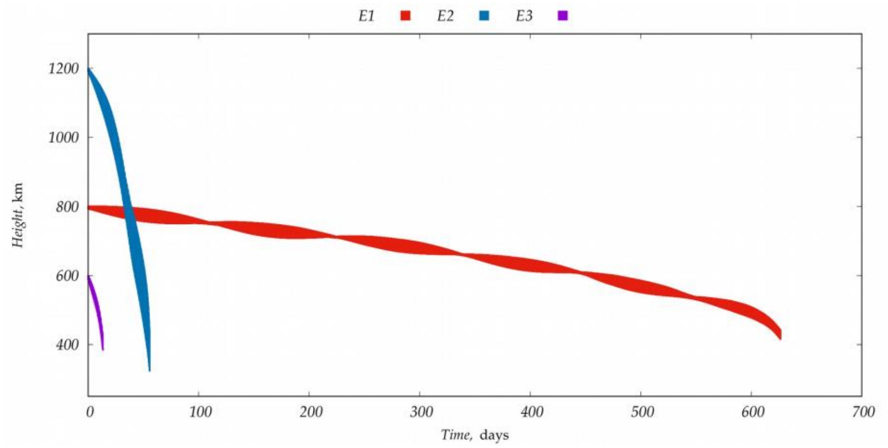

Figure 4: Apogee altitude evolution for the three configurations/cases with E.T.PACK deorbiting kit: Scenario E (High solar activity / Bare Electrodynamic Tether / Current Limiter Inactive).

3.3 Comparison with chemical propulsion

Disposal manoeuvres with chemical propulsion have the advantage of shorter operational time (in the range of one orbital period). With respect to tether and drag sail deorbiting strategies requiring dedicated modules, propulsive manoeuvres are usually performed with hardware already on-board the S/C and require only to save enough propellant for disposal at the end of the operational life. Two different scenarios have been considered, a Hohmann manoeuvre to lower the S/C orbit to an altitude of 400 km and a single-impulse manoeuvre to lower the perigee to 250 km. In both cases the propellant mass required to perform the disposal manoeuvre is calculated, considering values of the specific impulse of 350 s and 200 s. Table 5 shows the main results for the three configurations analysed. It can be noted that the Hohmann manoeuvre to 400 km requires less propellant with respect to the single impulse to 250 km. The mass budget for the propellant is not negligible, with about 6% of the S/C mass (81 kg) for Case 1 and more than 11% (18 kg) for Case 2.

Table 5: Orbit lowering to 400 km and perigee lowering to 250 km: required ΔV and propellant budget

Case	**Δv_{400}, m/s**	**m_{prop} (I_{sp}=350 s), kg (%)**	**Δv_{250}, m/s**	**m_{prop} (I_{sp}=200 s), kg (%)**
1	216.7	67.34 (6.1 %)	149.9	81.01 (7.3 %)
2	415.7	17.12 (11.4 %)	246.7	17.74 (11.8 %)
3	110.7	0.76 (3.2 %)	97.8	1.16 (4.9 %)

3.4 Comparison with electrical propulsion

Electrical propulsion allows to perform non-impulsive manoeuvers with high-specific-impulse and low-thrust actuators. Different technologies belong to the family of electric propulsion [9]; among them, plasma thrusters, arcjets, ion thrusters, and Hall thrusters, each of them with different application ranges and performances. For sake of simplicity, in this analysis a thruster with average performance parameters is considered; the selected values are extrapolated from online datasheets of available products. For all configurations, the thruster specific impulse is 1000 s; the power consumption and the thrust are respectively 0.6 kW and 5 mN for cases 1 and 2 and 0.06 kW and 0.5 kN for case 3.

