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Terminator Tether™: A Spacecraft Deorbit Device

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This paper investigates the use of passive electrodynamic tether drag as a method for quickly removing spent or dysfunctional spacecraft from low Earth orbits (LEO). The fundamental physical principles underlying the operation of an electrodynamic drag Terminator Tether™ are developed, some practical considerations are discussed, and calculations of the area-time product are made for spacecraft orbits representative of those that will be used in the LEO satellite constellations of the next few decades. These calculations indicate that electrodynamic drag can remove a spacecraft from a typical 700–2000-km LEO constellation orbit within a few months using a Terminator Tether system massing less than 3% of the spacecraft dry mass. Although the tether increases the cross-sectional area of the satellite system during the deorbit phase, the electrodynamic drag is so many times greater than atmospheric drag at these altitudes that the total area-time product can be reduced by several orders of magnitude, reducing the risks of collisions with other satellites. Concerns regarding tether survivability can be solved by using a multiline, fail-safe Hoytether™ construction. The Terminator Tether may thus provide a cost-effective method of mitigating the growth of debris in valuable constellation orbits.

Nomenclature

A_S = cross-sectional area of spacecraft
a = orbit semimajor axis
$\boldsymbol{B}$ = magnetic field vector
B_E = magnetic field at magnetic equator of the Earth, 31 μT = 0.31 G
B_H = magnetic-field horizontal component
B_R = magnetic-field radial component
B_T = magnetic-field transverse to both tether and spacecraft velocity vector
C_D = aerodynamic drag coefficient
d = density of tether material
$\boldsymbol{E}$ = electric field induced in tether
F_D = drag force on tether
F_E = electrodynamic force on tether
F_G = total gravity gradient force on tether
F_{GB} = gravity gradient force on tether from ballast mass
F_{GD} = gravity gradient force on tether from deployer mass
F_{GT} = gravity gradient force on tether from tether mass
G = Newtonian gravitational constant, 6.67×10^{-11} m^3/kg s^2
h = orbital altitude
I = current induced in tether
i = orbit inclination with respect to spin equator of the Earth
$\boldsymbol{L}$ = tether length vector
M_E = mass of the Earth, 5.98×10^{24} kg
M_S = mass of spacecraft
m_B = mass of ballast
m_D = mass of deployer
m_T = mass of tether
P = power
R = resistance
R_C = resistance of current control resistor
R_E = radius of the Earth, 6378 km
R_T = resistance of tether
$\boldsymbol{r}$ = radius vector from center of the Earth
T_E = electrodynamic torque on tether
T_G = total gravity gradient torque on tether
T_{GB} = gravity gradient torque on tether from ballast mass
T_{GD} = gravity gradient torque on tether from deployer mass
T_{GT} = gravity gradient torque on tether from tether mass
t = time
V = voltage induced in tether
$\boldsymbol{v}$ = velocity vector
v_M = velocity with respect to the Earth's magnetic field
v_o = orbital velocity
α = angle of tether from local zenith
β = coordinate angle from magnetic pole of the Earth
Γ = gravity gradient at radius r from the Earth, GM_E/r^3
θ = coordinate angle around magnetic pole of the Earth
λ = orbit inclination with respect to magnetic equator of the Earth
ρ = atmospheric density
σ = resistivity of conductive tether material
ϕ = tilt of the Earth magnetic pole from the Earth spin pole, 11.5 deg
ω_E = angular velocity of the Earth, 2π rad/day

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 of their useful lives. The terms Terminator Tether™, Remora Remover™, Hoytether™, and Hoytape™ are trademarks of Tethers Unlimited, Inc. When a spacecraft fails, or has completed its mission and is no longer wanted, the Terminator Tether device, weighing a small fraction of the mass of the host spacecraft, will deploy itself below the host spacecraft. The device will provide a means of providing electrical contact with the ambient plasma at both ends of the tether to enable the tether to transmit current 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 the orbit to deorbit far more rapidly than it

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would from atmospheric drag alone. Whereas a defunct spacecraft left in its orbit can take hundreds or thousands of years to deorbit because of 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 Sec. I.A of this paper, we will begin by discussing the orbital debris problem motivating the development of the Terminator Tether. We then describe in Sec. I.B the basic concept of electrodynamic tether drag and review the results of past experiments related to this concept. In Sec. II we will develop analytical methods for predicting the effectiveness of Terminator Tether systems for deorbiting spacecraft from various orbits. We will describe methods of optimizing the electrodynamic drag on the spacecraft, while stabilizing the electrodynamic tether libration. In Sec. III 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, in Sec. IV we describe two implementations of the Terminator Tether concept for reducing the LEO debris environment.

A. Motivation: Orbital Debris in LEO

Currently, the U.S. Space Command tracks roughly 7000 objects in LEO. Fewer than 400 of these objects are operational spacecraft. The rest are spent rockets and derelict spacecraft.[1] 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 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."[2] 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 upper stage 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 U.S. Department of Defense 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, the ESA, the United Kingdom, India, France, Italy, and the United States.

