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ELECTRON EMITTER FOR SMALL-SIZE ELECTRODYNAMIC SPACE-TETHER USING MEMS TECHNOLOGY

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ABSTRACT

Adjustment of the orbit of a spacecraft using the forces created by an electro-dynamic space-tether has been shown as a theoretic possibility in recent literature. Practical implementation is being pursued for larger scale missions where a hot filament device controls electron emission and the current flowing in the electrodynamic space tether. Applications to small spacecraft, or space debris in the 1–10 kg range, possess difficulties with electron emission technology, as low power emitting devices are needed. This paper addresses the system concepts of a small spacecraft electrodynamic tether system with focus on electron emitter design and manufacture using micro-electro-mechanical-system (MEMS) technology. The paper addresses the system concepts of a small size electrodynamic tether mission and shows a novel electron emitter for the 1-2 mA range where altitude can be effectively affected and other orbit parameters can be controlled for small sized missions, without on-board propulsion.

1. INTRODUCTION

Space debris is an ever-growing concern for satellite developers, launch companies and other space explores. The possibility of de-orbiting space debris or spacecraft using an electrodynamic tether has been at the drawing board of theorists for decades [1] and a small number of full-scale space experiments were made [2]. With an electrodynamic tether properly applied, energy of the spacecraft would be converted to a drag force from interaction between the Earth B-field and currents flowing in a wire that collects free electrons from the plasma in a low earth orbit. The fact that no additional propulsion need be carried for de-orbiting is a salient feature that makes electro-dynamic space tethers promising in our long-term exploration of Space.

The well-known principle of an electrodynamic tether is to provide forces distributed over its length proportional to the current it carries and the B-field. Current is produced as a result of absorbing free electrons from the plasma. If the tether is a non-isolated conductor, this happens through the Debye shield along parts of the tether. Electrons are emitted from other parts and from electron emitters. As the spacecraft travels through the thin plasma surrounding Earth, electrical charge builds up on the tether and the spacecraft. If left alone, currents running to and from the spacecraft will soon reach equilibrium. The net electrodynamic force imposed on the system in this mode is very small. In order to increase the currents, it is necessary to shift the potential of the spacecraft and tether. Since emitting electrons is a simple approach, an electron emitter is fitted on the spacecraft or at the far end of the tether. When increasing the current to increase forces, control of electron emission becomes essential since the current will also impact the stability of the electro-dynamic tether. Instability would appear as rotation of the tether about the spacecraft or in the tether making spirals along its length. Both phenomena would jeopardize the generation of de-orbiting forces. Adequate control of tether current, by contrast, would avoid instability and offer control of each of the orbit parameters.

The basic physics of an electrodynamic space-tether system was treated in [1,2] and references herein. Recent studies [3] and development efforts [4,9] have brought electro-dynamic tethers much closer to overcome the difficulties encountered in the first full-scale experiments [2]. A recent study, initiated by NASA, aim at development of electro-dynamic space tether technology for larger spacecraft. Applying the technique to space debris and Pico-satellites with mass 10 kg and below is a real challenge to downsizing and perhaps to re-thinking established techniques for electrodynamic tethers and their components. Control of current in an electrodynamic tether was pursued in [4] using a hot filament emitter. Theoretical studies on emitter configuration for electro dynamic tethers design was pursued in [9] and different coating materials were investigated in [8,10,11,12] showing that Mo, MoSi and CoSi will enhance emission properties. The free oxygen contained in the space plasma may prove to be a major concern for field effect emitters. Oxidation of the emitting tips will either significantly reduce the emitted current or completely stop it. Oxygen resistant emitter surfaces are thus desirable.

This paper presents a system designed for demonstration on a Pico-satellite [5,13,14] with limited available power, in the 1-2 W range. Fundamental factors affecting the performance for an electrodynamic space tether are first introduced. Results on stability and orbit control are put into perspective and a novel design of an electron emitter aimed at the 1mA range is presented. The new design was driven by tight budgets to energy and size. Electron emission through hot filament emitters, though technically simple, will drain too much on available power, and alternatives need be developed. We thus report current results on the development of a space compatible field effect emitter design using MEMS technology.

