ATTITUDE CONTROL SYSTEM DESIGN FOR THE SOLAR DYNAMICS OBSERVATORY

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ABSTRACT

The Solar Dynamics Observatory mission, part of the Living With a Star program, will place a geosynchronous satellite in orbit to observe the Sun and relay data to a dedicated ground station at all times. SDO remains Sun-pointing throughout most of its mission for the instruments to take measurements of the Sun. The SDO attitude control system is a single-fault tolerant design. Its fully redundant attitude sensor complement includes 16 coarse Sun sensors, a digital Sun sensor, 3 two-axis inertial reference units, 2 star trackers, and 4 guide telescopes. Attitude actuation is performed using 4 reaction wheels and 8 thrusters, and a single main engine nominally provides velocity-change thrust. The attitude control software has five nominal control modes—3 wheel-based modes and 2 thruster-based modes. A wheel-based Safehold running in the attitude control electronics box improves the robustness of the system as a whole. All six modes are designed on the same basic proportional-integral-derivative attitude error structure, with more robust modes setting their integral gains to zero. The paper details the mode designs and their uses.

THE SDO MISSION

The Solar Dynamics Observatory mission is the first Space Weather Research Network mission, part of NASA's Living With a Star program, which seeks to understand the changing Sun and its effects on the Solar System, life, and society. To this end, the SDO spacecraft will carry three Sun-observing instruments to geosynchronous orbit: Helioseismic and Magnetic Imager (HMI), led by Stanford University; Atmospheric Imaging Assembly (AIA), led by Lockheed Martin Space and Astrophysics Laboratory; and Extreme Ultraviolet Variability Experiment (EVE), led by the University of Colorado. Links describing the instruments in detail may be found through the SDO web site.[1]

The basic mission is to observe the Sun for a very high percentage of the 5-year mission (10-year goal) with long stretches of uninterrupted observations and with constant, high-data-rate transmission to a dedicated ground station to be located in White Sands, New Mexico. These goals guided the design of the spacecraft bus that will carry and service the three-instrument payload. The spacecraft is due to be launched in April 2008, and it has been designed and will be integrated at the NASA Goddard Space Flight Center (GSFC). Overarching design goals for the bus are geosynchronous orbit, near-constant Sun observations with the ability to fly through eclipses, and constant high-gain antenna (HGA) contact with the dedicated ground station. A three-axis stabilized attitude control system (ACS) is needed both to point at the Sun accurately and to keep the roll about the Sun vector correctly positioned with respect to the solar north pole, as provided by Carrington elements.[2] This roll control is especially important for the magnetic field imaging of HMI.

ACS DESIGN OVERVIEW

The mission requirements have several general impacts on the ACS design. Both the AIA and HMI instruments are very sensitive to the blurring caused by jitter. Each has an image stabilization system (ISS) with some ability to filter out high frequency motion, but below the bandwidth of the ISS the control system must compensate for disturbances within the ACS bandwidth or avoid exciting jitter at higher frequencies.

Within the ACS bandwidth, the control requirement imposed by AIA is to place the center of the solar disk no more than 2 arcsec, 3σ, from a body-defined target based on the four guide telescopes (GT) that accompany the instrument. This body-defined target, called the science reference boresight (SRB), is determined from the post-launch orientation of the GTs by averaging the bounding telescope boresights for pitch to get a pitch SRB coordinate, and by averaging the bounding boresights for yaw to get the yaw SRB coordinate. The location of this

SRB in the 0.5-deg field-of-view for each GT then becomes the central target for each telescope; one GT is selected for use as the ACS controlling guide telescope (CGT) at any given time. To simplify the ACS Science mode design, fine Sun-pointing is effected based on this SRB for all three instruments, not just AIA, when the Sun is within the linear range of the CGT.

In addition to limiting jitter, HMI science requires averaging several observations, making the instrument sensitive to low frequency motion that induces differential motion between each observation. This requires the spacecraft attitude to be stable about the roll axis to approximately 10 arcsec over a ten-minute period.

Instrument calibrations require that the spacecraft point the SRB up to 2.5 degrees in pitch and yaw away from the center of the Sun, placing the Sun outside the field-of-view of the guide telescopes. In such instances, when the GTs cannot provide the definitive target for the ACS, on-board attitude determination combined with ephemeris prediction of the Sun direction must provide the definitive target.

EVE is capable of observing the Sun with less dependence on attitude control. However, the ground data processing needs for calibrations result in the most strict attitude knowledge requirements for the mission: [35, 70, 70] arcsec, 3σ, of knowledge with respect to the center of the solar disk. In addition to driving the ACS sensor selection, the knowledge requirements, which have their effect primarily during Inertial mode calibrations, drive the accuracy requirements for the solar ephemeris. See [3] for more information on SDO instrument calibrations from an ACS perspective. Only the guide telescopes are capable of providing arcsecond-level accuracy for the Sun center, and the Sun would not be within the fields-of-view of the GTs during the off-pointing calibrations required by EVE.

The need to achieve and maintain geosynchronous orbit (GEO) drives the need for high-efficiency propulsive systems and appropriate attitude control. The main engine provides high specific impulse for the maneuvers to attain GEO, while smaller ACS thrusters manage the potential disturbance torques of that larger engine and provide capability for much smaller adjustment burns once on orbit. A large solar profile means that solar radiation pressure is a potentially large torque disturbance, such that magnetic torquers at GEO altitude would be incapable of managing the angular momentum of the system. The momentum buildup drives the ACS to use thrusters to manage vehicle momentum. The demanding data capture budget for the mission, however, requires SDO to avoid frequent thruster maneuvers, which interrupt observations. These factors combine to require thruster-based momentum and orbit management burns every four weeks at the most frequent.

HARDWARE COMPLEMENT

The SDO ACS hardware complement is single-fault tolerant; Figures 1 and 2 highlight the locations and orientations of the ACS components. Two main processors carry virtually identical copies of the command and data handling and ACS software, and two identical attitude control electronics (ACE) boxes carry Coldfire processors with contingency ACS software and other hardware interface cards; the ACE structure allows reaction wheels to be commanded by the Sun-pointing Safehold independent of the Mil Std 1553 data bus.