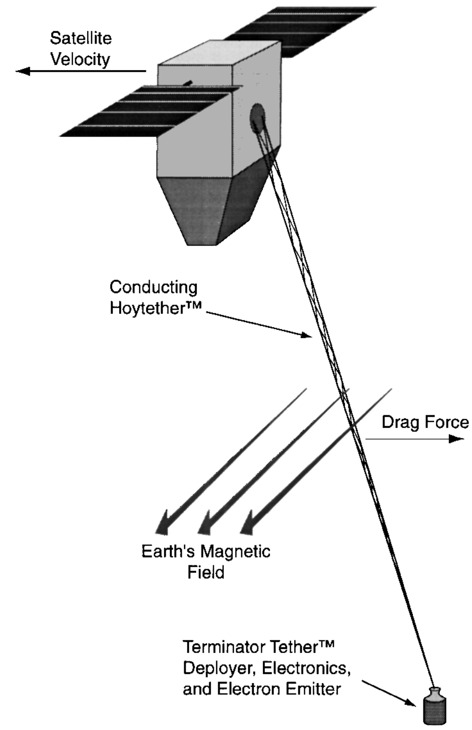

Fig. 1 Terminator Tether concept.

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 committed to deorbiting their satellites at the end of their operational lifetimes.[3,4] 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 postmission satellites and launch/dispenser rocket stages.

B. Summary of Concept

The electrodynamic drag concept for deorbit of LEO spacecraft is illustrated in Fig. 1. The idea of using electrodynamic drag to remove unwanted spacecraft from orbit was invented by Joseph P. Loftus of NASA Johnson Space Center in June 1996. A first-order analysis published by Robert L. Forward in July 1996[5] found that a conducting tether hanging down from a spacecraft orbiting above the magnetic equator will generate electrical power in the tether given by the equation

$$P = m_T (v_M B_T)^2 / \sigma d \tag{1}$$

where v_M is the velocity of the tether with respect to the magnetic field component B_T transverse to both the velocity and the tether. 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 10-kg tether of aluminum with a resistivity of 27.4 nΩ-m and density of 2700 kg/m^3, hanging from a spacecraft orbiting over the magnetic equator at an altitude of 1000 km, with a velocity of 6814 m/s relative to the Earth's transverse magnetic field of 20 μT, the power dissipated is 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.[6]

Experimental Confirmations of Induced Power Levels

Power levels of the magnitude estimated in the preceding paragraph have been measured in an orbital space experiment, the

TSS-1R mission carried out on the Shuttle Orbiter in 1995. 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 upward, 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 3850 W. Most of this energy went into the electrical arc, which burned through the tether, causing it to break and stopping 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, whereas 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 electrodynamic drag force can be obtained from a low-mass conducting tether attached to a host spacecraft, provided the ends of the conductor can exchange sufficient numbers of electrons with the surrounding space plasma.

Experimental data from the TSS-1R data also produced the amazing 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 m^2 of bare surface area of the Italian spacecraft were sufficient to collect the 1.1 A of electron current.

Planned Flight Demonstration of Electrodynamic Drag Deorbiting

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,[7] NASA Marshall Space Flight Center has formed a team, which includes the present authors, to embark on a new flight experiment.[6] The Propulsive Small Expendable-Tether Deployer System (ProSEDS) mission is scheduled for a piggyback flight on a Delta II launch of an Air Force global positioning satellite in August 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 flight experiment is essentially a demonstration of the Loftus electrodynamic drag deorbit concept and the first step in the development of a Terminator Tether.

The ProSEDS mission is presently baselined to use a 5-km-long copper wire massing 18 kg, a 10-km-long nonconducting tether, and a 25-kg ballast mass on the end of the tether. The total of 15 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 nearly at right angles to both the direction of the spacecraft motion in the nominal east-west direction and the Earth's near-equatorial magnetic field in the nominal north-south 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 commanded 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, the Terminator Tether. It 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[8] made of aluminum wire. 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 burn up 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, whereas a shorter tether length means a lower collision cross-section area-time product during deorbit. The Terminator Tether would autonomously maintain contact with ground control during the deorbit phase through its own telemetry system. Ground controllers can thereby control the rate of descent by commanding changes in the amount of current allowed to flow through the tether. In this manner the 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 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 of 17.0 nΩ-m, a density of 8933 kg/m^3, and a specific conductivity of $1/\sigma d = 6585$ m^2/Ω-kg. Aluminum has a resistivity of 27.4 nΩ-m, which is significantly greater than that of copper, but it has a much lower density of 2700 kg/m^3. As a result, aluminum's specific conductivity of 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 of 32.5 nΩ-m, density of 1850 kg/m^3, and a specific conductivity of 16,630 m^2/Ω-kg, 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 that 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.

