A FLIGHT EXPERIMENT OF ELECTRODYNAMIC TETHER USING A SMALL SATELLITE: AS THE FIRST STEP FOR DEBRIS REMOVAL

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Abstract

The active removal of existing space debris is the one of the most proactive strategies available to suppress space debris growth. The Aerospace Research and Development Directorate, Japan Aerospace Exploration Agency, is investigating an active space debris removal system that employs highly efficient electrodynamic tether (EDT) technology as its orbital transfer system. As the first step toward the realization of this debris removal system, an EDT flight experiment using a small satellite is planned to establish and demonstrate EDT technology, and to measure EDT characteristics such as electron emission and collection in the space plasma. In this paper, the results of precise numerical simulations for mission analysis are presented, including available electric currents, orbital changes, tether stability and deployment dynamics, considering small satellite characteristics.

1. Introduction

Space debris is steadily on the increase, and effective, practicable mitigation measures are essential to ensure safe space development in the future. Space debris left in orbit is dangerous since it can pose a serious collision risk to operating space systems, and moreover a huge quantity of smaller debris could be generated by mutual collisions between large objects. If the rate of increase in debris generated by on-orbit collisions overcomes the rate of debris reduction through re-entry due to atmospheric drag, the amount of debris in orbit will increase exponentially as the result of mutual collisions. Some evolutionary models predict that such "collisional cascading" has already started in some crowded regions such as in the 900–1000 km altitude band, and the effect of mutual collisions will be apparent within a few decades even if no new objects are launched[1)-3)].

To prevent this unpleasant scenario from occurring, the active removal of large space debris—that is, defunct or malfunctioning satellites and rockets—is one of the most proactive strategies and should be started as early as possible. Collisional cascading starts with an increase in the amount of small debris objects that are difficult to observe from the ground[4)], so it will be too late to delay debris removal until we observe many collisions of trackable debris objects. Three on-orbit collisions involving trackable debris have already occurred, and it is estimated that collisions involving items of debris larger than 1 cm occur more than once a year[5)]. It is difficult to remove small-sized debris, and so countless small debris objects may become a great threat to space activities, especially manned space flight. At high altitudes, where atmospheric drag is too small to decrease the debris population, removal is the only solution. Debris removal started at an early stage can make the space environment cleaner than it is today, so it will not merely maintain the level of today's environment, but will lower future costs of debris countermeasures such as debris protection bumpers and debris collision avoidance maneuvers, not to mention the risk of losing whole spacecraft.

With this aim, the Japan Aerospace Exploration Agency (JAXA) has been studying a debris removal system that captures and disposes of spacecraft that have ended their missions or have malfunctioned[6)-9)]. A service satellite dedicated to debris removal would rendezvous with a debris object and capture it for transfer into a lower altitude disposal orbit. However, it is unfeasible to transfer large debris objects from the useful, crowded regions (800-1,500 km alt.) to a disposal orbit (e.g. 630 km alt.) using a conventional propulsion system owing to the large propellant requirement. In this respect, electrodynamic tether (EDT) systems are very promising, since they are able to generate a sufficiently large thrust to conduct orbit transfers within a realistic time period without the need for much propellant, by utilizing interactions with the Earth's magnetic field.

Although many on-orbit experiments of tether systems including electrodynamic tethers have been conducted, EDT thrust generation has not yet been demonstrated. Thus, the first step toward the realization of the debris removal system is to establish EDT technology, and JAXA is planning an EDT flight experiment using a small satellite in the near future. In this paper, the dynamics of an EDT on a small satellite is studied by precise numerical simulations for certain aspects of mission analysis, such as available electric currents, orbital changes, tether stability, and deployment dynamics, considering the characteristics of small satellites.

2. Electrodynamic Tether (EDT)

Orbital transfer is one of the key technologies for debris removal. It is necessary to transfer debris to orbits where the orbital lifetime is at most 25 years, since most international debris mitigation guidelines require spacecraft in Low Earth Orbit (LEO) to re-enter within this time. Although the orbital lifetime of an object depends on its Area-to-Mass (A/M) ratio, a 630 km circular orbit is the standard disposal orbit altitude assuming objects with an average A/M of 0.01 m^2/kg.