The ACS sensor complement includes 16 coarse Sun sensors (CSS), 3 two-axis inertial reference units (IRU), 2 star trackers (ST) and 1 digital Sun sensor (DSS). The ST and DSS combine to provide two-out-of-three single-fault tolerance fine attitude determination. Any one of the 4 AIA guide telescopes may be selected as the ACS CGT. Control is actuated using reaction wheel assemblies (RWA) and attitude control thrusters. Orbit change maneuvers can be accomplished using either these thrusters or a main engine; the main engine will be used nominally for all long maneuvers performed in achieving geosynchronous orbit from the launch orbit.

Sixteen Adcole CSSs are grouped into primary and backup sets of eight sensors, each set providing the ability to calculate a sun vector. The primary set interfaces to one ACE and the backup set interfaces to the other. Each CSS outputs a current proportional to the cosine of the sun angle relative to the boresight. The CSSs are located at the outermost corners of the two solar panels. They are directed in an isometric pattern, parallel to the diagonals of a cube aligned with the spacecraft axes; with each eye covering a 170-degree cone field of view, each set of eight eyes provides full 4π-steradian coverage.
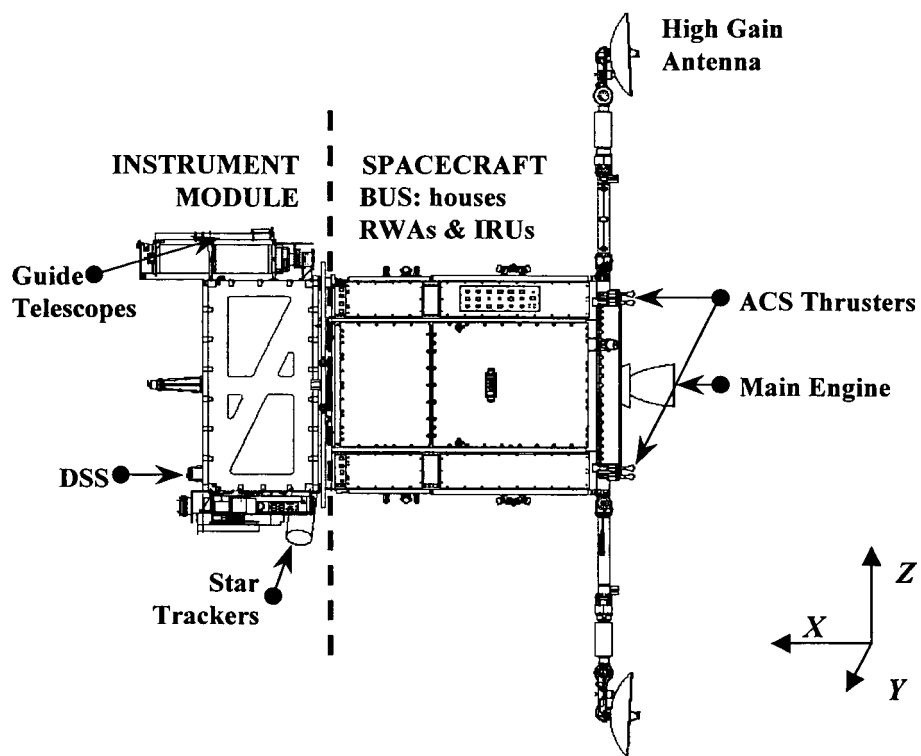
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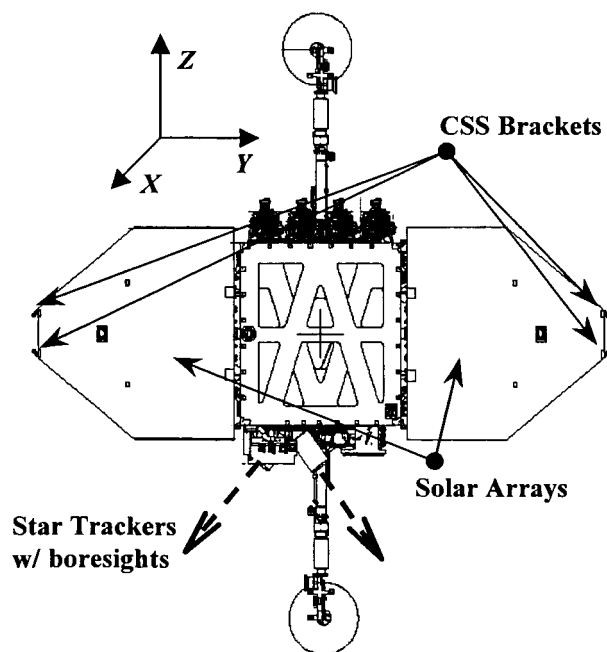**Figure 1: Solar Dynamics Observatory side view**

Figure 2: SDO Solar Array View

The Adcole DSS comprises an optics head and a separate electronics box providing a 1553 data interface. The electronics box is mounted inside the Faraday cage created by the spacecraft bus module. The DSS head with its 32-deg square FOV is mounted on the instrument module with its boresight along the spacecraft X axis, nearly aligned with the Sun during observations. Adcole has designed the DSS calibration parameters so that the accuracy is 0.24 arcminutes within 10 deg of the boresight, and diminishes to 3 arcminutes as the Sun moves towards the edges of its FOV. This DSS calibration scheme provides higher accuracy attitude determination over the range of the instrument calibration maneuvers.

The IRUs are TARA-1T units provided by Kearfott, each sensing rate in two orthogonal directions with an accuracy of 0.5 arcsec/bit. While the accuracy does degrade at very high rates, the sensor maintains the correct rate polarity up to 10 deg/sec, well above the highest expected vehicle rates of 2 deg/sec. With three units, the rate sensing capability of the IRUs is single-fault tolerant, and the failure detection and correction (FDC) system is capable of reconfiguring rate selection autonomously in the event of an IRU failure.

The STs are 2 A-STR autonomous star trackers provided by Galileo Avionica. They are positioned so that their boresights are 76 deg apart and approximately parallel to the YZ plane. In nominal Sun-pointing attitude, bright light sources are well outside the 55-deg Sun exclusion zone provided by the A-STR shade. Additionally, this configuration provides all necessary attitude knowledge and redundancy about the X axis and combines with the DSS to provide adequate knowledge and redundancy for the Y and Z axes as well. Each A-STR provides 1553 quaternion output with an accuracy of 90 arcsec about the boresight and 30 arcsec about the transverse axes.

