APPLICATION OF THE TERMINATOR TETHER™ ELECTRODYNAMIC DRAG TECHNOLOGY TO THE DEORBIT OF CONSTELLATION SPACECRAFT

Robert L. Forward, Robert P. Hoyt
Tethers Unlimited, Inc.
8114 Pebble Ct., Clinton, WA 98236
phone/fax: +1-206-306-0400

Chauncey Uphoff
Fortune-Eight Aerospace Industries, Inc.

The Terminator Tether™ is a small, lightweight system that will use passive electrodynamic tether drag to rapidly deorbit spacecraft from low Earth orbit. Studies of the application of electrodynamic drag to the deorbit of constellation satellites indicates that the Terminator Tether™ offers significant mass savings compared to conventional rocket-based deorbit systems. Moreover, because it uses passive electrodynamic drag to achieve deorbit, it can deorbit the spacecraft even if the host has lost power and control functions. Numerical analyses of the performance of the Terminator Tether™ indicate that a 5 to 10 km long conducting tether weighing only 2% of the host spacecraft mass can deorbit a typical constellation satellite within a few months. Although the tether increases the total collision cross-sectional area of the satellite system during the deorbit phase, the electrodynamic drag is so many times greater than atmospheric drag at constellation altitudes that the tether can reduce the collisional Area-Time product for the satellite by several orders of magnitude. The Terminator Tether™ thus can provide a low-cost and reliable method of mitigating the growth of debris in valuable constellation orbits.

I. INTRODUCTION

This paper investigates the use of a highly survivable, conducting electrodynamic tether for use as a "Terminator Tether™" for removing unwanted Low-Earth-Orbit (LEO) spacecraft from orbit at the end their useful lives.[1,2] When a spacecraft fails, or has completed its mission and is no longer wanted, the Terminator Tether, weighing a small fraction of the mass of the host spacecraft, will be deployed. At both ends of the tether, a means of providing electrical contact with the ambient plasma will be provided to enable current to be transmitted to and from the ionospheric plasma. The electrodynamic interaction of the conducting tether moving at orbital speeds across the Earth's magnetic field will induce current flow along the tether. The resulting energy loss from the heat generated by the current flowing through the ohmic resistance in the tether will remove energy from the spacecraft. Consequently, the orbital energy of the spacecraft will decay, causing it to deorbit far more rapidly than it would due to atmospheric drag alone. Whereas a defunct spacecraft left in its orbit can take hundreds or thousands of years to deorbit due to atmospheric drag, a spacecraft with a Terminator Tether can be deorbited in weeks or months. The Terminator Tether thus is a low-mass means of reducing both the risk of spacecraft fratricide and the amount of orbital space debris that must be coped with in the future.

In the first section of this paper, we will begin by discussing the orbital debris problem motivating the development of the Terminator Tether. We will then describe the basic concept of electrodynamic tether drag, and review the results of past experiments related to this concept. In the second section, we will develop analytical methods for predicting the effectiveness of Terminator Tether systems for deorbiting spacecraft from various orbits. In the third section, we will describe methods of optimizing the electrodynamic drag on the spacecraft, while concurrently stabilizing the electrodynamic tether libration. In the fourth section, we examine the effectiveness of the Terminator Tether for reducing the Area-Time-Product for orbital decay of LEO spacecraft, and compare it to conventional deorbit methods. Finally, we describe two implementations of the Terminator Tether concept for reducing the LEO debris environment.

I.A. Motivation: Orbital Debris in LEO

Currently, the US Space Command tracks roughly 6,000 objects in LEO. Less than 300 of these objects are operational spacecraft. The rest are spent rockets and derelict spacecraft.[3] In addition, there are countless numbers of debris objects too small to be tracked; these objects result from explosions of rocket stages and fragmentation of spacecraft. These objects pose a growing risk to operational spacecraft. Moreover, in the near future, a number of companies will begin deploying telecommunications constellations with tens or even hundreds of satellites. These satellites will have operational lifetimes of approximately 5-10 years. Unless proper measures are taken to remove these satellites from orbit at the end of their lives, the debris population in LEO may grow exponentially, making many orbital slots useless.

NASA Safety Standard

NASA and other agencies have begun to address this problem. The current status of efforts to mitigate the orbital debris population is expressed in the NASA Safety Standard NSS 1740.14 *Guidelines and Assessment Procedures for Limiting Orbital Debris.*[4] The relevant portion of the Standard starts on page 6-3:

General Policy Objective - Postmission Disposal of Space Structures.

Item 6-1: "Disposal for final mission orbits passing through LEO: A spacecraft or upperstage with perigee altitude below 2000 km in its final orbit will be disposed of by one of three methods."

The method of interest relevant for this paper is the atmospheric reentry option:

Option a: "Leave the structure in an orbit in which, using conservative projections for solar activity, atmospheric drag will limit the lifetime to no longer than 25 years after completion of mission. If drag enhancement devices are to be used to reduce the orbit lifetime, it should be demonstrated that such devices will significantly reduce the area-time product of the system or will not cause the spacecraft or large debris to fragment if a collision occurs while the system is decaying from orbit."

The NASA standard applies only to NASA spacecraft and even then only to completely new spacecraft designs. New versions of existing designs are to make a "best effort" to meet the standard, but will not be required to change their design to do so. The DoD has adopted the NASA standard with the same provisos. An Interagency Group report has recommended that the NASA Safety Standard be taken as a starting point for a national standard. It is NASA's recommendation to the Interagency Group that the safety requirement be phased in only as we reach consensus internationally. This consensus is being sought through the International Debris Coordination Working Group, whose members are Russia, China, Japan, ESA, UK, India, France, Italy, and the US.

Thus, although the NASA Safety Standard in its present form is not a "Law", the existence of the standard means that at some time in the future a similar requirement may be imposed on all spacecraft. In fact, most of the satellite constellation companies have already acknowledged that, even without regulatory requirements, they must take proactive steps to prevent orbital debris from contaminating their valuable orbital slots. Several, including Teledesic, Iridium, and GlobalStar, have publicly committed to de-orbiting their satellites at the end of their operational lifetimes.[5,6] For many of the satellite constellations currently under development, the Terminator Tether can provide a low-cost, low-mass, low-Area-Time-Product, reliable, and safe means for deorbiting post-mission satellites and launch/dispenser rocket stages.

I.B. Summary Of Concept

The electrodynamic drag concept for deorbit of LEO spacecraft is illustrated in Figure 1. The idea of using electrodynamic drag to remove unwanted spacecraft from orbit was first discussed by Joseph P. Loftus of NASA Johnson Space Center in June 1996.[1] A first-order analysis published by Robert L. Forward in July 1996[2] found that a conducting tether with mass m_T orbiting above the magnetic equator through a transverse magnetic field of strength B_T at a velocity with respect to the magnetic field v_M, will generate an electrical power P in the tether given by the equation:

$$P = \frac{m_T \left(v_M \, B_T \right)^2}{rd} \tag{1}$$

where r is the resistivity and d the density of the conducting tether material. This power is converted into heat by the resistance of the tether and radiated away into space, extracting kinetic energy from the host spacecraft. For a m_T=10 kg tether of aluminum with resistivity of r=27.4 nΩ-m and density d=2700 kg/m³, orbiting over the magnetic equator at an altitude of 1000 km, at a velocity v_M=6814 m/s relative to the Earth's transverse magnetic field B_T=20 μT, the power dissipated is P=2510 W! This energy loss in the form of heat must necessarily come out of the kinetic energy of the host spacecraft. For a typical example, a 1000 kg spacecraft in a 1000 km high orbit subjected to an energy loss of 2510 J/s from a 10 kg tether (1% the mass of the host spacecraft) will be deorbited in a few weeks. Similar conclusions have been reached by many others, including members of the NASA/MSFC ProSEDS team.[7,8,9,10]
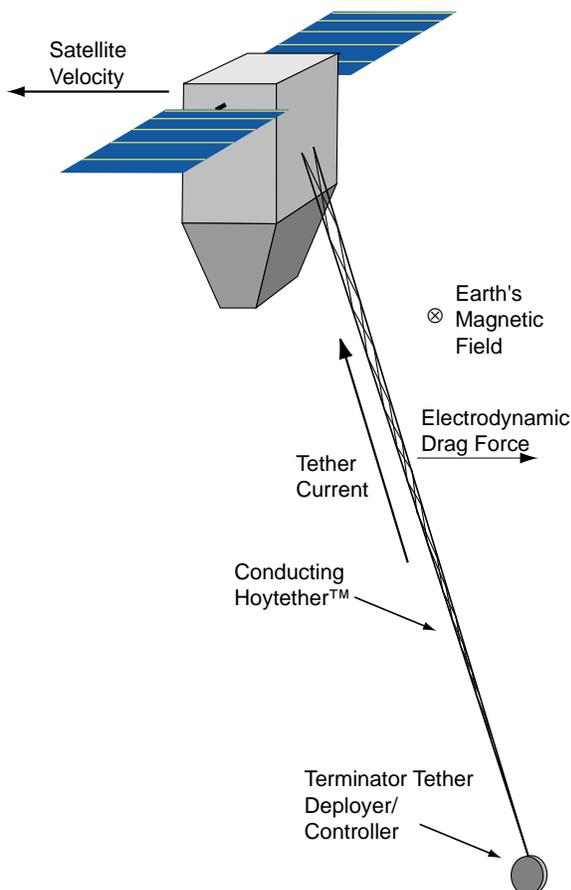
Figure 1. The Terminator Tether™ concept.