Table 6: Electrical propulsion thrusters parameters

Reference Configuration	S/C mass, kg	Specific impulse, s	Power, kW	Thrust, mN
Case 1	1100	1000	0.6	5.0
Case 2	150	1000	0.6	5.0
Case 3	24	1000	0.06	0.5

For the three cases, the deorbiting time is calculated considering a continuous thrust always direct along the tangent of the orbit. The orbit decay can be calculated as:

$$\frac{dh}{dt} = \frac{2(R_e + h)^2}{\mu} \cdot \frac{F_T}{m_{sat}} \sqrt{\frac{\mu}{R_e + h}} \tag{1}$$

where R_e and μ are Earth radius and gravitational parameters, h is the orbit altitude, F_T the thrust, and m_{sat} the satellite mass [26]. The manoeuvre is considered completed when the orbital altitude reaches 400 km, for compatibility with previous simulations; in fact, it shall be mentioned that while a tethered satellite has a large area and may re-enter in few days from such altitude, a spacecraft with average ballistic coefficient may still remain in orbit for many months at this altitude.

The electric propulsion system mass m_e can be estimated using empirical relations available in literature, considering the propellant mass m_p, the inert mass m_i, and the power generation mass m_{PW}. In first approximation, considering that the propellant mass flow rate is related to the thrust and the specific impulse with the relation $\dot{m} = F_T/(g_0 \cdot I_{sp})$, the system mass can be calculated as:

$$m_e = m_p + m_i + m_{PW} = m_p(1 + k_i) + m_{PW} = \dot{m}\, T\, (1 + k_i) + \alpha P \tag{2}$$

where T is the deorbiting time, k_i is the inert mass fraction, α is the inverse specific power (the mass requested to produce 1 unit of power), and P is the power consumption. Typical values for k_i and α are respectively of 0.12 and 20 kg/kW [26].

Table 7 shows the re-entry time for the three cases considered, as well as the propulsion system mass and mass fraction. It shall be mentioned that in this analysis the mass budget considers a propulsion system fully dedicated to the re-entry manoeuvre; in case the thruster is designed to operate also during the spacecraft life, only the propellant mass required for the manoeuvre should be considered in eqn. (2). It can be noted that the re-entry time is in general comparable to the result obtained for tethered systems. In cases 1 and 2 tether systems can be competitive in terms of system mass: the mass budget for electric propulsion is larger than the mass allocated for E.T.PACK deorbiting kit (2% of the spacecraft mass, to a minimum of 15 kg). Lastly, it should be mentioned that employing electrical propulsion requires the host spacecraft to always maintain a precise attitude control and a constant power supply during the whole re-entry manoeuvre, that implies an additional propellant mass consumption and an added control complexity.

Table 7: Orbit lowering to 400 km with electrical propulsion: manoeuvre time and propulsion system budget

Case	$t_{re\text{-}entry}$, hours	Propulsion system mass	mass fraction, %
1	13245 (1.5 y)	39.2	3.6
2	3467 (144 d)	19.1	12.7
3	1476 (61 d)	1.5	6.3

3.5 Comparison with drag sail

Drag sails have been proposed and tested in a wide range of missions [10][11]. Among the most recent advancements, the ADEO drag sail family is currently under development under the direct supervision of the European Space Agency by a large team of European stakeholders [12]. To date, two different size sails have been proposed [13], whose parameters are reported in Table 8. The two models are designed respectively for microsatellites and an in-orbit demonstration [14] (ADEO-M) and larger spacecraft (ADEO-L).

Table 8: main parameters for ADEO sails [13]

Sail model	Area, m^2	Mass. kg
ADEO-L	25	7-10
ADEO-M	2.5	1.5

These two sail models have been assessed to deorbit the three proposed configurations, with the ADEO-L employed for Cases 1 and 2, and the ADEO-M for Case 3. Deorbit simulations both with high and low solar activities have been performed with the NASA Debris Assessment Software (DAS 2.0.2), considering as simplifying hypothesis no spacecraft tumbling. Despite that assumption, results presented in Table 9 show that the large sail is not able to deorbit the first two configurations/cases in 25 years or less, while results are more promising for Case 3 (from 8 months to 3.4 years). Figure 5 shows the orbit decay evolution for high solar activity for both Case 1 and Case 3.

