

MEETING SCIENCE REQUIREMENTS FOR ATTITUDE DETERMINATION AND CONTROL IN A LOW-POWER, SPINNING NANOSATELLITE

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Abstract: *this paper describes the attitude determination and control system for a nanosatellite (30 kg), using geomagnetic field data and solar panels as sun sensors, applied to a spinning nanosatellite (Penn State University's LionSat). LionSat will map plasma densities in the ram and wake of the vehicle's path, using two hybrid plasma probes that rotate with the spacecraft. Attitude will be determined using a combination of voltage outputs from contiguous solar panels and a three-axis magnetometer (TAM). To correct for spin-axis drift due to orbital plane nodal regression (approximately five degrees per day), two magnetic torque rods will provide the necessary control actuation. To avoid corrupting the TAM and plasma probe measurements, periodic quiescent intervals for the torque rods are required. Initial design concepts for this mission employed an extended Kalman filter, implemented onboard to predict attitude during the passive control intervals; making use of digital sun sensors as part of the attitude determination scheme; however, budget limitations have resulted in the need to employ the body-mounted solar panels (sides and end caps) to estimate the sun vector. Simulation results indicate that the solar-panel sun sensors (in conjunction with the TAM) can produce the required attitude knowledge for processing the scientific data; the pointing accuracy for spin-axis control (nominally within 5 degrees of orbit-normal) is degraded to 10 degrees accuracy, but this is still within acceptable limits for the science.*

1 Introduction

Penn State University's Local Ionospheric Measurements Satellite (LionSat) was a participant in the University Nanosat-3 program co-sponsored by the American Institute of Aeronautics and Astronautics (AIAA), the National Aeronautics and Space Administration's Goddard Space Flight Center (NASA/GSFC), the U.S. Air Force Office of Scientific Research (AFOSR), and the U.S. Air Force Research Laboratory Space Vehicles Directorate (AFRL/VS).

As seen in Fig. 1, the satellite has an octagonal shape with a height of 0.4 m, an average radius of approximately 0.2 m, and a mass of 30 kg. For purposes of making scientific measurements, the satellite is required to spin about its maximum moment-of-inertia axis, designated as z_B . For an octagonal shape with uniform mass distribution, the moments of inertia have the relation $J_{x_B} = J_{y_B} > J_{z_B}$, which would make it a prolate spinner. In the case of spin stabilization with spin about the z_B -axis, energy dissipation would result in shifting its spin axis to a major moment of inertia axis (x_B or y_B). To prevent this, LionSat's mass will be distributed to make the spacecraft an oblate spinner, with an inertia ratio of $J_{z_B} / J_{x_B} > 1$. For its scientific measurements, the satellite is required to rotate with a rate of 10 rpm, while maintaining its spin axis as close to orbit-normal as possible.

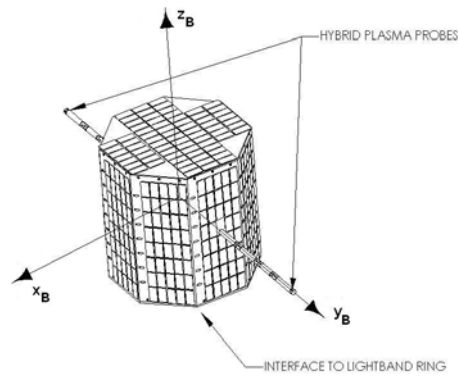


Figure 1. LionSat.

The spin axis should be maintained within 5 degrees of the orbit-normal direction, and azimuth angular position of the scientific probe should be known to within 10 degrees. To reduce complexity and to avoid problems with gyroscopic devices on spinning satellites, no gyroscopic instruments will be used. Instead, we are using three-axis magnetometer (TAM) measurements to find the angular velocity of the satellite. This angular velocity will be used to find the angular momentum vector of the satellite and will be integrated to get an attitude representation in terms of quaternions. Since the angular measurements have some errors, the quaternions will also accumulate errors, therefore frequent updates will be applied with the Triad (Black, 1964; Shuster, 2000) algorithm using the sun vector and the geomagnetic field vector. We will use the term *measurement* to include collectively the physical sensing and the estimation of quantities based upon the sensed data.

2 Overview of attitude determination and control system (ADCS)

The main goal of the attitude determination and control system is to maintain the satellite's attitude as desired at all times. Attitude sensors, algorithms to process the sensor output, control logic, and control hardware work together to achieve this.