Experimental Confirmations of Induced Power Levels

Power levels of the magnitude estimated in the previous paragraph have been measured in a <u>real</u> orbital space experiment, the TSS-1R mission carried out on the Shuttle Orbiter in 1995.[11] In that experiment, a large Italian spacecraft, 1.6 m in diameter, was deployed upward from the Shuttle Orbiter at the end of a conducting copper wire tether covered with electrical insulation. As the tether was slowly deployed upwards, a series of measurements were made of the open circuit voltage induced in the tether by its motion through the Earth's magnetic field. The voltage between the end of the tether and the Orbiter ground varied from zero volts at the start to 3500 V when the amount of tether deployed approached its maximum length of 20 km. Periodically, the end of the tether was connected either to one of two different electron guns, which supplied contact to the surrounding space plasma, or to the Orbiter ground, which proved to be a surprisingly good plasma contactor via a combination of ion collection and secondary electron emission. The current flow through the tether was deliberately limited by control circuits and the current capacity of the electron guns, but power levels of 1800 W were reached.

The tether was intended to have a fully deployed length of 20 km, but at a deployed length of 19.5 km, when about 3500 V was being induced at the end of the tether inside the Orbiter reel mechanism, a flaw in the insulation allowed an electrical spark to jump in an uncontrolled manner from the tether to the Orbiter ground. With no control circuits to keep the current level down, the current flow jumped to 1.1 A and the total power generated was P=3850 W. Most of this energy went into the electrical arc, which burned through the tether, causing it to break and halting the experiment. This experiment showed that large areas of bare conducting material, such as that provided by the Italian spacecraft and the Orbiter spacecraft, can collect amperes of current, while thousands of volts of potential can be generated by sufficiently long tethers moving at orbital speeds.

Thus, both theory and experimental data indicate that significant amounts of electro-dynamic drag force can be obtained from a low mass conducting tether attached to a host spacecraft, <u>provided</u> the ends of the conductor can

3

exchange sufficient numbers of electrons with the surrounding space plasma.

Experimental data from the TSS-1R data also produced the result that the efficiency of a bare metal surface in "contacting" the space plasma is many times better than the standard theory would predict. The 8 square meters of bare surface area of the Italian spacecraft were sufficient to collect the 1.1 A of electron current.

Flight Demonstration of Electrodynamic Drag

Because of the results of the TSS-1R experiment, and because of recent theoretical studies that indicate that a bare wire can easily collect electrons,[12] Les Johnson, Nobie Stone, Chris Rupp, and others at NASA Marshall Space Flight Center have formed a team, which includes the present authors, which is embarked on a new flight experiment.[10] The experiment is scheduled for a piggy-back flight on a Delta II launch of an AF Global Positioning Satellite in early 2000. The goal of the experiment is to demonstrate that electrodynamic drag from a wire moving at orbital speeds through the Earth's magnetic field will create a large enough electrodynamic drag force to deorbit the >1000 kg Delta II second stage in a few weeks. This is essentially a demonstration of the Loftus electrodynamic drag deorbit concept and the first step in the development of a Terminator Tether.

The ProSEDS (Propulsion Small Expendable-tether Deployer System) mission is presently baselined to use a 5 km long copper wire massing 18 kg, a 20 km long nonconducting tether, and a 25 kg ballast mass on the end of the tether. The total of 25 km of tether length and the 25 kg ballast mass on the end will provide enough gradient force to keep the tether aligned near the zenith, so that the direction of the current in the tether is at right angles to both the direction of the spacecraft motion in the nominal EW direction and the Earth's near-equatorial magnetic field in the nominal NS direction.

An important feature of the ProSEDS experiment is that it is designed to be completely self-powered. It uses a battery to initiate deployment and to power up the plasma contactor, but once current is flowing through the tether, some of the power is tapped off and used to recharge the battery. The battery in turn powers the current control electronics, the telemetry system, and the plasma contactor. The

ProSEDS mission will not be designed to allow ground control changes in operation, primarily because of the increase in cost associated with that option.

Terminator Tether™

In this paper we propose a commercialized version of the ProSEDS experiment, which would consist of a small, low-mass deployer/controller package containing a large collecting area, short length, highly-survivable, multiline space tether, such as a Hoytape mesh[13] made of aluminum wire, as a "Terminator Tether" for upper stages and LEO spacecraft, especially the expected multitude of LEO constellation satellites and their upper stage launcher/dispensers. The Terminator Tether would be deployed when the host vehicle is no longer working or no longer wanted. The electrodynamic drag from the Terminator Tether would rapidly remove the unwanted vehicle from the constellation orbit altitude and a few weeks later complete the deorbit of the host vehicle from space by burnup in the upper atmosphere of the Earth. For a Terminator Tether to be of maximum usefulness for constellation satellites, it would be desirable to minimize the mass and the length of the tether, while at the same time maximizing the electrodynamic drag force. A lower tether mass means more mass for revenue producing transponders, while a shorter tether length means a lower collision cross-section Area-Time-Product during deorbit. Since the proposed Terminator Tether would autonomously maintain contact with ground control during the deorbit phase, and ground control can control its rate of descent, a Terminator Tether can avoid the larger spacecraft with well-known and predictable orbits, thus decreasing the probability of a collision below that predicted using the Area-Time-Product alone.

Electrodynamic Tether Constraints

The electrodynamic tether is assumed to be made of a conducting metal, and have a length L, density d, resistivity r, and cross-sectional area A that is constant along the length of the tether. If the tether is a single round wire of diameter D, then the cross-sectional area is $A=\pi D^2/4$. Because of the micrometeorite and space debris hazard, however, it is likely the tether will be made up of redundantly interconnected multiple lines whose individual cross-sectional areas add up to A. Given these assumptions, the tether mass is

4

then $m_T=dLA$, while the end-to-end tether resistance is $R=rL/A=rdL^2/m_T$.

Specific Conductivity Parameter: The choice of the metal conductor to be used in a space tether is determined by a combination of low resistivity (high conductivity) and low density, with cost, strength, and melting point as secondary considerations for certain applications. Copper has a resistivity $r=17.0$ nΩ-m, a density $d=8933$ kg/m^3, and a "specific conductivity" of $1/rd=6,585$ m^2/Ω-kg. Aluminum has a resistivity $r=27.4$ nΩ-m, which is significantly greater than that of copper, but it has a much lower density of $d=2700$ kg/m^3. As a result, aluminum's "specific conductivity" of $1/rd=13,500$ m^2/Ω-kg is twice the conductivity per unit mass of copper. Silver, because of its higher density and higher cost, is not competitive as an electrodynamic space tether even though its resistivity of 16.1 nΩ-m is slightly less than that of copper. An alternate candidate material would be beryllium, with a resistivity $r=32.5$ nΩ-m, density $d=1850$ kg/m^3, and a specific conductivity of $1/rd=16,630$ m^2/Ωkg, slightly better than that of the much cheaper aluminum. Beryllium also has a higher melting point at 1551 K than aluminum at 933 K, so some of its alloys may be a preferred material for some electrodynamic applications despite its higher materials cost. Unfortunately, despite decades of metallurgical research by the nuclear power industry, highly ductile alloys of beryllium have not been found, so it is difficult to make it into wire. As a result, because of its high specific conductivity, low cost, and ready availability in ductile wire form, we will assume for this paper that the electrodynamic tether will be made of aluminum wire.

Typical Resistance Values: To be competitive, the mass of the tether needs to be a small fraction of the mass of the host spacecraft it is required to deorbit. Since a typical constellation satellite has a mass of about 1000 kg, a typical Terminator Tether with a mass that is 2% of the host spacecraft mass would consist of a deployer/ controller package with a mass $m_D=10$ kg, containing an aluminum tether with a mass $m_T=10$ kg with a volume of $m_T/d=LA=3.70$x10^{-3} m^3. If this 10 kg of aluminum were formed into a tether with a length of $L=2$ km and a cross-sectional area of $A=1.85$ mm^2, then the end-to-end resistance of the tether would be $R=rL/A=rdL^2/m_T=29.6$ Ω. A longer tether would

have a proportionately smaller cross-sectional area and a higher resistance; for example, a 5 km long tether with the same mass would have a resistance of 185 Ω.

II. TERMINATOR TETHER ANALYSIS AND OPTIMIZATION

II.A. Changing a Spacecraft Orbit Using Electrodynamic Tether Propulsion

To determine the effectiveness of the Terminator Tether system for de-orbiting a spent spacecraft, we will now develop analytical tools for predicting the time required for a electro-dynamic tether to deorbit a spacecraft from a specified altitude h and inclination i.