To transfer an object to such an orbit using a conventional propulsion system with an Isp of 200 sec., a spacecraft in an 800 km orbit will require about 4% of its mass in propellant, and one in a 1400 km orbit will require about 13% of its mass. Such propellant requirements make it unfeasible for a removal spacecraft with conventional propulsion to remove more than one debris object from LEO. We therefore propose the electrodynamic tether (EDT), which uses interaction with the geomagnetic field to generate a drag force, as a highly efficient propulsion system for lowering orbits.

The principle of EDT thrust is as follows (Fig. 1). An electromotive force (EMF) is set up within a conductive tether deployed from a space system as it moves through the geomagnetic field in its orbit round the Earth. If a plasma contactor is attached to each end of the tether, they emit and collect electrons, closing the circuit via the ambient plasma, and an electric current flows through the tether. The tether then generates a Lorentz force via interaction between the current and the geomagnetic field which acts opposite to the direction of flight. An EDT can thus provide deceleration without propellant, and shows promise as a high efficiency propulsion system for debris de-orbit. Furthermore, the thrust of the EDT is so low that it does not have to be as firmly attached to the target as a conventional propulsion system, so attaching it to a debris object by a robot arm will be less challenging. Trade-offs between different propulsion systems are shown in Table 1.

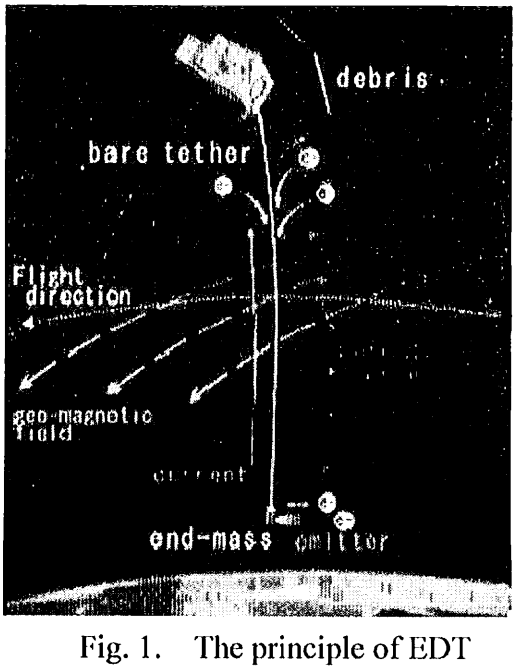

Fig. 1. The principle of EDT

Table 1 Trade-offs between propulsion systems for debris removal.

Method	Merits	Demerits
Chemical thruster	-established technology	- low Isp - difficult to fix to debris object
Ion thruster	- high Isp	- high electrical power requirement
Solid rocket motor	- established technology - compact	- generates numerous slag/dust debris - difficult to fix to debris object
Air bag	- simple - no electrical power	- huge size required for heavy debris - debris impact risk
EDT	- high Isp - easy to attach to debris object	- debris impact risk (mitigate by using a net tether)

3. EDT Demonstration using a small satellite

Up to the present, many on-orbit experiments of tether systems including electrodynamic tethers have been conducted, but thrust generation by an EDT has yet to be demonstrated. A flight experiment using a small satellite is therefore planned[10] as low cost means to prove the technology. We plan to use a small satellite with a total mass of around 50 kg to confirm the generation of EDT thrust and to obtain measurements of EDT characteristics such as electron emission and collection in the space plasma. The orbit for the experiment has not yet been decided; however, it is most likely to be a sun synchronous orbit (SSO) with the altitude of approximately 700 km.

After the small satellite is deployed from its launcher's upper stage, it will release a daughter satellite (an end-mass with an on-board reeled tether) using a spring-loaded mechanism while simultaneously deploying the tether, since the other end of the tether remains connected to the mother satellite. Table 2 shows target specifications of the on-orbit experiment and Fig. 2 shows the arrangement of the principal system elements. Key components are the electron emitter, the electron collector, and the tether itself. Electron collection from the space plasma is performed by a bare tether, a conductive wire without insulation that collects electrons directly from the ambient plasma when the tether has a positive electrical potential by EMF. Electron emission to the space plasma is primarily by a hollow cathode (HC), but a field emission cathode (FEC), which is promising as a future electron source, will also be carried (Fig. 3). The HC has high electron emission, but it is large and heavy, and requires a relatively high electrical power to operate. An FEC, on the other hand, is a more efficient electron source that uses the phenomenon of field emission of electrons from the surface of a solid material. Electric fields concentrate at the sharp tips of emitters on the surface and when a voltage is applied between a gate and the emitters, electrons are emitted by the tunnel effect. An FEC can emit electrons at lower electrical power than an HC, but at the current technological maturity its electron emission is comparatively low. In our study, a carbon nanotube (CNT) FEC is adopted because it is simple and durable in a low vacuum environment[11].

