Attitude Control for Chip Satellites using Multiple Electrodynamic Tethers

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This paper explores controlling the attitude of chip satellites using multiple short, semi-rigid electrodynamic tethers. The motivation for this work is the recent emergence of microscale spacecraft as a next-generation paradigm for science, exploration, and commercial space activity. Preliminary results show this propellantless approach to be feasible in terms of expected current, voltage, power, mass, and tether rigidity. The proposed attitude-control architecture simultaneously accommodates forces, which may be used for orbit control. Further work is needed to examine the robustness of similar multi-tether systems with regards to perturbations such as those caused by atmospheric drag, solar pressure, and charging effects.

Nomenclature

- $\beta$: Ratio of effective sheath to gyroradius
- $\eta_{\text{ram}}$: Ram collection fraction
- $\lambda_D$: Debye Length [m]
- $\phi_s$: Potential Bias [V]
- $\psi$: Dimensionless potential bias
- $\varepsilon_0$: Permittivity of Free Space [F/m]
- $B$: Local Magnetic Field Vector [T]
- $l_t$: Tether Vector Length [m]
- $A$: Jacobian Matrix
- $A^+$: Moore-Penrose Pseudoinverse of Jacobian Matrix
- $A_{\text{ram}}$: Ram collection area [$m^2$]
- $I_e$: Electron current [A]
- $I_{\text{ram}}$: Ram current [A]
- $I_{th}$: Thermal current [A]
- $J$: Current Magnitude [A]
- $k$: Boltzmann constant [J/K]
- $m_e$: Mass of electron [kg]
- $n_0$: Density of electrons [$m^{-3}$]
- $n_i$: Ion density [$m^{-3}$]
- $q_e$: Charge of electron [C]
- $q_i$: Ion charge [C]
- $r_{pr}$: Probe radius [m]
- $S_o$: Outer Sheath Radius [m]
- $T_e$: Electron Temperature [eV]
- $V_p$: Plasma potential [V]
- $v_r$: Ram velocity [m/s]

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I. Introduction

The high cost of access to space drives designs towards highly reliable, low quantity architectures. By making use of contemporary trends in commercial approaches to electronics, Microelectromechanical Systems (MEMS), and other microscale integrated technologies, spacecraft can be made much smaller, cheaper, and mass-producible to make space much more accessible for a range of applications. Spacecraft-on-a-chip technology stands not only to provide a very low unit-cost making it feasible for swarm applications, but also to open new avenues for space science.

This femto-satellite regime has small enough size and mass to benefit from novel applications of physics. This technology looks particularly promising for missions such as interplanetary exploration, magnetic field and atmospheric measurements, and various remote sensing applications, as well as possible commercial applications. For chip satellites to be truly viable as a scientific technology platform, they need basic capabilities such as communications, computation, and actuation, all with very low power, mass, and size requirements.

Fortunately, propellantless actuation becomes much more feasible at very low mass scale. Effects such as atmospheric drag, solar pressure, and Lorentz forces are significant in this regime, opening new possibilities for positioning and attitude control as well as energy harvesting. The length-scale regime of chip satellites also allows electrodynamic tethers (EDTs) to be particularly effective as lightweight, simple actuation. This work investigates the feasibility of using multiple electrodynamic tethers on a spacecraft-on-a-chip satellite to provide attitude control, without the traditional reliance on gravity gradient effects to stably orient the tether.

Using multiple electrodynamic tethers, allows much flexibility in control schemes. While electrodynamic tethers can experience forces only perpendicular to the magnetic field, the field direction can vary substantially throughout any orbit, especially in high inclination orbits. Magnetorquers have been shown to effectively stabilize attitude in three axes using time-varying control schemes. A single electrodynamic tether system was explored in Bell, et al. to provide longer lifetimes for small satellites in Low Earth Orbit, assuming the tether is stably aligned with the local vertical. A multiple electrodynamic tether design allows actively-controlled, stable attitude, in addition to the ability to counteract aerodynamic drag. The ability to stabilize in many orientations will make chip satellites more useful as an experimental platform.

Multiple electrodynamic tethers can successfully stabilize chip-satellite attitude for a variety of potential orbits, in many orientations. Gravity gradient and tether dynamics are expected to be minimal compared to the other forces for this length scale. By simulating the dynamics of a chip satellite with multiple electrodynamic tethers and implementing an attitude controller, this paper demonstrates that such a design can be a useful solution in actuating chip satellites.