The sun vector in the body-fixed coordinate frame is derived from estimates of the sun's direction relative to three contiguous solar panels and the solar panel on one end cap. Because the voltage output from each panel (side or end-cap) depends upon the sun's azimuth α and/or its elevation ε (Figure 2), it is possible to estimate these angles using the voltage information from sets of solar panels (the Appendix provides the required equations). Simulations indicate that this can be achieved with errors of 5 degrees (modeled as the 3σ -error level in the Kalman filter).

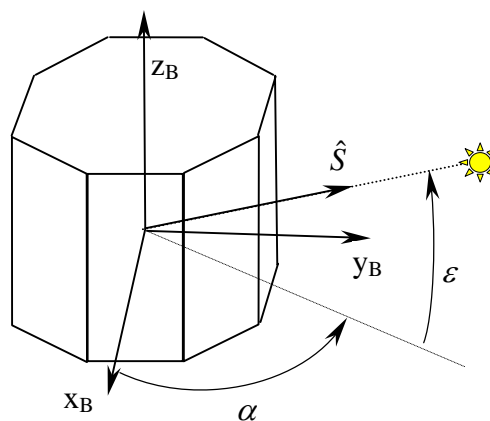


Figure 2. Determining sun's position using solar panels.

The ADCS makes two vector measurements each satellite rotation. These two vector measurements will be processed to find the attitude representation with respect to the inertial frame. Assuming we know the satellite's orbital elements, we can find the attitude representation of the satellite with respect to the orbit. Unfortunately, specific orbital elements will be defined by the primary payload of the launch vehicle, and because this satellite is a secondary payload, LionSat will have essentially the same orbit as the primary payload. During Earth eclipse, solar panel output for estimating the sun's direction will not be available, so the controller will have only one

measurement for the attitude information. We attempted to apply the magnetometer-only method (Natanson *et al.*, 1990) to determine its attitude; however, that method failed to meet the attitude determination requirements for LionSat. Instead, we use the magnetometer measurements to find angular velocities with an extended Kalman filter (EKF), and then integrate the rate to predict the attitude quaternion.

Figure 3 shows a block diagram for the ADCS system. Attitude control will generate commands for the control action, and these commands will be produced with attitude information from either the attitude determination or attitude prediction algorithms. Outputs from the attitude determination, $\hat{\omega}$ and \hat{q} , correspond to the updated angular velocity and quaternions, respectively, with data either from the solar panels, the magnetometer, or from both. The angular velocity and quaternion ($\hat{\omega}^-$ and \hat{q}^-) from the attitude predictions correspond to projection ahead without complete attitude data because either the sun is not available or the magnetometer is off due to magnetic torque rod operations.

Control torque to change orientation will be provided with two independently-operated magnetic torque rods, which are installed parallel and normal to the spin axis. Simple control logic will turn the rods on and off, changing their polarities if necessary.

Since the control hardware and sensors use the magnetic field, once the control hardware activates, no reliable measurements of the geomagnetic field can be made. Therefore, the control logic will use only propagated attitude and angular velocities, then periodically update the attitude as necessary, while keeping the control hardware off.

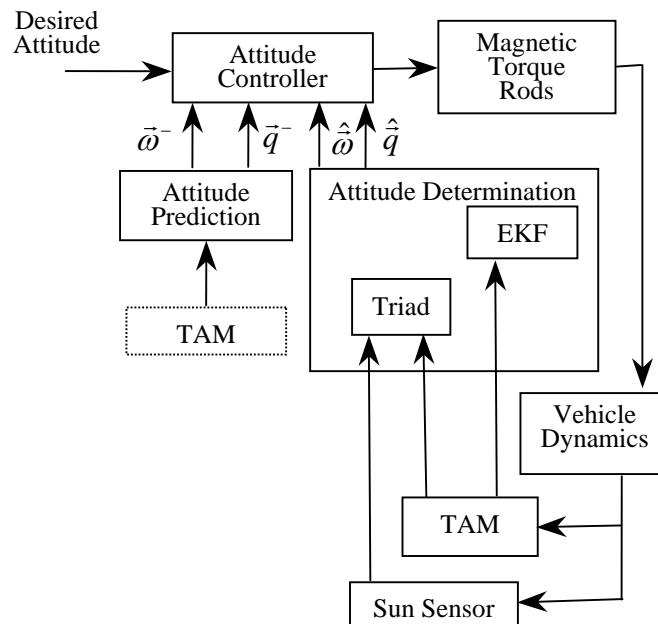


Figure 3. Attitude determination and control.

