Attitude Determination and Control Subsystem for Spartan Lite Spacecraft

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Abstract. This paper describes the Attitude Control Subsystem (ACS) for the Spartan Lite spacecraft bus. The performance requirements, the subsystem configuration, the components, the attitude determination algorithm, and the control law (modified from the Small Explorer spacecraft control law) are presented. Subsystem performance is predicted using the Treetops™ Dynamic Simulator. The ideal performance (obtainable with an ideal attitude sensor) is identified to potential users of the spacecraft bus as an indicator of performance approachable with additional user-provided fine sensors.

Introduction

Spartan Lite is a small, three-axis stabilized, non-retrievable satellite designed to be launched from a Hitchhiker ejection system mounted on the side wall of the Shuttle[1]. The design goal for Spartan Lite is to produce a low-cost fine pointing spacecraft bus capable of supporting celestial, solar, and some nadir/zenith missions, for a lifetime of up to 18 months. A single-string design approach is used, certain commercial off-the-shelf components have been selected, and existing spacecraft designs and software are reused wherever possible. System modularity, a feature of this design, provides the flexibility to accommodate modifications and improvements. To further reduce the cost, gyros are not included in the design. Instead, spacecraft body rates are derived from the attitude solution using the onboard computer. Figure 1 shows a view of the deployed spacecraft. In the stowed configuration, the solar arrays are folded along the sides of the spacecraft. Figure 1 also shows the location of the ACS components and defines the spacecraft body axes.

Requirements

The ACS is designed to accommodate different Spartan Lite missions. The pointing and slewing requirements were derived by polling potential users, and considering system cost limitations. The baseline ACS provides three-axis attitude determination and control for z-axis Sun-pointing. The instrument boresight may or may not be co-aligned with the z-axis depending on the mission. The spacecraft is capable of full sky coverage, with the mission target timeline limited by the user's management of the spacecraft power budget and by the characteristics of the mission specific sensor complement. The baseline ACS requirements are listed in Table I

Table I. ACS Pointing and Slewing Requirements with Baseline ACS Components (3σ)

	Accuracy	Jitter	Slewing	Settling Time
Pitch/Yaw	3 arcmin	30 arcsec	50 deg/min	3 min
Roll	6 deg	90 arcsec	N.A.	N.A.

Pointing performance beyond the baseline is attainable using fine pointing error measurements derived from the user's instrument, or additional (user-provided) sensors, such as a gyro and/or star tracker. Requirements were also set for the pointing performance of the spacecraft bus using an ideal sensor (i.e. three axis information with zero noise, no misalignment and infinite bandwidth). These requirements are listed in Table II.

Table II. ACS Pointing Requirements with Ideal Sensor (3σ)