Assumptions:

Circular Orbit

We will assume that the spacecraft trajectory is a nearly circular spiral which can, for each orbit, be approximated by a circular orbit with radius r.

Tether Orientation

We will assume that there is a balance between the electrodynamic drag on the tether and the gradient forces on the tether which causes the tether to hang at an angle α from the local vertical, with the rotation in the direction opposite to the velocity vector. In reality, variations in the electrodynamic forces along the tether length will likely cause the tether to hang in a curved manner, and variations in the drag force during an orbit will cause the tether to librate around some equilibrium point, but for this analysis we will assume that it hangs straight at the specified angle.

The tether length vector can thus be expressed as

$$\mathbf{L} = L \left(\hat{\mathbf{r}} \cos\alpha + \hat{\mathbf{v}} \sin\alpha \right). \qquad (2)$$

Current Collection

We will assume that the Terminator Tether system provides sufficient current contact with the ionospheric plasma to transmit the full current possible between the tether and the ambient plasma. Consequently, we will ignore the limitations in the tether current level that may occur due to ionospheric plasma density variations between the day and night sides of the Earth.
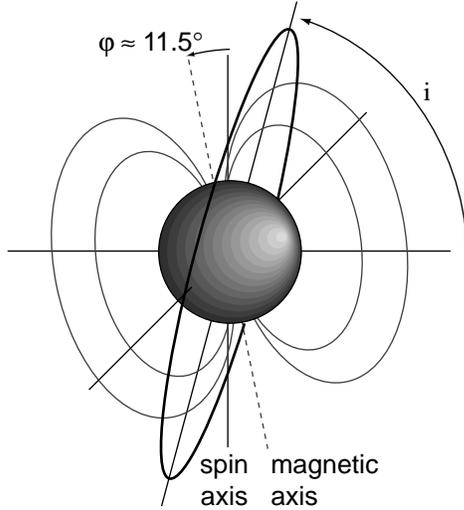
Figure 2. Tilted-dipole approximation to the geomagnetic field.

Figure 3. Spacecraft orbit in the reference frame aligned with the magnetic axis.

Geomagnetic Field Model:

To first order, the Earth's magnetic field can be approximated by a magnetic dipole with the magnetic axis of the dipole tilted off from the spin axis by approximately $\varphi = 11.5°$, as illustrated in Figure 2. For this analysis, we will ignore the 436 km offset of the dipole center from the Earth's center. At any given point, the magnetic field can be expressed as consisting of two components, a vertically-oriented component:

$$B_V = \frac{B_E R_E^3}{r^3} \sin \Lambda \qquad (3)$$

and a North-South oriented horizontal component:

$$B_H = \frac{B_E R_E^3}{r^3} \cos \Lambda \qquad (4)$$

where $B_E = 31 \ \mu T = 0.31$ gauss is the strength of the magnetic field on the magnetic equator at the surface of the Earth, $R_E = 6378$ km is the radius of the Earth, r is the radial distance of the point from the center of the Earth, and Λ is the magnetic latitude starting from the magnetic equator.

Reference Frame

In order to make the calculations tractable, we will perform the calculations in a reference frame that is rotated so that it has its z axis aligned with the axis of the Earth's magnetic dipole. The inclination λ of the spacecraft orbit with respect to the geomagnetic frame will vary

from $\lambda = i - \varphi$ to $\lambda = i + \varphi$ once a day as the Earth rotates. For the analysis in this paper we will neglect the slight variation of λ during a single orbit, and average the effects of the rotation over many orbits. For simplicity, we also choose the orientation of the reference frame so that the ascending node of the orbit lies on the *x* axis.

Expressed in Cartesian coordinates, the geomagnetic field is given by

$$
\mathbf{B} = \frac{B_E R_E^3}{r^3} \begin{bmatrix} 3xz/r^2 \\ 3yz/r^2 \\ 3z^2/r^2 - 1 \end{bmatrix}
$$

$$
= \frac{B_E R_E^3}{r^3} \begin{bmatrix} 3\sin\lambda\sin\theta\cos\theta \\ 3\sin\lambda\cos\lambda\sin^2\theta \\ 3\sin^2\lambda\sin^2\theta - 1 \end{bmatrix}, \qquad (5)
$$

In this rotated frame, the circular orbit of the spacecraft can be parameterized in terms of the angle θ around the orbit normal as

$$
\mathbf{r} = \begin{bmatrix} x \\ y \\ z \end{bmatrix} = r \begin{bmatrix} \cos\theta \\ \cos\lambda\sin\theta \\ \sin\lambda\sin\theta \end{bmatrix}, \qquad (6)
$$

and the velocity as

$$
\mathbf{v} = v_O \begin{bmatrix} -\sin\theta \\ \cos\lambda\cos\theta \\ \sin\lambda\cos\theta \end{bmatrix}, \qquad (7)
$$

6

where the magnitude of the orbital velocity is given by:

$$v_O = \sqrt{\frac{G M_E}{r}}, \qquad (8)$$

where $G = 6.67 \times 10^{-11}$ m^3/kg-s^2 is Newton's gravitational constant, and $M_E = 5.976 \times 10^{24}$ kg is the mass of the Earth.

The motion of the tether across the geomagnetic field induces an electric field in the reference frame moving with the tether[14]

$$\mathbf{E} = -\,\mathbf{v} \times \mathbf{B}. \qquad (9)$$

Consequently, in the reference frame of the tether there is a voltage along the tether

$$V = \mathbf{E} \bullet \mathbf{L}. \qquad (10)$$

After some trigonometric yoga, Eq. 10 reduces to

$$V = \frac{L\, B_E\, R_E^3\, v_O}{r^3}\, \cos\alpha \cos\lambda$$
$$= L\, B_T\, v_O\, \cos\alpha\ . \qquad (11)$$

where $B_T = B_E R_E^3 \cos\lambda / r^3 = B_H(\Lambda = \lambda)$ is the tangential component of the magnetic field at right angles to both the velocity vector and the tether.[15]

If the Terminator Tether system provides a means for the tether to make electrical contact with the ambient space plasma, such as a hollow cathode plasma contactor, field emission device, or a bare wire anode, this voltage will cause a current to flow through the tether conductor. If the total resistance of the Terminator Tether system, including tether resistance, control circuit resistance, plasma contact resistance, and parasitic resistances, is R, the current in the tether will be

$$\mathbf{I} = \frac{V}{R}\hat{\mathbf{L}}. \qquad (12)$$

If, as will be the case most of the time, the electron current is leaving the space plasma and entering the tether along an appreciable length of the tether near the end, then Eq. 12 needs to be replaced with an integral of the current along the length of the tether.

The reaction of this current with the geomagnetic field will induce a Lorentz force on the tether. Integrating this force along the

length of the tether, the net electrodynamic force $\mathbf{F}_E$ on the tether system is

$$\mathbf{F}_E = L(\mathbf{I} \times \mathbf{B}) = \frac{V}{R}(\mathbf{L} \times \mathbf{B})$$
$$= \frac{-L^2\, B_E^2\, R_E^6\, v_O\, \cos\alpha\, \cos^2\lambda}{R\; r^6}$$
$$= \frac{-L^2\, B_T^2\, v_O\, \cos\alpha}{R}\ . \qquad (13)$$

The drag force F_D on the tether is the component of the electrodynamic force $\mathbf{F}_E$ that is parallel to the velocity vector $\mathbf{v}$,

$$F_D = \mathbf{F}_E \bullet \hat{\mathbf{v}} = F_E \cos\alpha$$
$$= \frac{-L^2\, B_E^2\, R_E^6\, v_O\, \cos^2\alpha\, \cos^2\lambda}{R\; r^6}$$
$$= \frac{-L^2\, B_T^2\, v_O\, \cos^2\alpha}{R}\ . \qquad (14)$$

Using Lagrange's planetary equations and the assumption that the orbit is nearly circular, the time rate of change of the orbital semi-major axis a can be found to be

$$\frac{\partial a}{\partial t} = \frac{-2L^2\, B_E^2\, R_E^6\, \cos^2\alpha \left\langle \cos^2\lambda \right\rangle}{M_S\, R}\left(\frac{1}{a^5}\right), \quad (15)$$

where M_S is the total mass of the spacecraft (including the tether system), and $\left\langle \cos^2\lambda \right\rangle$ is the average of $\cos^2\lambda$ as λ varies over one day due to the rotation of the Earth:

$$\left\langle \cos^2\lambda \right\rangle = \frac{1}{16}\left(\begin{matrix} 6 + 2\cos 2i + 3\cos\left[2(i-\varphi)\right] \\ + 2\cos 2\phi + 3\cos\left[2(i+\varphi)\right] \end{matrix}\right). \quad (16)$$

Taking the reciprocal of Eq. 15 and integrating from the initial to the final orbit radius, we obtain an estimate of the total time required for a Terminator Tether to deorbit a spacecraft:

$$\Delta t = \frac{M_S\, R}{12 L^2\, B_E^2\, R_E^6\, \cos^2\alpha \left\langle \cos^2\lambda \right\rangle}\; a^6 \Big|_{a\ initial}^{a\ final}. \quad (17)$$

It should be noted that if current can flow in only one direction in the system, then the calculation of $\left\langle \cos^2\lambda \right\rangle$ must be handled differently for orbits with inclinations greater than 78.5°. This is due to the fact that for such high inclination orbits, the spin of the Earth will

rotate the magnetic dipole so that the spacecraft's orbit will actually move in the retrograde direction relative to the magnetic field during a portion of the day. Consequently, the voltage will reverse direction for a part of the day.

