Satellite attitude control using electrodynamic booms

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Abstract: This paper proposes the use of electrodynamic booms for satellite attitude control. By manipulating the magnitude and direction of electrical currents flowing through each boom, the induced Lorentz force acting on the booms can be harnessed as thrust and torque. This can potentially combine the spacecraft propulsion and attitude control systems into one package. This work presents the mathematical formulation of the electrodynamic torque and examines a feedback control algorithm to regulate the spacecraft angular velocity and orientation with respect to an inertial frame. The control algorithm is tested in simulation for several mission scenarios.

Keywords: electrodynamic booms; attitude control; feedback control; magnetic actuation.

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1 Introduction

The electrodynamic boom system, also called integrated structural electrodynamic propulsion, extends the basic concept of electrodynamic tethers and replaces the conductive cable with current carrying booms. An orthogonal set of booms can generate thrust and torque in almost any direction, enabling the propulsion and the attitude control systems to be combined into one package. Spacecraft propulsion using electrodynamic booms has been studied by several authors. Voronka et al. (2006) studied the feasibility and performance of electrodynamic booms. They concluded that such systems are technologically within reach and have performance comparable to other electric propulsion systems, but engineering challenges remain. Matthew and Voronka (2008) developed open loop and feedback controllers for orbit manoeuvring. Wong and Damaren (2010) also studied the orbital manoeuvring problem and developed a two-part open-loop algorithm that accounts for atmospheric drag. Lawrence (1992) investigated two possible boom configurations for the precision orbital tracking vehicle and developed a control algorithm for position and attitude control relative to another satellite using electrodynamic propulsion and attitude control.

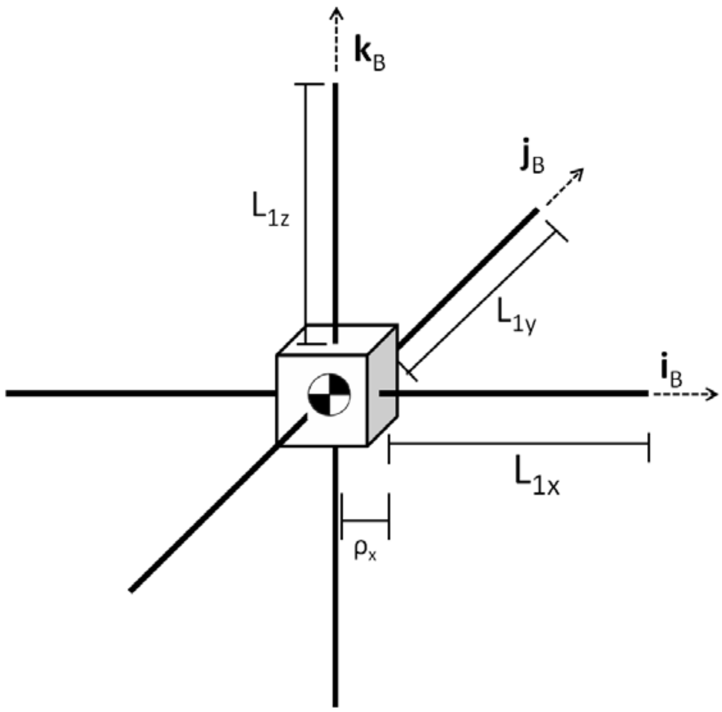
Torque for spacecraft attitude control can be provided by manipulating the Lorentz force generated by individual booms. Similar to the magnetorquer, the amount of obtainable torque depends upon the orientation of the booms relative to the magnetic field. Previous work on magnetic actuators can lend insight into the design of attitude controller using electrodynamic booms, as both types of actuators have time-varying dynamics and the control system cannot guarantee three independent torques at all times. For spacecraft operations near its desired attitude, Wisniewski and Markley (2001) and Lovera et al. (2002), amongst other authors, exploited the quasi-periodicity of the system to develop and analyse the stability and performance of control laws. The general formulation of the magnetic attitude control problem is studied by Wisniewski and Blanke (1999), Damaren (2002), and Lovera and Astolfi (2004).

This paper demonstrates the concept of controlling spacecraft attitude using electrodynamic booms. The mathematical expressions relating the electrodynamic torque to the input electrical currents are first derived, and a non-linear control law is developed to regulate the spacecraft attitude relative to the Earth centred inertial (ECI) frame. Several numerical examples would show the utility of this control and determine the amount of current required.

