Use of Electrodynamic Tethers for Satellite End of Life Deorbit Maneuvers

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This paper provides an overview of the principles of Electrodynamic Tether use for spacecraft propulsion. As a demonstration of tether effectiveness as a propulsive device, a sample mission of the Orbcomm Generation 2 constellation was examined for the potential of Electrodynamic Tethers to be used as a low thrust propulsive device for deorbiting of the spacecraft. Tethers were found to be able to deorbit the spacecraft well under the lifetime limits with potential mass savings.

I. Nomenclature

A = area

a = semi-major axis [km]

B = magnetic field vector [nT]

C_d = drag coefficient

d/dt = time derivative

E = electric field vector [N/C]

EDT = electrodynamic tether

F = force vector

h = angular momentum [N*m*s]

i = inclination [deg]

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I = electric current [Amps]

L = Tether Length [meters]

m = mass [kg]

M_0 = mean anomaly [deg]

n = mean motion [deg/sec]

p = semi-parameter [km]

R = orbital radius vector [km]

r = orbital radius magnitude [km]

R_{elec} = electrical resistance [ohms]

t = time [sec]

T = orbital period [sec]

u = Argument of Latitude [deg]

V = volt [Vdc]

v = orbital velocity magnitude [km/sec]

θ_{GST} = Greenwich sidereal time [deg]

v = true anomaly [deg]

ρ = atmospheric density [kg/m^3]

Ω = Right Angle of the Ascending Node [deg]

Ω_{elec} = electrical resistivity [ohms/m]

ω = Argument of Perigee [deg]

II. Introduction to Orbital Debris

A. What is Orbital Debris?

Orbital debris is any manmade object in orbit. In a broader sense, orbital debris can be taken to include all objects in earth orbit, both man-made and natural. This would expand the definition to include micrometeoroids that have been caught in orbit. This broader definition would include all of the following objects:

· Rocket Fairings
· Rocket Bodies
· Satellites

· Sensor Covers
· Tools
· Micro-meteors
· Paint Chips
· Collision Debris
· Iridium 33/Cosmos 2251 Collision (2009)
· Destroyed Satellite Debris
· FENGYUN 1C ASAT test (2007)

B. Danger to Current Spacecraft

The danger of the orbital debris environment is in the risk of impact with operational space craft. Even though most of the debris is very small, the very high orbital velocities mean that they still have a very high kinetic energy that can do significant damage.

The earliest suspected break up due to a collision with orbital debris was Kosmos 1275. On July 21, 1981 the Kosmos-1275 broke up into 90 trackable pieces (with thousands more un-trackable pieces likely). The investigation of the break up concluded that the most likely cause of the break up was an impact with a piece of untracked orbital debris. Since they could not recover the spacecraft, the exact cause could not be found. The conclusion that it was destroyed by debris was based on the fact that Kosmos-1275 had no propellant onboard (it was gravity gradient stabilized) and was located in an orbital regime that had a high concentration of orbital debris. Unfortunately, impacts like this are not only a result of orbital debris but actually make the problem much worse [1].

The Space Shuttle Program provided a more visible look into the dangers of the orbital debris environment. During the shuttle program the front windows were replace 46 times due to damage cause by impacts with paint chips and other small debris. The photo below shows the damage caused by a paint chip that impacted the Challenger window during STS-11. The paint chip was estimated to be only .02 mm in diameter but caused a pit in the window 100 times its size.