Four Goodrich E-Wheels serve as the usual attitude control and momentum management actuators. They are arranged in a symmetric, pyramidal configuration about the X axis, with spin axes forming 30-deg angles with the YZ plane of the spacecraft. At nominal 28V bus inputs, the momentum capacity for each RWA is 70 Nms, and torque capability is 0.25 Nm. However, due to operational constraints, the maximum commanded torque is limited by software to 0.2 Nm.

The eight attitude control thrusters are designed and built by AMPAC-ISP, and they are placed in four pairs at the aft corners of the vehicle to provide attitude control redundancy. To achieve disturbance rejection in roll, the thrusters are canted inward about the Z axis at an angle of 10 deg to the XY plane. The primary thrusters are commanded in pairs to effectuate X-, Y- and Z-axis control torques. The thrust provided by each unit is nominally 5 lbf. These thrusters are placed and sized to balance any disturbance torques, especially those from firing the main engine, which is an Aerojet R-4D model bi-propellant engine producing 110 lbf of thrust.

ATTITUDE DETERMINATION AND TARGETING

The attitude determination function of the ACS flight software design is filled predominantly by an extended Kalman filter (EKF) with heritage from the Tropical Rainfall Measurement Mission (TRMM) and the Wilkinson Microwave Anisotropy Probe (WMAP).[4,5] The six-state filter takes the measurements from the IRUs, the two star trackers, and the DSS as inputs, and provides an attitude quaternion and a drift bias correction for the in-use IRU axes as outputs. The Kalman filter can be disabled, and in this case the attitude solution is then simply the quaternion from one of the star trackers or the attitude propagated using the IRU rate measurements, depending on which option is selected by ground command.

SNR Target

To simplify attitude determination relative to the Sun itself, the software is designed to target quaternions in either an inertially-fixed reference frame—geocentric inertial, or GCI—or a Sun-referenced frame based on ephemeris calculations of the direction of the Sun relative to the spacecraft and the solar north pole—solar north reference, or SNR. The SNR frame defines its X axis as the apparent Sun unit vector (Sun vector corrected for velocity aberration). The SNR Y axis is defined as the normalized cross product of the solar north pole, which is approximated by an inertially constant unit vector determined by the Carrington elements, and the apparent sun vector. The SNR Z axis then completes the right-handed triad. Thus, the targeting functions of the flight software, when Sun-referenced attitude is the goal, can calculate a SNR frame target quaternion. The instrument boresights are all aligned to the spacecraft X axis, so that the nominal attitude target for science pointing is the identity quaternion. With the SNR frame defined in this way, all measures of accuracy for scientific or engineering analysis purposes can be determined with ease from the performance relative to the SNR target. The target calculation also allows

multiplication by a delta quaternion, which allows the spacecraft to be pointed away from the default target by a known angle; this capability greatly simplifies the calibration maneuvers that must be performed throughout the mission.

On-Board Ephemeris

The SDO onboard ephemeris predicts the locations of the Sun, Moon, spacecraft, and ground station in geocentric inertial coordinates, referred to the mean-equator-and-equinox of J2000 (GCI, mean-of-J2000). Each object's velocity is derived from differencing successive position vectors and dividing the result by the ephemeris task sample time (nominally 1 second). The solar ephemeris is obtained from [6]. Its accuracy is better than 2 arcseconds during the ten-year SDO mission lifetime and has been validated by the JPL DE405 ephemeris. The lunar ephemeris is based on the "Low-precision formulae for geocentric coordinates of the Moon" published in [7]. This algorithm was implemented by the Hubble Space Telescope onboard lunar ephemeris with accuracy better than 0.36 deg; SDO has adopted the same implementation.

The spacecraft ephemeris is not computed from analytical models but obtained from numerical integration of the acceleration model using a standard fourth-order Runge-Kutta routine. To meet accuracy requirements of 2.9 km in both radial and transverse directions after seven days of propagation, the acceleration model includes several geopotential terms (J2, J3, J4, C22, and S22), Sun and Moon gravitational effects, and the solar radiation pressure effects. Determining the locations of the two dedicated ground stations is performed by transformation from geodetic to GCI, mean-of-J2000 coordinates.

ATTITUDE CONTROL MODES

SDO ACS comprises six software modes corresponding to different primary control goals; together these modes are capable of meeting all mission objectives. Five modes operate in the main processor software: Sun Acquisition (SunAcq), Inertial, Science, DeltaH and DeltaV. One mode, Safehold, operates solely in the ACE software.

Sun Acquisition and Safehold both acquire and hold the sun using CSS, IRU and RWA. Inertial mode calculates control errors based on the attitude determination solution from the Kalman filter (ST, DSS, IRU) and the IRU rates and corrects using the RWA. Science adds the CGT information to its controller suite. DeltaV uses IRU, IRU-propagated attitude determination, the main engine and ACS thrusters to modify orbit velocity. To unload vehicle momentum, DeltaH uses the same attitude control suite as DeltaV, with the addition of actively controlling the momentum stored in the RWAs.

Figure 3 shows a mode transition diagram for the six modes; transitions between modes not directly connected in the diagram are not permitted by the flight software. Certain transitions are labeled as autonomous; during nominal operations these transitions will occur when certain objectives are met, but the transitions may be commanded directly as well.

The attitude control modes on SDO are based on simple attitude error regulation. In the CSS modes (*i.e.* Sun Acquisition and Safehold), approximate attitude error is obtained by crossing the measured Sun vector in the body frame with the desired Sun vector. In all other modes, the attitude error is obtained from an error quaternion (as explained in the Inertial mode section). Science mode uses this quaternion-calculated error only for roll control; the CGT signals are used to direct the spacecraft X axis to the Sun target directly.

All main-processor controllers are based on a generic proportional-integral-derivative (PID) structure also described in the Inertial mode section; the integral component is only active in the high-accuracy modes (*i.e.* Inertial and Science modes), so that the other mode controllers are effectively proportional-derivative (PD) regulators. Attitude limiting is available in software for all main-processor modes to provide easily adjustable overshoot suppression; it is used in SunAcq, Inertial and Science modes, but is unnecessary in the thruster modes (*i.e.* DeltaH and DeltaV) and is disabled by setting the attitude limits very high relative to the operational range.