Deorbit Times for Constellation Satellites

To illuminate the utility of using the Terminator Tether system to remove dead and obsolete satellites from useful orbital slots, we have used the equations developed above to estimated the time required for a tether massing only 2.5% of the total satellite mass to deorbit satellites from typical Big- and Little-LEO constellation orbits. In these calculations, we have assumed that the control and parasitic resistances in the system equal half of the tether resistance. Table 1 compares the predicted time required for a Terminator Tether to deorbit a satellite from the orbits used by several existing or planned constellation systems to the time required for the satellite to deorbit under the influence of atmospheric drag alone. The times for atmospheric decay are rough estimates based upon the assumption of a $10\,m^2$ satellite cross-sectional area, a drag coefficient of 2.0, and use of the 1977 Jaccia static atmosphere model[16] for the mean exospheric temperatures (see Section III.A). As Table 1 shows, a Terminator Tether massing only a very small percentage of the total system mass can deorbit a satellite within a few weeks to a few months, many orders of magnitude faster than the satellite would deorbit due to atmospheric drag alone.

It should be noted that the deorbit time estimates given above are based upon a theoretical model that does not include considerations for ionospheric plasma density effects. These effects may increase deorbit times for higher altitudes, where the plasma is more tenuous. More accurate calculations including these and other effects are currently underway.

It also should be noted here that the true measure of the effectiveness of a deorbit method is not just whether it reduces the orbital lifetime compared to atmospheric drag decay, but whether it reduces the product of the orbital lifetime times the collision cross-sectional area of the spacecraft. This Area-Time-Product provides a measure of the risk of the defunct spacecraft colliding with another spacecraft during its orbital lifetime. In Section III, we will show that the Terminator Tether can significantly reduce the Area-Time-Product for most LEO orbits.

Inclination Change

While Table 1 shows that a Terminator Tether system can rapidly deorbit a spacecraft from orbits with inclinations below about 75°, the rate of altitude drop for a near-polar orbit is low. Because electrodynamic tether propulsion can be used to change the inclination of an orbit,[18] we explored the possibility of decreasing the deorbit time for a polar satellite by modulating the tether current to reduce the orbit inclination, thus bringing the satellite into an orbit with a more favorable interaction between the velocity and magnetic field vectors. For a nearly polar orbit, however, the rate of inclination change

Table 1. Deorbit times from example constellation orbits using a Terminator Tether system with an aluminum tether massing 2.5% of the spacecraft mass.

Constellation	Altitude (km)	Inclination (degrees)	Deorbit Time, no TT (Derelict)	Initial Orbit Decay Rate (km/day)	Deorbit Time, with Terminator Tether
Orbcomm 1	775	45	100 years	44	11 days
Orbcomm 2	775	70	100 years	11.6	41 days
LEO One USA	950	50	600 years	32	18 days
GlobalStar	1390	52	9,000 years	22.3	37 days
Skybridge	1475	55	11,000 years	18.5	46 days
FaiSat	1000	66	800 years	13.5	45 days
Iridium	780	86.4	100 years	2.1[17]	7.5 months[17]
M-Star	1350	47	7000 years	27	28 days
Celestri	1400	48	9000 years	26	32 days
Teledesic	1350	~85	7000 years	1.7[17]	17 months[17]

Note: All spacecraft are assumed to have an effective drag cross section of $10\,m^2$.

achievable by a passive tether system turns out to be very small. The rate of inclination change is given by the rate of orbit precession caused by the net electrodynamic torque T_T on the orbit in the transverse direction perpendicular to both the line of nodes (the x axis, in this case) and the orbit axis:

$$\frac{\partial i}{\partial t} = \frac{T_T}{\Omega},\qquad(18)$$

where $\Omega = \mathbf{r} \times \mathbf{p} = M_S\,(\mathbf{r} \times \mathbf{v})$ is the angular momentum of the satellite in its orbit. This torque results primarily from the out-of-plane forces on the tether when the satellite is near the equator; because the velocity vector for polar orbits is nearly parallel to the magnetic vector when the satellite is at the equator, this force is rather small. Averaging the torque on the orbit due to the electrodynamic force given by Eq. 13 over one orbit, we obtain

$$\frac{\partial i}{\partial t} = \frac{L^2\,B_E^2\,R_E^6\,\cos^2\alpha\,\langle\sin 2\lambda\rangle}{4 M_S\,R}\left(\frac{1}{r^6}\right).\qquad(19)$$

If we choose an orbit such as the one used by the Iridium constellation (780 km altitude, 86.4° inclination), the maximum inclination change that a tether massing 2.5% of the spacecraft mass could cause would be only about 0.35° per year. Consequently, modulating the tether current to maximize the inclination change rate will not result in a significant improvement in the overall deorbit rate.

II.B.   Maximizing Electrodynamic Drag

Because the electrodynamic forces act perpendicular to the tether, the tether will trail behind the spacecraft. In fact, it is necessary for the tether to hang at an angle behind vertical for the electrodynamic forces to decelerate the spacecraft. The hang angle of the tether depends upon the balance between the electrodynamic drag force, which tends to pull the tether back, and the gradient force, which tends to restore the tether to a vertical orientation. Because the gradient force decreases as the tether libration increases, if the electrodynamic drag is too large, this balance can become unstable, resulting in loss of control of the tether system. In this section, we analyze the drag torque to gradient torque balance and develop a means of not only optimizing the electrodynamic drag but also stabilizing the tether libration.

Force and Torque Balance Analysis

We will now calculate the forces and torques on the tether and, using the fact that the electrodynamic and gradient torques on the tether must balance each other out to achieve a stable tether orientation angle, calculate some optimum values for some of the Terminator Tether parameters.

Electrodynamic Force and Torque

As discussed in previous sections, both theory and experiment show that: provided the conducting tether is moved rapidly through the Earth's magnetic field in order to generate a voltage across it, and provided good contact is made with the space plasma, we will have a conducting tether that has a current flowing through it. When a wire (moving or not) carrying a current I is embedded in a magnetic field $\mathbf{B}$, there will be an electrodynamic force $\mathbf{F}_E$ generated on each element of the wire. The electrodynamic force will be at right angles to both the magnetic field vector and the length vector of the wire, with a magnitude given by Eq. 13:

$$F_E = -\frac{B_T^2\,L^2\,v_o\,\cos\alpha}{R}.\qquad(20)$$

The electrodynamic force is always at right angles to the conductor, and stays at right angles to the conductor as the angle α varies, as shown in Figure 4. Assuming that the electrodynamic drag force is applied uniformly along the length of the tether, we can make the simplifying assumption that the integrated force is effectively applied at right angles to the center of mass of the tether at the point $L/2$ as shown in Figure 4. The electrodynamic torque on the tether is:

$$T_E = F_E\,\frac{L}{2} = \frac{B_T^2\,L^3\,v_o\,\cos\alpha}{2R}.\qquad(21)$$

Gradient Forces and Torques

When a tether and its ballast end mass are deployed from a host spacecraft, the gravity gradient force field of the Earth, combined with the orbital centrifugal gradient force field will cause the tether to deploy either up or down from the host spacecraft, depending upon the direction in which the ballast mass is ejected from the host vehicle. In the Terminator Tether system, the tether will be deployed below the host spacecraft.