	Accuracy	Jitter
Pitch/Yaw	10 arcsec	3 arcsec
Roll	10 arcsec	3 arcsec

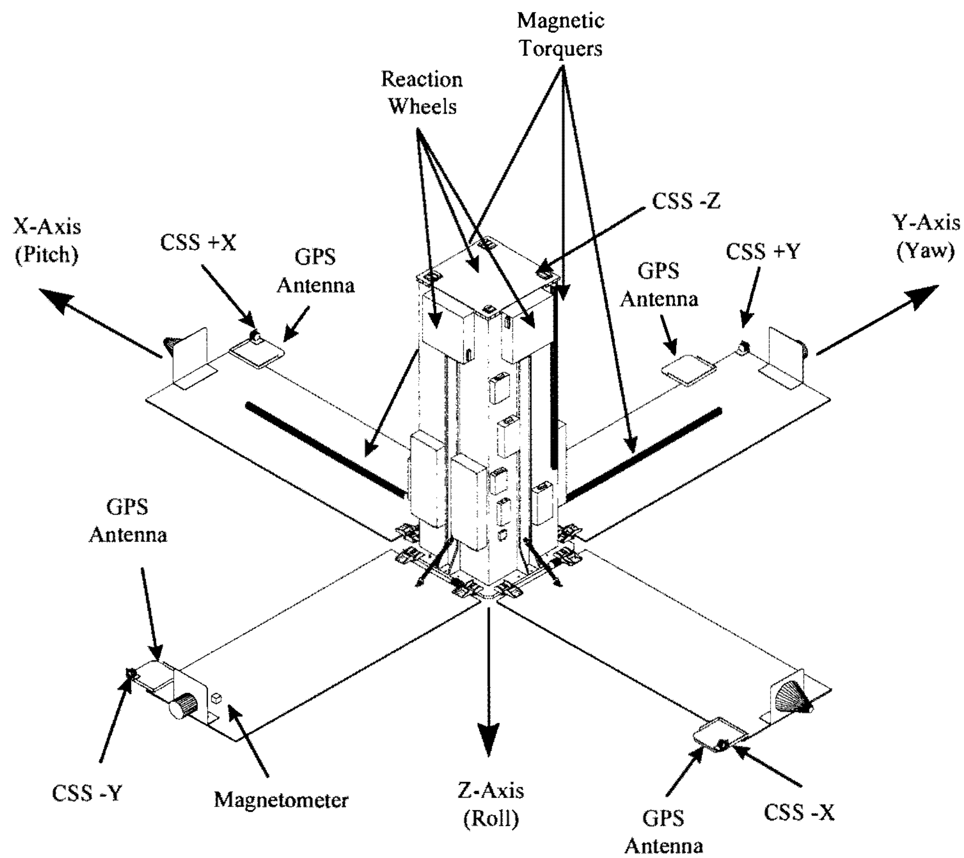

Figure 1. The Spartan Lite Spacecraft Showing Component Locations and Axis Definitions

Accuracy requirements do not take into account attitude sensor to instrument misalignments which typically can be reduced with on-orbit calibration.

It is shown later in this paper that the simulated pointing performance greatly exceeds both the baseline and ideal sensor ACS requirements shown in Tables I and II.

Components/Hardware

A block diagram of the ACS is shown in Figure 2. The hardware consists of portions of the spacecraft Central Unique Electronics (CUE), a sensor complement, and an actuator complement. Component selection was based on requirements, availability, heritage, system compatibility, and, primarily, total system cost.

Three of the cards housed in the CUE perform ACS related functions. These cards are the processor card, the housekeeping card, and the magnetic torquer driver card. All ACS computations are carried out by the processor card. The CUE processor card is the main spacecraft computer and is responsible for all sub-system software including command and data handling, power, instrument processing, and ACS. The CUE processor is a low cost, commercial-off-the-shelf single board computer featuring a 75 to 200 MHz Pentium CPU, 8 to 48 Mbytes DRAM, 2 to 4 Mbytes flash memory, two RS232 serial ports, a real-time clock, and a CompactPCI™ backplane. The CompactPCI™ backplane forms the primary backplane for the CUE box. The baseline operating system for the CUE processor is Vxworks™ and the code is written in C++, C and some assembly. The CUE processor card interfaces to the housekeeping card via the CompactPCI™ backplane. The housekeeping card is a custom built, in-house design based on previous NASA-GSFC designs that performs most of the analog and digital I/O for the spacecraft including performing analog to digital conversions for the ACS sensors. The magnetic torquer driver card is built in-house and utilizes designs flown on previous NASA-GSFC missions.