2 System modelling

The electrodynamic satellite of interest can be modelled as an arbitrary body with several rigid appendages extending from the body. The development of the attitude control scheme and subsequent analysis of the system are performed using two reference frames. The first is the ECI frame that is fixed at the centre of the planet, with the $\mathbf{x}_I$ axis pointing towards Ares, the $\mathbf{z}_I$ axis aligned with the rotation axis of the body, and $\mathbf{y}_I$ completing the triad. The second frame is the spacecraft body frame located at the mass centre of the satellite with $\mathbf{i}_B$, $\mathbf{j}_B$ and $\mathbf{k}_B$ aligned with the principal body axes. The orientation of the body frame relative to the inertial frame is parameterised by a set of quaternions ($\mathbf{q}$, q_4). This work assumes that the electrodynamic spacecraft has six rigid booms and they are aligned with the $\pm\,\mathbf{i}_B$, $\pm\,\mathbf{j}_B$, and $\pm\,\mathbf{k}_B$ directions respectively (Figure 1). This is not a requirement for a realistic multi-boom system and there could be some advantages in using a non-aligned setup. The boom lengths are represented by L_{ix}, L_{iy}, L_{iz} for i = 1, 2, and the booms attachment points are located a distance of ρ away from the spacecraft mass centre.

Figure 1 Spacecraft model

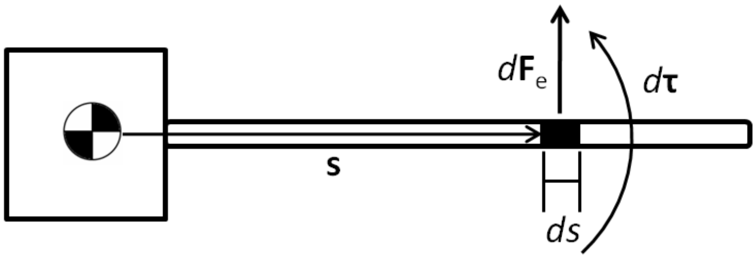


The Earth's magnetic field is idealised as a simple non-tilted dipole fixed at the centred of the planet and the magnetic axis $\mathbf{m}$ is aligned with $\mathbf{z}_I$. This implies that the magnetic equator is coincident with the geographic equator. The magnetic field expressed in the inertial frame is given by

$$\mathbf{B}_I = -\frac{\mu_m}{R^3}\left[3(\mathbf{r}\cdot\mathbf{m})\mathbf{r} - \mathbf{m}\right] \tag{1}$$

where μ_m is the magnetic moment of the dipole, R is the radial distance from the centre of the planet to the spacecraft, and $\mathbf{r}$ is the unit vector pointing from the centre of the planet towards the spacecraft. The geomagnetic field measured in the body frame is given by $\mathbf{B} = [B_x\ B_y\ B_z]^T$, and $\mathbf{B}$ is related to $\mathbf{B}_I$ through the rotation matrix $\mathbf{R}_{BI}$ such that $\mathbf{B} = \mathbf{R}_{BI}\mathbf{B}_I$. The Lorentz force acting on a boom segment extending in the $\mathbf{e}_s$ direction is given by $d\mathbf{F}_e = I\mathbf{e}_s \times \mathbf{B}ds$, where ds is the segment length and I is the current flowing through the segment. The total Lorentz force acting on the boom can be calculated by integrating $d\mathbf{F}_e$ over the length of the beam. Similarly, the torque about the spacecraft mass centre generated by the boom segment is given by $d\boldsymbol{\tau}_m = \mathbf{s} \times I\mathbf{e}_s \times \mathbf{B}ds$ where $\mathbf{s}$ is the position vector of the segment relative to the centre of mass (Figure 2).

Figure 2 Differential electrodynamic force and torque



The electrodynamic torque $\boldsymbol{\tau}_{jm}$ acting on the j^{th} boom can be calculated by integrating $d\boldsymbol{\tau}_{jm}$ over the length of the boom such that,

$$\boldsymbol{\tau}_{jm} = \int_{\rho_j}^{\rho_j + L_j} I_j \mathbf{e}_j \times (\mathbf{e}_j \times \mathbf{B}) ds = \left(\frac{L_j^2}{2} + \rho_j L_j \right) I_j B \left(\mathbf{e}_j \times (\mathbf{e}_j \times \mathbf{b}) \right) \tag{2}$$

where $\mathbf{e}_j$ is a vector tangential to the j^{th} boom, ρ_j and L_j are the offset distance and the length of the j^{th} boom respectively and I_j is the total current flowing through that segment in the $\mathbf{e}_j$ direction. Adding together the torques generated by the six booms, the total electrodynamic torque $\boldsymbol{\tau}_m$ is given by

$$\begin{aligned} \boldsymbol{\tau}_m = & \int_{\rho l 1x}^{\rho_{1x}+L_{1x}} I_{1x} \mathbf{i}_B \times (\mathbf{i}_B \times \mathbf{B}) ds + \int_{\rho_{2x}}^{\rho_{2x}+L_{2x}} I_{2x} (-\mathbf{i}_B) \times (\mathbf{i}_B \times \mathbf{B}) ds \\ & + \int_{\rho_{1y}}^{\rho_{1y}+L_{1y}} I_{1y} \mathbf{j}_B \times (\mathbf{j}_B \times \mathbf{B}) ds + \int_{\rho_{2y}}^{\rho_{2y}+L_{2y}} I_{2y} (-\mathbf{j}_B) \times (\mathbf{j}_B \times \mathbf{B}) ds \\ & + \int_{p_{1z}}^{\rho_{1z}+L_{1z}} I_{1z} \mathbf{k}_B \times (\mathbf{k}_B \times \mathbf{B}) ds + \int_{\rho_{2z}}^{\rho_{2z}+L_{2z}} I_{2z} (-\mathbf{k}_B) \times (\mathbf{k}_B \times \mathbf{B}) ds \end{aligned} \tag{3}$$