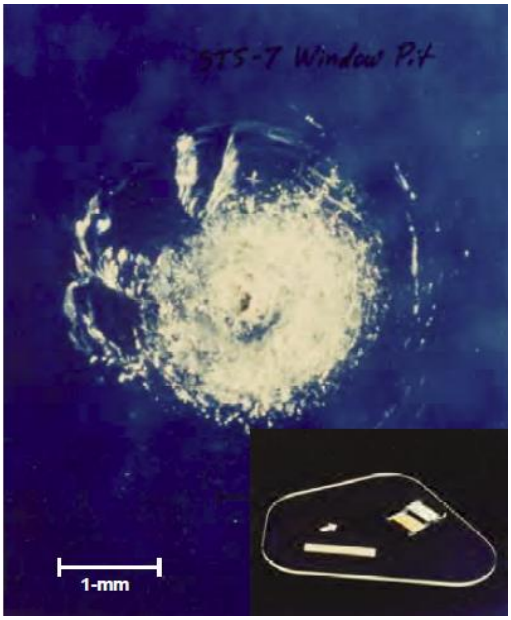


Figure 1. Challenger Window damage due to impact with paint chip during STS-7 [2]

C. Orbital Debris Environment

Since the launch of Sputnik on October 4 1957, the orbital debris environment has been getting worse at an exponential rate. The total number of objects has increased exponentially to nearly 18000 trackable objects today. Figure 2 below shows the growth of the number of trackable objects (note: the chart does not include several major recent events such as the FENGYUN 1C ASAT test in 2007 and the Iridium 33/Cosmos 2251 collision in 2009 that have contributed 3000-4000 of the 18000 total). The total number alone does not give the whole picture however since certain orbits are far more polluted than others. The two aforementioned events have created two basically inhospitable orbits since there is such a high concentration of debris. As time moves on, the debris clouds diffuse and some of the debris reenters the Earth's atmosphere making these orbits usable (albeit at higher risk) again.

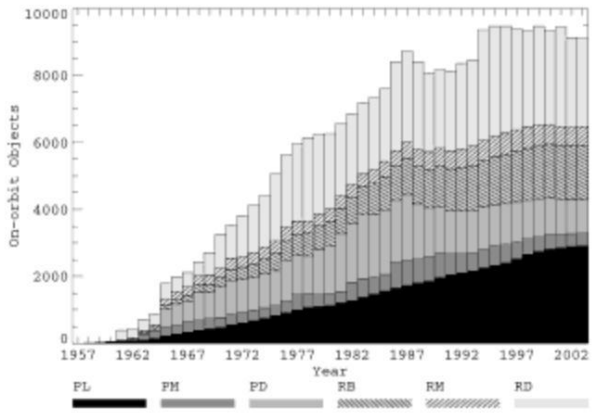


Figure 2. Historic Evolution of Trackable on Orbit Objects [3]

To combat the increase in the number of debris objects, the U.S. Government has enacted policies that require satellite designers and operators to take steps to minimize the creation of new debris. The most significant of these regulations is the 25 year limit on orbital decay of debris (which includes satellites after their operation mission) [4].

III. Mitigation of Orbital Debris Creation Using Electrodyanmic Tethers

Since many satellites cannot deorbit naturally within a reasonable timeframe, it is necessary to perform a maneuver to accelerate the deorbit process. The traditional method to do this is to use a chemical thruster to provide a quick deltaV that causes the satellite to reenter the Earth's atmosphere. This method generally requires significant propellant mass to perform this maneuver. Since it as end of life maneuver any mass required for deorbit operations is compounded by the launch and fuel costs to get that mass into orbit and perform station keeping holding the extra mass. For this reason, it is advantageous to come up with new lighter options for performing deorbit maneuvers.

Electrodyanmic Tethers (EDT) provides the potential for low mass propellant-less maneuvers. In the general sense, EDTs convert electric and magnetic energy into kinetic energy that is used to provide a thrust to the spacecraft. Depending on the configuration, EDTs can be used to provide a boosting or deboosting (i.e. orbit lowering) force.

