Investigating the Feasibility and Mission Enabling Potential of Miniaturized Electrodynamic Tethers for Femtosatellites and Other Ultra-small Satellites

Iverson C. Bell, III, Brian E. Gilchrist, David Liaw, Vritika Singh, Kyle A. Hagen, Chen Lu, James W. Cutler
The University of Michigan
Ann Arbor, Michigan 48109, USA; (734) 763-6230
ICBell@umich.edu

Sven G. Bilén and Jesse K. McTernan
The Pennsylvania State University
University Park, Pennsylvania 16802, USA; (814) 863-1526
SBilen@engr.psu.edu

ABSTRACT

The success of nanospacecraft (1–10 kg) and the evolution of the millimeter-scale wireless sensor network concept have cultivated interest in small, sub-kilogram scale, “smartphone”-sized ultra-small satellites, either as stand-alone spacecraft or as elements in a maneuverable fleet. Many of these are envisioned to have a flat geometry and can have a high area-to-mass ratio, which results in a short orbital lifetime in low Earth orbit due to atmospheric drag. Here, we update previous trade studies in which we investigated the use of a very short (few meters), semi-rigid electrodynamic tether for ultra-small satellite propulsion. The results reveal that an insulated tether, only a few meters long and tens of micrometers in diameter, can provide 10-g to 1-kg satellites with complete drag cancellation and the ability to change orbit. Further, a few meter tether could serve as a communications antenna. We also provide a description of the Miniature Tether Electrodynamics Experiment (MiTEE) being planned. The goal of MiTEE will be to demonstrate and study miniature electrodynamic tether capabilities in space.

INTRODUCTION

The promise and success of nanospacecraft (1–10 kg) and the evolution of the millimeter-scale wireless sensor network concept have generated interest in small, sub-kilogram scale, “smartphone”-sized satellites, either as stand-alone spacecraft or as elements in a maneuverable fleet. Miniaturized spacecraft at the levels of fully monolithic semiconductor integrated circuits (10–100 mg) or hybrid integrated circuits (10–300 g) are the next frontier in satellite miniaturization, made possible by advances in integrated circuit and microelectromechanical systems (MEMS) technology. Effectively, this architecture can be thought of as a small “satellite-on-a-chip,” or ChipSat. The name “ChipSat” has been adopted from the University of Surrey system-on-a-chip program focused on miniaturizing the small satellite platform. ChipSats can be in the picosatellite (100 g–1 kg) or femtosatellite (<100 g) mass categories. In this paper, we consider other types of ultra-small satellites as well, using “ultra-small” broadly for any satellite with a mass ≤1 kg.

ChipSats are an attractive platform because they can be less costly to manufacture in bulk and boost into orbit because of their low mass and small size. As a result, it may be possible to launch them in large numbers, possibly enabling unique mission capabilities. A few missions achievable by a coordinated ChipSat fleet are:

- earth observation and global monitoring of forest fires, earthquakes, tsunamis, etc. and mapping other planetary bodies;
- making simultaneous, distributed measurements of basic plasma properties in the ionosphere; and
- high risk missions where some portion of the satellites may be lost.

References 8, 10, and 11 provide other examples of possible ChipSat missions.

The flat ChipSat wafers have an inherently high area-to-mass ratio. Although this feature can be exploited, it can result in a short orbital lifetime in low Earth orbit (LEO) due to atmospheric drag, ranging from a few days to a few hours depending on altitude and solar conditions. Propulsion is also needed to overcome drag. In this paper, we update previous trade studies in which we investigated the use of a very short (few meters), semi-rigid electrodynamic (ED) tether for femtosatellite propulsion.
SATELLITE SIZE CONSIDERATIONS

The first step in the trade study was to establish the size, shape, and mass of adequately representative satellites to be utilized. Satellite size and shape are important for several reasons. First, they help establish the level of atmospheric effects, which in turn establishes the required thrust for drag make-up using the miniature tether (including the tether system’s drag effect). Next, the gravity-gradient force, which causes tension in a tethered system and a restoring torque along the local vertical, is proportional to the mass of the satellites. Finally, size and shape influence the electrical power that can be generated by surface mounted photovoltaic (PV) cells and used for propulsion.