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. Because 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 of 10 kg, containing an aluminum tether with a mass of 10 kg and a volume of 3.70×10^{-3} m^3. If this 10 kg of aluminum were formed into a tether with a length of 2 km and a cross-sectional area of 0.85 mm^2, then the end-to-end resistance of the tether would be 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

A. Lowering a Spacecraft Orbit Using an Electrodynamic Tether

To determine the effectiveness of the Terminator Tether system for deorbiting a spent spacecraft, we will now develop analytical tools for predicting the time required for an electrodynamic tether to deorbit a spacecraft from a specified altitude and inclination. 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. We will assume that there is a balance between the electrodynamic drag on the tether and the gradient forces on the tether that cause 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 oscillate 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

$$\boldsymbol{L} = L(\hat{\boldsymbol{r}}\cos\alpha + \hat{\boldsymbol{v}}\sin\alpha) \tag{2}$$

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 because of ionospheric plasma density variations between the day and night sides of the Earth.

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 $\varphi = 11.5$ deg, as illustrated in Fig. 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[9] as consisting of two components, a component in the radial direction,

$$B_R = \left(2B_E R_E^3 / r^3\right)\cos\beta \tag{3}$$

and a component in the horizontal or magnetic north-south direction,

$$B_H = \left(B_E R_E^3 / r^3\right)\sin\beta \tag{4}$$

where, as shown in Fig. 3, β is the magnetic latitude starting from the magnetic pole, with $\beta = 0$ deg at the north magnetic pole and $\beta = 90$ deg at the magnetic equator.

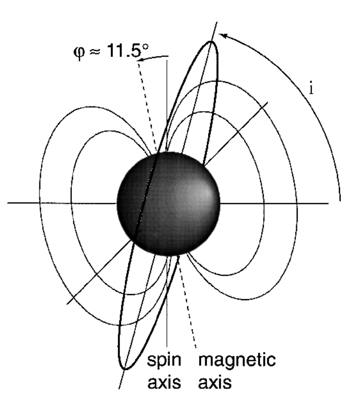

Fig. 2 Tilted-dipole approximation to the geomagnetic field.

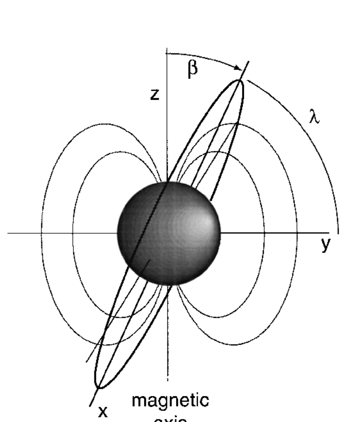

Fig. 3 Spacecraft orbit in the reference frame aligned with the magnetic axis.

Orbit Reference Frame

To make the calculations tractable, we will perform the orbit calculations in a reference frame that is rotated so that its z axis is aligned with the axis of the Earth's magnetic dipole as shown in Fig. 3. 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. In this rotated frame, the circular orbit of the spacecraft can be parameterized in the geomagnetic field reference frame in terms of the orbit inclination angle λ and the angle θ around the z axis as

$$\boldsymbol{r} = \begin{bmatrix} x \\ y \\ z \end{bmatrix} = r\begin{bmatrix} \cos\theta \\ \cos\lambda\sin\theta \\ \sin\lambda\sin\theta \end{bmatrix} \tag{5}$$

and the velocity as

$$\boldsymbol{v} = \begin{bmatrix} v_x \\ v_y \\ v_z \end{bmatrix} = v_o\begin{bmatrix} -\sin\theta \\ \cos\lambda\cos\theta \\ \sin\lambda\cos\theta \end{bmatrix} \tag{6}$$

where the magnitude of the orbital velocity is given by

$$v_o = \sqrt{GM_E/r} \tag{7}$$

Expressed in Cartesian coordinates, the geomagnetic field is given by

$$\boldsymbol{B} = \frac{B_E R_E^3}{r^3}\begin{bmatrix} 3xz/r^2 \\ 3yz/r^2 \\ 3z^2/r^2 - 1 \end{bmatrix} \tag{8}$$

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

$$\boldsymbol{E} = -\boldsymbol{v} \times \boldsymbol{B} \tag{9}$$

Note that in Eq. (9) the correct velocity vector to use is the relative velocity $v_M = v_o - \omega_E r\cos\lambda$ between the orbiting spacecraft and the geomagnetic field because 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.

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

$$V = \boldsymbol{E}\cdot\boldsymbol{L} \tag{10}$$

After some trigonometric yoga Eq. (10) reduces to

$$V = \left(LB_E R_E^3 v_o / r^3\right)\cos\alpha\cos\lambda = LB_T v_o\cos\alpha \tag{11}$$

where $B_T = B_E R_E^3 v_o \cos\lambda / r^3 = B_H(\beta = \pi/2 - \lambda)$ is the transverse component of the magnetic field at right angles to both the velocity vector and the tether. [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 $B_H(\beta = \pi/2)$ to a smaller value of $B_H(\beta = \pi/2 - \lambda)$ at the northernmost portion of an orbit with geomagnetic inclination λ. The variation in horizontal magnetic field strength $B_H(\beta)$ with magnetic latitude β on the Earth and the variation in the angle at which the velocity vector crosses B_H combine to produce a constant transverse magnetic field $B_T = B_H(\beta = \pi/2 - \lambda)$ over the entire orbit.]