Control torques are calculated in the spacecraft body coordinate system and may be filtered with up to a fourth-order discrete filter, with a different design possible in each mode and for each axis of control. The filtered torques are then distributed to the actuators using a torque distribution matrix defined by the orientation of the actuators with
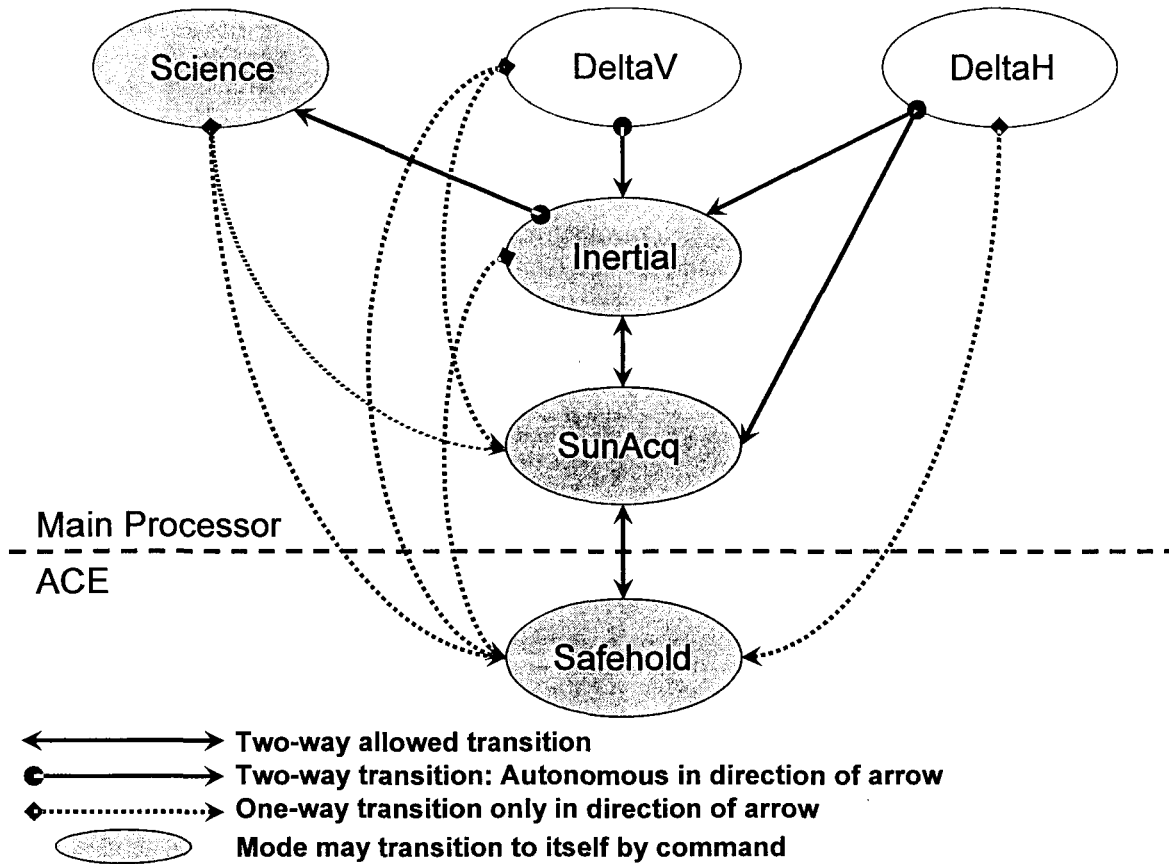
Figure 3: Allowed mode transitions for SDO ACS software

respect to the spacecraft. In the case of RWA-actuated modes, the RWA torques are then passed through logic that distributes the RWA momentum to minimize passing through zero wheel speed. For the thrusters, additional logic is used to convert the torque commands into on-off pulse profiles for each control cycle. In all cases, the actuator commands are scaled such that the maximum commanded torque does not exceed a database specified limit and the direction of the torque vector remains constant.

The remainder of the paper describes the modes in detail.

Inertial Mode

Inertial mode has two main responsibilities: it acts as a staging mode between the other main processor modes, particularly for thruster mode entry, and it is used to perform science instrument calibration maneuvers. All of the other main processor modes have the capability to enter this mode either by ground command, FDC command, and/or autonomous transition. Inertial mode can either hold the spacecraft at a target orientation or slew the spacecraft to a different orientation. The desired quaternion, q_c, by default, is the on-board calculated SNR frame quaternion. The desired quaternion can also include multiplication of the SNR quaternion by a ground-commanded delta quaternion, which allows the spacecraft to be pointed away from the default target by a known angle. Furthermore, the desired quaternion can also be a ground-commanded inertial quaternion. If the desired quaternion is not close to the current estimated spacecraft orientation, a slew will be executed to move the spacecraft to the desired orientation. Attitude control is performed using RWA torques, which are computed using a proportional-integral-derivative (PID) controller in terms of attitude and rate errors.

The attitude errors, θ_e, are calculated by taking the vector part of the quaternion difference between the desired quaternion, q_c, and the estimated quaternion, $\hat{q}$, and assuming small angles:

$$q_e = \pm q_c^{-1} \otimes \hat{q}$$
$$\theta_e = 2\bar{q}_e$$

(1)

The sign of the above equation is chosen such that the scalar component of q_e, which should be close to unity for small error quaternions, is positive. This selection of sign ensures the spacecraft will rotate through the smallest angle.

The estimated quaternion is computed in one of three ways: by an onboard, linearized, extended Kalman filter (EKF); by propagating the previous estimated quaternion using IRU measurements; or by flying directly off of a star tracker quaternion. The default attitude determination method during Inertial mode is the EKF, which is similar to the one used on WMAP. It is a six-state filter that uses IRU, DSS, and ST measurements as inputs, and estimates the attitude errors and gyro drift bias errors.

The commanded rates in Inertial mode are always zero; therefore, the rate error vector, $\vec{\omega}_e$, is the negative of the body rate vector, $\vec{\omega}_{BI}$, measured by the IRU:

$$\vec{\omega}_e = -\vec{\omega}_{BI}$$

(2)

The commanded control torque in the body coordinate system is calculated as follows:

$$\bar{T} = I\left(k_d \vec{\omega}_e + k_p \vec{\theta}_e + k_i \int \vec{\theta}_e\right)$$

(3)

where I is the SDO inertia tensor, and k_d, k_p, and k_i are the derivative, proportional, and integral gains, respectively. The integral term is limited to prevent wind-up and disabled if the attitude and/or rate errors for any axis exceed database specified limits to ensure absolute stability during slews. When the integrator is disabled, its input is set to zero such that the integral term in the above equation remains constant.