Spacecraft near and below 100 g are a relatively new and evolving architecture, so they may assume a variety of shapes and sizes over time. There are, however, several detailed ChipSat design studies that may serve as guiding examples. The masses, size, and shape of a few pico- and femtosatellites are listed in Table 1.

A few observations from Table 1:

- With an exception of WikiSat V4.1, none of the ChipSat dimensions exceed 10 cm. This feature allows the spacecraft to be stored inside a CubeSat or integrated into its structure.

**Sensor Size Considerations**

Sensors size and use requirements can also influence the satellite bus size. Table 2 shows a brief list of possible sensor technologies that have been identified for ultra-small satellites.

It has been suggested that coordinated, controllable groups of femtosatellites could perform remote sensing missions, so two visible imaging sensors are provided in Table 2. Antennas can also be used to make radio-based remote sensing measurements. The Cyclone Global Navigation Satellite System (CYGNSS) constellation of nanosatellites, for example, will use reflected Global Positioning System (GPS) signals to measure ocean surface wind speed. Thus, a GPS receiver can function as a scientific instrument.

The miniaturized ElectroStatic Analyzer (MESA) is capable of providing in situ electron and ion density and temperature measurements, potentially enabling detection of ionospheric depletions or “plasma bubbles.” Particles in the 0–20 eV range can be detected. The use of MESA would require attitude control because the instrument needs to be oriented in the ram direction to measure ions. MESA has flown on the MISSE-6, MISSE-7, and FalconSat 5 missions. At a smaller scale, the Flat Plasma Spectrometer (FlaPS) is capable of analyzing the energy and angular distributions of ions and electrons in the 10 eV–50 keV range. The instrument is described as a single “pixel” that could be combined with other FlaPS pixels. FlaPS was launched on the FalconSat 3 mission.

Langmuir probes have also been suggested for small satellites. Langmuir probes are common plasma diagnostic tools for measuring plasma potential and electron and ion density and temperature. The probe size and shape are influenced by a variety of factors, including ambient conditions, spacecraft surface area for return current, and the sensitivity of the current measurement equipment on board the spacecraft.

The Gas Chromatography Chip is an example of a MEMS-based “micro gas chromatograph” (μGC) designed for terrestrial applications. However, a similarly sized μGC could be considered for in situ atmospheric studies.
The satellites considered in this trade study range from 10 g to 1 kg. The largest 1-kg satellite has roughly the same dimensions as a 1U CubeSat. At the next mass level below this, we consider two 100-g × satellites: one thin, flat planar satellite and one cubic satellite. The planar satellite is representative of a ChipSat designed on a single PCB. This shape would offer large faces for mounting solar cells and low drag if attitude could be maintained. The small cube satellite has the same dimensions as the “1Q PocketQub.”14 The smallest satellite considered is a 10-g satellite which is inspired by both SpaceChip and Sprite. The dimensions are shown in Table 3 and Fig. 1.

Table 3: Satellite Dimensions

<table>
<thead>
<tr>
<th>Description</th>
<th>Size</th>
<th>Drag Area</th>
</tr>
</thead>
<tbody>
<tr>
<td>1-kg satellite</td>
<td>10 cm×10 cm×10 cm</td>
<td>100 cm²</td>
</tr>
<tr>
<td>100-g cubic satellite</td>
<td>5 cm×5 cm×5 cm</td>
<td>25 cm²</td>
</tr>
<tr>
<td>100-g planar satellite</td>
<td>10 cm×10 cm×1.25 cm</td>
<td>12.5 cm²</td>
</tr>
<tr>
<td>10-g satellite</td>
<td>2 cm×2 cm×0.5 cm</td>
<td>1 cm²</td>
</tr>
</tbody>
</table>

Fig. 1: Relative size of each of the satellite in Table 3

ELECTRODYNAMIC TETHER BACKGROUND

An electrodynamic tether, or ED tether, is a long bare or insulated conductor that connects two satellites. The tether is typically much longer than the satellites. Figure 2 shows an illustration of the basic components in the concept. Current conducted by the tether is collected by the satellite at one end of the tether while the satellite at the opposite end emits electron current. Final circuit closure occurs in the ambient plasma, satisfying Kirchhoff’s Voltage Law.