Table 2 Target specifications of EDT demonstration flight.

EDT system mass	20 kg
Required electric power	20 W
Tether length	1 km
Electron collector	Bare tether
Electron emitter (primary)	Hollow cathode
Electron emitter (secondary)	Field emission cathode
Orbit (TBD)	700-km SSO

A single cord tether would be susceptible to being severed by collisions with even very small debris objects and micrometeoroids, and could be severed within a short period of time in crowded orbits[12], so a net tether is proposed[13]. A net tether is expected to have a longer lifetime because the spaces between the cords that result when one cord of the net is longer than the others plays important role in surviving debris impacts (Fig. 3 bottom). It is also expected that a multi-cord bare tether will have a greater electron collection capability[14].

These key components have been fabricated and are now being evaluated[8) 15) 16)]. Their measured characteristics are used in the numerical simulations for mission analysis described in the following section.

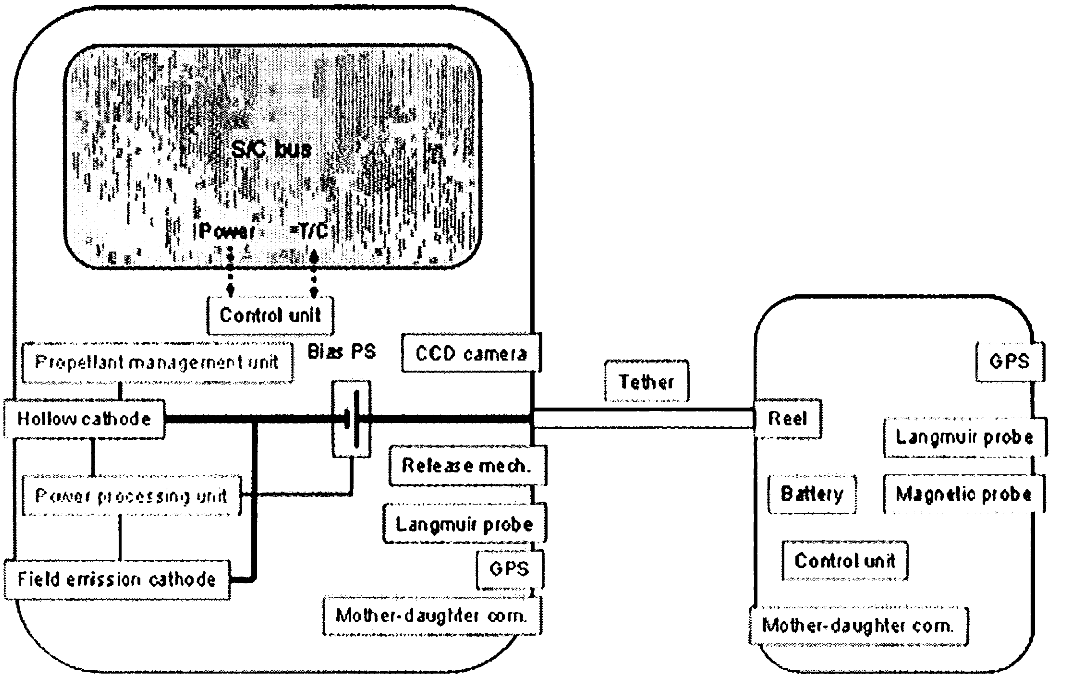

Fig. 2 The arrangement of EDT equipment

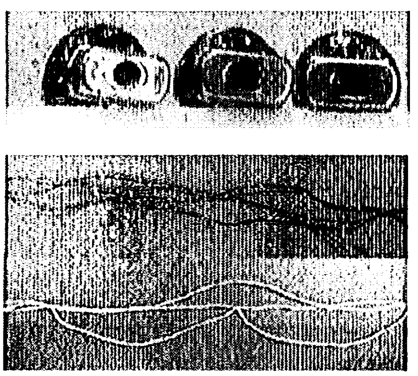

Fig. 3 FEC (top) and net tether (middle and bottom). Net tether with different lengths of cord (bottom) has spaces between the cords.