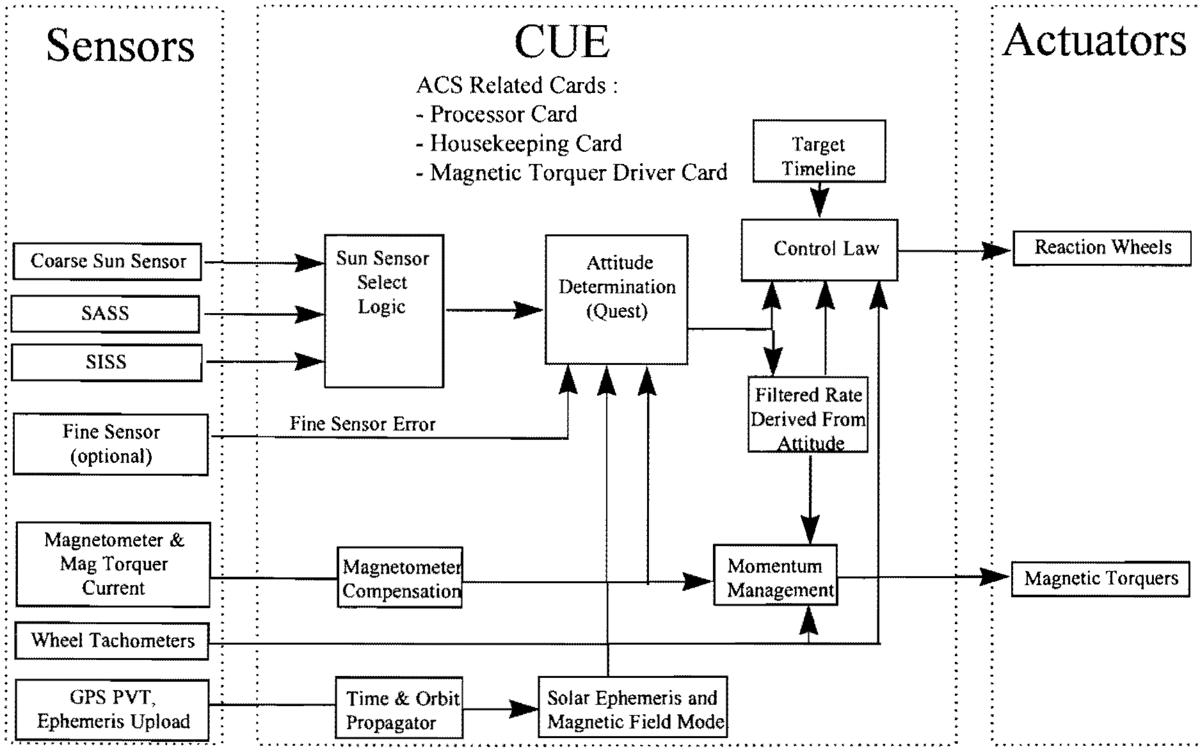

Figure 2. Spartan Lite ACS Block Diagram

The baseline sensor complement consists of a three-axis magnetometer (TAM) and Sun sensors. The TAM is a miniaturized, fluxgate, analog magnetic field sensor made at NASA-GSFC. The Sun sensors include : 1) Six coarse Sun sensors (CSS) configured to provide 4π steradian Sun vector information. 2) One moderate accuracy sensor (the Spartan Acquisition Sun Sensor or SASS), used as a staging sensor to overcome potential acquisition disruption from the effects of the Earth's albedo. 3) One fine sensor (the Spartan Intermediate Sun Sensor or SISS). Characteristics of these sensors are shown in Table III. The SISS performance was obtained from actual flight data from Spartan 201.

Table III. Sun Sensor Characteristics

Sensor	Number	Axes	Field of View (deg)	Accuracy	1σ Noise (arcsec)
CSS	6	2	± 85 each	2 deg*	~100
SASS	1	2	± 35	0.5 deg	~40
SISS	1	2	± 12	10 arcsec	0.5

* Assumes no Earth albedo effects.

A Navstar global positioning system (GPS) receiver is used for position, velocity, and time information and will be capable of providing three-axis attitude and rate information by the time the spacecraft is integrated. Since GPS attitude determination is based on interferometry, the GPS antennas are configured as far apart from each other and as orthogonally as possible. To mininize multipath, each GPS antenna has a 7 inch ground plane and an unobstructed 2π steradian field of view. The GPS receiver is a low cost, space qualified, light weight unit designed and built at NASA-GSFC.

The actuator complement consists of three reaction wheels and three magnetic torquers. The wheels are aligned with the spacecraft body axes. Each wheel has a maximum torque output of 0.14 $\text{N}\cdot\text{m}$, and a maximum momentum capacity of 8.6 $\text{N}\cdot\text{m}\cdot\text{sec}$. The wheels are designed and fabricated at NASA-GSFC. The magnetic torquers, also aligned with the spacecraft body axes, have redundant coils and provide a linear dipole moment of up to 110 $\text{amp}\cdot\text{m}^2$.

The payload instrument boresight is aligned in the +Z direction. Therefore, for solar pointing missions the SASS and SISS will be co-aligned with the instrument and the solar cells will be mounted on the +Z face of the solar panels. For stellar pointing missions, the SASS will point in the -Z direction and the solar cells will be mounted on the -Z face of the solar panels.

Table IV. Operational Modes

MODE	SENSORS	ACTUATORS	FEATURES
Digital Safehold	CSS SASS TAM	Magnetic Torquers Reaction Wheels	- No Ephemeris - Sun Pointing Based on Attitude Error from TAM and Sun Sensors - Momentum Management - Momentum Bias
Inertial Sun Point	CSS SASS SISS TAM GPS Fine Sensor	Magnetic Torquers Reaction Wheels	- Attitude Determination Based on Ephemerides and QUEST - Targeting Based on Computed Sun Position - Momentum Management - Momentum Bias
Science Point	CSS SASS SISS TAM GPS Fine Sensor	Magnetic Torquers Reaction Wheels	- Attitude Determination Based on Ephemerides and QUEST - Targeting Based on User-Uploaded Timeline - Momentum Management - Momentum Bias
Magnetic Calibration	TAM	Magnetic Torquers	N.A.

Operational Modes

The spacecraft has four modes of operation: Digital Safehold, Inertial Sun Point, Science Point and Magnetic Calibration. Table IV shows the sensors and actuators used in each mode, and basic ACS design features.

Digital Safehold is a non-inertial Sun-pointing mode that uses CSS, SASS and TAM measurements to coarsely orient (~5°) the solar arrays towards the Sun and dampen the spacecraft body rates.

The Inertial Sun Point mode uses an inertial attitude solution, obtained by appropriately filtering the CSS, SASS, SISS and TAM measurements, and the momentum based control law to perform the same function described above, but within the accuracy specified in Table I.