When an ED tether conducts current, it interacts with the planetary magnetic to produce a thrust force. This force is expressed as

\[ \mathbf{F}_{\text{Lorentz}} = \int_{0}^{L} I_{\text{tether}} d\mathbf{L} \times \mathbf{B}, \]  

where \( I_{\text{tether}} \) is tether current, \( d\mathbf{L} \) is a differential segment of tether with total length \( L \), and \( \mathbf{B} \) is the magnetic field.

In this paper, we assume that the tether conductor is insulated, straight, and perpendicular to the ambient magnetic field so the thrusting force is maximized. The magnitude of the maximum generated force is

\[ \mathbf{F}_{\text{Lorentz}} = I_{\text{tether}} L \mathbf{B}. \]  

The magnitude represented by Eq. 2 can nearly be achieved at low inclination orbits for a tether that is aligned along the local vertical by the gravity-gradient force, which causes tension in the tether and a torque that orients the entire system along the local vertical. The gravity-gradient force can be approximated by27

\[ F_{\text{gravity-gradient}} \approx \frac{3m\mu L}{R_0^3}, \]  

where \( m \) is the total mass, \( R_0 \) is the distance from the spacecraft center of mass to the Earth’s center, and \( \mu \) is the standard gravitational parameter of Earth, \( 3.986 \times 10^{14} \text{ m}^3\text{s}^{-2} \).

Drag is the dominant perturbation for satellites at the ChipSat scale in LEO.28 The magnitude of the drag force is given by

\[ F_{\text{drag}} = \frac{1}{2} \rho C_d A v^2, \]  

where \( \rho \) is the neutral atmosphere density, \( A \) is the cross sectional area of the dual spacecraft and the tether, \( v \) is the velocity of the spacecraft relative to the co-rotating atmosphere, and \( C_d \) is the drag coefficient, often assumed to be 2.2.27 We assume that the tether produces thrust necessary for drag make-up when Eq. 2 is equal to Eq. 4.
SYSTEM CONCEPT DESCRIPTION

The concept uses a very short (several meter), thin semi-rigid ED tether for propulsion, which keeps the overall mass low and provides enough thrust to overcome atmospheric drag in orbit using electrical energy from solar cells. We consider tethering two nearly identical satellites that work together as one element. Each satellite is equipped with a solar panel, power supply, cold cathode electron emitter, and is capable of collecting electrons on the surface. The current direction can be reversed so the ED tether is capable of boosting and deboosting. The identical mass and size of each satellite is also intended to reduce drag torques, which could rotate the system.

![Diagram of ED tether propulsion system](image)

**Fig. 2:** A diagram showing the core components of the tether propulsion system.12

TETHER DESIGN CONSIDERATIONS

The ED tether is intended to be a “semi-rigid” structure. The effective length of the tether will be reduced if the tether retains residual bending from storage. To prevent this, the tether will need to have some degree of flexibility. However, some rigidity is needed after deployment because the gravity gradient may not provide tension as it does with conventional tethers at the 100–1000 m length scale. Thus, the tether will need to be “semi-rigid”.

The ED tether considered here has a Monel™ core to carry current and provide the needed rigidity. A thin layer of Teflon™ provides insulation. The tether’s radius increases with length to provide a higher area moment of inertia and thus rigidity at longer lengths. The tether mass also increases with length, but this is small compared to the satellites’ masses.29

TRADE STUDY ENVIRONMENT ASSUMPTIONS

The altitudes considered are 400 km, 500 km, and 600 km in a circular, equatorial orbit. Following the same assumptions made in Ref. 30, the electron density was determined by averaging electron densities calculated at these altitudes at the equator using the International Reference Ionosphere-2007 (IRI-2007) model. This was done for January 1, 2000, which was a day with high solar activity in solar cycle 23 (F10.7D = 126). The neutral density was similarly taken from the Mass-Spectrometer-Incoherent-Scatter (MSIS-E-90) model. Atmosphere and ionosphere assumptions are summarized in Table 4. The assumed spacecraft velocity relative to the Earth’s co-rotating atmosphere is 7.5 km·s⁻¹.