II. Chip Satellite Propulsion

One of the major challenges of very small spacecraft is closed-loop attitude and orbit control. Propellant-based solutions have limited usefulness for chip-sized spacecraft; current MEMS-based microthruster technologies are intended for spacecraft of several kilograms. Solutions such as using solar pressure and magnetic fields show promise because they do not require consuming mass, out of an already tiny mass budget. Propellantless propulsion allows much longer useful lifetimes and net delta-V for very lightweight spacecraft.

There are a variety of propellantless technologies for very small scale spacecraft. Magnetorquers can provide propellantless attitude control but cannot supply a torque about the magnetic field, and thus offer instantaneous attitude control only about two axes at each point in the spacecraft’s orbit. At this size scale, Lorentz forces from static charge can provide substantial thrust, and can be controlled through maintaining voltages on conductive structures. An example is discussed in detail in Atchison and Peck (2007).

A single tether can provide some passive attitude stabilization using the gravity gradient to align to the local vertical, in addition to using the electrical properties for orbital maneuvers. However, spacecraft with a single tether can achieve only one of two stable attitudes and requires thoughtful design and damping to limit librations, especially if the tether thrust is used to maintain or deorbit the spacecraft. There are also a number of interesting applications for bare electrodynamic tether spacecraft, including Jupiter missions and depleting the Earth radiation belts. For chip-satellites, aerodynamic drag is a significant limitation due to the high surface area to mass ratio, so an EDT may be a highly successful solution for orbit maintenance.

A multi-EDT system can provide both net forces in the plane perpendicular to the magnetic field and
torques in three dimensions. Voronka, et al. (2006) suggest a similar propulsion system for larger-scale systems. Their solution requires much longer EDTs and support structures for each tether, but allows the spacecraft to maintain orientations independent of the local vertical.¹⁰ The chip satellite size scale allows for short, unsupported tethers while still providing this flexibility in orientation. By applying the multi-tether concept to chip satellites, we gain not only an inexpensive testbed for larger systems, but also greatly increased functionality of these fentosatellites, through both attitude control and orbital maneuvers. Here we focus on the feasibility of multi-EDT attitude control, and simulate a six-EDT system.

A. Current in the Ionosphere

Since EDTs function by completing a current loop through the ionosphere, their function is highly dependent on charge collection mechanisms and the plasma-tether interface. This section briefly outlines the passive charge collection mechanisms that a multi-EDT chip satellite (chipsat) concept may exploit.

The ionosphere is a region of atmosphere ionized by incoming solar radiation, extending from 50 km out to approximately 500 km, and can be modeled as having numerous layers, conventionally labeled E through F, where each is characterized by different ionizing species.¹¹ The ionosphere is also subject to substantial variability caused by many factors, including solar flares, magnetic storms, solar and lunar tides, and latitude-dependent effects such as plasma convection.¹² In addition to daily cycles, such as the F1 layer disappearing at night, these effects can drastically alter the ionospheric conditions at a given altitude. Peak electron density occurs at roughly 300 km, in the F2 layer, which is dominated by atomic oxygen ions, and has the highest electron density of the ionosphere, at 300 km. The electron density here has daily extremes of around 10⁵⁰ to 10⁵² Ne/m⁻³ and both the ions and electrons are at a temperature (T_e) of ~1eV.¹³

1. Charge Collection

There are several mechanisms by which current is conducted through a conducting body moving through plasma. Since the atmospheric plasma is conductive and effectively grounded, simply applying a voltage across a wire longer than a few Debye lengths apart will allow current to flow. The Debye length is defined as the characteristic shielding distance in a plasma and is given by

$$\lambda_D = \sqrt{\frac{\varepsilon_0 * T_e}{q * n_0}}$$  \hspace{1cm} (1)

A wire with a potential across it in a conductive medium such as the ionosphere attracts charge carriers at each contact surface. Since a quasi-neutral plasma such as the ionosphere is comprised of both free electrons and ions, each species acts as charge carriers in different physical processes. Since electrons are much less massive than atomic oxygen ions, their thermal velocity is also much higher. In our region of interest, the spacecraft speed through the ionosphere is between these thermal velocities, and the spacecraft thus interacts with the charge carriers differently. The major currents are summarized in Figure 1 However, charge collection by a probe in a flowing, magnetized plasma is a notoriously challenging problem. So, instead of explicit modeling, a number of approximation models give bounding estimates for current produced by a given potential bias, incorporating the geometry of the contacts.⁴