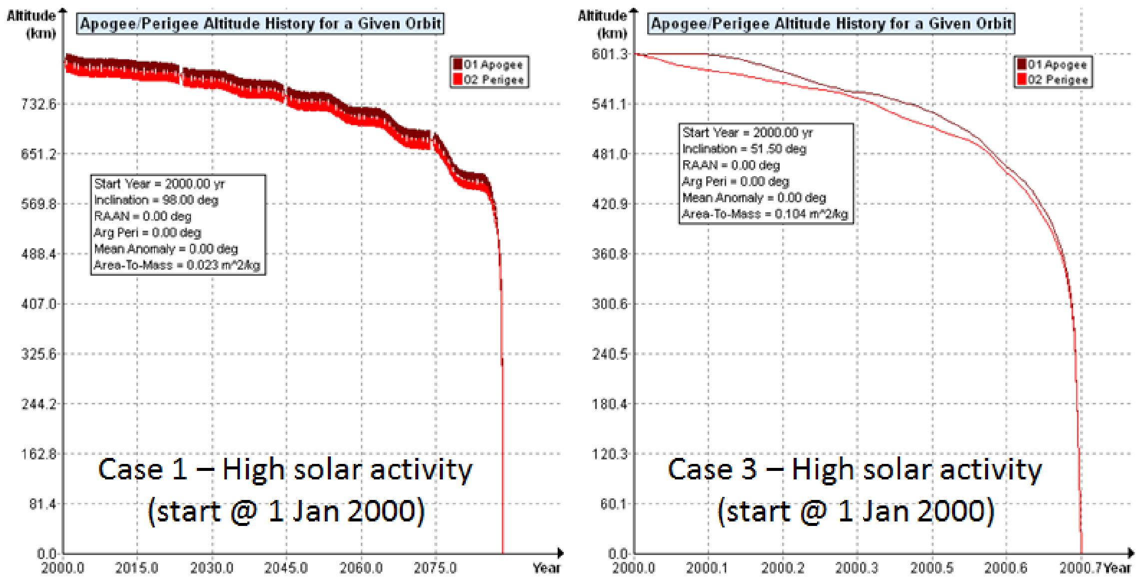

Figure 5: Orbit decay (high solar activity) with ADEO drag sail for case 1 (left, 88 years for re-entry) and case 3 (right, 8.4 month for re-entry).

Table 9: Area to mass ratio, re-entry time, and mass fraction for ADEO sail applied to the selected configurations/cases

Case	**Sail model**	**A/m ratio, m²/kg**	**$t_{re\text{-}entry}$, years (solar max)**	**$t_{re\text{-}entry}$, years (solar min)**	**mass fraction, %**
1	ADEO-L	0.0227	88	90	0.6-0.9
2	ADEO-L	0.1667	>100	>100	4.7-6.7
3	ADEO-M	0.1042	0.7	3.4	6.3

To better assess the drag sail technology, a scaling formula to adapt the ADEO sail to the proposed configurations/cases is reported in Eq. (3), tuned on the mass and area parameters of the ADEO–L and ADEO–M models. Relating the sail module mass to the area allows to better compare the sail technology with the electrodynamic tether for similar disposal kit masses (up to 2% of the whole S/C mass, with a minimum of 15 kg); it shall be underlined that the proposed formula cannot be applied to Case 3, as with a mass fraction of 2% the estimated sail area would be negative.

$$A_{SAIL} = 3.36\, m_{SAIL} - 2.54 \tag{3}$$

Table 10 shows the scaled sail data and the simulated re-entry time for both high and low solar activities. It can be noted that for the first configuration the re-entry time is consistently lower but still over 25 years; details on the orbit decay are depicted in Figure 6. For Case 2 the re-entry time is again longer than one century, indicating that the spacecraft orbit is too high for this re-entry technology.

Table 10: Area to mass ratio, re-entry time, and mass fraction for the scaled sail applied to selected configurations; for Case 3 miniaturization to 2% of mass fraction is not feasible nor significative.