The effect of a conductive tether to lower the orbit of a spacecraft is discussed and it is shown that even tether currents in the mA range can bring a pico-satellite sized spacecraft or space debris down within reasonable time.

2. ELECTRO DYNAMIC TETHER

Conductive tethers in space experience the effects of free electrons and ions in the plasma encountered in low Earth orbits. Currents flowing in the tether will interact with the Earth's B field to give net forces on the satellite. Only currents that do not form a closed circuit within the spacecraft will give rise to net forces, the Ampere forces distributed along the tether. With the tether moving in plasma of charged particles, electrons could be absorbed at part of the tether and emitted elsewhere along the tether. Accurate control of the current flowing in the tether would require controlled emission of electrons.

Moving a thin conductor in the Earth's B field has two basic effects. The electromotive force induced is

$$V_{emf} = \oint (\mathbf{v} \times \mathbf{B}) \cdot d\mathbf{s}. \qquad (1)$$

where V_{emf} is electromotive force, **v** is local velocity, **B** the magnetic field vector and d**s** the integration element along the length of the conductor. Provided electron absorption and emission can take place, the electromotive force will drive a certain current through the wire.
When a current **i** flows in the conductor, each element of the tether will be subjected to the Ampere force

$$\begin{aligned} d\mathbf{F}(s) &= \mathbf{i}(s) \times \mathbf{B} = -i(s)(\mathbf{B} \times d\mathbf{s}) \\ \mathbf{F} &= -\int_0^L i(s)(\mathbf{B} \times d\mathbf{s}) \end{aligned} \qquad (2)$$

In calculating net forces on the tether, gravity gradient (micro gravity) effects need be included. If a satellite with main body mass m_{sat} had a straight-line tether connection to an end mass m_{end}, and distances from the system's centre of gravity (COG) to the COG of the satellite and the end body are l_{sat} and l_{end}, respectively, and the system has an angle θ to vertical, gravity forces acting on the two bodies are approximately

$$\begin{aligned} F_{gg,sat} &= 3\omega_o^2 m_{sat} l_{sat} \cos\theta \\ F_{gg,end} &= 3\omega_o^2 m_{end} l_{end} \cos\theta \end{aligned} \qquad (3)$$

2.1 Stability

When the tether current is sufficiently high, ampere forces might exceed those of the gravity gradient such that a rotating motion could be the result. There is hence an upper value for the allowable current in this semi-static setting. In the case of a tether with mass m_t, the limiting value for the current is [1]

$$I_* = \frac{3\omega_o^2}{B_\perp}\left(m_{end} + \frac{1}{2}m_t\right) \qquad (4)$$

A current above this value will make the tether rotate about the satellite.

Dynamic effects are crucial for accurate analysis of the stability and eigenmodes need be considered for the combined tether-spacecraft system. Using the partial differential equations for tension **T** and position **R**, with tether density ρ and material elasticity and stiffness parameters T and E result in Eq. 5. Stability was investigated in [1] and [14]. The results showed that stability require the tether current to be less than a value given in Eq. 6 where *G* is the gravitational constant and *l* the length of the tether. The critical currents are functions of the tether's mass and length. For a length of 500 m and a total mass of 0.1 kg $I_{**} \simeq 2mA$ [14].