The Science Point mode uses the same attitude determination and control algorithms as the Inertial Sun Point mode to point the payload toward targets uploaded as quaternion timelines by the user. Targets can be either solar, stellar, or, if gyros are added, non-trackable objects.

The Magnetic Calibration mode is used to determine the magnetic field contamination produced by the magnetic torquers. The known field contamination is then removed from the magnetometer measurements.

Attitude Determination

In Inertial Sun Point and Science Point modes the attitude determination algorithm produces a quaternion estimate of the spacecraft three-axis orientation in the Earth-Centered Inertial (ECI) frame. Unless such an algorithm is incorporated into a user-provided three-axis attitude sensor, the QUEST (Quaternion Estimation) algorithm[2], resident in the onboard computer, is used. Similar to the TRIAD[3] method, QUEST requires two reference frame vectors and two body frame (or measurement) vectors. One of the advantages of QUEST over TRIAD is that it allows for optimal weighting of sensor information.

For the Inertial Sun Point mode, and for the Science Point mode where the target is the Sun, the reference vectors are the Sun and the earth's magnetic field vectors in the ECI frame. They are determined from the current spacecraft position (produced by the GPS receiver or the onboard orbit propagator), and from the onboard geomagnetic field (8^{th} order IGRF) and Sun models.

The measurement vectors are provided by the Sun sensors and the TAM. The Sun sensor signals are pre-processed so that the signal from the highest accuracy sensor available is always used. The SASS signal is available when the Sun appears in its field of view and the SISS signal is only used when the Sun is within the SISS's proportional range (~3°). This results in a smoother hand-over from the SASS during acquisition and improves the transient ACS performance. During eclipse, when the Sun reference is missing, the QUEST algorithm is disabled.

Spacecraft attitude is coarsely maintained by passive control. Passive control results from the angular momentum bias about the Sun pointing roll axis. This reduces the pitch and yaw drift due to disturbance torques. Wheel speed during eclipse is maintained equal to the last daylight tachometer measurement. The Sun must be re-acquired after orbit sunrise.

Due to the higher accuracy of the SISS, which is the nominal Sun sensor during Sun pointing (after acquisition/re-acquisition transients), the Sun vector measurements are more heavily weighted than the relatively noisy and less accurate TAM measurements. A weight ratio of 100/1 was found optimum by means of ACS performance simulation.

It is noted that, in when using QUEST for Sun pointing, the three-axis attitude reference becomes underdetermined when the Sun vector aligns closely with the Earth's magnetic vector. Although this condition is temporary, it generally occurs every orbit and produces large roll attitude errors. This produces pitch and yaw errors through wheel dynamic coupling. Without reducing the transverse wheel biases (useful in preventing zero-speed crossings) this effect can be reduced by appropriate control law design, as discussed in the following section.

Spacecraft body rates are continuously computed from the attitude solution. This is done by determining the Euler axis and angle for the motion between each sampling step from the difference in the current and previous estimated quaternions. The Euler angle is then differentiated to give the rate about the Euler axis. The calculated rates are low-pass filtered (1 Hz cut-off frequency) before being used in the control law algorithm.

For reliability and simplicity, QUEST is not used in Digital Safehold. While in sunlight, pitch and yaw Sun pointing errors are obtained directly from the SASS, and the corresponding body rates are derived from these by differentiation. Roll rates are damped with a B-dot control law. The passive control strategy used in Inertial Sun Point is adopted in eclipse.

Attitude Control

Digital Safehold uses a PD (Proportional-Derivative) controller to provide spacecraft stabilization and a minimum pointing accuracy of ~5° to the Sun, in order to guarantee survival power to the spacecraft. Active control during this mode occurs only during the daylight portion of the orbit.

Inertial Sun Point/Science Point Control

The Inertial Sun Point and Science Point modes use a modified version of the momentum based control law designed for the Small Explorer (SMEX) series of spacecraft[4]. The original version was designed to support a 4-wheel system. The general features of the original control law, modified for a system with 3 wheels, are described here. The control law operates at 10 Hz and consists of two parts integrated into a single algorithm. The first is used for slewing control and the second is used for fine pointing. In both cases, the momentum based control law continuously calculates the desired angular momentum for the reaction wheels. During slewing the desired angular momentum vector is determined according to the equation (1) :

$$\vec{h}_{command} = \vec{H}_{system} - \sqrt{2 \cdot a_{command} \cdot \Phi} \cdot (I \cdot \hat{e}) \quad (1)$$