Completing the integrals in equation (3) and writing the result in matrix form, $\boldsymbol{\tau}_m$ can be simplified as

$$\boldsymbol{\tau}_m = \mathcal{L}_x^{\times} (\Delta \boldsymbol{I}_{\mathbf{x}}^{\times}) \mathbf{B} + \mathcal{L}_y^{\times} (\Delta \boldsymbol{I}_{\mathbf{y}}^{\times}) \mathbf{B} + \mathcal{L}_z^{\times} (\Delta \boldsymbol{I}_{\mathbf{z}}^{\times}) \mathbf{B} \tag{4}$$

$$\mathcal{L}_x^{\times} = \left(\frac{L_x^2}{2} + \rho_x L_x \right) \mathbf{i}_{\mathbf{B}}^{\times}, \mathcal{L}_y^{\times} = \left(\frac{L_y^2}{2} + \rho_y L_y \right) \mathbf{j}_{\mathbf{B}}^{\times}, \mathcal{L}_z^{\times} = \left(\frac{L_z^2}{2} + \rho_z L_z \right) \mathbf{k}_{\mathbf{B}}^{\times} \tag{5}$$

$$\Delta \mathbf{I}_{\mathbf{x}}^{\times} = (I_{1x} - I_{2x}) \mathbf{i}_{\mathbf{B}}^{\times}, \Delta \mathbf{I}_{\mathbf{y}}^{\times} = (I_{1y} - I_{2y}) \mathbf{j}_{\mathbf{B}}^{\times}, \Delta \mathbf{I}_{\mathbf{z}}^{\times} = (I_{1z} - I_{2z}) \mathbf{k}_{\mathbf{B}}^{\times} \tag{6}$$

$\mathbf{i}_{\mathbf{B}}^{\times}$, $\mathbf{j}_{\mathbf{B}}^{\times}$ and $\mathbf{k}_{\mathbf{B}}^{\times}$ represent the cross product matrices of $\mathbf{i}_B$, $\mathbf{j}_B$, $\mathbf{k}_B$. The attachment points of opposite booms are assumed to have the same offset from the spacecraft mass centre, such that $\rho_{1x} = \rho_{2x} = \rho_x$, $\rho_{1y} = \rho_{2y} = \rho_y$, and $\rho_{1z} = \rho_{2z} = \rho_z$. Further manipulating equation (4) and isolating ΔI_x, ΔI_y, and ΔI_z, it can be shown that

$$\boldsymbol{\tau}_m = \begin{bmatrix} 0 & B_y \mathcal{L}_x & B_z \mathcal{L}_x \\ B_x \mathcal{L}_y & 0 & B_z \mathcal{L}_y \\ B_x \mathcal{L}_z & B_y \mathcal{L}_z & 0 \end{bmatrix}^T \begin{bmatrix} \Delta I_x \\ \Delta I_y \\ \Delta I_z \end{bmatrix} = \boldsymbol{\gamma}^T \mathbf{u} \tag{7}$$

where $\mathbf{u}$ is the control input. Υ is a time-varying matrix as B_x, B_y, and B_z change with the location and orientation of the satellite. This means that the control input required to generate a desired torque also varies with time, similar to conventional magnetic torque rods. The spacecraft power system can choose to generate the desired current difference ΔI by using changing the magnitude and the direction of the currents within each boom. Pure torque is generated when currents of the same magnitude flow in opposite directions, while a mix of thrust and torque is created when currents of different magnitude flow in the same direction. Pure thrust is generated when currents of the same magnitude flow in the same direction.