Basically, considering the spin-stabilized attitude with the axisymmetric configuration, the spin axis will stay fixed in the inertia frame. Anticipated external or internal torque will cause precession of the spin axis, but not cause any secular drift in the inertial frame. Therefore, the major reason the spin axis drifts from the desired orientation (orbit-normal) is the regression of the ascending node of the orbit due to Earth-oblateness. At 400 km altitude and 52 degrees inclination, we have a regression rate of 5 degrees per day. Program requirements dictate (as a worst-case) an initial deployment with the spin axis in the orbital plane; consequently, the ADCS must first reorient the vehicle by 90 degrees (Fig. 4).

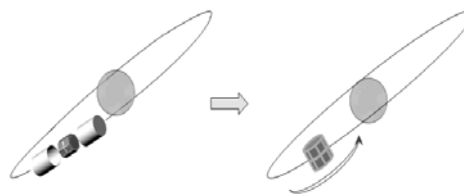


Figure 4. Spin-axis reorientation.

Once the desired attitude is achieved after separation from a launch vehicle, very little control action will be necessary. Therefore, for most of the time in the orbit, sensing with the magnetometer will be possible.

3 Angular velocity measurement

The most crucial task for the successful simulation depends on the accuracy level of the angular velocity measurements which are used to project the quaternions ahead. Therefore, the accuracy level of the measurement of the angular velocity decides the accuracy of the attitude control. Angular measurements are done with the EKF, and to prevent any possibility of divergence of the EKF, it resets itself whenever it starts a new measurement right after the magnetic torque rods are used for attitude control. Simulations indicate the EKF converges to an estimate of the angular velocity within 50 seconds. Measurement errors for angular velocities about the transverse axes (x_B, y_B) fall within 0.05 degrees per second, while the error levels for spin rate (ω_3 about the z_B -axis) are found to vary from 0 to 0.2 degrees per second, with a worst case of 0.5 degrees per second. We use a modified version of the method described by Tortora and Oshman (Tortora and Oshman, 2000) to estimate the angular velocity.

4 Control scheme

Shigehara (Shigehara, 1972), using the angular momentum vector \vec{H} as a control error function, proposed the control logic we are applying to LionSat. It permits spin rate and spin axis reorientation control with two magnetic dipoles. This control law uses a switching function to change the polarity of the dipoles to generate the desired torque. The switching functions are defined as

$$S_{orient} = \vec{E} \cdot (\hat{z}_B \times \vec{B}), \begin{cases} \vec{U} = \beta^2 \hat{z}_B & \text{if } S_{orient} > 0 \\ \vec{U} = -\beta^2 \hat{z}_B & \text{if } S_{orient} < 0 \end{cases} \quad (1)$$

(spin - axis orient. control)

$$S_{spin} = \vec{E} \cdot (\hat{x}_B \times \vec{B}), \begin{cases} \vec{V} = \beta^2 \hat{x}_B & \text{if } S_{spin} > 0 \\ \vec{V} = -\beta^2 \hat{x}_B & \text{if } S_{spin} < 0 \end{cases} \quad (2)$$

(spin - rate control)

where \vec{E} is the angular momentum error (difference between desired and actual), \vec{B} is the geomagnetic field vector, \hat{x}_B and \hat{z}_B are unit vectors aligned with the respective body-fixed axes, β^2 is the maximum magnitude of magnetic moment generated by a torque rod, and \vec{U} and \vec{V} are the magnetic moment vectors along the \hat{z}_B and \hat{x}_B axes, respectively.