After the control torques are calculated, they are filtered using a second-order elliptical filter. The resulting filtered torques are distributed and limited as discussed above.

Science Mode
Science mode is based primarily on regulation of GT error signals and is used predominantly for science operations; however, GT calibrations and one science instrument calibration maneuver about the X axis are also performed in Science mode. The spacecraft roll attitude is controlled as in Inertial mode by using the IRU and Kalman Filter. These two methods combine to provide a three-axis commanded torque vector, which is actuated using the RWAs. Transitions into and out of Science mode are nominally via Inertial mode. The only exception is that FDC or ground can command the spacecraft from Science mode to Sun Acquisition or Safehold. When the CGT is in its acquisition range, the software automatically transitions from Inertial mode into Science mode to maximize time spent in Science mode. This autonomy can be disabled, and is disabled on entry into Inertial mode to prevent the software switching back and forth.

In Science mode, the spacecraft Y and Z attitude is controlled using the CGT measurements. The controller acts both to null the IRU-sensed rates and the CGT errors such that the SRB of the spacecraft points toward the Sun. Each GT measures the offset from the center of the solar disk along the Y and Z axes. Because the Science mode controller acts to zero the attitude errors, biases are added to the GT measurements such that when the GT measurement plus bias is zero, the spacecraft points at the SRB. All four GTs are processed in this manner, but the spacecraft only uses the CGT errors in the controller. Because the CGT errors are the offsets from the SRB along the CGT Y and Z axes, the pitch and yaw spacecraft attitude errors are calculated as follows:

$$\theta_{e,y} = S_z$$
$$\theta_{e,z} = -S_y$$

(4)

where S_y and S_z are the CGT errors rotated into the body frame. Figure 4 is a schematic of the geometry of the guide telescope FOV and the GT axes by which the errors are calculated.

As in Inertial mode, the roll axis attitude error is calculated as the quaternion difference between the desired quaternion, q_c, and the estimated quaternion, $\hat{q}$, assuming small angles. The estimated quaternion comes from the EKF with IRU, DSS, and ST measurements as inputs.

The desired rates in Science mode are zero; therefore, the rate error vector, $\bar{\omega}_e$, is the negative of the body rate vector, $\bar{\omega}_{BI}$, measured by the IRU:

$$\bar{\omega}_e = -\bar{\omega}_{BI}$$

(5)

After the control torques are calculated, they are filtered using a second-order elliptical filter on the X axis, a second-order low pass filter on the Y axis, and a third-order elliptical filter on the Z axis. The resulting filtered torques are distributed to the four RWAs as in the other wheel-based modes.

Figure 5 shows a single simulation run in which Inertial mode performs a slew and then transitions automatically to Science mode when CGT errors are within limits. The first plot shows guide telescope errors and how they behave on transition into Science mode. The second highlights Inertial mode slew performance with attitude errors in degrees. The third highlights Science performance in the same simulation run with attitude errors in arcsec.
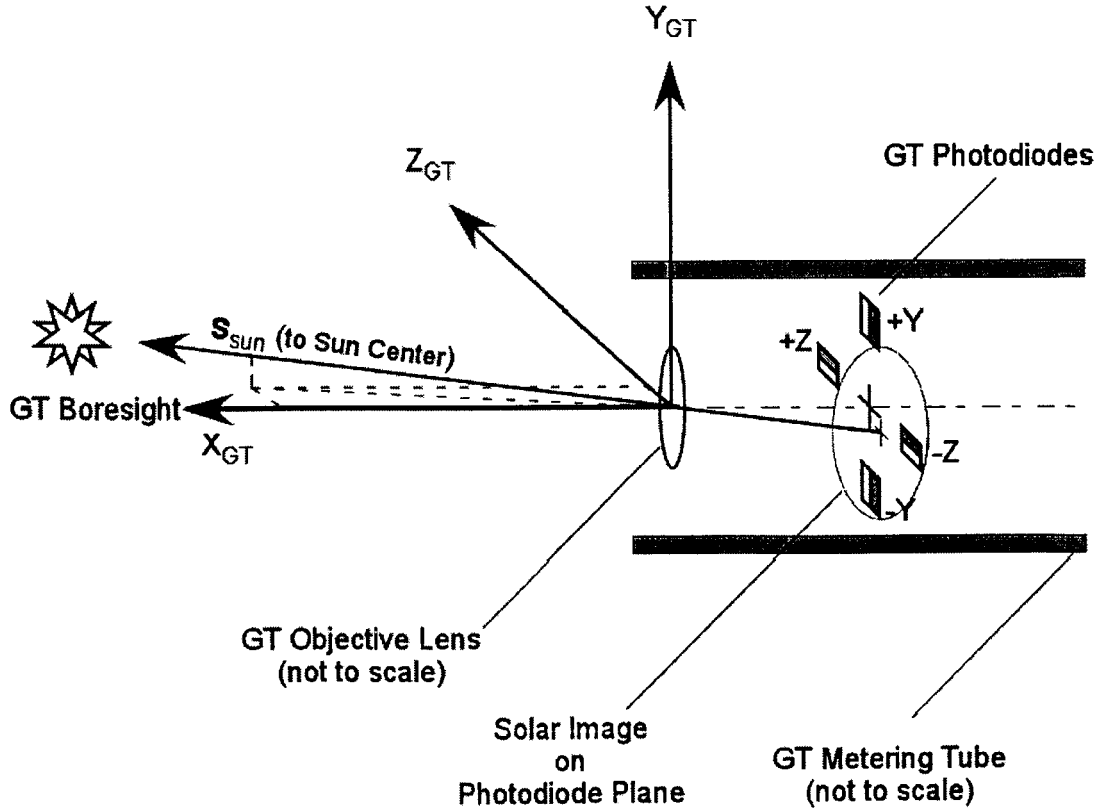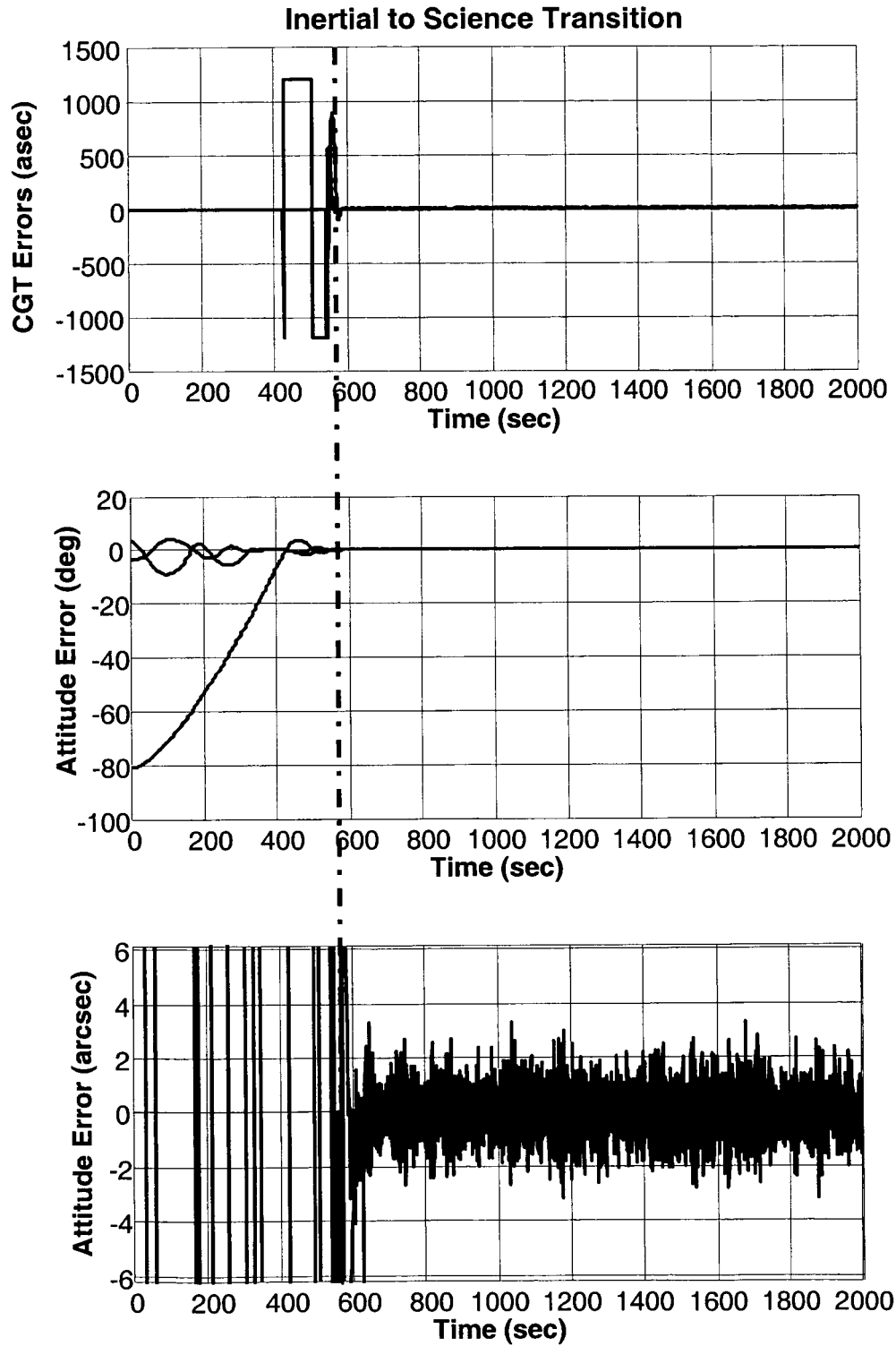