The attitude dynamics of the spacecraft is represented by Euler's equation

$$\mathbf{J}\dot{\boldsymbol{\omega}} + \boldsymbol{\omega}^{\times}\mathbf{J}\boldsymbol{\omega} = \boldsymbol{\tau}_{gg} + \boldsymbol{\tau}_m \tag{8}$$

where $\mathbf{J}$ is the inertia matrix of the satellite, $\boldsymbol{\omega}$ is the inertial angular velocity vector expressed in the body frame and $\boldsymbol{\tau}_{gg}$ represents the gravity-gradient torque. As the attitude is parameterised by quaternions, the kinematic differential equation is given by

$$\dot{\mathbf{q}} = \frac{1}{2}\mathbf{S}\boldsymbol{\omega} = \frac{1}{2}\begin{bmatrix} q_4 & -q_3 & q_2 \\ q_3 & q_4 & -q_1 \\ -q_2 & q_1 & q_4 \end{bmatrix}\boldsymbol{\omega} \tag{9}$$

$$\dot{q}_4 = -\mathbf{q}^T\boldsymbol{\omega} \tag{10}$$

Equations (7), (8), (9) and (10) together give a description of the satellite attitude dynamics under the influence of the electrodynamic and gravity gradient torques.

3 Attitude control law

A non-linear feedback control law is developed to calculate the currents required to regulate the states of the satellite from any initial condition. Consider the attitude control law where the control input $\mathbf{u}$ is calculated using

$$\mathbf{u} = -\Upsilon\left(K_d\boldsymbol{\omega} + K_p\mathbf{q}\right) \tag{10}$$

where $K_d > 0$ and $K_p \geq 0$ are the derivative and proportional gains respectively, $\boldsymbol{\omega}$ is the spacecraft angular velocity vector, $\mathbf{q}$ is the vector part of the quaternion and Υ is given by equation (7). The control law adds damping to the system dynamics when $K_d > 0$ and $K_p = 0$, taking energy away from the system. Setting both K_d and K_p greater than zero makes $\mathbf{u}$ have the form of a PD law that can drive the system states to zero for suitable choices of control gains. Substituting equation (10) into equation (7), the control torque is given by

$$\boldsymbol{\tau}_m = -\Upsilon^T\Upsilon\left(K_d\boldsymbol{\omega} + K_p\mathbf{q}\right) \tag{11}$$

$$\Upsilon^T\Upsilon = \begin{bmatrix} B_x^2\left(\mathcal{L}_y^2 + \mathcal{L}_z^2\right) & B_xB_y\mathcal{L}_z^2 & B_xB_z\mathcal{L}_y^2 \\ B_xB_y\mathcal{L}_z^2 & B_y^2\left(\mathcal{L}_x^2 + \mathcal{L}_z^2\right) & B_yB_z\mathcal{L}_x^2 \\ B_xB_z\mathcal{L}_y^2 & B_yB_z\mathcal{L}_z^2 & B_z^2\left(\mathcal{L}_x^2 + \mathcal{L}_y^2\right) \end{bmatrix} \tag{12}$$

The determinant of the $\Upsilon^T\Upsilon$ matrix is $4(B_x B_y B_z \mathcal{L}_x \mathcal{L}_y \mathcal{L}_z)^2$ and the matrix has full rank only when B_x, B_y and B_z are all non-zero, otherwise the system becomes under actuated. As the satellite is constantly changing its orientation with respect to the local magnetic field lines these periods of under actuation are usually short lived, but in special situations the controller can have very little control authority. An example of this is a satellite in a near-equatorial orbit with its attitude is closely aligned with the ECI frame. In this case, B_x and B_y have small magnitudes for long periods of time and very little torque can be generated about $\mathbf{x}_B$ and $\mathbf{y}_B$.

The global asymptotic stability of the attitude control law is difficult to prove due to the time-varying nature of the $\Upsilon^T\Upsilon$ matrix, especially when both K_p and K_d are non-zero. Some researchers in magnetic actuation such as Wisniewski and Blanke (1999) have exploited the quasi-periodic nature of the magnetic field to apply Floquet theory, demonstrating the linear stability of the spacecraft about a desired orientation. Other authors such as Lovera and Astolfi (2004) used an averaging method to show that the satellite is controllable on average when its angular rate is below a critical value. This work is mainly concerned with the feasibility of using electrodynamic booms to regulating the spacecraft states from arbitrary initial conditions and spacecraft location. Standard non-linear proofs of stability are very difficult to apply in these situations, and therefore numerical simulations are used to verify that the control law is stable for several scenarios.