To have a nutation damping effect, the generated torque should have the opposite direction to $\vec{\omega}_{12}$ (the component of $\vec{\omega}$ in the x_B - y_B plane). For each nutation ($\vec{\omega}_{12}$ makes one revolution around the spin axis), the magnetic torque $\vec{U} \times \vec{B}$ can be assumed constant in magnitude and ψ is the angular position of $\vec{\omega}_{12}$ in the x_B - y_B plane. Then, in order to act as a nutation damper, $\oint (\vec{U} \times \vec{B}) \cdot \vec{\omega}_{12} d\psi$ must have a negative value, which means that the net torque for each nutation has the opposite direction of $\vec{\omega}_{12}$. But since $\oint \cos\psi d\psi = 0$, it follows that $\oint (\vec{U} \times \vec{B}) \cdot \vec{\omega}_{12} d\psi = 0$, meaning that no damping effect can be expected while generating torque for spin reorientation. Therefore, an additional switching function used mainly for nutation damping is introduced, defined as

$$S_{damp} = \vec{\omega}_{12} \cdot (\hat{z}_B \times \vec{B}), \begin{cases} \vec{U} = -\beta^2 \hat{z}_B & \text{if } S_{damp} \geq 0 \\ \vec{U} = \beta^2 \hat{z}_B & \text{if } S_{damp} < 0 \end{cases} \quad (3)$$

(nutation damping)

This switching function doesn't involve the error vector used in Eqs. (1) and (2), but uses only angular velocity. This is important because nutation damping can be activated at any attitude, even when the error vector cannot be calculated (to determine if the angular momentum vector exactly matches the desired vector). This switching function will be used for both spin-axis reorientation and spin rate change. Once the magnitude of $\vec{\omega}_{12}$ reaches a

predetermined maximum limit, for instance 3 deg/sec, S_{orient} and S_{spin} will be deactivated and the torque rod parallel to \hat{z}_B will be controlled by S_{damp} to make $\oint (\vec{U} \times \vec{B}) \cdot \vec{\omega}_{12} d\psi < 0$. As $|\vec{\omega}_{12}|$ decreases to the lower limit, for example 0.3 deg/sec, the normal attitude correction resumes.

5 Simulation results

Simulation studies of up to 11 orbits indicate that the proposed measurement and control schemes will permit LionSat to meet the scientific requirements in varying degrees. In these simulations, the vehicle is deployed into a 400 km. altitude circular orbit, with inclination of 52 degrees; at the time of deployment, the orbital plane is parallel to the sun vector.

Figure 5 shows the behavior of the vehicle's angular momentum vector during the initial reorientation maneuver (assuming that the satellite is deployed with its spin axis lying in the orbital plane). The controller brings the \vec{H} vector to within 2.5 degrees of orbital-normal in 60000 s. (approximately 11 orbits). During this maneuver, it was found that the controller's damper mode is critical for preventing large excursions of the spin axis from the angular momentum vector. At the end of the reorientation maneuver, the spin axis is within 2 degrees of the \vec{H} vector (and within 4.5 degrees of orbit-normal).

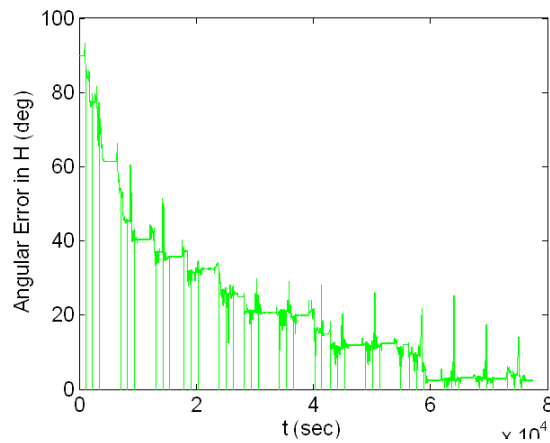


Figure 5. Error in angular momentum during reorientation maneuver (spin rate is 30 deg/s).

After achieving the desired orientation, the vehicle then enters a spin-up maneuver to increase its rate to the required 60 deg/s. Figure 6 indicates that this procedure takes approximately 3000 s. to complete (the clock time is reset to zero after the initial reorientation); the maneuver begins at $t = 1000$ s. in this simulation because the vehicle is in eclipse until $t = 980$ s.

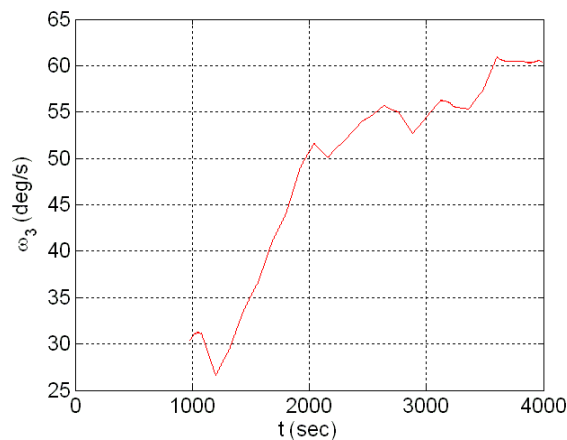


Figure 6. Spin rate during spin-up maneuver.