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 given by

$$\boldsymbol{I} = (V/R)\hat{\boldsymbol{L}} \tag{12}$$

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 on the tether system is

$$\boldsymbol{F}_E = L(\boldsymbol{I} \times \boldsymbol{B}) = (V/R)(\boldsymbol{L} \times \boldsymbol{B})$$

$$= -L^2 B_E^2 R_E^6 v_o \cos\alpha \cos^2\lambda \big/ Rr^6 = -L^2 B_T^2 v_o \cos\alpha \big/ R \tag{13}$$

The drag force on the tether is the component of the electrodynamic force that is parallel to the velocity vector:

$$F_D = \boldsymbol{F}_E \cdot \hat{\boldsymbol{v}} = F_E \cos\alpha = -L^2 B_E^2 R_E^6 v_o \cos^2\alpha \cos^2\lambda \big/ Rr^6$$

$$= -L^2 B_T^2 v_o \cos^2\alpha \big/ R \tag{14}$$

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

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

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

$$\langle\cos^2\lambda\rangle = \tfrac{1}{16}\{6 + 2\cos 2i + 3\cos[2(i - \varphi)] + 2\cos 2\phi$$

$$+ 3\cos[2(i + \varphi)]\} \tag{16}$$

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:

$$\Delta t = \left(M_S R \big/ 12L^2 B_E^2 R_E^6 \cos^2\alpha \langle\cos^2\lambda\rangle\right) a^6 \Big|_{a_{\text{initial}}}^{a_{\text{final}}} \tag{17}$$

If current can flow in only one direction in the system, then the calculation of $\langle\cos^2\lambda\rangle$ must be handled differently for orbits with inclinations greater than 78.5 deg. This is because 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.

The decay time predicted by Eq. (17) is quite rapid for spacecraft at low altitudes and low inclinations. For an Orbcomm 1 spacecraft at 775-km altitude and 45-deg inclination, the deorbit time with an aluminum tether massing 2.5% of the host mass would be only 11 days, whereas the deorbit time using air drag alone is over 100 years. For a Globalstar or Teledesic type spacecraft at 1400-km altitude and 52-deg inclination, the electrodynamic drag deorbit time is 37 days, whereas the air drag deorbit time would be over 8000 years. Spacecraft in higher inclination orbits will take longer to deorbit because the spacecraft's orbit carries it more along the geomagnetic field lines than across them. For example, Eq. (17) predicts that a spacecraft at 1000-km altitude and 50-deg inclination can be deorbited in only 18 days, whereas a spacecraft at 1000-km altitude and 66-deg inclination will require 45 days to deorbit. If the spacecraft's orbit is greater than 78.5 deg, then during portions of each day the spacecraft will cross the magnetic field lines in a retrograde direction. During that retrograde portion of the trajectory, the voltage induced in the tether is reversed. In this scenario it is desirable to arrange for electron emitters at both ends of the tether so that the electrodynamic drag can operate over the entire orbit. For a specific example, a bidirectional electrodynamic drag aluminum tether massing 2.5% of an Iridium spacecraft at 780-km altitude and 86.4-deg inclination will deorbit the spacecraft in 7.5 months, compared to 100 years for air drag deorbit alone.

B. Maximizing Electrodynamic Drag

Because the electrodynamic forces are perpendicular to the tether, the tether will tend to trail behind the spacecraft. In fact, it is necessary for the tether to hang at an angle behind the vertical for the electrodynamic forces to decelerate the spacecraft. The hang angle of the tether will depend 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 angle 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 of stabilizing the tether angle.

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 preceding sections, both theory and experiment show that, provided the conducting tether is moved rapidly through the Earth's magnetic field 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 is embedded in a magnetic field, there will be an electrodynamic force 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):

$$F_E = -L^2 B_T^2 v_o \cos\alpha \big/ R \tag{18}$$

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 Fig. 4. Assuming that the electrodynamic drag force

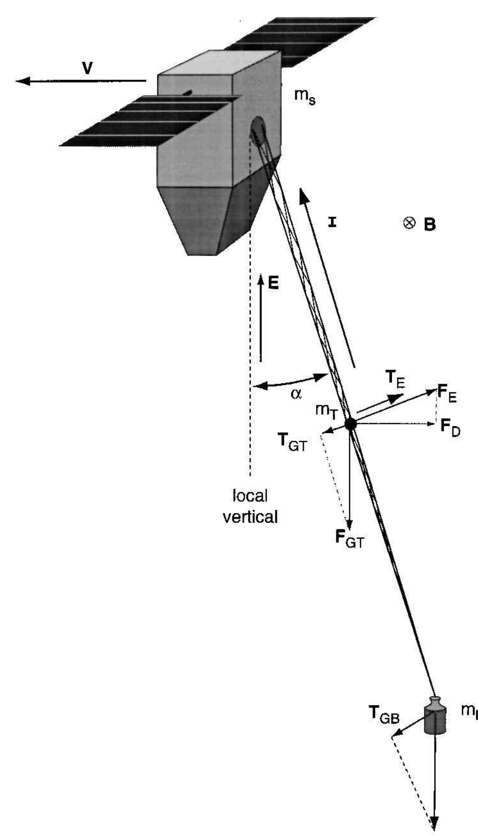

Fig. 4 Gradient and electrodynamic forces and torques on the tethered system.