2. Drift Current

Plasma contacts with negative voltages relative to the ambient plasma primarily collect current through electron-drift effects. The high relative speed of electrons allows them to perturb the charge concentration around a biased plasma contact, either being attracted or repelled. This sheath region develops a potential gradient from the contact bias to the unperturbed plasma, and its size depends on temperature, Debye length, and potential bias. One model for sheath-size is the Child-Langmuir Law:⁷

$$S_o = \frac{\sqrt{2}}{3} \lambda_D \left(\frac{2 \phi_s (3/4)}{T_e}\right)$$  \hspace{1cm} (2)
According to the Rubinstein-Laframboise model, the electron thermal current is then

\[ I_{th} = n q_e 4 \pi r_p^2 \sqrt{\frac{kT_e}{2\pi m_e}} \]  

and the total current, in terms of the thermal current, the dimensionless potential bias \( \psi = \frac{2eV_0}{T_e} \) and the ratio of the effective sheath radius to gyroradius of electrons \( \beta \), is

\[ I_e = I_{th} \left( \frac{1}{2} + \frac{2 \sqrt{\psi}}{\beta \sqrt{\pi}} + \frac{2}{\beta^2 \pi} \right) \]  

This model assumes that the sheath thickness is small relative to the Debye length, and that the potential bias is large and positive. If the contactor has a large positive voltage, the plasma magnetization term becomes much more relevant as the effective sheath increases in size. There are a number of other theories for modeling this current at high potentials, but since the chip-sat scale allows operation at relatively low potentials, the deviation from the Child-Langmuir law is less marked than for more traditional, high-power EDT systems.

3. Ram Current

In the ionosphere, the orbital speed can easily exceed the thermal velocity of the ions, leading to charge collection on the windward faces as the spacecraft sweeps through. This produces a ram current, which may be calculated as follows.

\[ I_{ram} = \eta_{ram} n_i q_i v_r A_{ram} \]  

If the surface is adequately connected to mobile electrons, then this current will not necessarily result in a positive charge build-up.

4. Other Currents

Photoemission is another mechanism of current collection. It occurs on any conductive surface, when an incident photon excites an electron enough to be ejected, although depending on the energies it may be recollected. Other passive mechanisms include secondary electron emission, thermionic emission, and impact vaporization, but each has relatively insignificant effects on total current. Finally, electrons or ions may be actively expelled, using a number of devices. Examples include hollow-cathodes, electron field emission, and hot cathode technologies. However, the low power and small size of a multi-EDT chip satellite system make passive collection much more feasible here.

B. Electrodynamic Tethers

Single electrodynamic tether systems have been used successfully, but primarily for orbit raising and lowering. They are particularly attractive as a method for passive deorbiting of spacecraft in low inclination orbits, since the current induced by orbital velocity provides an almost directly opposing force. The basic principle for electrodynamic tethers is straightforward: charged particles moving through a magnetic field experience a force, a principle that also applies to current moving through a conductor. With electrodynamic tethers, the circuit is closed through the ionosphere, thus allowing only the charges traveling in the tether to exert a force on the spacecraft.

In an inertial reference frame, an infinitesimal piece of current \( Jd\vec{l} \) in a magnetic field \( \vec{B} \) experiences a force

\[ d\vec{F} = Jd\vec{l} \times \vec{B} \]
The tether thus experiences a total force

$$\vec{F} = J \int \hat{\vec{r}} \times \vec{B} dl$$

(7)

Except in special cases, these forces also impart a net torque onto the spacecraft.

$$\vec{\tau} = J \int \hat{\vec{r}} \times (\hat{\vec{r}} \times \vec{B}) dl$$

(8)

See Figure 2 for a diagram showing the relative directions of force and torque produced by a single tether. With multiple tethers, we can produce any force in the plane perpendicular to the magnetic field, and any torque in three dimensions. Conventional single-tether systems use gravity gradient effects to stabilize the orientation of the tether and to maintain its extension. This architecture makes them vulnerable to a number of instabilities, including out of plane librations, which requires relying on the orbital rotation of the magnetic field to provide damping.3

All electrodynamic tether systems convert energy between electrical and kinetic. In their standard use as deorbiting devices, they use the magnetic field to pull energy out of the orbit, expending that power through electrical resistance. However, they can also inject energy into the orbit, provided by solar cells or other sources.