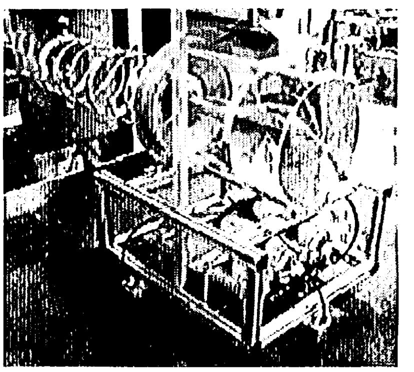

Fig. 4 Reel and deployment mechanism

In order to confirm thrust generation, the orbital motion of the EDT system will be measured using the Global Positioning System (GPS). Langmuir probes and magnetic probes are also equipped for evaluating EDT characteristics such as the influences of the geomagnetic field on EMF and thrust generation, and the relationship between plasma properties and tether current. The total mass required for this EDT demonstration is considered to be about 20 kg, including about 7.5 kg for the end-mass.

4. Numerical simulation of EDT

Precise numerical simulations are conducted for some aspects of flight experiment mission analysis, such as available electric currents and Lorentz force, and tether deployment dynamics[13].

4.1 EDT Model

To take into account its flexibility, the tether is modeled as a lumped mass by dividing it into point masses connected by segments consisting of a spring and viscous damper. The equation of motion of each point mass is formulated in a coordinate system with its origin at the center of mass (C.M.) of the system rotating around the Earth (Fig. 5). The equation of the motion of i-th point mass in this rotating coordinate system is as follows:

$$
\begin{aligned}
\ddot{x}_i &= (r + y_i)\ddot{\theta} + 2(\dot{r} + \dot{y}_i)\dot{\theta} + x_i\dot{\theta}^2 \\
&\quad -\mu_e x_i\left[x_i^2 + (y_i + r)^2 + z_i^2\right]^{-\frac{3}{2}} + \frac{Q_{xi}}{m_i} \\
\ddot{y}_i &= -\ddot{r} - x_i\ddot{\theta} - 2\dot{x}_i\dot{\theta} + (r + y_i)\dot{\theta}^2 \\
&\quad -\mu_e (y_i + r)\left[x_i^2 + (y_i + r)^2 + z_i^2\right]^{\frac{3}{2}} + \frac{Q_{yi}}{m_i} \\
\ddot{z} &= -\mu_e z_i\left[x_i^2 + (y_i + r)^2 + z_i^2\right]^{-\frac{3}{2}} + \frac{Q_{zi}}{m_i}
\end{aligned}
\tag{1}
$$

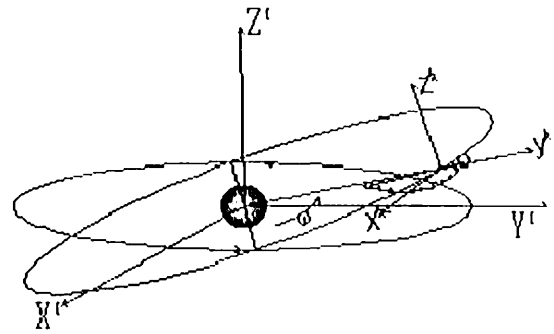

Fig. 5 Rotating coordinate system around the Earth (X^h, Y^h, Z^h)

where θ, r, μ_e are the orbital angular velocity of the C.M. of the tether system, the radius of the orbit, and the gravitational constant of the Earth respectively. m is the mass of the point, and Q is the total force acting on it. Orbital perturbations caused by the Lorentz force, atmospheric drag and geo-potential are taken into account using Gauss's variational equations of motion.