Figure 4: Guide telescope geometry

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Figure 5: Slew and automatic transition from Inertial to Science mode (marked by dashed line).

DeltaV

DeltaV mode is the orbital velocity change (ΔV) thruster-based control mode. It can function either with the main engine on or without. In main engine burns, the ACS thrusters on-pulse as needed for control, and in burns without the main engine, the ACS thrusters provide the velocity change and off-pulse for disturbance rejection. The IRU system provides attitude knowledge in this mode, and only the ACS thrusters are used for actuation as the RWAs are allowed to run down due to drag. SDO has eight thrusters—four primary and four redundant, so that only one set of four is used during nominal operations. All ACS thrusters are canted 10 deg about the Z axis. The main engine, which nominally provides ΔV only in the x-direction, is located in the center of the ACS thruster suite.

The longest expected ΔV maneuver will be about 50 minutes in duration, a capability which places some particular demands on the DeltaV mode. The maneuver time is long enough both for the attitude to have to change throughout the maneuver and for the mass properties to significantly change as well. The propellant makes up about one third of SDO's total mass, and the 50-minute burn would use well over half of that total propellant. DeltaV mode utilizes a PD controller to hold the spacecraft to a commanded quaternion while executing the ΔV maneuver. The attitude target is provided either in the DeltaV mode entry command or by a target quaternion command, and this target may be changed during a maneuver. Although the PD controller is not designed specifically for tracking, it is quite capable of acquiring new targets throughout a long maneuver.