Again, the damper mode was found to be critical for assuring stability in the spin-axis pointing. It is possible to hold the spin-axis to within 16 deg. of orbit-normal during the spin-up (Fig. 7).

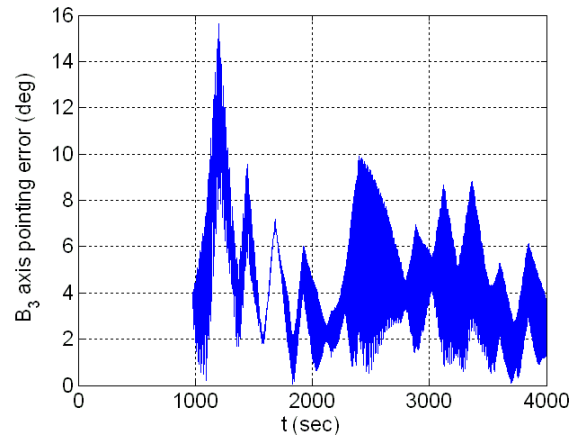


Figure 7. True spin-axis pointing error during spin-up.

As seen in Figs. 8-10, the required magnetic torques to achieve the required final spin rate are all of order 10^{-3} N·m. Note that the intervals of zero torque permit the Kalman filter to reset, using complete information from Triad updates.

Each torque rod generates a magnetic moment of $10 \text{ A}\cdot\text{m}^2$, drawing 1 W of power. This is well within the allotted power budget for attitude control of LionSat, which will generate a total of 20 W (allocated to the plasma probes, their control circuitry, and the onboard processor).

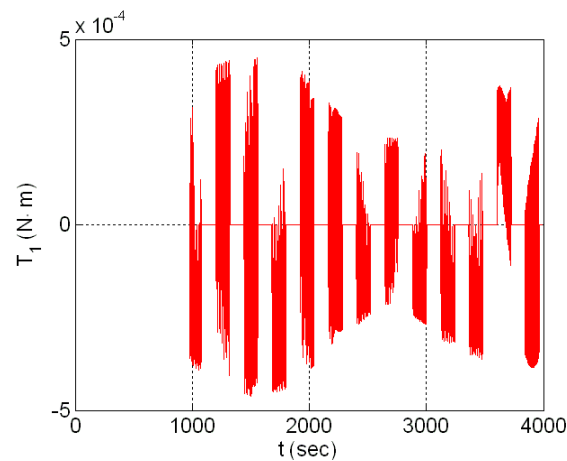


Figure 8. Magnetic torque (x_b -axis) during spin-up.

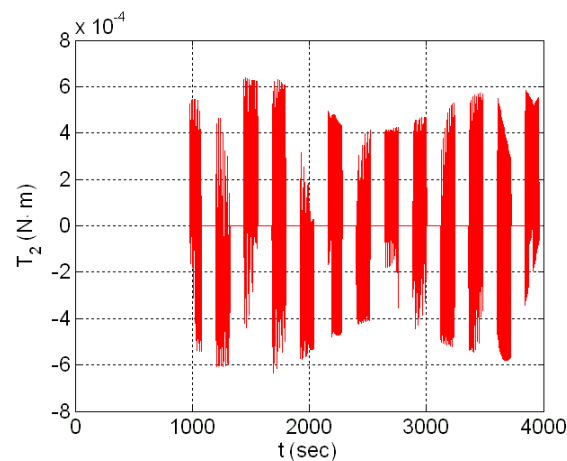


Figure 9. Magnetic torque (y_b -axis) during spin-up.

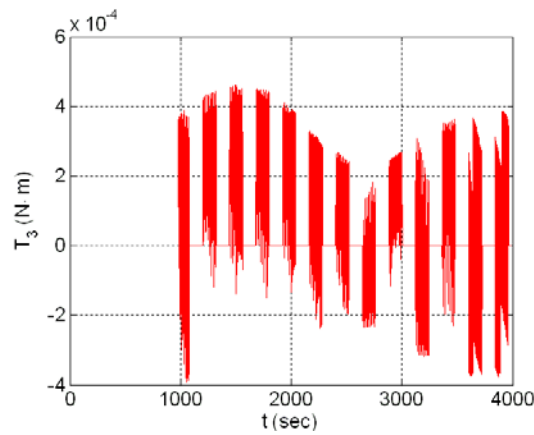


Figure 10. Magnetic torque (z_b -axis) during spin-up.