To model electron collection by a bare tether, the two-dimensional Orbital Motion Limit (OML) theory as written below is used:

$$
\frac{dJ}{dx} = en_p d\left(\frac{2e\phi(x)}{m_e}\right)^{\frac{1}{2}} \tag{2}
$$

where $J(x)$, $\Phi(x)$ e, n_p, d, and m_e, are current, potential, the elementary electron charge, the plasma density, tether diameter, and the electron mass, respectively. The following models are used: IGRF 2000 (International Geomagnetic Reference Field) (10 *10) for the geomagnetic field, IRI2001 (International Reference Ionosphere) for the plasma density, NRLMSISE-00 (NRL Mass Spectrometer, Incoherent Scatter Radar Extended Model) for atmospheric density, and EGM96 (Earth Gravitational Model) (10 *10) for the Earth's geo-potential field. Thermal calculations are carried out by considering the direct solar flux, albedo, the Earth's infrared emission, Joule heat, electron collection heat, and aerodynamic heat.

The details of the flight experiment, including orbit and schedule, are not yet fixed, but the simulation results presented here assume the mass of mother satellite as 40.5 kg and that of the end-mass as 7.5 kg. A 700 km circular orbit with an inclination of 98 deg is assumed. The date of the experiment is assumed to be 2011, and the F10.7 flux of the last solar cycle (that of the previous 11 years) is used. A 1 km-long net tether is assumed with the following characteristics: equivalent diameter 2 mm, line density 2 g/m, electrical resistance 0.0485 Ω/m, and tensile modulus 1.4×10^{10} N/mm^2.

4.2 Numerical simulation results

Figures 6–11 show some numerical simulation results. The electron emitter is located on board the mother satellite and so the end-mass needs to be deployed at the nadir end when the orbital inclination is greater than 90 deg, which is the opposite direction to that of an orbital inclination less than 90 deg. The axis of the geomagnetic field is inclined with respect to the Earth's axis of rotation by about 11.5 deg, so the geomagnetic inclination (measured from the geomagnetic equatorial plane) varies periodically within a day (Fig. 7). The available tether current varies depending on factors including plasma density, geomagnetic field, and the libration angle of the tether (Fig. 8). If it is assumed that the maximum possible number of electrons given by Eq. (2) are collected, the average available current is about 0.09 A, and the average Lorentz force is about 0.8 mN, as is shown in Fig. 8 and Fig. 9. This average Lorentz force is more than ten times greater than atmospheric drag, and the semi-major axis of the orbit is expected to be decrease by about 2.4 km/day, compared to a decrease of 0.1 km/day from atmospheric drag alone (Fig. 10).

When the torque generated by EDT thrust is large compared with the gravity gradient torque, the tether may start tumbling and tether current control (by switching it on and off) is needed to prevent this[13)17)]. However, in the present mission simulations, the EDT torque is not so large that the tether starts tumbling, and current control is not required (Fig. 11).