Incorporating the decoupled inertia tensor into the control gains reduces sensitivity to changes in the total spacecraft inertia. Also, since the mass of propellant is so large compared to the spacecraft dry mass, propellant slosh can cause difficulty; therefore, a settling burn of 20 sec is currently planned for all main engine maneuvers so that the propellant may settle before the main engine is applied. For further discussion of slosh analysis, please see [8].
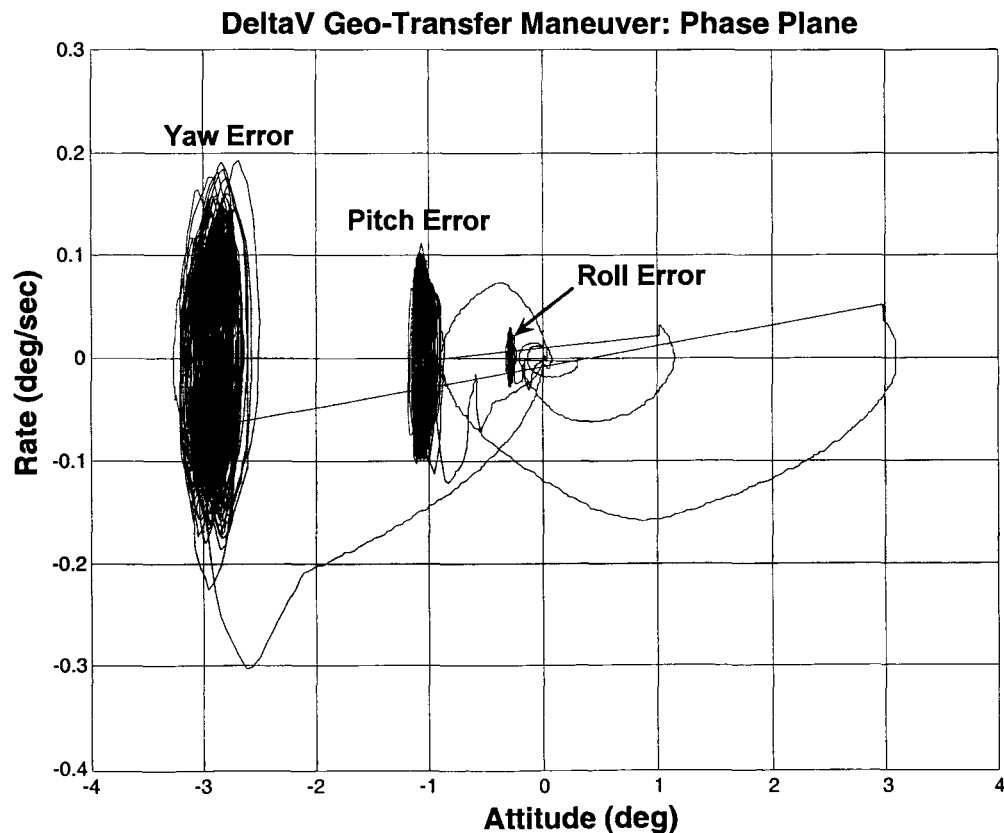

Figure 6: DeltaV Mode Simulation of Main Engine Maneuver toward Geosynchronous Orbit

The outputs of the controller are thruster count commands, equivalent to the number of thruster pulses in a given control cycle. Each thruster is capable of delivering up to four 50-msec pulses during each 200-msec control cycle. The exit criterion for the DeltaV mode is based on spending a prescribed amount of time in the mode; the controller also monitors an on-board estimate of the ΔV delivered for ground telemetry and failure detection purposes.

The transition between Inertial and DeltaV mode is strictly ground commanded, and the transition back to Inertial mode is autonomously enforced, but may be commanded if necessary. To transition to DeltaV mode, the spacecraft flight software needs the following commands and information: (1) desired attitude target for DeltaV burn, (2) burn duration, including any settling burn length, (3) whether the main engine will be used, and (4) desired set of thrusters to be used for the maneuver. In preparation for a nominal ΔV maneuver, the spacecraft uses Inertial mode to slew into an inertially-fixed initial ΔV orientation. The spacecraft waits in Inertial mode until the DeltaV command is accepted, at which time the thrusters begin to fire. After reaching the desired burn time, the software exits DeltaV mode, disables the thrusters, and returns to Inertial mode.

Nominal burn configurations have been simulated using a high-fidelity Simulink model; the simulations include both main engine burns and ACS thruster-only burns. Figure 6 shows the results of a main engine maneuver simulation with maximum main engine disturbance torques. In this case, as in all other simulations, the DeltaV mode maintains the attitude to within 5 degrees.

DeltaH

The purpose of the DeltaH mode is management of the spacecraft system angular momentum. Nominally, the system momentum needs only to be managed approximately once every four weeks, though allowance is made for more frequent DeltaH burns during early operations at lower altitude. During these monthly momentum unloads, the attitude is kept relatively constant (to within 5 degrees of entry attitude), so that science observations may resume as expediently as possible after the maneuver.

DeltaH is the only control mode that commands the thrusters and the RWAs at the same time: During a momentum unload maneuver, the DeltaH Mode provides a command torque to drive the reaction wheels to a desired wheel speed. The wheel commands are processed in a similar manner to the other wheel-controlled modes. As the wheels slow down, they impart their momentum onto the body. In turn this momentum is removed by the thrusters, which are commanded such that body rates are reduced and attitude excursions are controlled. As with DeltaV mode, DeltaH uses the IRU as its primary sensor. The DeltaH attitude target is defined as the estimated spacecraft attitude upon DeltaH mode entry. The small-angle quaternion difference between the IRU-propagated quaternion and the target quaternion provides the attitude error signals to the PD controller. At launch vehicle separation, if the angular rates exceed the capacity of the RWAs, DeltaH can be used to null the rates while unloading any momentum in the wheels. In this contingency situation, the proportional gain of DeltaH mode will be decreased by a factor of ten, so that rapid rate damping is the only effect of the controller. This rate nulling will provide much more effective reduction in the system momentum at the expense of maintaining attitude.

Similar to the DeltaV mode, DeltaH utilizes a PD controller to hold the spacecraft attitude (except for the launch vehicle separation contingency) while executing a maneuver. Because the main engine is never fired during DeltaH, the ACS thrusters are on-pulsed during all DeltaH maneuvers. The output of the mode is similar to that for DeltaV— thruster counts per control cycle. The exit criterion for the DeltaH mode is based on the calculated system momentum error; when the wheel momentum plus the body momentum is within 3 Nms of the target momentum, the mode exits to the mode from which it was entered.