In the absence of electrodynamic and atmospheric drag, the equilibrium direction of the

Figure 4. Gradient and electrodynamic forces and torques on the tethered system.

where the gradient field strength $\Gamma=\omega_o^2=v_o^2/r^2=GM_E/r^3$. The strength of the force depends not only on the ballast mass m_B and the strength of the gradient field 3Γ, but also the radial component of the distance of the ballast mass from the center of mass of the spacecraft, which is $L\cos(\alpha)$. As shown in Fig. 4, this force acts in the vertical direction along the radius vector leading from the ballast mass towards the center of the Earth. The component of this gradient force that is at right angles to the tether, given by $F_{GB}\sin\alpha$, will produce a torque T_{GB} on the tether that tends to restore the tether toward the vertical, lessening the angle α.

$$T_{GB} = L\,F_{GB}\,\sin\alpha = 3\,\Gamma\,m_B\,L^2\sin\alpha\,\cos\alpha \quad (23)$$

The tether mass m_T also contributes to the gradient force and torque. If we assume that the tether has a uniform cross section, then we can replace the distributed mass of the tether with an equivalent point mass m_T placed at the center of mass of the tether, which is the point $L/2$ along the tether, and a distance $L/2\cos\alpha$ in the radial direction. The gradient force due to the tether mass is then:

$$F_{GT} = \frac{3}{2}\,\Gamma\,m_T\,L\,\cos\alpha \quad (24)$$

While the gradient torque is:

$$T_{GT} = \frac{L}{2}F_{GT}\,\sin\alpha = \frac{3}{4}\,\Gamma\,m_T\,L^2\,\cos\alpha\,\sin\alpha \quad (25)$$

The total gradient torque attempting to restore the tether to its vertical orientation is then:

$$T_G = T_{GB} + T_{GT}$$
$$= 3\,\Gamma\left(m_B + \frac{m_T}{4}\right)L^2\,\cos\alpha\,\sin\alpha \quad (26)$$

It is important to notice the variation of the total gradient torque as the tether angle α is varied. Since the gradient force is always in the radial or vertical direction, there is no torque on the tether when the tether is vertical, as would be the case when there are no aerodynamic or electrodynamic drag forces. Once the drag forces become important and start to apply torque to the tether, increasing the tether angle α, those drag torques causing an increase in tether angle α will be opposed by a rising gradient torque which will attempt to decrease the tether angle. The gradient torque reaches its maximum at $\alpha=45°$, where $\sin\alpha=\cos\alpha=0.707$ and $\sin\alpha\cos\alpha=0.50$. When this angle is reached, we are at a point of

tether would be exactly along the vertical, since the combined gradient field is a maximum in that direction. Because the tether will generate a significant amount of electromagnetic drag, however, the actual equilibrium position of the tether with respect to the local vertical will be at some angle α lagging behind the spacecraft motion in the plane of the orbit, as shown in Figure 4. In the following analysis, we find there is an optimum angle for α that produces the largest electrodynamic drag force on the host spacecraft, minimizing its deorbit time.

The combined vertical gravity gradient and centrifugal gradient field 3Γ acting on the ballast mass m_B at the end of the tether of length L will produce a gradient force F_{GB} given by:

$$F_{GB} = 3\,\Gamma\,m_B\,L\,\cos\alpha, \quad (22)$$

rotational instability, for if there is a further increase in the electrodynamic drag force due to an increase in magnetic field strength or plasma density, causing an increase in current flow through the tether and causing the angle α to become greater than 45°, the gradient torque, instead of growing stronger to counteract the increased drag torque, will become weaker. The tether will become unstable and the angle α will go rapidly to 90°, where the drag force will also drop to near zero.

To restore control to the tether angle if the instability occurs, it will be necessary to turn off the electrodynamic drag forces by shutting off the current flow through the tether. The $\alpha=90°$ position for the tether and ballast mass is a gravitationally unstable orientation. After a time, slight fluctuations in the gravity field will allow the gradient force to slowly take over and restore the tether to the vertical orientation, which, unless it can be controlled in some way, is equally likely to be up or down. It would there-fore be desirable to maintain control of the tether angle so as to avoid the tether angle getting into the region of instability. To avoid this poss-ibility of tether instability, the ProSEDS mission planners are planning on using a large ballast mass and a long non-conducting tether in order to keep the gradient forces high. For Terminator Tether applications on commercial spacecraft, however, use of a large ballast mass will not be economically feasible. Consequently, the Terminator Tether will use feedback control on the tether current to maximize the drag force and stabilize the tether dynamics.

Torque Balance on a Stable Tether

The angle α of a stable tether is determined by the balance between the electrodynamic torque T_E attempting to increase the angle α and the gradient torque T_G attempting to decrease the angle α. Balance is achieved when the two torques are equal:

$$T_E = T_G = T_{GB} + T_{GT} \qquad (27)$$

or, using Eq. 21 and Eq. 26:

$$T_E = \frac{B_T^2\, L^3\, v_o}{2\,R}\ \cos\alpha$$
$$= 3\,\Gamma\left(m_B + \frac{m_T}{4}\right) L^2\ \sin\alpha\ \cos\alpha = T_G \qquad (28)$$

Simplifying Eq. 28, we obtain a relationship between the electrodynamic and gradient

parameters of the tether that must hold if the tether is to be in a stable equilibrium state.

$$\frac{B_T^2\, L\, v_o}{R} = 6\,\Gamma\left(m_B + \frac{m_T}{4}\right)\sin\alpha \qquad (29)$$

At first glance, it might seem that the optimum angle for the tether would be 45°, since at that angle the gradient torque is largest and therefore can counteract a larger electrodynamic drag force, despite the fact that at 45° the tether is at the onset of instability. The optimum angle, however, is that which maximizes the horizontal or drag component of the electro-dynamic force that opposes the host spacecraft motion, not the <u>total</u> electrodynamic force. This horizontal drag force is given by Eq. 14:

$$F_D = F_E\cos\alpha = -\frac{B_T^2\, L^2\, v_o}{R}\cos^2\alpha$$
$$= -\frac{B_T^2\, L^2\, v_o}{R_C + rdL^2\,/\,m_T}\cos^2\alpha \qquad (30)$$

This equation gives a maximum drag force for long tether length L, small tether resistance R and small tether angle α. But to maintain α near zero when there is a large drag force on the tether requires a large ballast mass or a very long tether. If a large ballast mass were available, such as might be obtained by cutting off a large portion of the host vehicle (a solar panel, for example), then this is a mode of operation which can allow the maximum electrodynamic force F_E that is available to produce the maximum drag force F_D. If, however, the amount of drag force that can be applied to the tether is limited by tether instability, as it is in the NASA/MSFC ProSEDS mission and the various Terminator Tether applications, then instead of looking at the electrodynamic limits to maximizing the drag force F_D, we want to look at the gradient limits to maximizing the drag force. To do this, we use Eq. 29 in Eq. 30 to obtain:

$$F_D = -\,6\,\Gamma\,L\left(m_B + \frac{m_T}{4}\right)\sin\alpha\,\cos^2\alpha \qquad (31)$$

This equation says that for maximum drag force on the host spacecraft, you want long tether length L, as well as massive ballast mass m_B and tether mass m_T. The equation also states that a small tether angle α (tether near vertical) is not optimum. If a very large ballast mass is available then it is possible to operate with α at

a small angle and get the maximum drag force available from the maximum electrodynamic force made possible by the available environmental parameters. More realistically, for any given ballast mass, it is better to operate the tether at the angle α that maximizes the drag force. We can determine that optimum angle by setting the partial derivative $\partial F_D/\partial\alpha=0$ and solving the resulting equation. When we do this, we find that the optimum angle for the tether that gives the maximum electrodynamic drag force F_D, while still keeping the tether torques balanced and under control, is $\alpha=\arctan(0.707)$ $=35.26°$. This angle is well below the angle of 45° where tether instability sets in. With this angle selected, we obtain an equation for the maximum stable drag force of:

$$F_D(max, \alpha = 35.26°) = -6\sin\alpha\cos^2\alpha\,\Gamma\,L\left(m_B + \frac{m_T}{4}\right)$$

$$= -2.31\,\Gamma\,L\left(m_B + \frac{m_T}{4}\right) \qquad (32)$$

The tether angle in a Terminator Tether will be controlled by controlling the current through the tether to compensate for variations in magnetic field strength and direction, plasma density (which affects the plasma resistance), and other factors, and thereby maintain the tether at an intermediate angle where both the electrodynamic and gradient forces are at an appreciable level and balance each other. This can be done in a number of ways, either by varying a control resistor or inserting stepped values of ballast resistors in series with the resistance of the tether, or by periodically interrupting the current through the tether to keep the average current at the desired value.

There are many ways to generate the sensing information needed to provide the feedback signals to the tether current controller. The simplest is to measure the drag acceleration on either the host spacecraft or the end mass with a set of accelerometers and maximize the deceleration force in the direction opposite to the host spacecraft motion. Another method would be to measure the current in the tether, and, knowing the tether resistance and the amount of control resistance, calculate the power being extracted and maximize that value. Alternate methods would be to use GPS receivers at both ends of the tether to measure the angle of the tether or an optical position sensor to measure the position of the ballast mass with respect to the

host spacecraft. These methods of controlling the drag force or the tether angle should also stabilize the tether oscillations that presently concern the ProSEDS mission planners.

Electrodynamic Drag Force and Power Levels

We will now estimate the magnitude of the electrodynamic drag force and power attainable from a Terminator Tether. If we assume the Terminator Tether is in orbit at an altitude of 1000 km, where the gradient field $\Gamma=0.99\times10^{-6}/s^2$, and the electrodynamic tether has a length of L=5 km, a mass of m_T=10 kg, a ballast mass of m_B=10 kg, and a tether tilt angle $\alpha=35.26°$. then the gradient-force-limited maximum allowable stable drag force using Eq. 32 is F_D=0.143 N. This is to be compared with the electrodynamic drag force obtainable from the aluminum tether moving at velocity v_M=6814 m/s with respect to the transverse magnetic field B_T=20 μT. If we assume the control resistor R_C=0 Ω, then the maximum available electrodynamic drag force using Eq. 30 is 0.246 N, which is more than the stable drag force of 0.143 N. The control resistance must be increased to lower the current flow through the tether and bring the electrodynamic torque down to a level where it will balance the gradient torque and leave the tether at the optimum angle to produce the stable drag force level of 0.143 N.