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 Fig. 4. The electrodynamic torque on the tether is

$$T_E = F_E(L/2) = L^3 B_T^2 v_o \cos\alpha / 2R \tag{19}$$

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. The direction desired depends on which end of the tether is connected to the electron emitter so that the induced voltage in the tether will produce an excess of electrons at the electron emitter end. For a typical prograde orbit moving in the direction of the rotation of the Earth, the electron emitting end will be the end closer to the Earth. The desired direction for the tether is selected by having the deployer eject the ballast mass in the desired direction. Once the ballast mass has been started in that direction, the combined gravity and centrifugal gradient forces will cause the ballast mass to continue to accelerate in the desired direction until it is brought to a halt by the full deployment of the tether.

If there were no electrodynamic or atmospheric drag, the equilibrium direction of the tether would be exactly along the vertical because the combined gradient field is a maximum in that direction. Because we expect a significant amount of electromagnetic drag, the actual angle 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 Fig. 4. In the following analysis we will find there is an optimum angle for α that produces the largest electrodynamic drag force on the host spacecraft, hastening its deorbit time.

The combined vertical gravity gradient 2Γ and centrifugal gradient field Γ acting on the ballast mass at the end of the tether will produce a gradient force given by

$$F_{\mathrm{GB}} = 3\Gamma m_B L \cos\alpha \tag{20}$$

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 and the strength of the total 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 toward the center of the Earth. The component of this gradient force that is at right angles to the tether, given by $F_{\mathrm{GB}}\sin\alpha$, will produce a torque on the tether that tends to restore the tether toward the vertical, lessening the angle α.

$$T_{\mathrm{GB}} = LF_{\mathrm{GB}}\sin\alpha = 3\Gamma m_B L^2 \sin\alpha\cos\alpha \tag{21}$$

The tether mass 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 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 caused by the tether mass is then

$$F_{\mathrm{GT}} = \tfrac{3}{2}\Gamma m_T L \cos\alpha \tag{22}$$

The gradient torque caused by the tether mass is

$$T_{\mathrm{GT}} = (L/2)F_{\mathrm{GT}}\sin\alpha = \tfrac{3}{4}\Gamma m_T L^2 \cos\alpha\sin\alpha \tag{23}$$

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

$$T_G = T_{\mathrm{GB}} + T_{\mathrm{GT}} = 3\Gamma(m_B + m_T/4)L^2\cos\alpha\sin\alpha \tag{24}$$

It is important to notice the variation of the total gradient torque as the tether angle α is varied. Because 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 $\alpha = 45$ deg, 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 catastrophic instability, for if there is a further increase in the electrodynamic drag force caused by an increase in magnetic field strength or plasma density, causing a increase in current flow through the tether and causing the angle α to become greater than 45 deg, 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 deg, where the drag force will also drop to near zero. It would therefore 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 possibility of tether instability, the ProSEDS mission planners are planning on using a large ballast mass and a long nonconducting tether to keep the gradient forces high. To keep the electrodynamic forces from getting too large, the mission planners are also planning on limiting the current flow through the tether to less than 0.5 A average.

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:

$$T_E = T_G = T_{\mathrm{GB}} + T_{\mathrm{GT}} \tag{25}$$

or, using Eq. (21) and Eq. (26),

$$T_E = \left(B_T^2 L^3 v_o / 2R\right)\cos\alpha = 3\Gamma(m_B + m_T/4)L^2\sin\alpha\cos\alpha = T_G \tag{26}$$

Simplifying Eq. (26), 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:

$$B_T^2 L v_o / R = 6\Gamma(m_B + m_T/4)\sin\alpha \tag{27}$$

At first glance it might seem that the optimum angle for the tether would be 45 deg because at that angle the gradient torque is largest and therefore can counteract a larger electrodynamic drag force, despite the fact that at 45 deg the tether is at the onset of instability. The optimum angle, however, is that which maximizes the horizontal or drag component of the electrodynamic force that opposes the host spacecraft motion, not the total electrodynamic force. The vertical component of the total electrodynamic force does not decelerate the spacecraft; it merely lifts the tether and spacecraft at right angles to the spacecraft velocity vector, in essence counteracting a portion of the Earth's gravity pull. As a result, the spacecraft orbital radius is a little less than it would be for a given spacecraft velocity. Only the horizontal component of the total electrodynamic force decelerates the spacecraft. This horizontal drag force is given by Eq. (14):

$$F_D = F_E\cos\alpha = -\frac{B_T^2 L^2 v_o}{R}\cos^2\alpha = -\frac{B_T^2 L^2 v_o}{R_C + \sigma d L^2/m_T}\cos^2\alpha \tag{28}$$