<table>
<thead>
<tr>
<th>Parameter</th>
<th>400 km Altitude</th>
<th>500 km Altitude</th>
<th>600 km Altitude</th>
</tr>
</thead>
<tbody>
<tr>
<td>Electron Temp.</td>
<td>0.11 eV</td>
<td>0.14 eV</td>
<td>0.15 eV</td>
</tr>
<tr>
<td>Magnetic Field</td>
<td>0.36 gauss</td>
<td>0.36 gauss</td>
<td>0.36 gauss</td>
</tr>
<tr>
<td>Gyroradius</td>
<td>2.2 cm</td>
<td>2.5 cm</td>
<td>2.6 cm</td>
</tr>
<tr>
<td>Neutral Density</td>
<td>5×10⁻¹⁵ g·cm⁻³</td>
<td>9×10⁻¹⁶ g·cm⁻³</td>
<td>2×10⁻¹⁶ g·cm⁻³</td>
</tr>
<tr>
<td>Electron Density</td>
<td>1×10⁶ cm⁻³</td>
<td>7×10⁵ cm⁻³</td>
<td>3×10⁵ cm⁻³</td>
</tr>
<tr>
<td>Debye Length</td>
<td>2 mm</td>
<td>3 mm</td>
<td>5 mm</td>
</tr>
</tbody>
</table>

ELECTRON COLLECTION

The electron collecting anode is a critical component of an ED tether circuit. To facilitate electron collection in our system concept, areas of the satellite can be coated with a transparent conductor, e.g., Indium Tin Oxide (ITO). However, estimating the electron collection current to the surfaces of a positively biased ChipSat is complex. The possible shape of the pico- and femtosatellite, the relative motion of the plasma, and the presence of an ambient magnetic field contribute to the complexity of predicting this current. Simplifying assumptions were made to estimate current in Ref. 30 and are summarized here.

**The Spherical Sheath Model**

Current collection models provide a relationship between the anode bias and the collected current for less complex geometries like spheres, infinite cylinders, and infinite plates.31 These satellites, however, are cuboids and this complicates estimating the collection current. We assume that the non-neutral sheath region between the immersed object’s surface and the ambient plasma will expand outward from the biased spacecraft surfaces and conceal the fine details of the electron
the collector’s geometry, allowing us to approximate it as a sphere. We estimate current collection by assuming that the anode collects current like a sphere with an equivalent diameter equal to each satellite’s longest edge. Reference 32 shows the expansion of an ion collecting sheath in a stationary plasma with sheath radius increasing with potential, where at high potentials the sheath surrounding a thin, circular plate resembles an oblate spheroid.

The Current–Voltage Characteristic

Reference 33 provides a strategy for extracting plasma parameters from empirical current collection measurements in LEO. The expression

$$I_{\text{WLP}} = \frac{I_{\text{nomal}}}{2} \left(1 + \frac{q(V_{\text{anode}} - V_{\text{gate}})}{kT_e} \right)$$

was fit to the Langmuir probe current–voltage \((I-V)\) sweeps of a 5-cm-radius sphere with varying values of the dimensionless parameter \(\beta\). The thermal current \(I_{\text{thermal}}\) is

$$I_{\text{thermal}} = A_{\text{prob}} n_e q \sqrt{\frac{kT_e}{2\pi m_e}} ,$$

and \(kT_e/q\) is the electron temperature in eV. We conservatively choose \(\beta = 0.75\) for our model, which is close to the apparent average \(\beta\) value observed in the time window in Ref. 33. This is more conservative than previous studies, in which \(\beta = 0.85\).\(^{30}\)