This maximum stable drag force F_D=0.143 N opposing the motion of the host spacecraft, assumed to be in an equatorial orbit with $\lambda=0$ and a velocity with respect to the magnetic field of $v_M=v_o-\omega_E r\cos\lambda$=(7350-536) m/s=6814 m/s, is equivalent to a deceleration power of:

$$P = F_D\,v_M = 975\,W \qquad (33)$$

Since, as pointed out in Eq. 1, the power generation capability of an electrodynamic tether is proportional primarily to its mass, the Terminator Tether will be designed to have a high conductivity tether with enough mass to exceed the design power levels needed for any particular initial orbit and host vehicle. The current through the tether would then be controlled at the gradient-limited maximum stable power level so as to maintain the tether at the optimum angle to give maximum stable drag. For example, the power level P that could be generated and dissipated in an electrodynamic tether can be obtained either by using Eq. 11 for the voltage induced across the tether and

dividing the square of the tether voltage V by the tether resistance R, or by using Eq. 14 or 30 for the electrodynamic drag force and multiplying it by the spacecraft velocity v_M with respect to the geomagnetic frame:

$$P = \frac{V^2}{R} = \frac{\left(B_T\, L\, v_M \cos\alpha\right)^2}{R_C + r\, d\, L^2 \big/ m_T} = F_D\, v_M \qquad (34)$$

Where R_C is the control resistor, and $R_T = rdL^2/m_T$ is the tether resistance. An aluminum tether of length L=5 km and mass of m_T=10 kg has a tether resistance R_T=185 Ω. A spacecraft in orbit at 1000 km altitude over the magnetic equator will have a velocity with respect to the magnetic field of v_M=6814 m/s, and will see a transverse magnetic field of B_T=20 μT. Using Eq. 34, we calculate that the above aluminum tether trailing at the optimum drag tether angle of α=35.26° has the ability to generate up to 1670 W of power if the control resistor is set to zero. A control resistor of R_C=132 Ω will bring the power level down to the desired 975 W. Variations in the control resistor would then be used to keep the tether stabilized at an angle of α=35.26°, despite variations in magnetic field strength and plasma density. Since B_T varies as cosλ, a 10 kg tether will suffice for orbit inclinations up to λ=40˚. For orbits with higher inclinations and therefore lower horizontal magnetic fields, a tether with a larger mass would be called for. Since the tether mass also determines the maximum gradient-limited drag force, the more massive tether would allow for a higher allowable stable drag force.

III. EFFECTIVENESS OF THE TERMINATOR TETHER FOR DEORBITING SPACECRAFT

III.A. Comparison with Atmospheric Decay

The most straightforward method of removing a spacecraft from orbit is to simply allow atmospheric drag to decay the orbit. For orbits above about 500 km, however, orbital lifetimes can be tens to thousands of years. The NASA Safety Standard discussed in Section II.A. states that if a drag-enhancement method is used to speed the deorbit of a spacecraft, it must also significantly reduce the total Area-Time-Product of the system. The use of a several-kilometer long tether will increase the cross-sectional area of the spacecraft system. Nonetheless, the effectiveness of electrodynamic drag is so many orders of magnitude greater than atmospheric drag for most LEO orbits that the total Area-Time-Product can be greatly reduced.

For a spacecraft decaying due to atmospheric drag alone in a near-circular spiral trajectory, the Area-Time-Product is given by

$$A_S \int dt = -\frac{M_S}{C_D} \int_{r_{initial}}^{r_{final}} \frac{dr}{\rho(r, T_\infty)\sqrt{\mu_e r}}, \qquad (35)$$

where A_S is the cross sectional area of the spacecraft, r is the average semimajor axis of the orbit, C_D is the coefficient of drag, and $\rho(r, T_\infty)$ is the atmospheric density as a function of the semimajor axis and the slowly-varying exospheric temperature, T_∞ (to account for solar variations). Thus, the relationships shown in Eq. 35 represent an upper and lower limit on the Area-Time product for a spacecraft descending under the influence of atmospheric drag.

The curves of Fig. 5 show what would happen if the exospheric temperature were constant for as long as it takes the spacecraft to descend to 250 km. Thus, the upper curve of Fig. 5 shows the Area-Time product in the case of minimum (static model-800 kelvin) density of the atmosphere if the exospheric temperature remained constant for the lifetime of the spacecraft. The lower curve of Fig. 5 shows the Area-Time product in the case of constant (1400 kelvin) exospheric temperature for the entire time of the spacecraft's descent. Thus, the 1400 and 800 kelvin curves bound the Area-Time product for thousands of years. Actual Area-Time products for real systems will be much closer to the 1100 kelvin curve because the exospheric temperature and the density at any altitude along the descent path will average out over many Solar cycles. The important thing to note about the atmospheric decay curves of Area-Time product is that they are independent of spacecraft cross-sectional area.

Figure 5 compares the Area-Time-Products for spacecraft with Terminator Tether systems to the Area-Time-Products for spacecraft deorbiting due to atmospheric drag alone. For these calculations we have assumed that the spacecraft mass 1300 kg, are in near-circular equatorial orbits, and have a coefficient of drag of C_D=2.0. In addition, we have used the 1977 Jaccia static atmosphere model for the exospheric temperatures.[14] The figure shows that the use of electrodynamic tether drag can reduce the deorbit Area-Time-

13

Product by several orders of magnitude. As a result, the Terminator Tether system can greatly reduce the risks of a decaying spacecraft colliding with another spacecraft.

As pointed out before, a well-designed Terminator Tether can lower the collision probability even further than the blind chance probability implied by the use of the Area-Time-Product criteria, by using ground control of the rate of descent to avoid collision with the larger objects in space with well-known and predictable orbits.

Note that Fig. 5 is conservative in two ways. First, the assumed cross-sectional area of the tether is much larger than its neutral drag cross section (the area presented to the "wind") and, second, the power generated in the tether is assumed to be constant at values that are considerably less than those to be expected in the range of altitudes shown in Figure 5. For example, a 5 km, 13 kg tether whose resistance is 143 ohms, would generate over 5000 watts at 622 km altitude if it had perfect contact with the plasma and if it were orbiting in the magnetic equator. In these examples, we have assumed that the same tether will generate only 2040 watts throughout its descent from 622 to 250 km altitude, although, in the ideal case, the power would increase with decreasing altitude. These assumptions are based on the power levels observed in the TSS-1R electrodynamic tether experiment. These lower power levels are thought to have resulted from incomplete contact with the plasma. As the technology matures, the higher theoretical values may be possible. The induced power values used in the calculations presented in Fig. 5 are the lower values, which we can be confident of, rather than the higher theoretical values.

Effects of Orbital Inclination

Fig. 5 also shows the effects of changing from an equatorial orbit to a polar orbit where the average values of inclination with respect to the geomagnetic equator have been applied according to Eqn. 16.
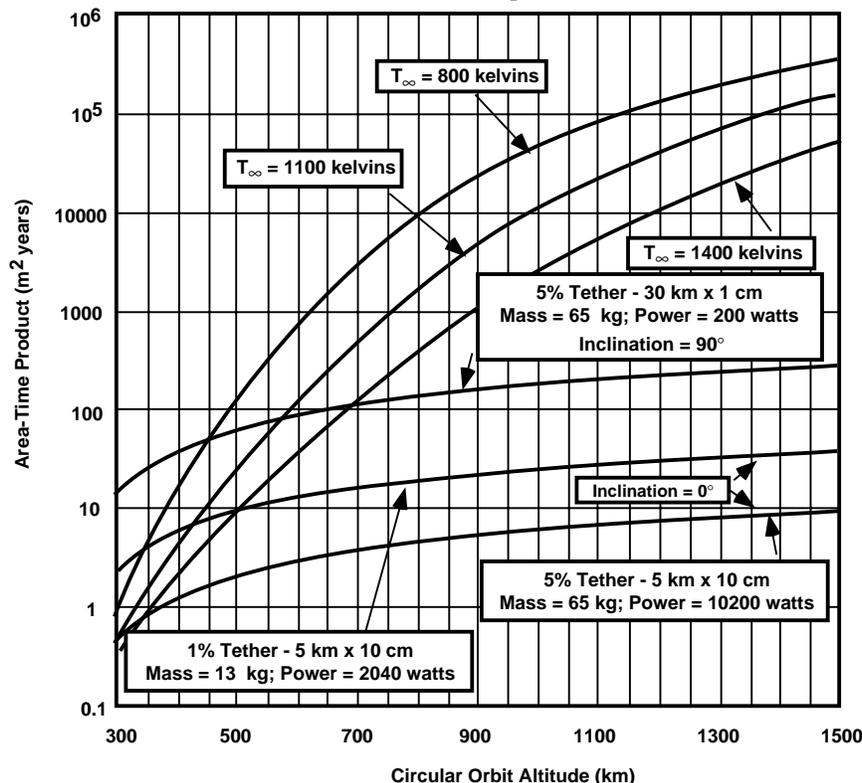
Area-Time Product for 1300 kg Spacecraft

Note: A∆t Scales Linearly with Spacecraft Mass

Figure 5. Area-Time Product vs. Initial Altitude for 1300 kg spacecraft at inclinations of 0° and 90°. Upper curves show results for atmospheric drag alone at mean and extremes of exospheric temperature. Lower three curves show results for Terminator Tether systems.