$$\rho(s)\left(\frac{\partial^2}{\partial t^2}\delta\mathbf{R} + 2\boldsymbol{\omega}\times\frac{\partial}{\partial t}\delta\mathbf{R} + \boldsymbol{\omega}\times(\boldsymbol{\omega}\times\delta\mathbf{R})\right)$$

$$+\rho(s)\left(\mu\frac{\delta\mathbf{R}}{R^3} - 3\mu\frac{\mathbf{R}}{R^5}\mathbf{R}\bullet\delta\mathbf{R}\right) = \frac{\partial}{\partial s}\delta\mathbf{T} + \delta\mathbf{F} \tag{5}$$

$$\frac{\partial}{\partial s}\delta\mathbf{R} = \delta\mathbf{T}\left(\frac{1}{T} + \frac{1}{E}\right) - \frac{\mathbf{T}}{T^3}\mathbf{T}\bullet\delta\mathbf{T}$$

$$\delta\mathbf{F} = \delta\left(\mathrm{i(s)}\left(\mathbf{B}\times\delta\mathbf{R}\right)\right)$$

$$I_{**} \simeq 1.65\left(I_*^2 l^2 \frac{B_\perp}{G}\omega_o\right)^{1/3} \tag{6}$$

2.2 Orbit correction

If the task of the conductive tether is end of life de-orbiting, the energy to lower the orbit is a key factor. Since total energy at altitude h, with μ being the gravitational constant, R radius of Earth and m the mass of the spacecraft:

$$\Delta E = \frac{m\mu}{2}\left(\frac{1}{R+h_0-\Delta h} - \frac{1}{R+h_0}\right) \tag{7}$$

In a 600 km altitude orbit, 4 J/kg is required to lower the altitude by 1 m. The order of magnitude of forces available form a 500 m long conductive tether, with 10 mA flowing in the tether, in a B field of 30000 nT, would be in the range

$$\begin{aligned} &|F| < 3*10^{-7}\ [N] \Rightarrow \\ &\left|\frac{dE}{dt}\right| = |\mathbf{F}\bullet v| < 2*10^{-3}\ [J/s] \Rightarrow \\ &m\left|\frac{dh}{dt}\right| = 2|\mathbf{F}\bullet v|\frac{(R+h_0)^2}{\mu} < 5*10^{-4}\ [\mathrm{m/s}] \end{aligned} \tag{8}$$

This result of decreasing the altitude by up to 2 m per hour for a 1 kg mass is indeed a result of idealizations. A correct estimate is obtained integrating the Ampere force using the spatial and temporal variation of **B** seen from the orbiting spacecraft. The orbit's inclination is decisive for the possibilities, as only the B filed component perpendicular to the velocity vector gives a force opposing motion. Nonlinear simulations [14] showed orbit corrections of about 25 % of this figure in a polar orbit with 98 deg inclination.
The above discussion has dealt with the case of a steady current flowing in the tether. It was shown in [3], that in fact each of the Kepler parameters of an orbit could be modified using appropriate time-varying control of the tether current. Control of the tether current could be achieved by active control of electron emission.

2.3 Tether-plasma interaction

When surrounded by the space plasma, the conductive tether will absorb electrons but a Debye shield [13] will be formed that limits electron absorption. The building of an electromotive force will create a potential along the conductive tether that will allow emission of electrons further up (or down) the tether. The sign depends on the sign of $\mathbf{v}\times\mathbf{B}$ in Eq. 1. This means an idle current will run in the tether. This current might have a magnitude that could bring the tether into the unstable region if I_{**} of Eq. 6 is exceeded [4].

If electron emission can be controlled, an operating current, as illustrated in Figure 1b, can superimpose the idle current, making it larger when appropriate and providing annihilation when this is advantageous. As an electron emitter is a diode-like device, tether orientation up or down must be arranged according to orbit inclination and the electron emitter be placed in the satellite or at the end mass according to mission profile. Alternatively, electron emitters placed at both ends of the tether could form a flexible orbit adjustment system.
The possibility to use a conductive tether system to perform orbit drift/parameter corrections was treated in [3] and [5]. Individual Kepler parameter corrections require that the tether current can be controlled according to local B field and the state (eigenmodes amplitudes) of the tether. Dynamic control of the current could assist to obtain optimal interaction with the B field at any time.