EDTs work creating a Lorentz force ($F = E\ x\ B$) by interacting with the Earth's magnetic field. A current is induced through a long conductive tether (current and tether length depends on the size of the spacecraft and force desired) inducing an electric field. This field then interacts with the Earth's magnetic field to generate a propulsive

force [5]. The net force is determined by the strength of the current, the length of the tether and the strength of the local magnetic field. Eq. (1) below shows the force equation for an electrodynamic tether:

$$F_{EDT} = \frac{I*L}{R} * R_{RSW}\ x\ B \qquad (1)$$

I is the tether current (+ = zenith pointing current), L is the tether length, R is the spacecraft radius vector in the RSW frame (satellite body frame: radial, in-track, cross-track), and B is the magnetic field vector. Based on this equation, it is shown that EDTs can be operated in a boost or deboost mode. Figure 3 below shows the two modes and the basic electrical and magnetic field parameters that are used for them. In the deboost mode, the current must flow in the zenith (radial, away from Earth) direction. Since the electron flow stems from the Satellite side of the tether, this mode can be driven either actively (i.e. using spacecraft power) or passively by collecting electrons from the ionosphere and using an electron emitter at the end mass to reemit the electrons. By placing the electron emitter at the spacecraft end of the tether, the current direction can be reversed. This causes the cross product to yield a force in the velocity vectors direction boosting the satellite into a higher orbit.

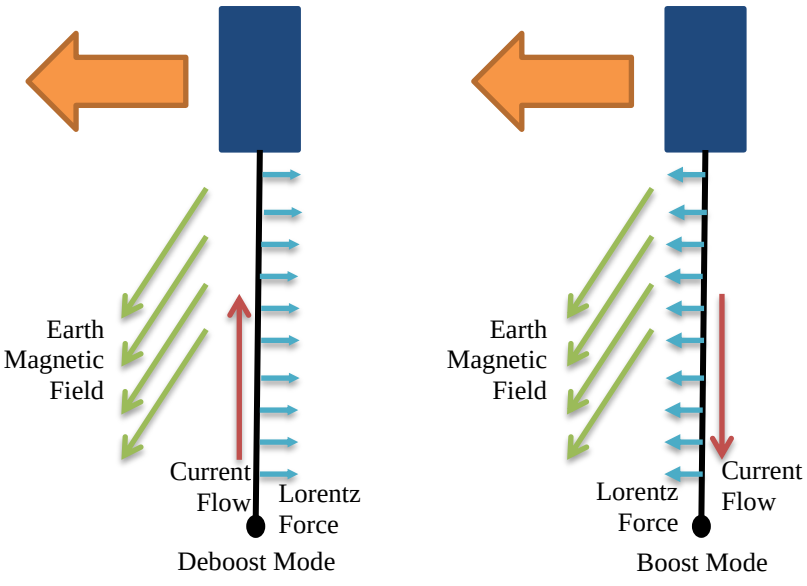


Figure 3. Diagram showing a simplified model of EDT operation in Deboost and Boost Modes.

When using an EDT in deboost mode, the operation can be carried out without actively driving a current. In this passive operational mode, the top half of the tether is un-insulated. As the bare conductive tether is dragged through the ionosphere, it picks up electrons. This induces a negative charge to build up on this half of the tether. Instead of an inert endmass, an electron emitter is placed at the end of the tether. The use of an electron emitter allows the collected electrons at the other end to flow down the tether and be reemitted into the atmosphere. Plasma waves in the atmosphere complete the circuit creating a net current along the wire from endmass to satellite (by convention current is defined as the flow of positive so it is in the opposite direction of electron flow).

An additional advantage of using an EDT is the fact that the long tether with a mass on the end provides gravity gradient stabilization for the spacecraft. This maintains the tether/spacecraft orientation holding the propulsive forces in the correct direction. This would reduce or eliminate (depending on the pointing requirements of the spacecraft) the need for additional attitude control systems including thrusters and momentum wheels.

Since EDTs do not consume any propellant, they have significant advantage over conventional chemical and electric propulsion systems. Since Mdot is zero, their I_{sp} is infinite. As long as the tether gets the current flow, it will provide a propulsive force with no mass usage. This effectively eliminates the current lifetime limitations for satellites based on the propulsions systems ability to provide deltaV.