**ELECTRON EMISSION**

As a complement to the electron collection required for current flow through the tether, electron emission needs to be well-defined in order to effectively neutralize the entire system. There are well-tested technologies for electron emission, such as a thermionic cathode or hot filament, and then there are new and innovative technologies for electron emission such as field emitter array cathodes. Field emitter arrays (FEAs), as the name suggests, use field electron emission to pull electrons out of the surface through quantum tunneling. This is accomplished by generating very large electric fields (on the order of single V·nm\(^{-1}\)) with very sharp tips in order to enhance the electric field around these tips. By arranging large arrays of such tips, the array is able to generate the electron emission current necessary to neutralize an electrodynamic tether system.

For the most part, these field emitter arrays are currently used for flat panel displays, and have not yet been space-tested with concerns that these very sharp tips necessary for electron emission will be dulled by the bombardment of plasma in the harsh space environment, reducing their emission capabilities. However, there are electron emission technologies that have been used in space such as the thermionic cathode, which emits electrons by thermionically heating up the filament, allowing the electrons to gain enough energy to overcome the work function to be emitted. The benefits of the field emitter array are that it has flat-panel scalability, meaning that it has a low profile and can fit very well into different faces of the small satellite, and it can take up as much area as is needed to achieve the current levels that are desired to be emitted. The field emitter array also has the capability to achieve a better current density or efficiency and also utilize the available power more effectively, both important considerations in the design of a small satellite.

FEA technology can be used to emit electrons at one end of the tether. The Fowler–Nordheim emission law for field emission is

$$I_{\text{cathode}} = \alpha_{\text{FN}} V_{\text{gate}}^2 \exp(-b_{\text{FN}}/V_{\text{gate}}) .$$

A Spindt cathode consists of an array of sharp-tipped, sub-µm-radius cones that emit electrons when the nearby gate is biased to \(V_{\text{gate}}\). The values used for Fowler–Nordheim coefficients in Eq. 7 are \(\alpha_{\text{FN}} = 0.03 \text{ A·V}^{-2}\) and \(b_{\text{FN}} = 487 \text{ V}^{34}\) The coefficients \(\alpha_{\text{FN}}\) and \(b_{\text{FN}}\) may change as the FEAC technology advances.

**POWER GENERATION AND POWER REQUIRED FOR THRUST**

The power required by the anode and cathode make up a majority of the electrical demand for the miniature tether application. The power dissipated in the tether is not dominant because it scales with resistance and the square of current, both of which are small. To estimate the current needed for drag make-up, the assumed ED thrust is set equal to the drag force.
It was previously estimated that there would be 4.4 mW available for propulsion per square centimeter of PV area, but this assumed that 90\% of the satellite power could be used for propulsion.\textsuperscript{30} Making a more conservative estimate that only 70\% of the generated solar power can be used for propulsion, we now estimate that 3.4 mW·cm\textsuperscript{2} are available for thrusting. We assume that the 100-g and 1-kg cubic satellites have solar panels on all six sides, three of which are exposed to the sun at any given time. We assume that the 10-g and 100-g thin, planar satellites have solar panels on the two largest sides, one of which has a view of the sun at a time. As a result, the 10-g satellite, the 100-g planar satellite, the 100-g cubic satellite, and the 1-kg satellite can generate 13.8 mW, 345 mW, 259 mW, and 1 W for propulsion, respectively. Our power generation estimates are consistent (on an order-of-magnitude basis) with the power generation estimates for the ChipSats in Table 1.