The natural conclusion is that an electron emitting device would be highly desirable for a system to de-orbit pico-satellites and space-debris in the 1-10 kg range. Ability to support currents in the range 0-10 mA with good efficiency would be sufficient.

2.3 Ejector

The ejector mechanism developed for the DTUsat tether experiment takes several factors into account. Primary the acceleration of the spool should not stress the tether wire unnecessary. Secondary the forces exerted on the S/C during deployment should go through center of mass. Tertiary the tether spool should remain attached to the end of the tether and thereby the S/C as no splitting of the S/C is allowed within the CubeSat concept. Quaternary the system should be light-weight though still robust since the mass of the tether wire is a significant portion of the total S/C mass.

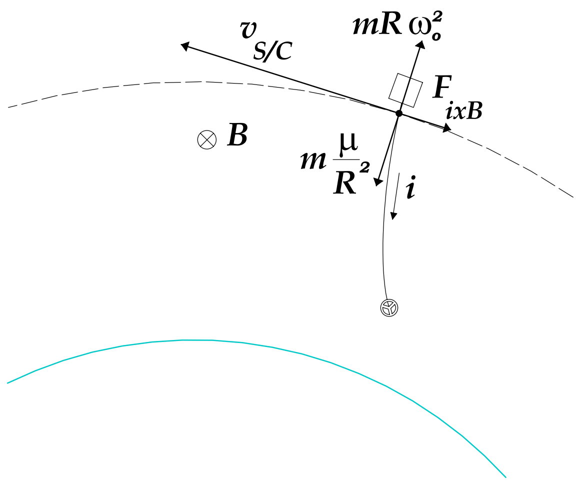

Fig. 1a. The principle of an electro dynamic tether. The figure shows the forces exerted on a S/C in orbit with a deployed tether (not to scale).

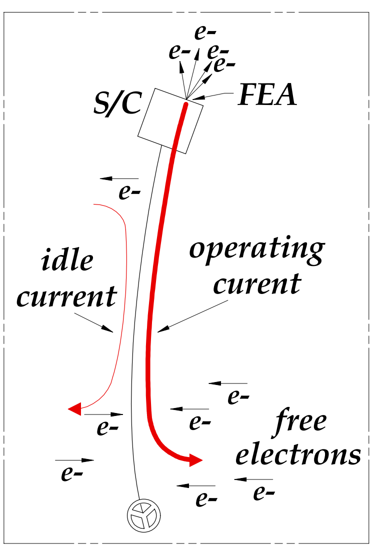

Fig. 1b. The figure shows a schematic representation of the electrons path in the S/C. When the FEA is inoperative only a small idle current runs in the tether.

Fig. 2 shows a CAD rendering of the ejector. When the tether is to be deployed the locking mechanism (not shown) releases the deployer arm (light green). The arm is rotated by the spring (light grey) around the pivot (middle green rod in the spring and rod assembly in the top of the figure). The arm pushes the spool (blue) towards the box opening (right side). As the spool is pushed out of the box it roles over a gear rack (orange in the bottom) which forces the spool to rotate. [13]

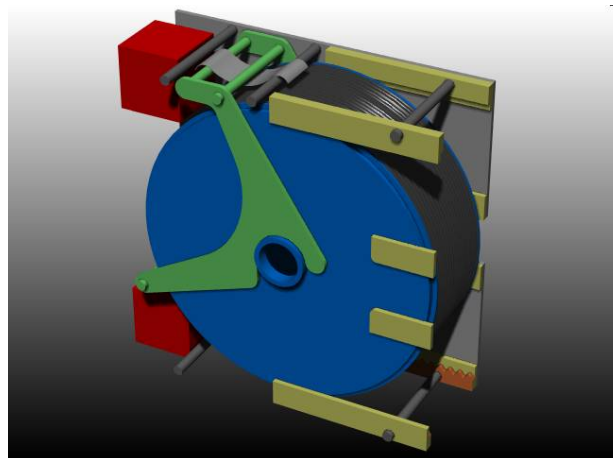

Fig. 2. A CAD rendering of the tether ejector mechanism. A spring loaded mechanism deploys the tether spool. During deployment the spool is both translated and rotated, thereby ensuring that the tether wire is not stressed unnecessary. By courtesy of Tøpholm.[13].