While EDTs have in design are very simple, there are some additional considerations that must be made with the overall orbit and spacecraft design that must be accounted for it they are to be used. EDTs are less effective at high inclinations. Due to the angle between the spacecraft velocity vector and the magnetic field vector when at high latitudes, there is increased cross track forces at the expense of propulsive force. This causes a coupling of forces that means that the satellite is unable to impart a pure boosting force. There will always be a cross track that alters the orbital parameters while boosting or deboosting. These cross track forces causes the inclination of the orbit to change (direction depends on boost/deboost mode) while the EDT is in use. In addition, the high voltages and currents induced across the tether can cause electrical damage to the spacecraft and/or tether system. While not a major problem, it does present an issue that must be addressed during design to ensure the safety of the tether and spacecraft from electrical arcing. This issue was shown during the TSS-1 (Tethered Space Satellite) experiment performed on the Space Shuttle. The high voltage differential between the tether and its attachment hardware caused an arc across the current regulator. The resulting current across the tether caused it to melt breaking the tether.

IV. Analysis of an EDT system for Deorbit Maneuvers

Since EDT forces are small relative to the other orbital forces, they cannot be treated as an instantaneous deltaV. As such they are treated as small force perturbations that also vary throughout the orbit. For these reasons, analysis of an Electrodynamic system becomes coupled propulsion and orbits problem.

To determine the orbit changes as the tether is used, the Gaussian Variance of Parameters (VOP) method was used accounting for non-conservative affects [6]. As such we start with the following equations:

$$\frac{da}{dt} = \frac{2}{n * \sqrt{1-e^2}} * \left[e * \sin(v) * F_R + \frac{p}{r} F_s \right] \tag{2}$$

$$\frac{de}{dt} = \frac{\sqrt{1-e^2}}{n * a} \left[\sin(v) * F_R + \left(\cos(v) + \frac{e + \cos(v)}{1 + e * \cos(v)} \right) * F_s \right] \tag{3}$$

$$\frac{di}{dt} = \frac{r * \cos(u)}{n * a^2 * \sqrt{1-e^2}} * F_W \tag{4}$$

$$\frac{d\Omega}{dt} = \frac{r * \sin(u)}{n * a^2 * \sqrt{1-e^2} * \sin(i)} * F_W \tag{5}$$

$$\frac{d\omega}{dt} = \frac{\sqrt{1-e^2}}{n * a * e} \left[-\cos(v) * F_R + \sin(v) * \left(1 + \frac{r}{p} \right) * F_S \right] - \frac{r * \cot(i) * \sin(u)}{h} * F_W \tag{6}$$

$$\frac{dM_0}{dt} = \frac{1}{n * a^2 * e} \left[(p * \cos(v) - 2 * e * r) * F_R - (p + r) * sind(v) * Fs \right] \tag{7}$$

The orbit may be propagated by first calculating each of these terms and then applying them over a period of time. In these formulas the forces is a specific force (i.e. force/mass) in the RSW (Vehicle frame R=radius direction, S = in-track direction, W = cross-track direction that completes the right hand rule. Although not explicitly determined, angular momentum *h*, argument of latitude *u*, and mean motion *n* can be found using the following equations

$$h = \sqrt{\mu * a * (1 - e^2)} \tag{8}$$

$$u = (M_0 + \omega) \tag{9}$$

$$n = \sqrt{\frac{\mu}{a^3}} \tag{10}$$

Since the force is dependent on the altitude, the solution must be calculated for a time interval and then iterated. For this calculation, this threshold was assumed to be 180km altitude. Since the drag force is small and does not vary with altitude very rapidly, a time of 10 orbits was chosen for each iteration. This allows for decent resolution while not overloading the computer system with too many data point.