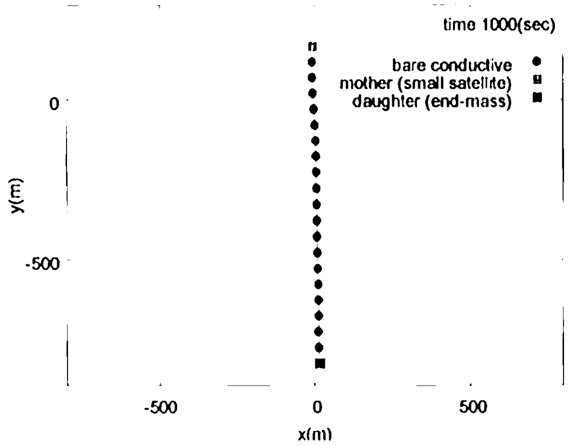

Fig. 6 The tether as a lumped mass model

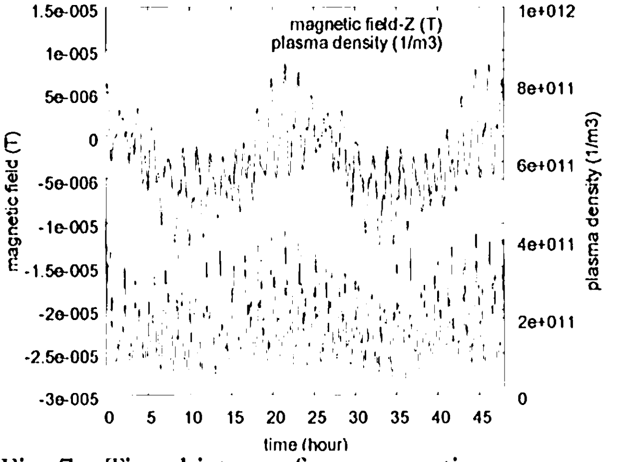

Fig. 7 Time history of geomagnetic field and plasma density

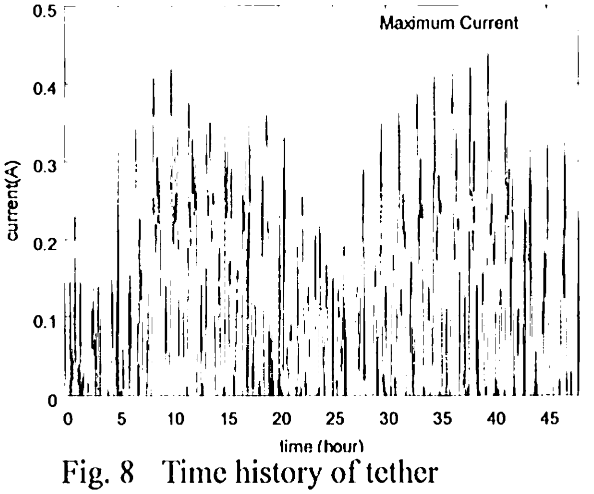

Fig. 8 Time history of tether current

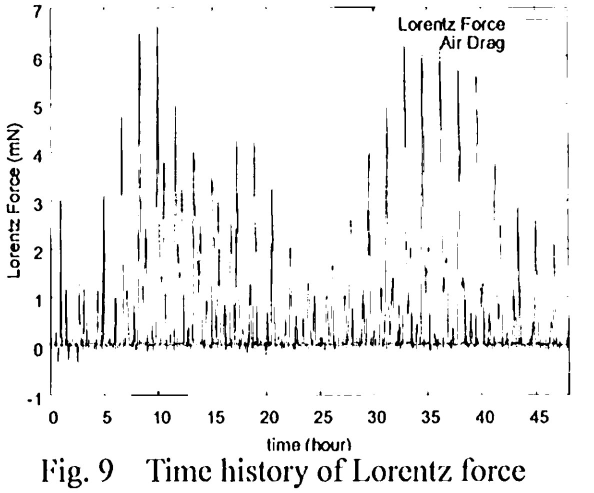

Fig. 9 Time history of Lorentz force and atmospheric drag

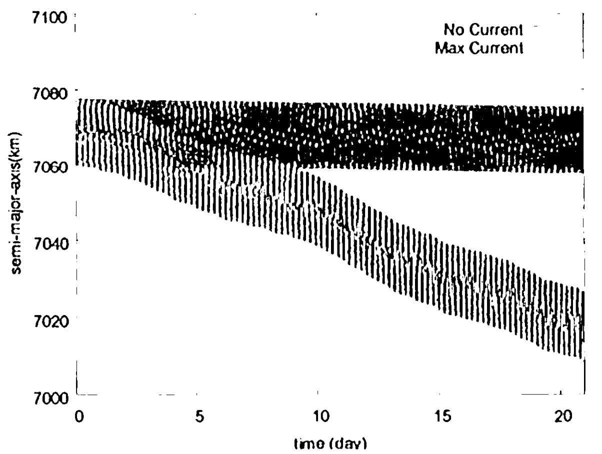

Fig. 10 Time history of semi-major axis with and without EDT thrust

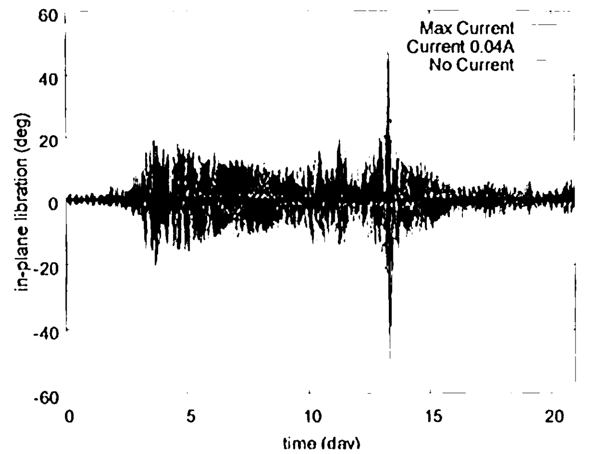

Fig. 11 Time history of in-plane libration without current control

If the electrical power and mass provided by the small satellite are limited, an HC cannot be used for reasons described previously. In such a case, the FEC is the only available emitter and the maximum tether current might be limited. Fig. 12 shows the time history of the Lorentz force when the available tether current is limited to 40 mA. In this case, the average Lorentz force is about two to three times greater than atmospheric drag. With such a small relative magnitude, it is not clear whether the effect of the Lorentz force can be distinguished from that of atmospheric drag; however, the orbital inclination time history can provide evidence of thrust generation. Fig. 13 shows the time history of the orbital inclination angle. It is almost constant without EDT thrust, but changes with EDT thrust.

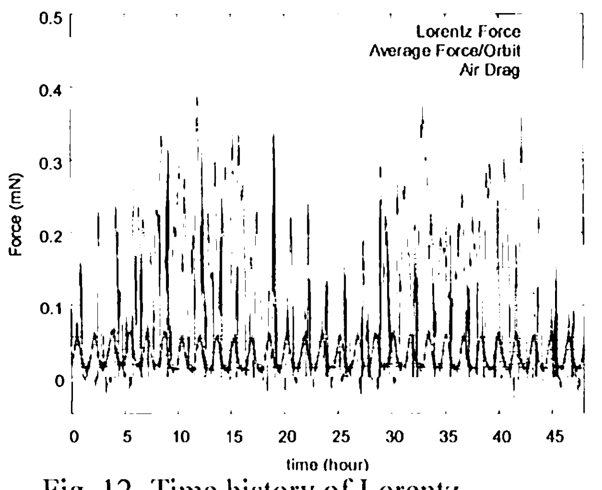

Fig. 12 Time history of Lorentz force and atmospheric drag

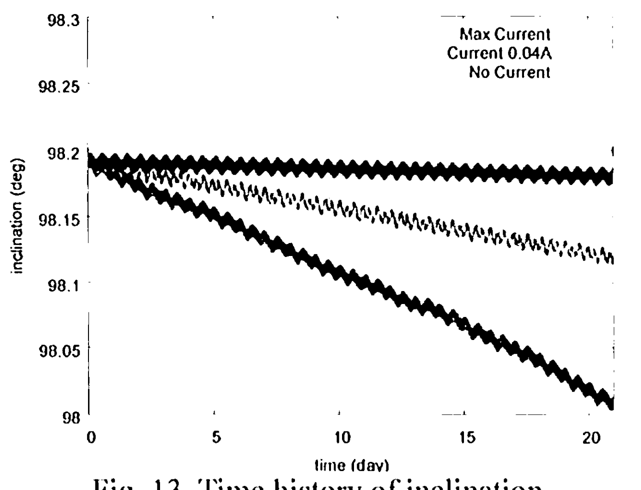

Fig. 13 Time history of inclination with EDT thrust

4.3 Deployment Dynamics