4 Simulation results

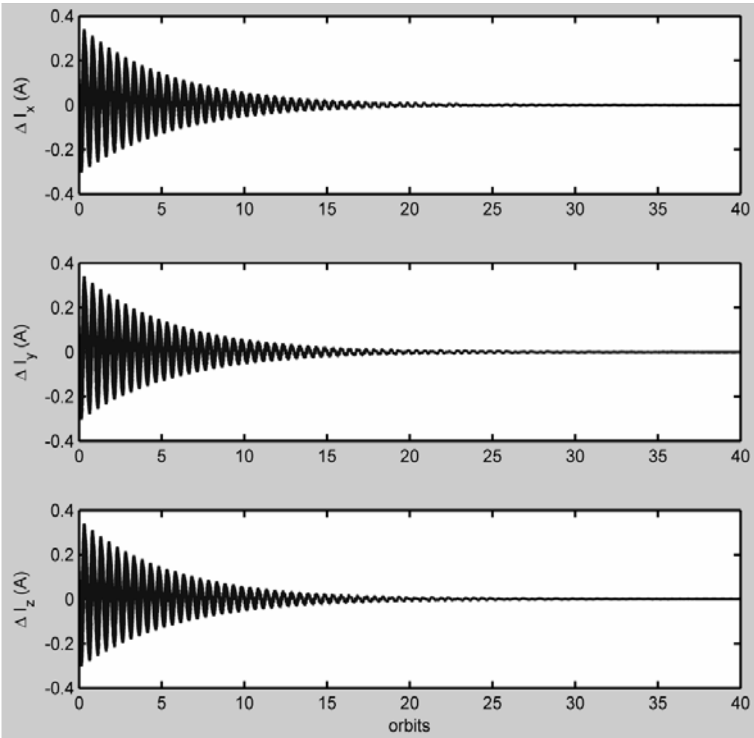
The stability of the attitude control law is difficult to assess using non-linear control theory. In this work, the performance and stability of the control law are tested for several scenarios using numerical simulations. Of interest are the relationship between the stability of the control law and the orbit inclination and the relationship between K_p and K_d. Assuming that the electrodynamic booms exert no net force on the spacecraft and neglecting other orbital perturbations, the orbital equation of motion is given by

$$\ddot{\mathbf{R}} = \frac{\mu}{R^3}\mathbf{R} = \mathbf{0} \tag{13}$$

where $\mathbf{R}$ is the position vector of the spacecraft in the inertial frame and μ is the gravitational parameter. The position of the satellite is then used to compute $\mathbf{B}_I$ using equation (1) with $\mu_m = 8 \times 10^6$ T · km^3. The inertia matrix of the spacecraft $\mathbf{J}$ is give by $\mathbf{J} = \text{diag}[100\ 100\ 100]$ kg · m^2, while the boom lengths and mass centre offsets are given by $L_x = L_y = L_z = 10$ metres, and $\rho_x = \rho_y = \rho_z = 1$ metre. This example satellite is axisymmetric and $\boldsymbol{\tau}_{gg}$ is zero. In all simulations, the satellite is in an 800 km altitude circular orbit and the spacecraft initial conditions are given by $\boldsymbol{\omega}_0 = [5\ -5\ 5]^T$ deg/s and $\mathbf{q}_0 = [0.01\ 0.03\ 0.2\ 0.9327]^T$.

The first three examples consider spacecraft detumbling for different orbit inclinations. In the first example, the spacecraft is in a 45 degrees inclination orbit and K_d is set to 1,500 A · s/Wb while K_p is zero. Figure 3 shows that the angular velocity of the satellite asymptotically approaches zero over a period of time, decreasing to less than 10^{-4} deg/s after 38 orbits. The maximum ΔI for each boom pair is 0.3417 A, recorded near at the start of the simulation. The currents also rapidly switch directions in response to the changing orientations of the booms. The quaternions are not driven to zero as this controller does not regulate spacecraft attitude. Figure 4 shows that the control torque required to detumble the spacecraft has order of magnitude of 10^{-4} Nm. Increasing the derivative gain reduces the time required to detumble the spacecraft at the cost of greater current demand.

Figure 3 Satellite detumbling, i = 45 deg, K_d = 1,500 A · s/Wb



As previously mentioned, equatorial orbits present a challenge to electrodynamic control. The primary reason is that only B_{Iz} is non-zero in these situations, and this reduces the magnitude of the available torque for the same input current. As the control law given by equation (10) takes Υ into account, less current is also commanded for smaller B_x, B_y and B_z. In the second example, the electrodynamic satellite is occupying a zero inclination orbit. For the same control gain in the first example, K_d = 1,500 A · s/Wb, Figure 5 shows ω_{xB}, ω_{yB} and ω_{zB} decreases to less than 10^{-4} deg/s after 132 orbits, vs. 38 orbits in the first example. The maximum ΔI is also lower at 0.1530 A. For the third example, K_d is set to 4,000 A · s/Wb and the time required to detumble the satellite is decreased to 50 orbits, and the time histories of the spacecraft angular velocity and control inputs are shown in Figure 6. The maximum ΔI is 0.4078 A is higher than the result from the first example.