At each time step, the EDT force must be recalculated. Since the force vector direction varies in the RSW frame the simplifying assumption of zero radial and cross track functions could not be made. As shown earlier, the Lorentz force is determined by the following equation:

$$F_{Lorentz} = \frac{I*L}{R} * R_{RSW}\ x\ B \tag{11}$$

where I and L are the current and length of the tether. The Length was varied to determine its effect on the effectiveness of using EDTs. When used in the passive mode, the current I is function of the voltage across the wire as shown below in Eq. () below.

$$I = V/\Omega_{elect} \tag{12}$$

where the voltage V is a function of the tether length and ionosphere properties. For purposes of this analysis, the ionosphere was assumed to be constant yielding a function for voltage as found in Eq. (18) below. The resistivity Ω_{elect} is determined by the resistance R of the tether material as well as its length and cross-sectional area A_{tether}.

$$V = .00002 * L \tag{13}$$

$$\Omega_{elect} = \frac{R_{elec}*L}{A_{tether}} \tag{14}$$

Additionally in Eq. () above, R_{RSW} is the radius vector transformed in the RSW coordinate frame as shown in Eq. (20) below. Lastly, B is the magnetic field vector what was modeled using a simplified set of equations shown below in Eq. (21).

$$R_{RSW} = \begin{matrix} cos(-\Omega) & sin(-\Omega) & 0 \\ -sin(-\Omega) & cos(-\Omega) & 0 \\ 0 & 0 & 1 \end{matrix} * \begin{matrix} 1 & 0 & 0 \\ 0 & cos(-i) & sind(-i) \\ 0 & -sin(-i) & cos(-i) \end{matrix} * \begin{matrix} cos(-[\nu+\omega]) & sin(-[\nu+\omega]) & 0 \\ -sin(-[\nu+\omega]) & cos(-[\nu+\omega]) & 0 \\ 0 & 0 & 1 \end{matrix}^{-1} * R_{IJK} \tag{15}$$

$$B = \begin{matrix} -2 * \frac{M}{R^3} * sin\ \omega + \nu \\ \frac{M}{R^3} * cos\ \omega + \nu \\ \frac{M}{R^3} * cos\ i \end{matrix} * sin\ i * \frac{1T}{10^9 nT} \tag{16}$$

where *M* is the Earth's magnetic dipole moment and *R* is the radius vector magnitude (value is the same for all reference frames). The last term multiplied by the entire vector is a unit transformation from Nano-Tesla to Tesla.

Putting all of this back together yields the equations for the normal force using an EDT in the RSW coordinate frame. Since drag force still exists, the drag term is still seen in the in-track term.

$$F_R = \frac{F_{Lorentz,R}}{mass} \tag{17}$$

$$F_S = \frac{F_{Lorentz,S}}{mass} - \frac{1}{2} * \rho * v^2 * C_d * Area \tag{18}$$

$$F_w = \frac{F_{Lorentz,W}}{mass} \tag{19}$$

Once the forces are calculated, the orbit can be propagated forward in time by the given 10 orbits using the following equation:

$$C_t = C_{t-1} + \frac{dC}{dt} * \Delta t \tag{20}$$

$$\Delta t = 10 * T \tag{21}$$

where *C* is replaced by each of the parameters captured in Eq. (1) though Eq. (6) and T is the orbital period. This entire process is repeated for each time step until the altitude reaches a minimum value at which the drag becomes the dominant force. For this analysis, this point was assumed to be 180km.

V. Sample Mission

To demonstrate the effectiveness of an electrodynamic tether at providing the deorbit maneuver, the Orbcomm Generation 2 (OG2) satellite constellation was used as a demonstration. This is an 18 satellite constellation of

142kg spacecraft located in a 650km circular orbit. Under natural drag conditions, these satellites do not meet deorbit time requirements so they currently use a deorbit burn to accelerate their reentry. To provide a baseline, the deltaV of this burn was calculated as 183m/s Using Tsiolkovsky's rocket equation, the mass of propellant required for this maneuver was calculated to be 14.59kg.