The scale of chip satellites allows much shorter, and therefore semi-rigid, tethers. This benefit allows the use of multiple, unsupported tethers magnetic forces to fully control the spacecraft orientation. Multi-EDT systems rely on the same physical processes, but can independently impart force and torque. Furthermore, with sufficiently many independent tethers, the spacecraft may be controlled in five of the six rigid-body degrees of freedom, excluding only force parallel with the magnetic field. This feature is a substantial improvement over other magnetic attitude control systems.

The net force and torque are each the sum of the force and torque due to each tether, shown in Equation 9.

$$\begin{bmatrix} \vec{F} \\ \vec{\tau} \end{bmatrix} = \begin{bmatrix} \hat{\vec{l}}_1 \times \vec{B} & \hat{\vec{l}}_2 \times \vec{B} & \cdots & \hat{\vec{l}}_n \times \vec{B} \\ \frac{1}{2} \hat{\vec{l}}_1 \times (\hat{\vec{l}}_1 \times \vec{B}) & \frac{1}{2} \hat{\vec{l}}_2 \times (\hat{\vec{l}}_2 \times \vec{B}) & \cdots & \frac{1}{2} \hat{\vec{l}}_n \times (\hat{\vec{l}}_n \times \vec{B}) \end{bmatrix} * \vec{J}$$

(9)

Using the Moore-Penrose pseudoinverse of the Jacobian matrix in Eq. (9) provides a closed-form solution for the current required in each tether to produce a desired net force and torque on the system.

$$\vec{J} = A^+ \begin{bmatrix} \vec{F} \\ \vec{\tau} \end{bmatrix}$$

(10)

The pseudoinverse matrix $A^+$ allows the controller to partition the mechanical dynamics from the electrical dynamics.

III. Simulation

To demonstrate that a multi-EDT system can provide attitude control for a chip satellite, we present a simplified simulation. The model uses a circular orbit and an altitude of 300 km. Conveniently, this orbit falls at the highest electron density in the F2 region of the ionosphere. Explicitly modeling the electron density would be prohibitively complex for the purposes of this work; so the model uses a quasi-neutral plasma, with an electron density of $10^2 N_e / m^{-3}$ and a Debye length of $2 \times 10^{-3} m$. More detailed values can be obtained from the International Reference Ionosphere-2007 (IRI-2007) model.
The model also uses a low inclination orbit, in order to highlight difficulties from special orientations relative to the magnetic field, such as when one or more tethers are closely aligned with the field lines. In single tether systems such as the design described in Bell, et al. (2011), this limitation is chosen to limit out-of-plane thrust which cause librations, but multi-EDT systems do not have this limitation since they can provide multi axis control.

The simulated system is .1 g, with six separate, 10 cm long electrodynamic tethers, each with a small spherical end-body for charge collection. For simplicity, the simulation models the case of tethers that are insulated along their lengths, with insignificance mass in the charge collectors at the ends. While uninsulated or partially insulated tethers have other advantages, as described in Sanmartin, et al. (1993) and Estes, et al., end-body collection allows the model to neglect current variation along the tether. Furthermore, the distance between plasma contacts means that interactions between tethers are minimal. The chip itself also has a plasma contact, so that each tether may have fully independent currents. The layout used in the simulation is shown in Figure 3. The system has six tethers arrayed in an off-center hexagon as shown by the dashed line, forming a hexagonal pyramid. This means that only one tether can be completely disabled through alignment with the magnetic field.

In this simulation, the tethers do not substantially affect the moment of inertia. With aluminum tethers at a radius 25 \( \mu \text{m} \) and a density of 2700 kg/m\(^3\) each tether will have a mass of only around 50 \( \mu \text{g} \). Since they are relatively short, and so much lighter than the chip, they do not contribute substantially to the moment of inertia. In future work, it may be advantageous to explicitly consider how nonsymmetric moments of inertia may be helpful in particular missions; however, this is beyond our current scope. The tethers are assumed to be quasi-rigid - they will dissipate energy, modeled with a Kane damping term, but do not deflect enough to substantially change the generated forces.

This simulation transforms between two reference frames, the non-inertial body frame of the chip satellite, where the tether directions and moment of inertia tensor are constant, and the Earth-centered inertial (ECI) frame, where the z axis is along the geographic north pole and the y-axis is perpendicular to the direction of the sun. The simulation uses the International Geomagnetic Reference Field (IGRF) to provide the magnetic field vectors at each timestep. The forces and torques are calculated in the body frame and then transformed to the orbit frame to solve the equations of motion. A variable-step Dormand-Prince solver is used for integration.