This equation indicates that to obtain maximum drag force it is desirable to have a 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 that can allow the maximum electrodynamic force which is available to produce the maximum drag force. If, however, the amount of drag force that can be applied to the tether is limited by tether instability, as it is in the ProSEDS mission and the various Terminator Tether applications, then instead of looking at the electrodynamic limits to maximizing the drag force we want to look at the gradient limits to maximizing the drag force. To do this, we use Eq. (27) in Eq. (28) to obtain

$$F_D = -6\Gamma L(m_B + m_T/4)\sin\alpha\cos^2\alpha \tag{29}$$

This equation shows that for maximum drag force on the host spacecraft one wants a long tether length L, as well as massive ballast mass m_B and tether mass m_T. The equation also indicates that a small tether angle α (tether near vertical) is not optimum. 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, which gives the maximum electrodynamic drag force while still keeping the tether torques balanced and under control, is $\alpha = \arctan(0.707) = 35.26$ deg. This angle is well below the angle of 45 deg, where tether instability sets in. With this angle selected we obtain an equation for the maximum stable drag force of

$$F_D(\max, \alpha = 35.26\ \text{deg}) = -6\sin\alpha\cos^2\alpha\ \Gamma L(m_B + m_T/4)$$

$$= -2.31\Gamma L(m_B + m_T/4) \tag{30}$$

The tether angle in a Terminator Tether can 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, but the simplest is to merely measure the drag acceleration on the host spacecraft 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 global positioning system 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 will also allow stabilization of the tether oscillations.

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 $\Gamma = 0.99 \times 10^{-6}\ \text{s}^{-2}$ and the electrodynamic tether has a length of 5 km, a mass of 10 kg, a ballast mass of 10 kg, and a tether tilt angle of 35.26 deg, then the gradient-force-limited maximum allowable stable drag force using Eq. (32) is 0.143 N. This is to be compared with the electrodynamic drag force obtainable from the aluminum tether moving at velocity 6814 m/s with respect to the transverse magnetic field of 20 μT. If we assume the control resistor 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 0.143 N opposing the motion of the host spacecraft, assumed to be in a magnetic equatorial orbit with $\lambda = 0$ and a velocity with respect to the magnetic field of 6814 m/s, is equivalent to a deceleration power of

$$P = F_D v_M = 975\ \text{W} \tag{31}$$

Because, 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 Eqs. (14) or (30) for the electrodynamic drag force and multiplying it by the spacecraft velocity v_M with respect to the geomagnetic frame:

$$P = \frac{V^2}{R} = \frac{(B_T L v_M \cos\alpha)^2}{R_c + \sigma d L^2/m_T} = F_D v_M \tag{32}$$

An aluminum tether of length 5 km and mass of 10 kg has a tether resistance of 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 6814 m/s and will see a transverse magnetic field of 20 μT. Using Eq. (32), we calculate that the preciding aluminum tether trailing at the optimum drag tether angle of 35.26 deg has the ability to generate up to 1670 W of power if the control resistor is set to zero. A control resistor of 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 35.26 deg, despite variations in magnetic field strength and plasma density. Because B_T varies as $\cos\lambda$, a 10-kg tether will suffice for orbit inclinations up to $\lambda = 40$ deg. For orbits with higher inclinations and therefore lower horizontal magnetic fields, a tether with a larger mass would be needed. Because the tether mass also determines the maximum gradient-limited drag force, the more massive tether would allow for a higher allowable stable drag force.

C. Terminator Tether Effectiveness in Deorbiting Spacecraft

Comparison with Atmospheric Drag 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 Sec. 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-kilometerlong 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 from atmospheric drag alone in a near-circular spiral trajectory, the area-time product is given by

$$A_S \int dt = -\frac{M_S}{C_D}\int_{a_{\text{initial}}}^{a_{\text{final}}} \frac{da}{\rho(a,t)\sqrt{\mu_e a}} \tag{33}$$

Figure 5 compares the area-time products for spacecraft with Terminator Tether systems to the area-time products for spacecraft deorbiting caused by atmospheric drag alone. Upper curves show results for atmospheric drag alone at mean and extremes of exospheric temperature. Lower three curves show results for Terminator Tether systems. 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 2.0. We have used the 1977 Jaccia static atmosphere model for the exospheric temperatures.[10] Figure 5 shows that the use of electrodynamic tether drag can reduce the deorbit area-time 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 its telemetry system to respond to ground-control commands to change the rate of descent

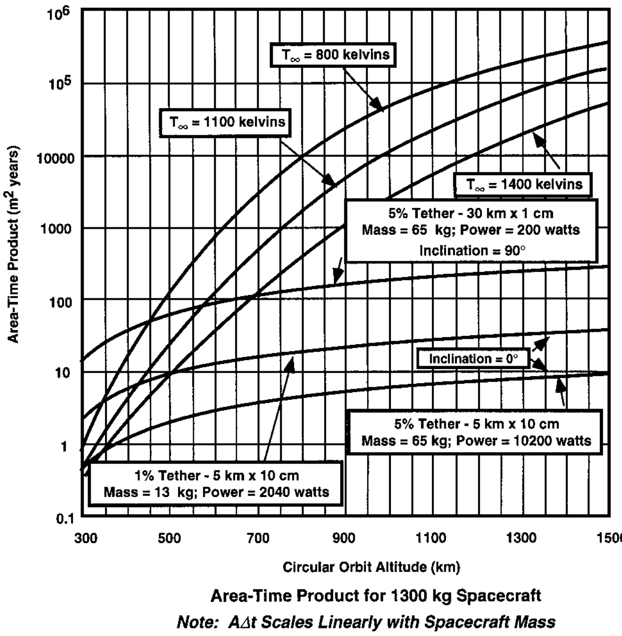

Fig. 5 Area-time product vs initial altitude.