Deployment is critical for the tether system, and the deployment dynamics need to be investigated by numerical simulation since it is difficult to conduct experiments of full deployment on the ground[18]. Stable tether deployment is made difficult not only by the large friction of the conductive tether but also by the small gravity gradient force provided by the small satellite. Moreover, the effect of tether coiling cannot be neglected when the tether tension is small, such as during deployment.

In the numerical simulations, tether deployment is modeled by adding point masses to above-mentioned lumped mass model. The coiling of the tether is modeled by a weak spring other than the spring mentioned in section 4.1 for modeling the tension of tether itself. The attitude motions of the mother satellite and the daughter satellite are also considered. Some examples are shown here for the cases of the EDT flight experiment.

The reel mechanism is composed of a fixed spool-type reel and a braking reel (Fig. 14). A spool-type reel is adopted because it has a simple structure and has yielded good results in the past. The first 900 m of the tether is wound on the spool-type reel and the last 100 m is wound on the braking reel. The tether wound on the spool-type reel will be payed out first. The deployment friction $F_{friction}$ is modeled as:

$$F_{friction} = 0.2 \times V_{deploy}^{\ 2} \ [\mathrm{N}] \qquad (3)$$

where V_{deploy} is the deployment velocity. Braking is applied to the terminal part of the tether by the rotary braking reel using the eddy current braking effect. The braking force can be written as:

$$F_{braking} = 2.0 \times V_{deploy} \ [\mathrm{N}]. \qquad (4)$$

These reels will be released from the mother satellite by a double helical spring at a velocity of about 0.8 m/s.