Figure 4 Body torque produced by the electrodynamic booms

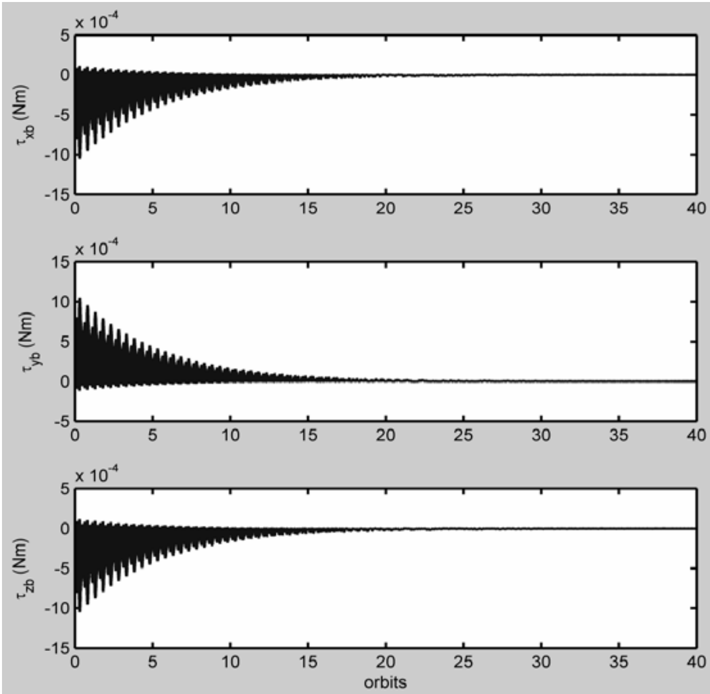


Figure 5 Satellite detumbling, $i = 0$ deg, $K_d = 1{,}500$ A · s/Wb

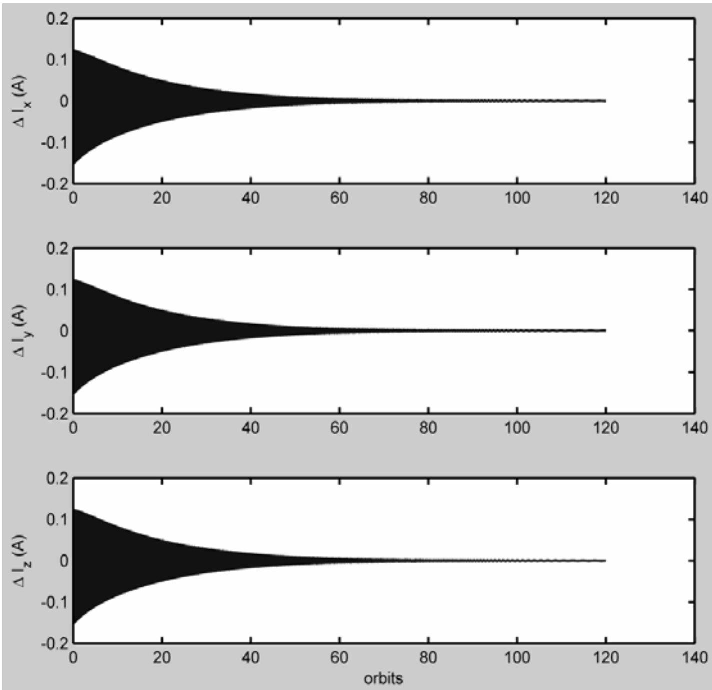
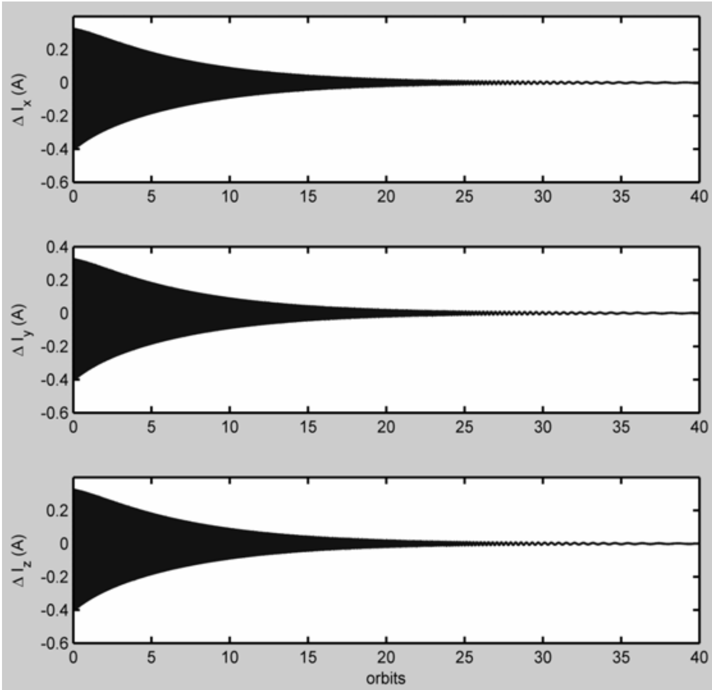