Figures 3a–d compare the estimated power demand to the estimated power generated for propulsion. Although very short ED tether lengths may be easier to store, they require a large current to overcome the atmospheric drag force on the ChipSat. Rigidity decreases with length, so a very long tether must have a relatively large radius to prevent bending or bowing. As a result, the drag due to the tether dominates over the drag due to the satellite, driving up the required current. The current is minimized when these two effects are balanced. This motivates us to choose a 4-m tether for the 10-g satellite, a 10-m tether for the 100-g planar satellite, a 15-m tether for the 100-g cubic satellite, and a 30-m tether for the 1-kg satellite. Tether lengths and radii are listed in Table 5. The 100-g planar satellite has a wider tether than the 100-g cubic satellite. This is because the planar satellite shows potential to overcome drag at 400 km, so it needs a thicker, and thus more rigid, tether to prevent bowing in the higher drag environment.
FORCE ESTIMATE

The drag force and the gravity-gradient force are the dominant forces that impact the dynamics of tethered ultra-small satellites. Figures 4a–d show the thrust, drag, and gravity-gradient force estimates for each satellite. All four satellites show potential to generate a drag make-up force at 500 km and 600 km altitude. Only the 100-g planar satellite is able to produce thrust forces on par with drag at 400 km using a short ED tether. The gravity-gradient force exceeds other forces at 500 km and 600 km. This suggests that the gravity gradient force will ensure a degree of stability.

It may also be possible for the 1-kg satellite to overcome drag at 400 km with an ED tether exceeding 30 m in length. There are examples of missions in which picosatellites used tethers exceeding 30 m. The gravity-gradient force for longer tethered systems can dominate over other perturbation forces and provide the tension force that reduces bowing and makes a thicker, semi-rigid tether unnecessary. Although this would be a departure from the short, semi-rigid tether system described in this study, it may be appropriate for the 1 kg satellite. For example, if the tether were 60 m long and 210 µm in diameter instead of 30 m long and 640 µm in diameter, thrust and gravity-gradient forces would double and the tether drag area would decrease by about one-third. Although storage of long tethers is a concern for the smaller ChipSats, the 60 m tether from our example would only occupy ~2 cm³ (without the spooling and deployment mechanism), which could be accommodated on board the 1 kg satellite.

It will be important to study the relative strength of the drag and gravity gradient torques in order to understand the resulting tether attitude. If the center of mass and the center of pressure are vertically displaced, the aerodynamic drag torque will rotate the system until the vector connecting the two points is parallel (or antiparallel) to the velocity vector. If the gravity-
gradient torque is strong enough, however, it will counteract this rotation and restore the tether to the local vertical.

**PERFORMANCE SIMULATION**

The software tool TeMPEST allows us to simulate an ED tether system in orbit. TeMPEST incorporates current geomagnetic field models, ionospheric and atmospheric conditions, plasma contactor modeling, and precise orbital calculations to predict propulsion performance. TeMPEST was used to generate Figs. 5a–d, which shows the altitude change for satellites with and without ED tether propulsion at 400-km, 500-km, and 600-km starting altitudes.

Figures 5a–d show early predictions for each individual satellite, showing rapid drag deboost without an ED tether (orange) and the actual boost capability at 500 km and 600 km (blue) for a pair of satellites connected by an ED tether. The simulated 100-g planar ChipSat is able to increase altitude as low as 400 km, which is consistent with results from previous sections. It should also be noted that the tether extends mission life even in the cases where the tethered satellite system is unable to overcome drag.

Although the altitude curves in Figs. 5a–d appear to widen, this simply represents increasing eccentricity of the satellite over time. This effect is particularly pronounced for ED tethers in the simulation that are continuously boosting. The thrust force increases in regions of the ionosphere where the electron density is higher, like the dayside during an orbit, and the uneven thrust in each orbit results in an increasing orbital eccentricity. However, boosting can be planned so the satellite orbit remains circular.

**OTHER BENEFITS: USING THE CONDUCTING TETHER AS AN ANTENNA**

An additional goal of this project was to investigate the feasibility of using an ED tether to enhance the antenna aperture. ChipSats have inherently small antenna apertures and low transmission power, but an insulated, gold-coated nickel alloy tether provides the potential for a long, directional wire antenna. The antenna radiation pattern was simulated using ANSYS® HFSS™ simulation software. The antenna can be modeled as an off-centered dipole if a short wire,
10’s of centimeters long, is attached to one of the tethered ChipSats. It was also found that at the CubeSat scale, the conducting satellite frame can be used in lieu of the short wire without degrading antenna performance. The radiation pattern is shown in Fig. 6. The z axis in Fig. 6 points in the nadir direction.