3. ELECTRON EMITTER

Electron emitter devices create free electrons and accelerate these in an electric field. Electrons may be generated thermally (hot cathode), form a gas/plasma (hollow cathode) or be directly extracted from a sharp tip by a strong electric field (cold cathode). Cold cathode emitters may be realized using MEMS technology [7,8,10,11] thereby reducing mass and size. MEMS technology furthermore facilitates structure duplication, which in term enables the arrangement of cathodes in arrays. Such cold cathodes are also referred to as field emitter arrays (FEA). For a small S/C with limited power available the FEA are desirable.

3.1 Design

The design we are implementing is based on the development of sharp silicon tips for AFM measurements [6]. We have added a gate layer for electron extraction and acceleration. See fig. 3. Furthermore we have multiplied the structure by a factor of 10^5 giving a total of 200.000 emitting tips pr. chip. The device is realized using one mask in a standard UV-photolithography line. The guaranteed minimum feature size of this line is 1,5 μm. We chose a minimum feature size of 2 μm in order to increase the yield.

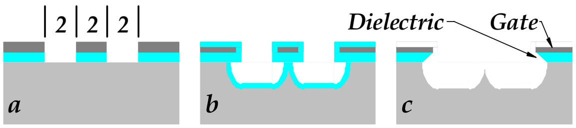

Fig. 3. The tip fabrication is based on the process sequence for AFM tip manufacturing. The minimum feature size in our design is 2 μm.

From literature we have that a field strength of 35 MV/m is necessary for emission turn-on to occur [8] when using bare silicon tips. With a tip to gate distance of 3,6 μm a minimum gate voltage of 126 V is required. We thus design for a maximum operating voltage of 200 V, which should give a reasonable safety margin. Thermally grown silicon dioxide has a breakdown voltage of $1*10^7$ V/cm hence a minimum of 2000 Å is needed. It should be noted that the calculations for turn on voltage does not take the geometry factors into account but are only used for initial assessment of the thickness of the dielectric layer.

Processing requirements and not the resistance requirements primarily dictate the thickness of the gate layer since the currents running in the gate layer are very small. Thermal oxidation and subsequent etching sharpen the tips, see fig. 3. This process will thin down any exposed silicon surface. Hence the gate layer needs to be sufficiently thick to withstand the thinning. The tip-pad is 2 μm wide and will be under-etched 0,9 μm leaving a 0,2 μm pillar to be sharpened by oxidation. Thus 0,1 μm or 1000 Å of gate layer will be consumed.

3.2 Fabrication

A thermal oxide layer of 5000 Å is first grown on the wafer; next a LPCVD poly-silicon layer of 5000 Å is deposited. The mask pattern is transferred to the wafer using negative photolithography. A chromium layer of 250 Å is then evaporated onto the wafer creating a positive etch mask after lift-off. Through a sequence of RIE processing the mask pattern is transferred to the wafer leaving all tip-pads standing on 0,2 μm pillars. After the chromium mask has been etched away and the wafer has been rinsed the pillars are sharpened to tips by thermal oxidation. Finally the oxide layer is removed in BHF leaving sharp emitter tips in each gate hole.
If the bare silicon tips were exposed to the environment in space they would soon deteriorate, mainly due to atomic oxygen. Molybdenum silicide is capable of withstanding oxidation and at the same time lowers the turn-on potential of the emitters. A 200 Å molybdenum film is evaporated onto the wafer immediately after the BHF etch. During Rapid Thermal Annealing (RTA) the molybdenum reacts with the silicon and forms a silicide.