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 Fig. 5. For example, a 5-km, 10-kg tether whose resistance is 185 ohms would generate over 4000 W 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 1570 W 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.

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 from 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 ($\approx$ 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 0.6–0.75. If the deorbit stage is required to perform its own attitude determination, the stage propellant mass fraction may be as low as 0.5.

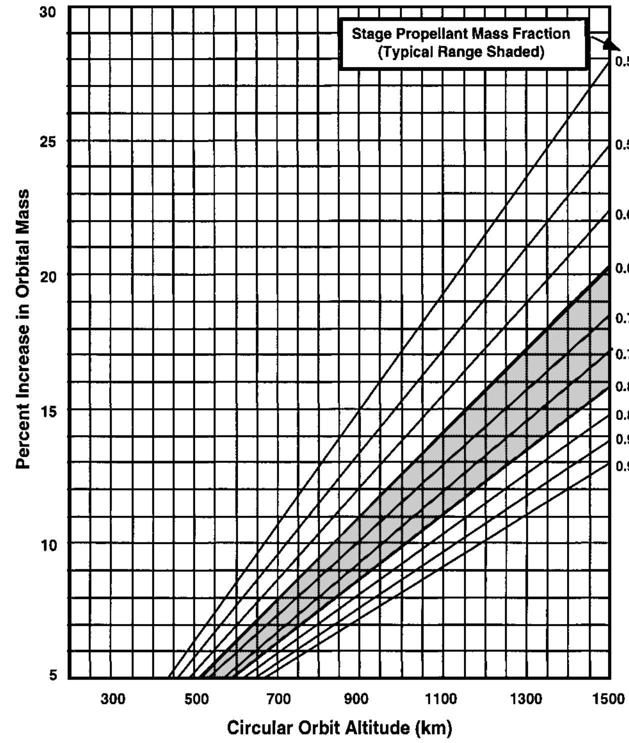

Fig. 6 Conventional solid rocket motor deorbit system percent mass increase vs altitude and stage propellant mass fraction (I_{sp} = 288 s).

Figure 6 shows that any solid-rocket deorbit system will require a mass allocation that is a 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 2–5% of the launch mass. The mass savings achieved with the Terminator Tether system can be used for additional revenue-producing transponders or for more station-keeping fuel to provide longer operational lifetimes.

III. Implementation

A. Tether Structure

The basic optimum structure for the electrodynamic tether in a Terminator Tether system would be one of the many types of Hoytethers™.[8] A multiline (2–10 primary line) Hoytape will provide the largest contact area with the plasma because both sides of the tape would be able to pass current to the plasma. If the spacing between the primary lines is chosen to be larger than twice the expected average Debye length of the plasma, then the effective maximum current collection area per unit length of the Hoytape is proportional to the width of the Hoytape mesh, not the diameter of the wires in the mesh. Thus, a Hoytape not only provides an assured longer life for the Terminator Tether, but very short lengths may also provide a very large current collection areas.

The deployer for the tether can deploy the Terminator Tether either down or up or both. The deployer could stay attached to the spacecraft as was done in the SEDS missions and as is planned for the ProSEDS mission, which will use a standard SEDS deployer. However, a better alternative would be to have the deployer ejected from the spacecraft with one end of the tether still attached to the spacecraft, reeling out tether as it leaves, as shown in Figs. 1 and 4. The empty deployer would then act as a ballast mass at the end of the Terminator Tether, adding to its electrodynamic drag force performance, as can be seen in Eq. (30).

The standard Terminator Tether will be a completely autonomous package with no connections to the host spacecraft except bolt holes and minimal electronic connections to enable monitoring of the host spacecraft. It will contain the electrodynamic tether and its deployer, communication, command, and control circuitry, a battery sufficient to operate the electronics during the deployment period, a small photovoltaic array to supply a trickle charge to the battery during the long waiting period prior to the deorbit command, a more robust battery charging circuit that pulls power off the current running through the tether, one or more methods, such as plasma contactors or field emission devices, to collect and eject electrons from the ends of the tether, one or more methods to control the current through the tether, and one or more methods, such as an accelerometer package, to determine the maximum electrodynamic drag and/or tether tilt angle.