Figure 6 Satellite detumbling, $i = 0$ deg, $K_d = 4{,}000$ A · s/Wb



Satellite attitude regulation can be performed by using positive proportional gain K_p in addition to a positive K_d value. However, the choice of the (K_d, K_p) gain pair is not arbitrary, as numerical simulations show that for an orbit with parameters given by [a, e, i], every K_d corresponds with a critical K_p where if exceeded the control law becomes unstable. It is difficult to predict the critical K_p in a general manner as there is an infinite combination of orbit and satellite design parameters. A simple procedure adopted in this work is to first select K_d as to accomplish the detumbling within a desired time, then gradually increase K_p from some small value until the simulation results show instability. The next three examples consider attitude regulation. For the fourth example, the satellite is again in a 45 degrees inclination orbit and the control gains are given by $K_d = 5{,}000$ A · s/Wb and $K_p = 10$ A/Wb. It can be seen in Figure 7 and Figure 8 that the spacecraft angular velocity decreases quickly, while the spacecraft orientation takes 35 orbits to settle. The maximum ΔI for this operation is 1.0388 A. Increasing K_p to 40 A/Wb while K_d remains at 5,000 A · s/Wb decreases the time required to 30 orbits, but Figure 9 shows that beat oscillation occurs when K_p is increases to 45 A/Wb. Further increasing K_p only worsens the situation. Numerous simulation runs have shown that a stable control law requires that K_p be at least two orders of magnitude small than K_d. This means the detumbling operation is the primary driver of the ΔI requirement and that the controller cannot make use the full capacity of the spacecraft power system for attitude regulation without inducing instability.

Figure 7 Stable eegulation, K_p = 10 A/Wb, K_d = 5,000 A · s/Wb

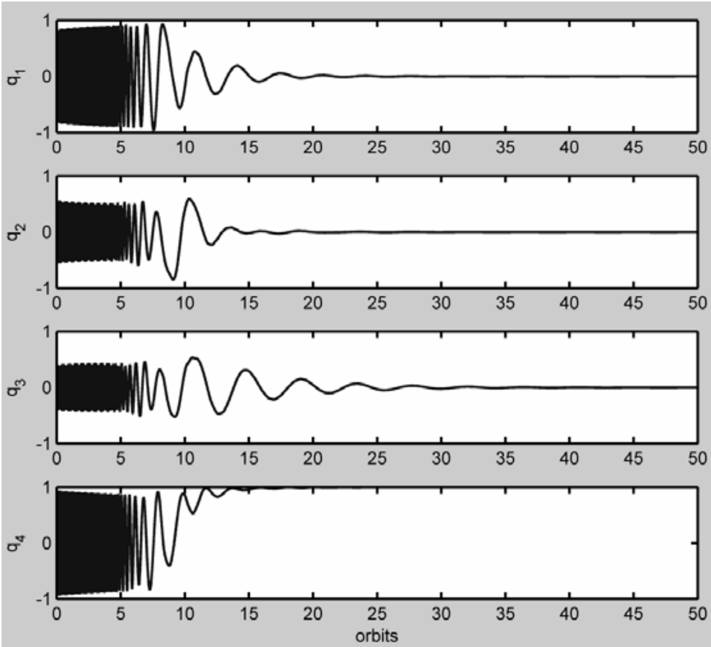
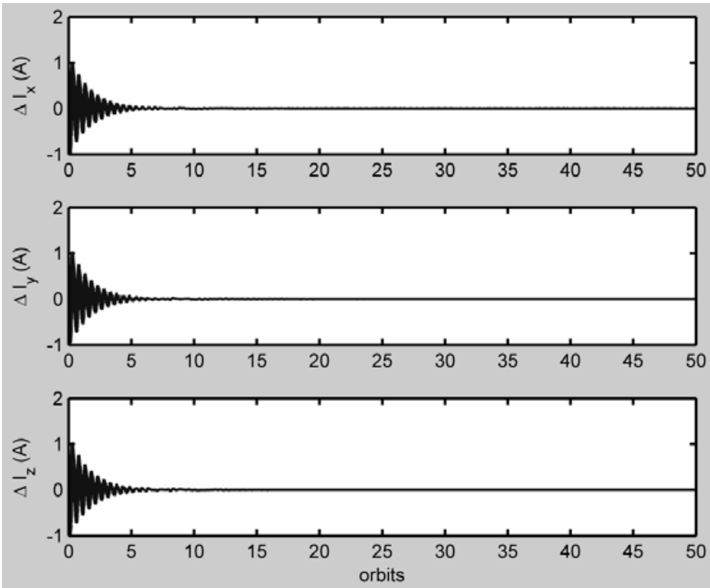


Figure 8 Current difference for regulation



For the last example, consider an equatorial satellite with the control gains set as K_d = 5,000 A · s/Wb and K_p = 0.1 A/Wb. Figure 10 shows that the electrodynamic booms are unable to align the satellite's body axes with the ECI frame even after 300 orbits. This is because the available torque about the $\mathbf{x}_B$ and $\mathbf{y}_B$ axes decreases as the spacecraft attitude rotates to match the ECI frame in those situations. The electrodynamic boom configuration analysed in this paper is unsuitable for use on inertial pointing satellites in equatorial and near-equatorial orbits, and different boom configurations should be considered.