4. CURRENT RESULTS

So far three batches of electron emitters have been made. The first two batches aimed at producing chips for DTUsat. Though the yield was high in the second batch, residues from the tip sharpening etch (BHF), caused the chips to short-circuit during evaporation of the molybdenum metal. See fig. 4.

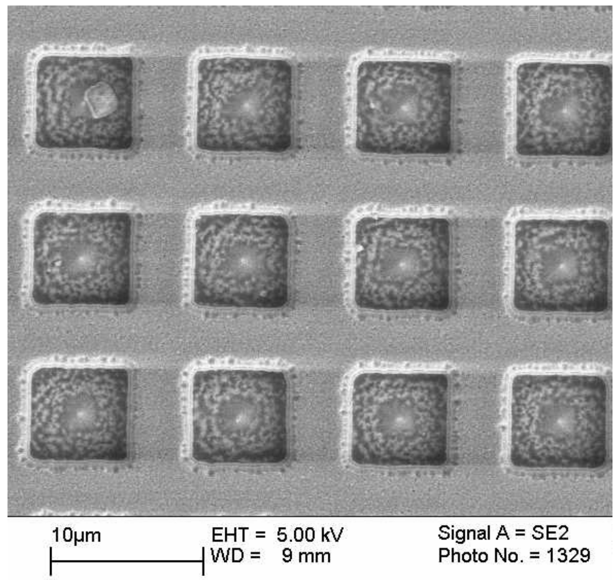

Fig. 4. A SEM micrograph of the first finished batch. Residues from the last etching step (the tip sharpening etch) caused the FEA to short circuit.

In the third batch the residue problem was solved and again the yield was high. Unfortunately the molybdenum and/or the silicide film were melted during RTA due to a defect temperature controller. However, we learned to control the under-etching even better and are able to produce sharper tips. See fig. 5.

4.1 Outlook

A fourth batch is under consideration. In this batch we will improve the mask layout to obtain a more rigid design. At the same time we will implement a short circuit protection. This will further reduce the risk of catastrophic device failure during the final metallization and increase the time to catastrophic failure in space. With the fourth batch based on lessons learned from previous batches and based on a new and improved mask design we anticipate to be able to perform measurements on and characterization of emitter devices.

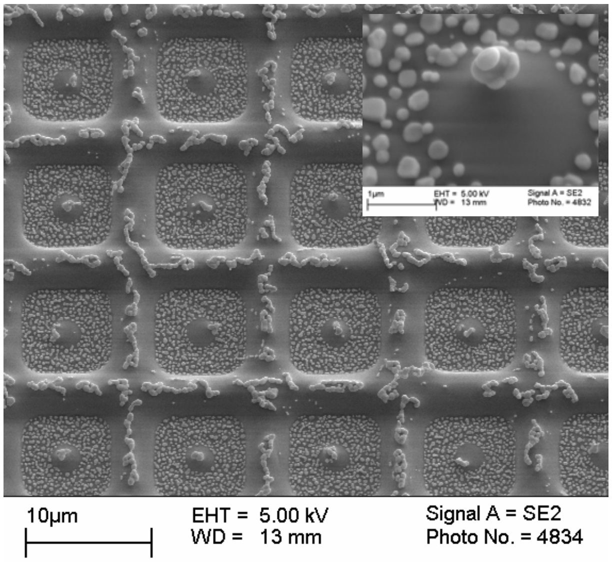

Fig. 5. A SEM micrograph of the latest batch, with a close-up view in the top right corner. Short circuiting etch residues was solved. A defect temperature controller caused the silicide to melt during formation.

5. CONCLUSION

A 500 m long conductive tether and an electron emission system providing a 1mA drain was found to be able to alter the altitude of a 10 kg space item by 1-2 km per month thereby we have devised a system for de-orbiting of small S/C in the 1-10 kg class. The system concept and design is focused on a low weight, low volume and low power solution. The MEMS based electron emitter was developed through three batches, a fourth batch is under consideration. With the fourth batch it is anticipated that emission characteristics can be recorded and endurance tests performed.

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