A. Tether Analysis

In looking at using EDTs as a propulsion system it is important to realize that the deltaV required to complete a maneuver is going to be significantly higher than using a chemical propulsion system and Hohmann transfers. As a low force propulsion method, the deltaV required to deorbit the spacecraft can be estimated as the difference in orbital velocities between the final and initial orbit. Since aerodynamic drag forces will become extremely dominant at altitudes below 180km this was selected as the target altitude. This yields a deltaV requirement of 265.2m/s. This is a 45% increase in deltaV over the traditional chemical thruster maneuver. However, the total propellant mass required to perform the maneuver using an EDT is zero making it an attractive option.

When performing analysis on the propulsive effectiveness of EDTs to deorbit the OG2 satellite, it is helpful to look at a range of tether lengths. Since the passive deboost mode is being looked at, the current is not an independent function but rather a derived function of the length of the tether as well. Looking at the EDT force function shown earlier in Eq. (11) it becomes apparent that both the length of the tether now has an increased effect of tether force since I is also a function of length. Longer tethers, although heavier, will provide a higher thrust force. Since the tethers do not consume propellant mass to provide deltaV, they are all capable of providing the necessary deltaV to deorbit the spacecraft. What becomes important then is the time of flight for the maneuver. Since chemical rockets provide a nearly instantaneous deltaV, they can cause reentry within hours of the initial deorbit burn. When dealing with low thrust systems, the time of flight becomes important due to component lifetimes or in this case deorbit lifetime regulations that limit the deorbit time to 25 years. When moving through LEO, the longer that an object stays in orbit, the higher the risk becomes of colliding with another space object. As a result, analysis was performed to look at the deorbit time based on the length of the tether (and subsequently the thrust provided by the tether). Figure 4 below shows the orbit decay of the OG2 utilizing a range of lengths of EDTs. The system was assumed to be passive so length affected both the current through the tether as well as the strength of the interaction with the magnetic field. By looking at the plot, it is apparent that any of the lengths analyzed

allow the satellite to decay in less than 2 years meaning even the shortest tether provides the necessary thrust capability.

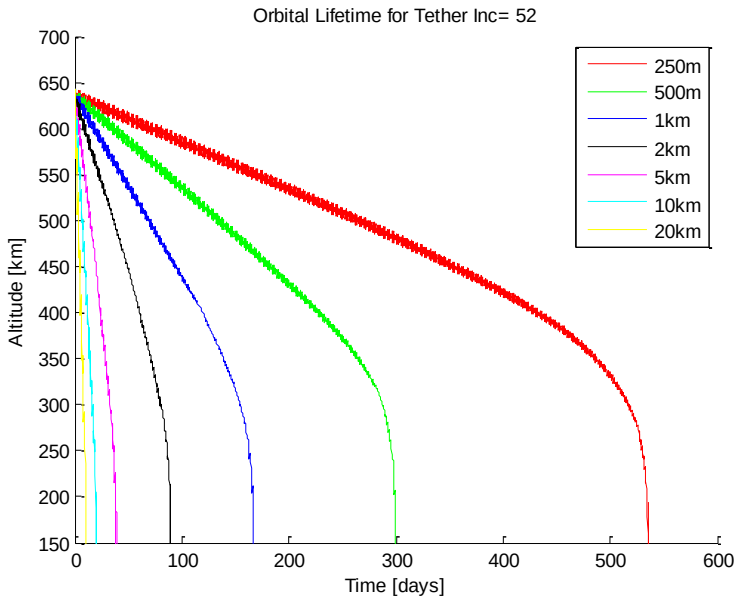


Figure 4. Orbital Lifetime for OG2 Spacecraft Deorbit using EDTs of various lengths.

Estimates for tether system mass run at ~2.5% of total satellite mass for a 20km tether. Since the OG2 could conceivably use a 250m tether system this percentage might be lower than that, but since the scalability of tethers is not well characterized, this assumption of ~2.5% is accepted. . Even maintaining this mass assumption this results in a tether mass of 3.55 kg. This would be a decrease in the propulsion system mass of 11kg which is approximately 7.7% of the total system mass, which is a very significant mass savings.