Figure 6: The 3D radiation pattern for a 10-meter-long tether radiating at 430 MHz

Simulations reveal that as the source gets closer to the end, the antenna is more directed and the pointing direction moves towards the direction of the tether. With a small resonator in the tether line at the proper location, the antenna can also be adjusted for frequency and gain independent of its overall length.

An improved communication link can be the basis for a sound command and control plan. Thus, in practice, we can consider a ChipSat constellation with these capabilities to be something more: a carefully managed “fleet” of organized, spatially reconfigurable, capable sensor platforms.

PLANS TO REFINE THE TRADE STUDY

Our next step is to revisit several of the study assumptions to enhance our understanding of the concept feasibility. We have made conservative electron collection estimates, for example, and we would like to improve our estimates by conducting experiments that capture characteristics of the LEO environment. We have also made simplifying assumptions for tether dynamics.

The complex LEO environment creates a current collection scenario that no analytical model the authors are aware of completely captures in our size scale and for our likely box-like geometries. However, we can refine our electron collection estimates by conducting experiments in which we capture the I–V characteristics of pico- and femtosat shaped probes in a flowing plasma. Plasma flow is important because satellites in LEO travel faster than the ion thermal speed and slower than the electron thermal speed, or at mesothermal speed. The high speed flowing plasma has a beam-like effect and creates a rarefied wake region behind the probe that impacts collection. The probes dimensions should be scaled so the probe dimension-to-Debye length ratio is near what is expected in LEO. Also, the voltage should be scaled so that the voltage-to-electron temperature ratio is near what is expected in LEO. Similar approaches have been taken in Ref. 36–40 to improve current collection estimates for anodes in LEO. An experimental test bed is currently being developed.

We have also assumed that the tethered system will be oriented along the local vertical, but it will be important to investigate the attitude dynamics of the system. The interactions of drag, gravity-gradient, and thrust forces and torques can cause complex behavior. We are planning on conducting more sophisticated studies and a microgravity flight to shed light on deployment dynamics. A technology demonstration mission in space could also capture the effects of different disturbance forces and torques on the tethered system.

MINIATURE TETHER ELECTRODYNAMICS EXPERIMENT (MITEE) SPACE MISSION DESCRIPTION

The Miniature Tether Electrodynamics Experiment (MiTEE) is a technology-demonstration mission concept that will utilize CubeSat capabilities to deploy a PicoSat/FemtoSat–tether system (50 g to 250 g deployed system) to assess the key dynamics and electrodynamics fundamental to the operation of a miniature electrodynamic tether system. Starting as a 1U CubeSat, a small body, equivalent to one-fourth of the 1U CubeSat or less, will be deployed from the main CubeSat body. The main body and the deployed body will be connected by an insulated, conducting tether. As the deployed tether system operates in orbit, tether current and thrust will be generated and the dynamics and electrodynamics will be studied.
MiTEE Spacecraft and Science

The MiTEE spacecraft will utilize a cathode and anode on the separate end bodies of the tethered system. The anode body will receive a positive bias, collecting electrons from the ionosphere, whereas the cathode will act as the electron emitter. A Langmuir probe will be used for characterizing ionospheric conditions and relating these conditions to the behavior of the tether system. Supplying the high voltages needed for the cathode and anode on the tethered end bodies is a major challenge for the spacecraft and has not yet been demonstrated on CubeSats to the authors’ knowledge. Thus, the high voltage power supply is a critical component, as it is needed to drive current through the tether.

Additionally, due to the CubeSat form factor and the resulting power limitations, power consumption for the spacecraft will need to be significantly reduced. The spacecraft will have solar panels for collection of power, and subsystems will be duty-cycled to optimize energy usage while still allowing the science goals to be accomplished. An Attitude Determination and Control System (ADCS) will also be required to reduce the detumbling time and reorient the spacecraft upon orbital insertion. The MiTEE mission is being designed to operate in a large range of orbits to satisfy CubeSat and mission objectives.