Figure 9 Unstable attitude when K_p =45 A/Wb exceeded

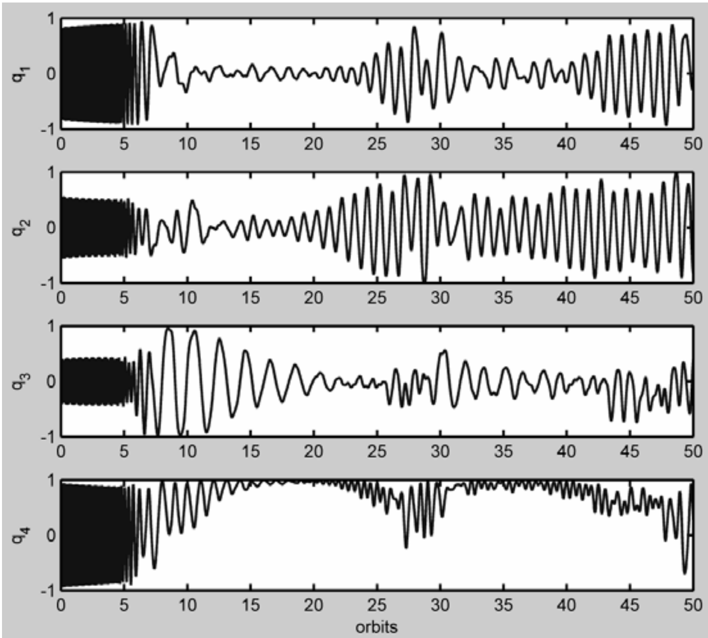
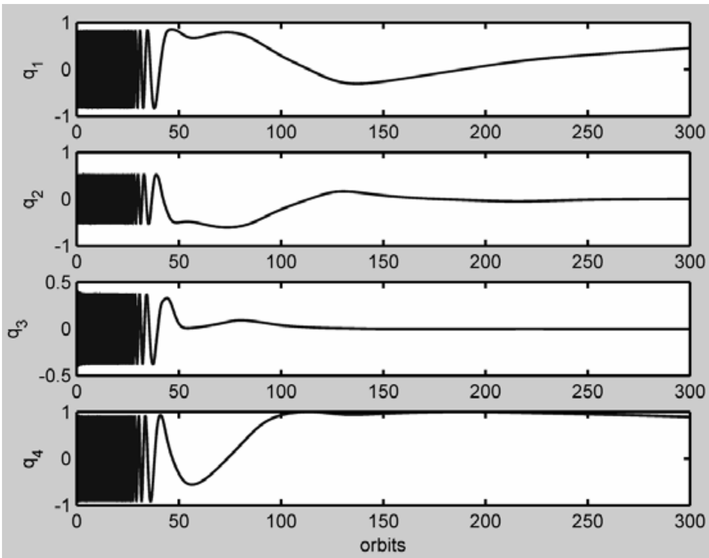


Figure 10 Satellite in equatorial orbit, K_d = 5,000 A · s/Wb, K_p = 0.1 A/Wb



5 Conclusions

The concept of satellite attitude control using electrodynamic booms is introduced in this paper. Electrodynamic booms have been previously introduced for satellite propulsion and this work studies whether the same booms can be use for attitude regulation. The mathematical formulation of the electrodynamic torque is derived, and a PD control law using spacecraft angular velocity and orientation as feedbacks is introduced for satellite detumbling and attitude regulation. Numerical simulations are used to verify the stability of the control law for several operational scenarios. The simulation results presented here show that attitude control using electrodynamic booms is a viable concept for non-equatorial orbits. Setting the proportional gain to zero and selecting a positive derivative gain in the control law would drive the spacecraft angular velocity to zero from its initial state, while setting using positive derivative and proportional gains regulates the spacecraft states with respect to the ECI frame. It is found that for a satellite in a particular orbit, every derivative gain selection has a corresponding maximum proportional gain. The obtained results show that the effectiveness of the electrodynamic booms is affected by orbit inclination and that the boom configuration studied in this paper performs worse in equatorial orbits. Previous work on propulsion using electrodynamic booms found that several amps are required for orbital manoeuvres. The current requirements for detumbling and attitude regulation are a fraction of those required for the propulsion tasks.

Future studies on this topic would use a more detailed geomagnetic model and develop new control laws enable continuous pointing at ground targets. Combined propulsion and attitude control manoeuvres in the presence of one or more magnetic fields are also an interesting area of study. Some authors have combined conventional reaction wheels with magnetic torque rods to form hybrid controllers. The same approach can be applied to electrodynamic booms in order to overcome controller under actuation.

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