where $\vec{h}_{command}$ is the desired wheel momentum, $\vec{H}_{system}$ is the total system momentum, $a_{command}$ is the maximum desired angular acceleration, Φ is the Euler rotation between the current and final orientations (later referred to as the angle-to-go), I is the spacecraft moment of inertia tensor, and $\hat{e}$ is the unit vector along the Euler rotation axis. The term $\sqrt{2 \cdot a_{command} \cdot \Phi}$ produces angular rate commands. At the beginning of a maneuver the spacecraft accelerates about the Euler axis until the commanded rate is achieved. As the angle-to-go approaches zero the commanded rate also approaches zero causing the spacecraft to decelerate and stop on target. Fine pointing control uses a conventional Proportional-Integral-Derivative (PID) controller. The desired angular momentum vector is determined according to equation (2) :

$$\begin{aligned}\vec{h}_{command} = \vec{H}_{system} &- \vec{K}_h \cdot \Phi \cdot (I \cdot \hat{e}) \\ &- \vec{K}_i \int \Phi \cdot (I \cdot \hat{e}) \cdot dt\end{aligned} \quad (2)$$

where $\vec{K}_h$ and $\vec{K}_i$ are the derivative and integral gain vectors, respectively.

During slewing, the desired wheel torques are derived according to equation (3) :

$$\vec{T}_{wheels} = \vec{K}_t \cdot (\vec{h}_{command} - \vec{h}_{wheels}) - \vec{\omega}_{body} \times \vec{H}_{system} \quad (3)$$

where $\vec{T}_{wheels}$ is the commanded torque, $\vec{K}_t$ is the proportional gain vector, $\vec{h}_{wheels}$ is the measured wheel momentum (obtained from tachometer measurements), and $\vec{\omega}_{body}$ is the derived body rate. The second term is a feedforward torque that accounts for the motion of the body frame. (During fine pointing, when the body rates are low, this term is neglected.) The maximum wheel torque at any given speed is constrained by current and power limits. The maximum is calculated at each control cycle and if the commanded wheel torque exceeds it then the calculated maximum is used.

The transition from slewing to fine pointing occurs when the Euler angle is less than a preset threshold. This is chosen so that the wheel torque commands before and after the transition do not change drastically and cause unwanted structural vibrations. The threshold for transition from fine pointing to slewing is set higher than that for the reverse transition. This provides hysteresis that reduces the risk of chattering back and forth between the two modes.

The criteria for the selection of the PID gains take into account performance requirements and environmental disturbances. Since the requirements for instrument boresight pointing are very stringent while the control about the boresight is rather lax, the selected control loop bandwidths for the baseline sensor complement are respectively 0.05 Hz for pitch and yaw, and 0.005 Hz for roll. The lower roll bandwidth attains two goals: 1) It limits pointing degradation due to attitude reference breakdown during times of Sun-vector and B-vector co-alignment. 2) It reduces the effects of TAM noise on pointing jitter. The bandwidth can be varied depending on mission requirements and user provided sensor performance. For the ideal sensor simulations, the bandwith was set at 0.2 Hz for all three axes. Linear stability analyses were performed for bandwidths ranging from 0.005 Hz to 0.20 Hz. The lowest gain and phase margins were 13 db and 57°, respectively.

Momentum Management

The purpose of momentum management is to keep disturbance torques from saturating the wheels and to maintain wheel momentum biases so that the wheels do not pass through zero speed during fine pointing. At zero or near-zero speed, knowledge of wheel torque and momentum is very poor resulting in spacecraft pointing errors. Momentum management is implemented by magnetic torquers. The Earth's magnetic field vector, measured by the magnetometer, and the desired wheel biases are used to calculate dipole commands for the magnetic torquers. The torques produced by the interaction of each dipole with the geomagnetic field unload the excess wheel momenta.

Performance Simulation

The performance of the closed-loop system was simulated using Treetops™ (a general purpose dynamic simulator program). High fidelity models of the sensors and actuators, the attitude determination and control algorithms, and models of the orbital environment were programmed into Treetops™. The major simulation parameters are shown in Table V. Simulated performance is presented for two cases. One is Inertial Sun Point using the baseline sensor complement, the other is performance with an ideal sensor. The latter is intended to provide potential users with performance limits that may be approached through the use of highly accurate sensors.

Table V. Treetops™ Simulation Parameters.

Spacecraft Moment of Inertia Tensor		$I = \begin{bmatrix} 32 & 0 & 0 \\ 0 & 32 & 0 \\ 0 & 0 & 23 \end{bmatrix} (\mathrm{kg \cdot m^2})$	
Orbit (circular)		Altitude : Inclination :	300 km 28 degrees
Sensors	SISS	Misalignment/Bias : Noise : Quantization :	0.01 degrees 0.5 arcsec 7 arcsec
	TAM	Misalignment/Bias : Noise : Quantization :	0.1 degrees 0.7 mGauss 0.34 mGauss
Actuators	Wheels	Quantization : Bandwidth :	1.4E-4 Nm 4 Hz
	Magnetic Torquers	Quantization : Bandwidth :	2.9E-2 $\mathrm{amp \cdot m^2}$ 0.5 Hz

Inertial Sun Point Mode Performance

The results of this simulation are shown in Figures 3 through 5.