Note that the area of the tether for the 90° orbit is only 300 m^2 while the area for the 0° examples is 500 m^2. If all factors were equal in the comparison, we should expect a factor of about 50 in AΔt as we change the inclination of the orbit from a daily average of 11°.5 to 82°, the equivalent average inclinations of the equatorial and polar orbits with respect to the geomagnetic equator. We find at the 1500 km level an increase in AΔt of a factor of about 30 from the lower tether curve of Fig. 5 to the upper tether curve showing the decay of a polar satellite from various altitudes at a fixed induced power level of 200 watts. If the area of the polar orbit tether were 500 m^2, the AΔt value at 1500 km would be about 500 m^2-years instead of 300 m^2-years.

Thus, we find that it is considerably more difficult to bring down a polar orbiter than ones in the low to middle inclinations. As a result, we expect to require longer tethers at the higher inclinations in order to generate enough potential difference along the tether to permit good contact with the ionospheric plasma and to provide a stable current along the tether. We have assumed, for the calculations shown in Fig. 5, that the tether angle α is always 0°. This corresponds to a situation where the tether tip mass is large as would be the case if a large piece of the spacecraft could be separated from the main spacecraft and used to hold the tether more nearly vertical.

Nevertheless, the Area-Time products of the tether decay examples are still less than 1% of the neutral drag decay values, even for polar orbits. These high-inclination and/or high-altitude orbits are typical of those under consideration for the Teledesic, M-Star, and Celestri constellations. Reasonable estimates of their mass and area show them to have neutral-drag lifetimes of several thousand years and AΔt values of 100,000 m^2-years. 5% Terminator Tethers, made to be somewhat longer than for lower inclinations, can bring these spacecraft down in one or two years with AΔt values of 300 to 500 m^2-years.

Furthermore, it is important to note that the built-in conservatism of using fixed power tethers yields a comfortable pad that can be realized if we design the tethers to operate at the maximum power levels available at the lowest altitude in the descent profile. The calculations and Figures in this paper have been based on power levels fixed at the highest altitude. Thus, the tether descent is not assumed to take advantage of the increased power levels available as the spacecraft altitude decreases. Preliminary calculations based on Eqn. 17 of this paper indicate that the Area-Time products and the descent times can be decreased by a factor of 3 simply by designing for the lower altitude power levels. This will be particularly helpful and easy to do for the high-inclination orbits where the available induced power is of the order of hundreds of watts rather than the kilowatt levels available at lower inclinations.

III.B.   Comparison with Solid Rocket Motors

The other conventional method of removing a spacecraft from a LEO orbit is to build into the spacecraft system a rocket mechanism capable of deorbiting the spacecraft. This method, however, requires that a significant fraction of the spacecraft's launch mass be dedicated to the propellant needed for deorbit.

If a spacecraft manufacturer were to use a rocket deorbit system, the design requirements for the system will be more stringent than those for ordinary spacecraft; the system must operate after many years on-orbit and when some or all other components of the spacecraft have failed. Moreover, a rocket deorbit system must be capable of proper operation under many kinds of anomalous situations, such as spacecraft tumbling due to attitude control failure, offset of center of mass, or lack of orbital position knowledge.

Figure 6 shows the percent additional solid-rocket propellant mass required to drop a spacecraft from a circular orbit at the specified altitude to a new orbit with a perigee of 200 km. At this altitude, atmospheric drag will remove a typical spacecraft from orbit in a few revolutions. The contours of constant stage propellant mass fraction range from low values of 0.5 up to the values associated with the best solid motors (≈0.93) that can be built without adding any extra hardware to the deorbit stage. An effective, independent stage to provide a retro ΔV of 50-325 m/s will almost certainly have a mass fraction on the order of 0.6 to 0.75. If the deorbit stage is required to perform its own attitude determination, the stage propellant mass fraction may be as low as 0.5.

The figure shows that a solid-rocket deorbit system will require a mass allocation that is a
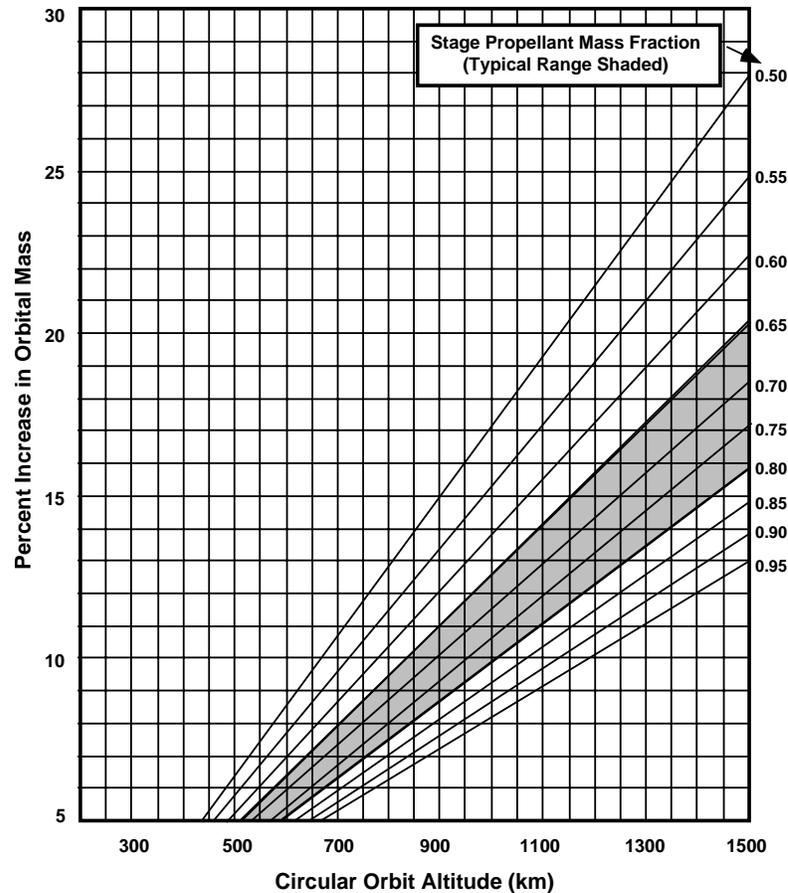
Figure 6. Conventional solid rocket motor deorbit system percent mass increase vs. altitude and stage propellant mass fraction (I_{sp}=288 sec).

significant fraction of the spacecraft's launch mass. For a spacecraft in a 1000 km orbit, a deorbit rocket system with a reasonable propellant mass fraction of 0.7 will consume nearly 13% of the vehicle's launch mass. A Terminator Tether system, however, can achieve deorbit of the spacecraft while requiring as little as 2 to 5% of the launch mass. The mass savings achieved with the Terminator Tether system can be used for additional revenue-producing equipment or for additional station-keeping fuel to provide longer operational lifetimes.

IV. IMPLEMENTATION

IV.A. THE TERMINATOR TETHER™

The standard Terminator Tether Deorbit System will be a small, lightweight, highly-autonomous package that is attached to the host vehicle before launch. Only minimal electronic interfacing will be required, likely limited to several circuits for monitoring the host status. The Terminator Tether unit will contain the tether and its deployer, an electronics control package, a power conversion circuit to enable the electronics to be powered by the tether current, a three-axis accelerometer, a radio receiver, a small battery, a small solar panel to keep the battery charged during the long dormant period, and an electron emission device such as a hollow cathode plasma contactor or a Spindt Cathode.

Dormant Phase: During the operational mission of the host spacecraft, the Terminator Tether will be dormant. Periodically, a timer circuit will wake the system up and it will check the status of the host. Multiple, redundant methods of observing host status will be used to prevent any premature activation. The system will also use the radio receiver to listen for activation signals from ground stations. If the host is defunct, or if the system receives an authentic activation signal, it will initiate the deployment and deorbit sequence.

Deployment: The tether could be deployed upwards from the host like the ProSEDS tether,

with the deployer and electronics control unit remaining on the host. However, for constellation satellite deorbit it is preferable to use a deployment scheme where the deployer and control unit are ejected downwards from the satellite, with the tether paying out as they fall. This scheme has several advantages. First, the tether is dropped below the satellite so that it will be out of the way of other satellites in the constellation. Second, the entire mass of the deployer and electronics packages serve as ballast mass, eliminating the need for a dedicated ballast mass. Third, the section of tether nearest the host can be made of a non-conductor, electrically insulating the host from the tether and preventing charging of the host spacecraft.