Case	**Scaled sail Area**	**A/m ratio, m²/kg**	**$t_{re\text{-}entry}$, years (solar max)**	**$t_{re\text{-}entry}$, years (solar min)**	**mass fraction, %**
1	71.4	0.0649	34.3	37	2
2	7.5	0.0503	>100	>100	2

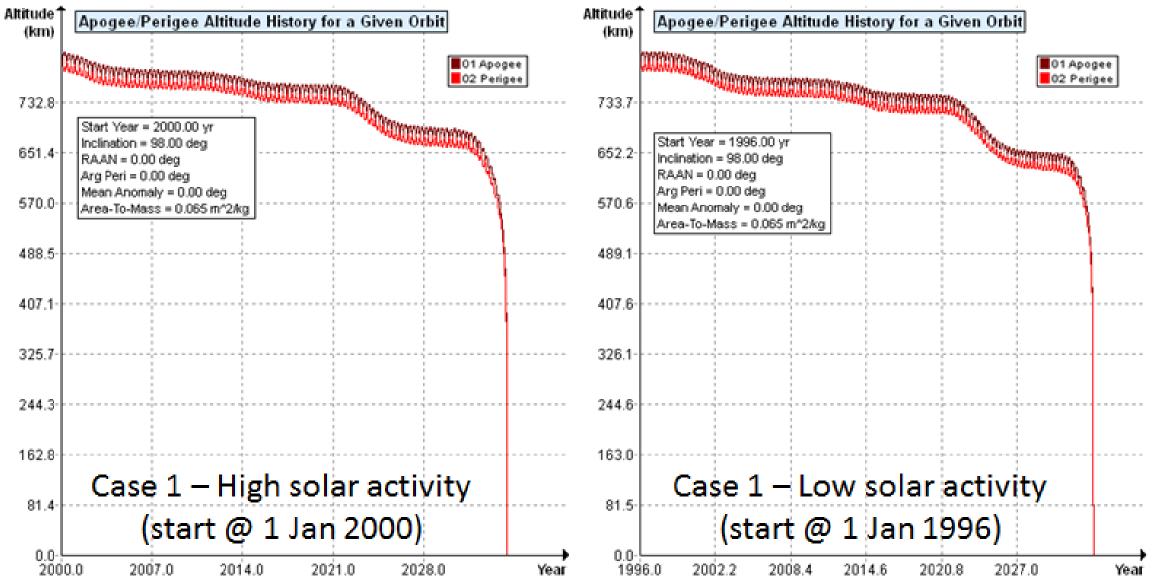

Figure 6: Orbit decay for high (left) and low (right) solar activity with scaled solar sail for case 1; in both scenarios the re-entry time is longer than 30 years.

The aforementioned data indicates that the drag sail performances are out-performed by E.T.PACK for both low and high solar activity.

4 CONCLUSIONS

In this paper the E.T.PACK kit has been presented and compared with propulsive systems and drag augmentation devices. The electrodynamic tether technology is promising with respect to the other options thanks to: (1) the reduced mass budget (under 2% of the host spacecraft, with a minimum of 15 kg), that makes it more attractive than chemical propulsion systems; (2) the wide range of operational orbits (up to the highest large constellations orbital altitudes that are unsuitable for drag sails); and (3) the possibility to perform collision avoidance manoeuvres through current switching during disposal. A suitable competitor of electrodynamic tethers is electric propulsion if the system were already present on-board the host spacecraft, as this system shows similar performances for the selected operational scenarios. However, in these cases, tether systems are still more attractive as they do not require either constant power supply or attitude control of the host spacecraft during the whole deorbiting. In particular, the E.T.PACK kit is suitable for high-altitude, small-size, large constellation spacecraft, that would require significant amounts of chemical propellant (more than 10% of their mass) and cannot be deorbited with drag sails.
In the next months further investigation will be conducted on the E.T.PACK performance, including a more detailed comparison with electric propulsion systems. In parallel, the development of the in-orbit demonstrator will lead to the E.T.PACK DKD flight verification.

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