Having achieved the desired orientation and spin rate, the controller now enters a mode of controlling spin rate, orientation, and nutation. As depicted in Fig. 11, when the satellite is able to use the Triad algorithm to obtain attitude estimates using both sun and magnetic field data, the Kalman filter is capable of providing high-level estimates of the azimuthal position (nominally, the angular position of the x_B -axis in the orbital plane). The estimation error is less than 4 degrees until Triad updates are not available (in this case, at approximately 1680 s., the magnetic torquers are activated and the output from the TAM is halted). Similarly, estimates of the spin-axis orientation (with Triad updates) have errors of order 3 degrees (Fig. 12). The true error in spin-axis orientation is held to within approximately 7 deg. (with Triad updates), shown in Fig. 13.

When Triad updates are unavailable (during intervals of magnetic torque rod operation or during eclipse), the error in estimated azimuthal position grows rapidly, as seen in Fig. 14, while the estimation error in spin-axis orientation is essentially unaffected (Fig. 15). The true spin-axis orientation remains well-controlled even in the absence of Triad updates (Fig.16). This is because the angle between the spin axis and the geomagnetic field, as well as the angle between the spin axis and the angular momentum error vector, remain relatively constant over this short interval of time; the spin-rate control law, Eq. (2), is thus insensitive to small errors produced by the attitude prediction algorithm.

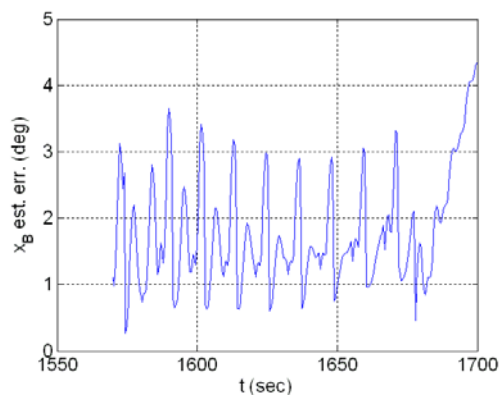


Figure 11. Estimation error in azimuthal position (angular position of the x_b axis).

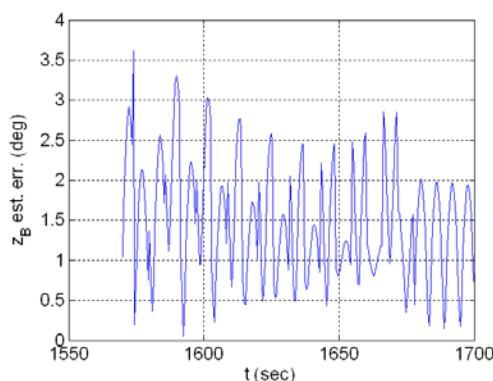


Figure 12. Estimation of error in spin-axis position (angular position of z_b -axis).

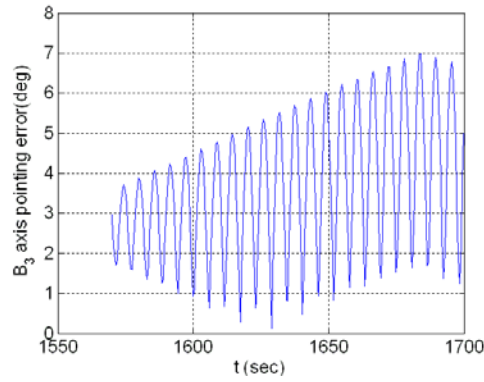


Figure 13. True error in spin-axis position (with triad).

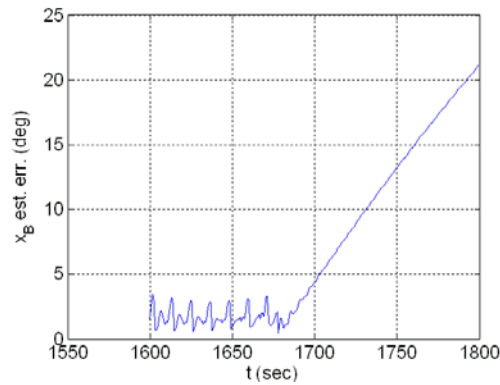


Figure 14. Estimation error in azimuthal position (without triad).