We chose a relatively simple controller for these initial investigations. The controller uses a PD feedback loop. The control gains were derived from the desired closed-loop natural frequency. A more sophisticated controller would provide better overall performance, but the intent here is to show that multi-EDT attitude control is indeed feasible.

As discussed above, the spacecraft interacts with electrons differently than ions, which can lead to overall charging as well as nonuniform charge distribution. Chip satellites also have a high surface-area-to-mass ratio, and thus are more than usually vulnerable to aerodynamic drag. Depending on the attitude relative to the orbital velocity, the chip satellite will experience a torque. However, this simulation does not incorporate these effects, in the interest of simplicity.

IV. Results

The simulation shows that a multi-EDT system can provide attitude stabilization while simultaneouesly performing orbital maneuvers for a chip-sized spacecraft. The chipsat was initialized with an angular velocity of 1 rad/s about an arbitrary axis. The controller attempted to enforce a zero angular velocity, a quaternion command of [-.02 .01 .03 1] relative to ECI, and apply 1 nN along the velocity vector to provide an orbit.
boost. The simulated results are summarized below. Figures 4 and 5 show that the simple PD controller can stabilize attitude, and point in a particular direction in a short amount of time, on the order of tens of seconds.

![Figure 4. Spacecraft attitude relative to the ECI coordinate system.](image)

![Figure 5. Spacecraft angular velocity in the ECI coordinate system.](image)

The multi-tether system can also provide independent net force while controlling attitude. Figure 6 shows the applied force. Since the controller tries to apply a constant force along the velocity vector, when the velocity is somewhat aligned with the magnetic field, the force magnitude is reduced, leading to the cyclic pattern shown. More sophisticated control may provide specific orbital adjustments.

This simulation focuses primarily on the interaction of the mechanical system with the magnetic field. However, the electrical subsystem of the multi-EDT system must also be feasible. Since the main body of the spacecraft is likely to be a single substrate, there are real limits on current and voltage. Figure 7 and 8 on the following page show the current in each tether during the beginning stabilization phase and the steady-state behavior in which the force is applied. Since each of the currents does not exceed a few mA, the design is feasible in many semiconductor technologies.

The periodic variations seen in Figure 8 on the next page result from the changing magnetic field as the spacecraft orbits. Furthermore, as the applied force causes the orbital radius to increase, the combined magnitude will also increase to maintain a constant force, since the strength of the geomagnetic field is decreasing. More complex orbital maneuvers can take this into account so that the currents do not exceed design specifications.

However, even with the redundancy of a six-tether design, the performance is still dependent on spacecraft orientation. Figure 9 shows the simulated response of the system with all parameters the same, except that the initial position of the tethers has been

![Figure 6. Spacecraft altitude relative to mean Earth radius.](image)
rotated by 90 degrees. One of the tethers becomes nearly aligned with the magnetic field, leading to large spikes in current. To keep the force and torques smooth, the currents need much larger magnitude. With a more advanced controller, these spikes can be artificially smoothed. For simplicity, this simulation does not consider many perturbing effects, such as aerodynamic drag or solar pressure. Additionally, the simulation would be improved by modeling the electronics, since internal capacitances and inductances may substantially affect the dynamics of the electromagnetic system.

V. Conclusion

This work demonstrates attitude stabilization, precise pointing, and orbit maneuvering using a multi-EDT system for a number of special cases. An important next step is to integrate a non-linear attitude control and also provide for arbitrary orbital maneuvers. This concept also merits a more comprehensive design approach to optimize tether placement and other physical parameters. The tethers will affect the moment of inertia of the spacecraft as well as the drag torques, and careful consideration may allow this interaction to be advantageous for higher efficiency in specific missions.

A multi-EDT chipsat has the potential for application in a variety of missions. For example, with attitude control, a chip satellite may be able to maintain sunpointing, allowing a solar cell to gather energy more efficiently. Alternatively, it could appropriately orient a sensor, increasing the the overall utility. It would also be advantageous to better characterize the power required, since the control electronics will provide another layer of non-linearities to the system. With careful electrical design, it may be feasible to redirect geomagnetically-induced currents for some amount of force and torque control. By natively taking advantage of magnetically...
induced voltages, a multi-EDT chip satellite may exhibit passive attitude control.

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References