B. Tether Sizing

In order to design the appropriate tethers system, it is important to be able to size the tether system to meet the system requirements. Since debris requirements are written in terms of deorbit lifetime, a simple relationship between tether length and orbital lifetime can be developed. Figure 5 below shows this relationship for tether lengths of 250m to 20km (the design envelope generally assumed for medium size satellites). Using statistical regression, an empirical equation for this function can be found giving a direct formula for sizing the tether based on the deorbit time requirements. Equation (22) below shows this equation; to size the tether, the lifetime limit could be entered

and the equation can be solved for the tether length. The equation found has an R^2 value of 0.9997 showing strong correlation between the power fit function and analysis data. For this sample mission, however, the entire design space falls under the time limit of 25 years so solving the equation this way will fall outside the usage limits for the fit equation. In this case, the conclusion becomes that the smallest tether in the design space is the appropriate size since it meets requirements with the lowest system mass.

$$Lifetime = 205.57 * L^{-0.888} \tag{22}$$

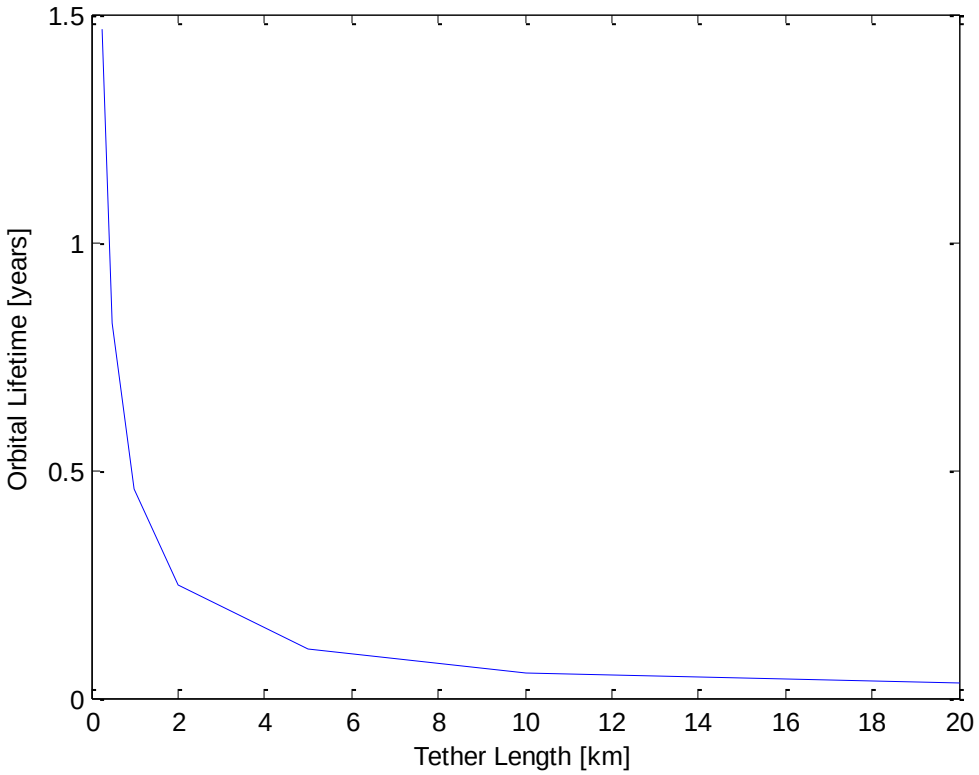


Figure 5. Orbital Lifetime of OG2 Satellite as a Function of EDT Length.

VI. Conclusions

While the use of Electrodynamic tethers must be evaluated for each specific mission, this analysis shows that the design option shows potential for propulsion system mass savings. This is especially so for small mass limited

spacecraft being operated in higher orbital regimes (500-1000km altitude orbits). In addition, the fact that EDTs provide propellantless thrust provides the possibility of using tethers for long term orbit maintenance for large spacecraft such as the ISS that would provide significant savings in refuel launches currently used to provide reboost thrust.

VII. References

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