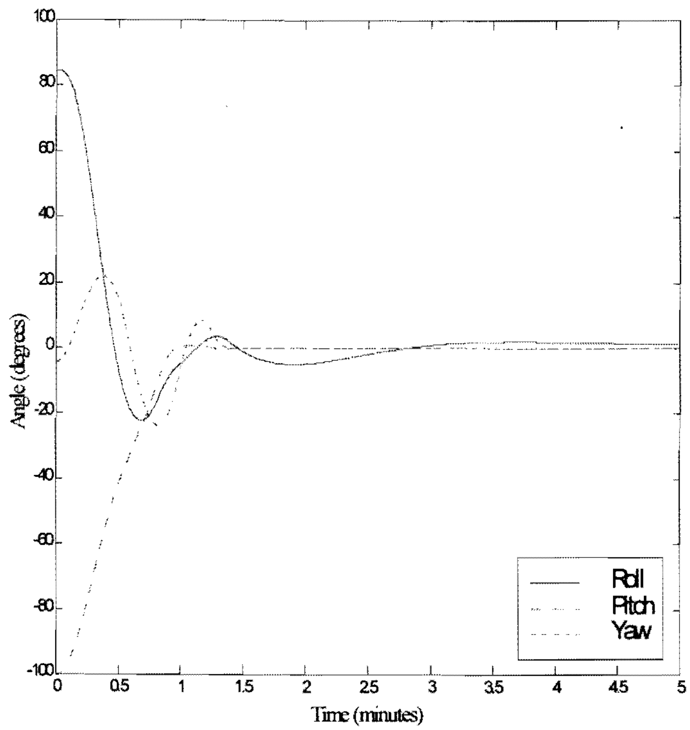

Figure 3. Sun Acquisition in Inertial Sun Point Mode

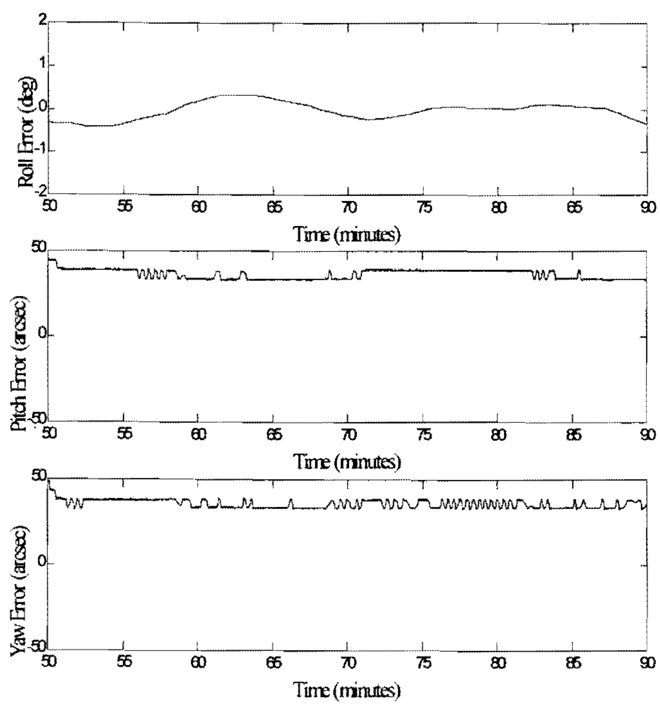

Figure 4. Fine Pointing Performance in Inertial Sun Point Mode

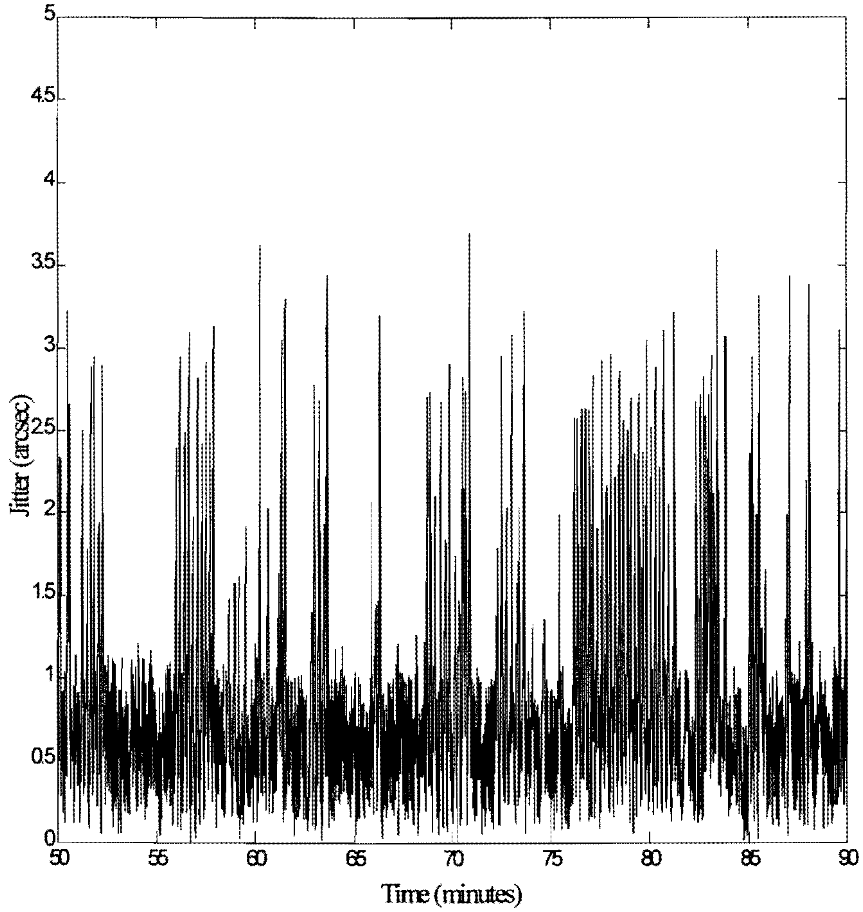

Figure 5. RSS of the Steady State Jitter in Inertial Sun Point Mode

Figure 3 shows a simulated large angle acquisition. The settling time is less then 3 minutes for the pitch and yaw axes, but longer for the roll axis due to its lower bandwidth. Figure 4 shows the typical steady state pointing performance. The accuracy about the pitch and yaw axes is better than 50 arcseconds. The average error is due to sensor bias and misalignment, the environmental disturbances, and the selected loop parameters (bandwidth and integral compensation). The ripple and jitter effects are due to sensor signal and wheel torque quantization, and to component noise. Jitter due to the TAM has been minimized by lowering the roll loop bandwidth. The lower bandwidth results in an accuracy on the order of 1° about the roll axis (excluding field model errors). Misalignment effects can be reduced with on-orbit calibration, after which the accuracy about the pitch and yaw axes could be under 8 arcseconds. Figure 5 shows the root-sum-square (RSS) of the jitter about the pitch and yaw axes. The peak (<4 arcsec) is significantly smaller than the specified requirements.

Performance with an Ideal Sensor

The same simulations were repeated using an ideal attitude sensor. The true attitude quaternion and the true spacecraft body rates were used to simulate the output of ideal sensors. The results, shown if Figures 6 and 7, represent the best possible pointing performance that the system can achieve, given the level of environmental and internal disturbances (noise, quantization, residual dynamic imbalance, tachometer errors and misalignment in the reaction wheels and magnetic torquers). The figures indicate a pointing performance limit of <0.2 arcsec and jitter of <0.1 arcsec (per-axis).