Operation: The tether will be deployed with the tether circuit closed, but with the electron emitters off. This will prevent charging of the Terminator Tether unit relative to the tether and eliminate any chances of arcing between the tether and the deployer. Once the tether is fully deployed, the electron emitters will be turned on gradually, allowing the tether current to rise slowly. The drag force, as measured by the accelerometers or other means, will start to increase and the tether will start to lag behind the host. The system will maintain the drag at a low level for several orbits while it determines the maximum and minimum voltage levels and the system's ability to collect and emit electrons. Then, it will increase and vary the tether current to maintain a maximum deceleration level. The system will also use information from the accelerometers to perform feedback control on the tether current to counteract tether oscillations that may arise due to variations in plasma density and magnetic field.

System Power: One of the major advantages of the Terminator Tether concept is that it can obtain its power from the tether current. Essentially, it will use some of the orbital energy of the host to power itself. Consequently, this deorbit method does not require that the host spacecraft solar panels and power supply be functional for deorbit.

Deorbit Control: As the Terminator Tether drags the host spacecraft down in a slowly decaying spiral, it will periodically pass through altitudes used by other constellations. At these altitudes, signals will be sent to the Terminator
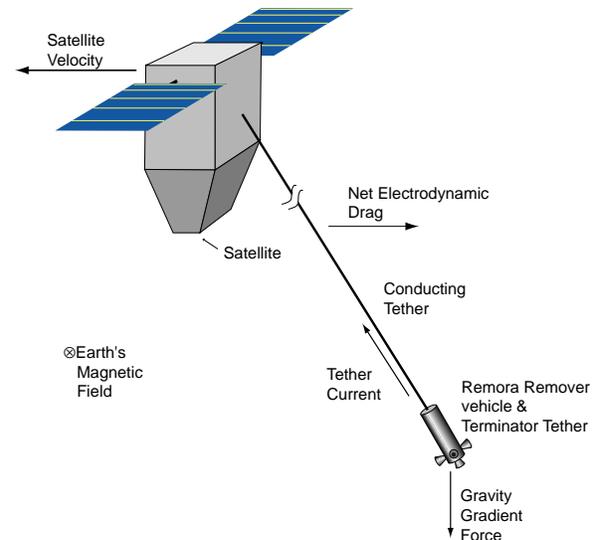
Figure 7. The Remora Remover™.

Tether system from ground stations to control its rate of descent to avoid any close approaches with other spacecraft.

IV.B. THE "REMORA REMOVER"™

The Terminator Tether concept, combined with anti-satellite technologies, can also provide a method of safely removing from orbit existing large objects such as derelict, rogue, or hostile spacecraft. This "Remora Remover" spacecraft would consist of a Terminator Tether attached to small intercept vehicle similar to the small "hit-to-kill" vehicles developed by the Space Defense Initiative Office. The Remora Remover intercept vehicle would rendezvous with a spacecraft that needs to be removed from space. Instead of hitting the spacecraft, however, the vehicle would rendezvous with the spacecraft and attach itself to the host spacecraft using a hooked net, harpoon, or adhesive "sucker." As illustrated in Fig. 7, the Remora Remover would then deploy the Terminator Tether, which would bring down both the derelict spacecraft and the intercept vehicle.

IV.C. TETHER SURVIVABILITY CONCERNS

The use of a long tether structure naturally raises concerns regarding the possibility of tether failure due to micrometeorite or debris impacts. Such concerns, however, can be allayed by using a survivable tether structure such as the multiline, failsafe "Hoytether" structure.[19] The Hoytether structure has several parallel lines that are periodically interlinked to provide redundant load- and current-bearing paths. This

redundancy enables the Hoytether to withstand many line cuts by small impactors while still maintaining the design load. As a result, it can provide very high reliability of survival for periods of years.

V.   CONCLUSIONS

By using electrodynamic drag to greatly increase the orbital decay rate of a spacecraft, a Terminator Tether system can remove unwanted objects from LEO rapidly and safely.   This technology uses a passive interaction with the Earth's magnetic field to generate drag, so no propellant or input power is required. Consequently, the Terminator Tether can deorbit even defunct spacecraft.   Using an analytical approach, we have developed methods for predicting Terminator Tether deorbit times from various orbits.   Using these methods, we have shown that tether systems massing just 2 to 5% of the total spacecraft mass can deorbit a typical communication satellite within several weeks or months, depending upon the initial orbit. It was shown that high-inclination orbits are more difficult to bring down than those in moderate to low inclination orbits. The Area-time products for polar orbits are about 50 times larger than the equivalent system in an equatorial orbit.   This still leaves nearly 3 orders of magnitude advantage in Area-Time product over neutral drag decay, even for polar orbits. The low mass requirements of a Terminator Tether system makes it highly advantageous compared to a conventional solid-rocket deorbit stage. Moreover, the drag enhancement provided by the electrodynamic tether technique is so large that the total deorbit Area-Time-Product can be reduced by several orders of magnitude compared to atmospheric drag alone, minimizing the long-lived orbital debris hazard created by a constellation spacecraft after their end-of-life. In addition, we have developed a method of optimizing the electrodynamic drag on the tether system by controlling the tether hang angle. This method also provides a simple method for stabilizing the tether libration.

Acknowledgements

REFERENCES AND NOTES

1. Loftus, J.P., *JLoftus@ems.jsc.nasa.gov*, personal communication via email Monday 10 June 1996 15:50:10.

2. Forward, R.L., "Electrodynamic Drag 'Terminator Tether'", Appendix K, in Final Report on NAS8-40690, 10 July 1996.

3. Office of Science and Technology Policy: 1995, *Interagency Report on Orbital Debris 1995*, The National Science and Technology Council

4. *NASA Safety Standard: Guidelines and Assessment Procedures for Limiting Orbital Debris.* (NSS 1740.14, August 95).

5. *Space News*, March 31-April 6, 1997, p 16.

6. *Space News*, Aug 4-10, 1997, p 30.

7. Estes, R., *restes@mars.harvard.edu*, personal communication via email Friday 31 May 1996 10:39:16.

8. Loftus, J.P. *JLoftus@ems.jsc.nasa.gov*, personal communication via email Wednesday 19 June 1996 08:20:39.

9. Lorenzini, E., personal communication, 13 March 1997.

10. Johnson, L., "Propulsive Small Expendable Deployer System Mission (ProSEDS)", *OAST Advanced Propulsion Workshop*, JPL, Pasadena, CA, 20-22 May 1997.

11. Gilchrist, B., personal communication, 12 May 1997.

12. Sanmartín, J.R., Martínez-Sánchez, M., Ahedo, E., "Bare Wire Anodes for Electrodynamic Tethers," J. Propulsion and Power, 7(3), pp. 353-360, 1993

13. Hoyt, R.P., Forward, R.L., "Electrodynamic Hoytethers for ISS and LEO Spacecraft", Appendix E of *High Strength-To-Weight Tapered Hoytether for LEO to GEO Payload Transport*, Final Report on NASA Contract NAS8-40690,10 July 1996.

14. Note that in Eq. 9, the correct velocity vector to use is the relative velocity $v_M = v_0 - \omega_E r \cos\lambda$ between the orbiting spacecraft and the geomagnetic field, since the geomagnetic field rotates with the Earth

at the rate of $\omega_E = 2\pi$ rad/day. For an equatorial orbit at an altitude of 1000 km, the velocity of the geomagnetic field is 0.536 km/s or only 7% of the orbital velocity of 7350 m/s. For nonequatorial orbits the difference is even smaller. We will ignore this small difference to keep the equations manageable.

15. By a geometric coincidence, the transverse magnetic field B_T and therefore voltage V given by Eq. 11 are both essentially constant over the entire orbit, despite the fact that the horizontal magnetic field varies from a maximum at the magnetic equator $B_H(\Lambda=0)$ to a smaller value of $B_H(\Lambda=\lambda)$ at the northernmost portion of an orbit with geomagnetic inclination λ. The variation in horizontal magnetic field strength $B_H(\Lambda)$ with latitude Λ on the Earth and the variation in the angle at which the velocity vector crosses B_N, combine to produce a constant transverse magnetic field $B_T = B_H(\Lambda=\lambda)$ over the entire orbit.

16. Jaccia, L.G., "Thermospheric Temperature, Density, and Composition: New Model," SAO Special Report 375, Mar. 1977.

17. The results for orbits with $i > 78.5$ assume that the tether system can carry current in both directions; if the tether system is designed to carry current in one direction only, the deorbit times will be roughly twice as long, due to portions of the day when the orbit is retrograde to the magnetic dipole.

18. Penzo, P.A., and Ammann, P.W., eds., *Tethers In Space Handbook, 2nd Edition*, NASA/OSF, 1989, p. 168-9.

19. Hoyt, R.P., Forward, R.L., "The Hoytether: A Failsafe Multiline Space Tether Structure," *Tether Technical interchange Meeting*, Huntsville, AL, Sept 9-10, 1997.