The Terminator Tether package will normally be powered down except for timing circuits, backed up by temperature sensors on the base-plate connection to the host spacecraft—a cold host means a dead host, and accelerometer signals—continued acceleration in free fall means a stuck thruster and a spacecraft out of control. Periodically the Terminator Tether electronics package will wake up, go through a self-check, listen for radio signals from the host spacecraft to determine if the host is still functioning, and make a status report to ground control through its telemetry system. If the Terminator Tether fails to report or reports a serious malfunction, then ground control still has the option of using the last portion of the stationkeeping propellant on the host spacecraft to deorbit the spacecraft. When the host spacecraft dies, or becomes obsolete, ground control can activate the deorbit sequence the next time the Terminator Tether checks in. The system will incorporate suitable safety features to prevent accidental or malicious activation of the deorbit sequence. The most reliable measure would be to have the Terminator Tether check for radio transmissions from the host spacecraft. If the host is still transmitting, no deorbit would be performed.

Once the deorbit sequence is initiated, the tether will be deployed with the current control circuit open. Although a voltage will be generated across the tether and can be measured between the end of the tether and the Terminator Tether ground, with no current flow there will be no drag. If needed, small amounts of current flow can be used to damp out any oscillations resulting from deployment. Once the tether is stabilized, the current control circuit would slowly allow the current to rise. The drag force, as measured by the accelerometers or other means, will start to increase, and the tether will start to lag behind. After a few orbits at low drag, to determine the maximum and minimum voltages experienced and the ease with which electrons are collected and ejected from the ends of the tether the current flow would be allowed to increase until the maximum deceleration level is reached. The current flow would then be varied as needed to maintain that maximum deceleration level while at the same time using phase-shifted rate feedback to cancel out any induced tether oscillations from the orbit going through regions of high and low plasma density on the dark and light hemispheres, or through regions of low or high magnetic field. As the host spacecraft starts its decent from the constellation, it will likely be necessary to have ground control vary the rate of descent to avoid fratricide with other spacecraft in the constellation. Of course, because ground control has control over the rest of the spacecraft in the constellation, their stationkeeping propellant systems could also be used to avoid the host spacecraft and its tether during the early phases of the deorbit process. After the host spacecraft is clear of the constellation, the deorbit process can proceed with little input from the ground, except for those orbital altitudes known to contain large spacecraft, when again ground control of the rate of descent should be sufficient to avoid collision.

It is not known at this time if the control of the rate of descent is sufficient to ensure that the host deorbits into one of the ocean basins. Because the strength of the magnetic field is stronger at lower altitude, there will be more electrodynamic drag force available. Whether that stronger control of the electrodynamic portion of the drag can compensate for the unpredictable portion of the variations in the atmospheric drag at low altitudes is unknown at this time.

B. Remora Remover™

The Terminator Tether concept, combined with antisatellite 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 illustrated in Fig. 7 will consist of a Terminator Tether attached to small seeker missile similar to the small hit-to-kill missiles developed by the Space Defense Initiative Office, which has evolved into the Clementine vehicle used for space exploration. Because, in this case, the host vehicle for the tether has operational electronics, the amount of specialized electronics needed to control the current in the tether would be minimal. The Remora Remover missile will hunt down a spacecraft that needs to be removed from space, but instead of hitting the spacecraft the missile would be programmed to rendezvous with the spacecraft and attach itself to the host spacecraft using a hooked net, harpoon, or adhesive sucker. The Remora Remover missile will then deploy the Terminator Tether, which would bring down both the derelict spacecraft and the missile.

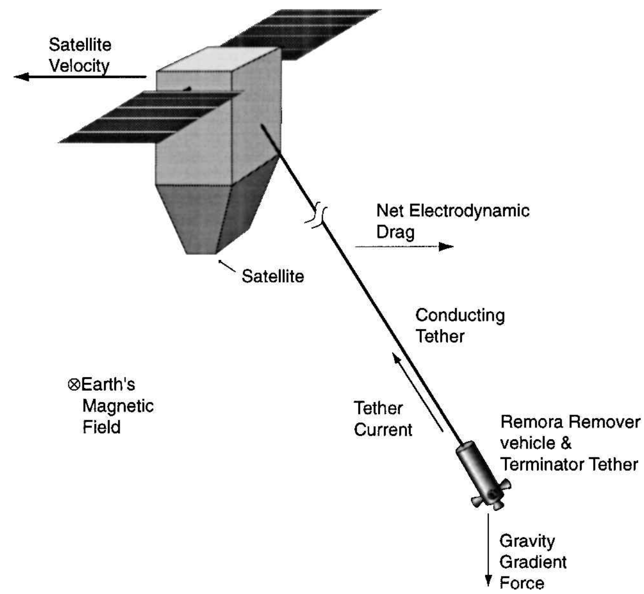

Fig. 7 Remora Remover.

IV. Conclusions

By using electrodynamic drag to increase greatly the orbital decay rate of a spacecraft, a Terminator Tether system can remove unwanted objects from LEO rapidly and safely. 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 2–3% of the total spacecraft mass can deorbit a typical communication satellite within several weeks or months, depending upon the initial orbit. The low-mass requirements of a Terminator Tether system make 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.

Acknowledgments

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