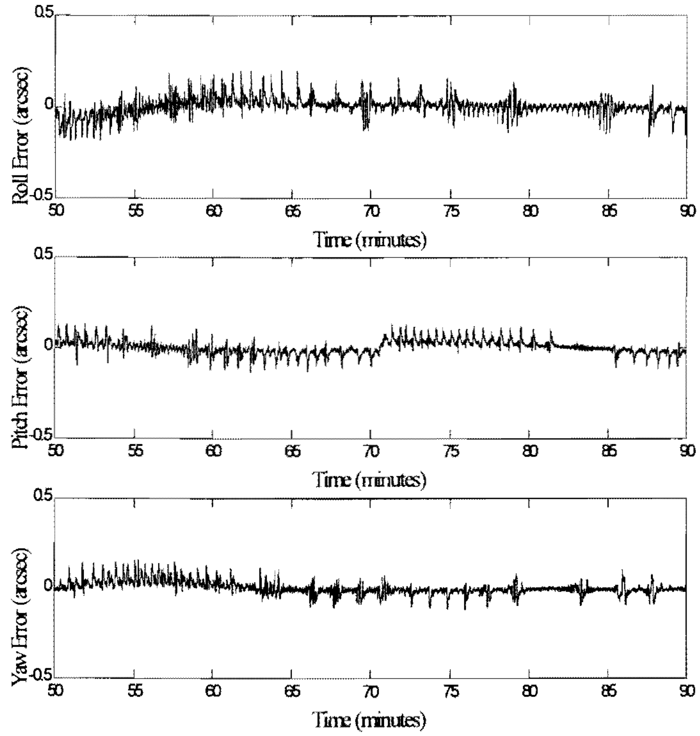

Figure 6. Steady State Performance Using Ideal Sensor

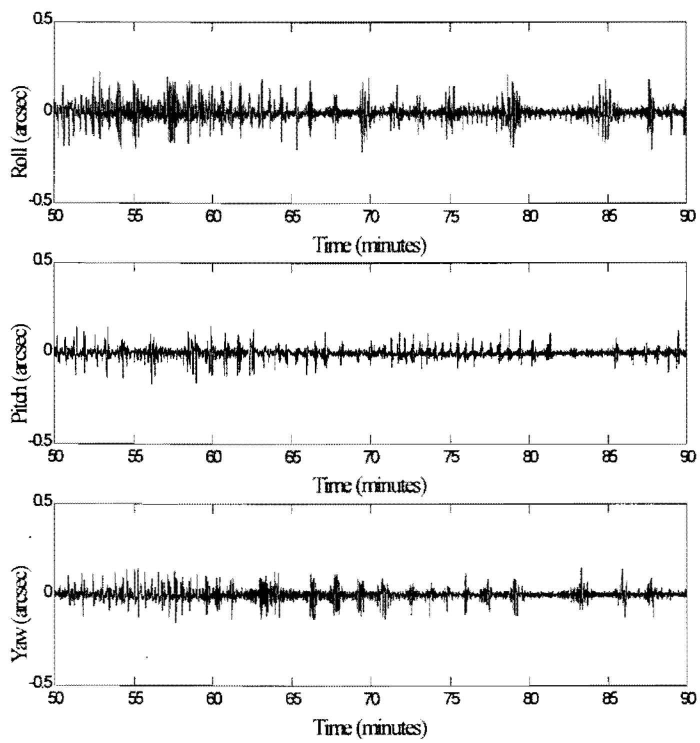

Figure 9. Steady State Jitter Using Ideal Sensor

Summary

The Spartan Lite attitude control subsystem is modularly designed to accommodate multiple Spartan Lite missions. The reuse of previous NASA-GSFC designs from the Small Explorer and Spartan programs and the use of a simple system architecture have minimized the total subsystem cost. The control laws are implemented in the form of a comprehensive algorithm that provides optimal slewing for large-angle maneuvers, and PID control for steady pointing. System performance, significantly exceeding the specified requirements, has been optimized by proper selection of loop parameters (bandwidth and integral compensation) and has been verified with the Treetops™ dynamic simulator.

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Attitude Determination and Control Subsystem for Spartan Lite Spacecraft

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Jim Morrissey received a B.S. in Chemistry from McGill University in 1990 and, after taking prerequisite courses in Physics and Mechanical Engineering at the University of Massachusetts, received a M.S. in Astronautics from The George Washington University in 1995. He is currently enrolled at the University of Maryland and is working towards a doctorate in Spacecraft Control Systems while employed as a co-op student in the Attitude Control and Stabilization Branch of the Special Payloads Division at the NASA Goddard Space Flight Center. In the past three years he has worked primarily on the Spartan series of satellites including Spartan 201-3, Spartan 204, Spartan 206 and Spartan 207.

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Tom Quinn received a B.S. in Electronics Engineering from Penn State University in 1990. He is currently employed as an Attitude Control Systems engineer in the Attitude Control and Stabilization Branch of the Special Payloads Division at the NASA Goddard Space Flight Center. Since starting at Goddard in the summer of 1990, he worked on the Small Explorer (SMEX) SAMPEX, FAST, SWAS, TRACE, WIRE and SMEX Lite series satellites of which SAMPEX and FAST are currently in orbit and operating successfully. He has also worked and is working on other projects including the Cosmic Ray Upset Experiment (CRUX), a miniaturized three axis magnetometer (TAM), Spartan 400, Spartan Lite, and the Gas and Aerosol Monitoring Sensorcraft (GAMS).

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Giacomo Porcelli received a doctoral degree in Industrial Engineering from the University of Naples (Italy) in 1957, and a Ph.D. degree in Electrical Engineering from the University of Pennsylvania in 1963. Throughout his professional life, he has been involved first with industrial automatic controls, later with aerospace controls (attitude, orbit), trajectory analysis/synthesis, mission analysis, to which he has devoted most of his career. His several employers include General Electric (Space Vehicle Div.), Westinghouse Astronuclear Div., the Fairchild Space Co., the International Telecommunication Satellite Organization (INTELSAT), where he managed the Spacecraft R&D section, and the Orbital Sciences Corp. He has worked on spacecraft programs such as Mariner Mars '71, ATS-6, INTELSAT 5 and 6. His hobby is classical music (plays piano).