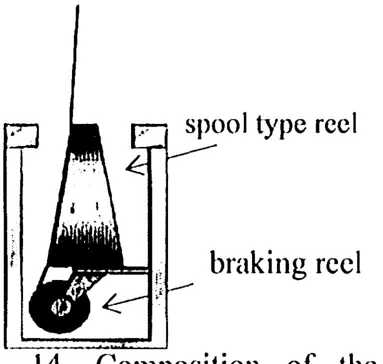

Fig. 14. Composition of the tether reel.

Figure 15 shows the time history of the form of the tether and Fig 16 shows the time history of tension during the deployment phase. Although only a simple braking law is applied, the figures show that the tether is deployed stably.

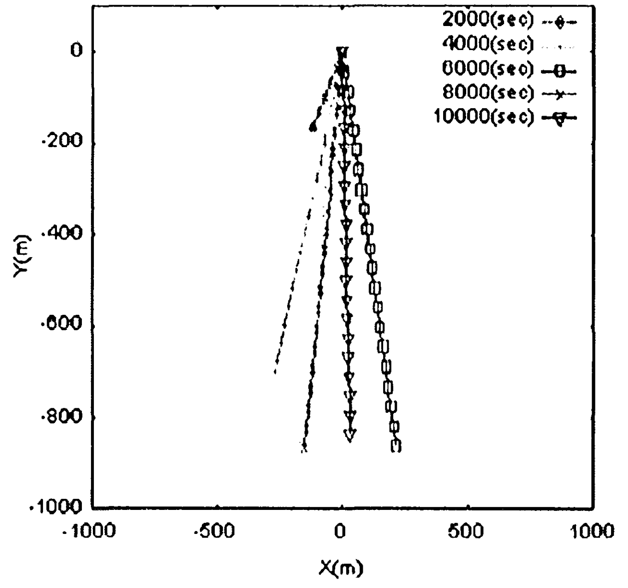

Fig 15 Time history of the form of the tether during deployment when the spring constant of the coiling is 5×10^{-4} N/m

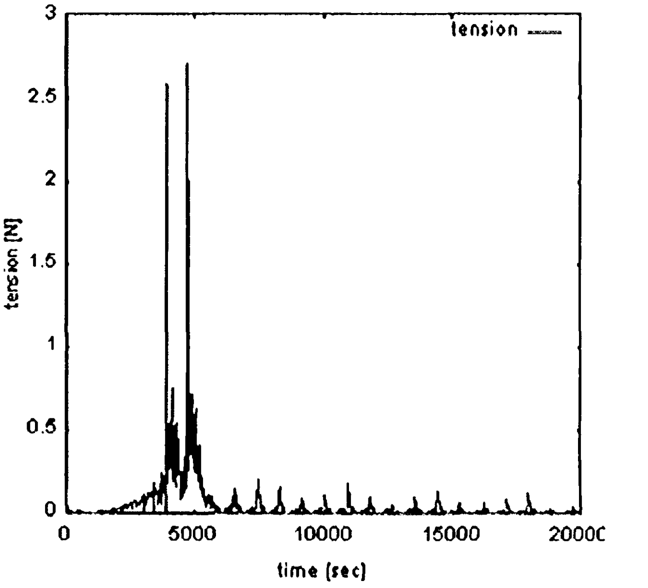

Fig 16 Time history of tether tension during and after deployment.

5. Conclusion

An active space debris removal system that employs highly efficient EDT technology for orbital transfer is being investigated as a means of suppressing space debris growth. As a first step towards realizing the debris removal system, an EDT flight demonstration is planned using a small satellite. Precise numerical simulations were performed for certain aspects of mission analysis, such as available electric currents, orbital changes, tether stability, and deployment dynamics. It was found that a 1 km tether on a 50 kg-sized small satellite in SSO can demonstrate thrust generation. After the small satellite flight experiment, up scaling of the EDT will be studied to develop a larger system for de-orbiting a rocket upper stage or a large satellite. Then, a Micro Remover to demonstrate debris removal will be developed. At the same time, an international framework should be discussed to move toward the final goal of debris removal to maintain the space environment for the next generation.

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