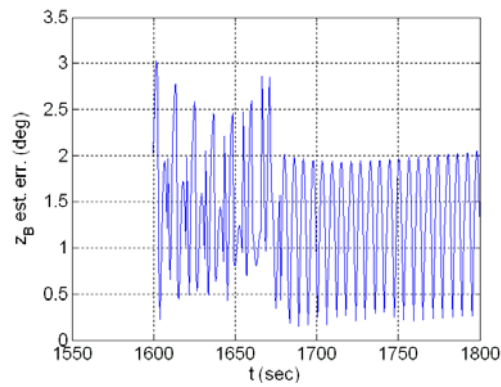


Figure 15. Estimation error in spin-axis position (without triad).

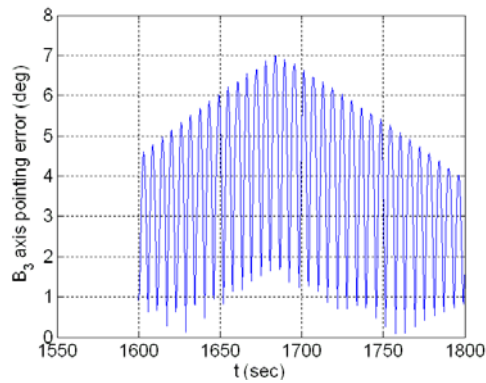


Figure 16. True position of spin-axis without triad updates.

6 Conclusions

Attitude determination and control for a low-budget nanosatellite is possible using a combination of magnetic and solar panel sensors and an extended Kalman filter. In the particular application of LionSat, attitude sensing and control functions must be alternated to avoid magnetic interference. For this mission, the original scientific requirements were 5-degree spin-axis pointing accuracy, and 10-degree knowledge of azimuthal position (to determine the angular location of the plasma probes during data acquisition). While the azimuthal angle knowledge requirement can still be met, a spin-axis control of no better than 10 degrees from orbit-normal is the best possible outcome when the sun angle is being estimated using voltage outputs from the solar panels.

7 Acknowledgment

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8 References

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Appendix

The SAPPHERE satellite also used solar-panel outputs to estimate the azimuth and elevation angles (<http://students.cec.wustl.edu/~sapphire/subsystem/adc/oddss/mathback.html>). The method proposed for LionSat differs in two significant ways: a nonlinear least-squares algorithm is employed, and voltage information from an end cap supplies additional information; consequently, the estimated angles for the sun vector have greater accuracy. The voltage output from each side panel can be modeled as $V_i = V_{0,i} \cos \alpha_i \cos \varepsilon$, where $V_{0,i}$ is the maximum voltage for panel i , α_i and ε are the azimuth relative to panel i and elevation, respectively, of the sun vector \hat{S} , as shown in Figure 2. To estimate the values of these two angles, voltages from side panels $i-1$, i , and $i+1$, along with the voltage from the illuminated end cap (V_e) can be used as follows. (Note that the octagonal geometry of the satellite separates the side panels by 45 degrees in azimuth.)

$$\begin{aligned} V_{i-1} &= V_{0,i-1} \cos(\alpha_i + 45 \text{ deg}) \cos \varepsilon \\ V_i &= V_{0,i} \cos \alpha_i \cos \varepsilon \\ V_{i+1} &= V_{0,i+1} \cos(\alpha_i - 45 \text{ deg}) \cos \varepsilon \\ V_e &= V_{0,e} \sin \varepsilon \end{aligned} \tag{A-1}$$

During operation, an onboard program identifies panel i as the one with the largest voltage output. Equations (A-1) are then solved using a standard nonlinear least squares algorithm that gives the best fit for α_i and ε . Simulations show that this process always converges in fewer than 10 iterations.

With a fixed angle Δ_i between panel i and the body axis x_B , it is then possible to determine the sun vector in terms of the body-axis system as

$$\begin{aligned} \hat{S}_x &= \cos(\alpha_i + \Delta_i) \cos \varepsilon \\ \hat{S}_y &= \sin(\alpha_i + \Delta_i) \cos \varepsilon \\ \hat{S}_z &= \sin \varepsilon \end{aligned} \tag{A-2}$$