Tether deployment and storage mechanisms are crucial for mission success and have the largest area of active research and testing. Standard CubeSat structures and components will be used in construction along with the use of a primary antenna and a radio on the deployer body. A secondary objective of the mission is to investigate the use of the tether as an antenna for satellite to ground communication.

The Next Phase of Mission Development

In the next phase of the design, high risk and critical components associated with the mission will be further investigated and risks will be reduced. Successful tether deployment is critical, so we will be investigating deployment mechanisms and dynamics while developing strategies to mitigate any risks we identify. A microgravity flight presents the opportunity to test and study deployment in reduced gravity, so we are planning to complete a microgravity flight proposal in 2013. Additional risks associated with tether dynamics, satellite orbit, and power generation will be analyzed with respective modeling. This phase will continue into 2014.

Mission development student team

The MiTEE mission is being developed by a team of approximately 20 students at the University of Michigan. The team was first formed in September 2012, drawing students from a variety of disciplines, including aerospace, electrical, mechanical, and systems engineering, and a range of academic years, from first year undergraduate students to doctoral students. Although the MiTEE team is student lead, it receives faculty guidance. The team will continue to form and refine the mission concept and design.

CONCLUSIONS

The miniature tether concept shows potential to enhance capabilities for a range of pico- and femtosatellite sizes. The trade study is summarized in Table 5. The results of the trade study and MiTEE can guide optimal system configurations and reveal new capabilities for utilizing ED tethers with ultra-small satellite technology. The capability of maneuvering in a controlled manner represents an opportunity for any constellation of pico- or femtosatellites to be more of a coordinated fleet rather than a swarm.
<table>
<thead>
<tr>
<th>Parameter</th>
<th>Trade Study Satellites</th>
<th></th>
<th></th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>10 g</td>
<td>100 g planar</td>
<td>100 g cubic</td>
<td>1 kg</td>
</tr>
<tr>
<td>Tether Length</td>
<td>4 m</td>
<td>10 m</td>
<td>15 m</td>
<td>30 m</td>
</tr>
<tr>
<td>Tether Radius</td>
<td>36 µm</td>
<td>116 µm</td>
<td>100 µm</td>
<td>320 µm</td>
</tr>
<tr>
<td>Available Power</td>
<td>14 mW</td>
<td>345 mW</td>
<td>259 mW</td>
<td>1 W</td>
</tr>
<tr>
<td>Tether Current</td>
<td>400 km</td>
<td>254 µA</td>
<td>5.5 mA</td>
<td>3 mA</td>
</tr>
<tr>
<td></td>
<td>500 km</td>
<td>237 µA</td>
<td>5.1 mA</td>
<td>2.8 mA</td>
</tr>
<tr>
<td></td>
<td>600 km</td>
<td>177 µA</td>
<td>4 mA</td>
<td>1.9 mA</td>
</tr>
<tr>
<td>Thrust</td>
<td>400 km</td>
<td>30 nN</td>
<td>1.7 µN</td>
<td>1.4 µN</td>
</tr>
<tr>
<td></td>
<td>500 km</td>
<td>28 nN</td>
<td>1.6 µN</td>
<td>1.2 µN</td>
</tr>
<tr>
<td></td>
<td>600 km</td>
<td>21 nN</td>
<td>1.2 µN</td>
<td>0.9 µN</td>
</tr>
<tr>
<td>Gravity-Gradient Force</td>
<td>74 nN</td>
<td>1.9 µN</td>
<td>2.9 µN</td>
<td>58 µN</td>
</tr>
</tbody>
</table>

**AKNOWLEDGEMENTS**

The authors gratefully acknowledge support from AFSOR grant FA9550-09-1-0646, the National Science Foundation Graduate Student Research Fellowship under Grant No. DGE 1256260, and the Michigan Space Grant Consortium.

**References**