In simulations of the DeltaH contingency momentum unload after large launch vehicle separation rates of [1,2,2] deg/sec the thrusters reduce the system momentum in less than 3 minutes, well within the requirement of 15 minutes. Figure 7 shows a high-fidelity simulation of the contingency maneuver.
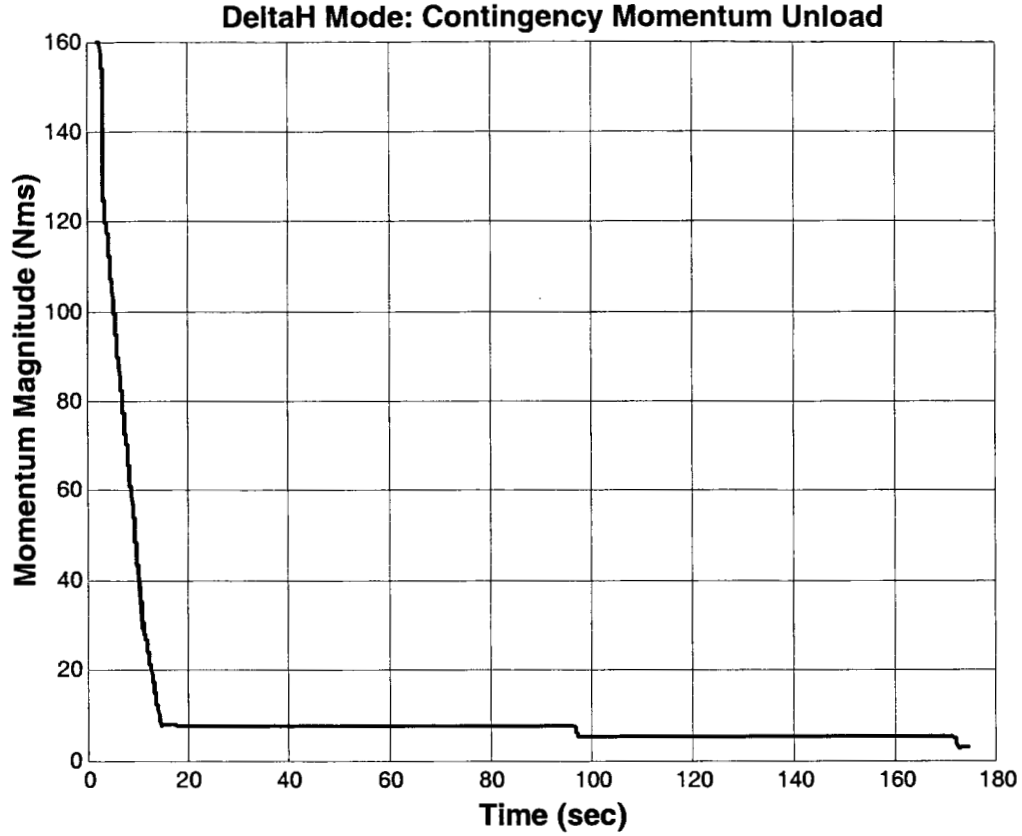
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DeltaH Mode: Contingency Momentum Unload

Figure 7: SDO momentum magnitude for a DeltaH tip-off contingency maneuver

Sun Acquisition

The Sun Acquisition mode, abbreviated as SunAcq, places and maintains the spacecraft in a power-positive orientation with the solar arrays within 15 deg of the Sun. The mode meets its requirement to recover from potential launch vehicle separation rates of up to [0.5, 0.6, 0.6] deg/sec within 30 minutes.

SunAcq uses CSSs and IRUs for sensors and RWAs for actuators. A PD controller is used, with the proportional law dependent on the Sun vector derived from the CSS measurements, and the derivative component dependent on the rates measured by the IRU. The SunAcq control law is

$$T = K_p \left(\hat{s} \times \hat{s}_d \right) + K_d \omega \qquad (6)$$

where $\hat{s}$ is the Sun vector in the spacecraft body frame, $\hat{s}_d$ is the desired Sun vector, T is the control torque, K_p is the proportional gain, K_d is the derivative gain, and ω is the body angular rate measured by the IRUs. During periods of eclipse, the SunAcq controller only controls the angular rate, providing torque commands $T = K_d \omega$.

Figure 8 gives the SunAcq controller response during possible launch vehicle separation cases, with initial rates up to [0.5, 0.6, 0.6] deg/sec.
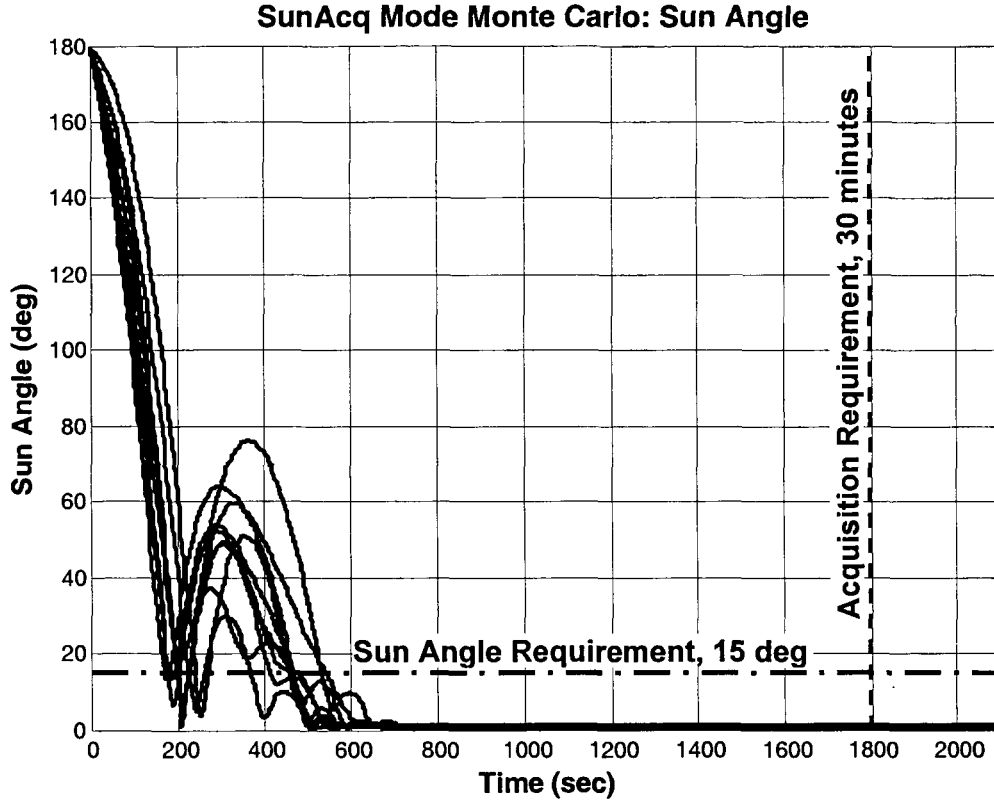
Figure 8: Sun Acquisition simulations starting from 180-deg Sun angle
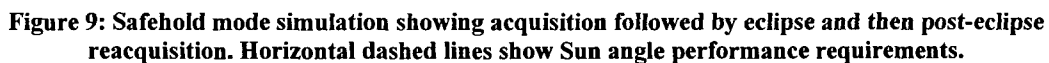
Safehold

The flight software for SDO Safehold is located in the ACE and is responsible for keeping the spacecraft in a thermally safe and power positive attitude, as well as remaining within HGA contact. Because of the high percentage of time spent observing the Sun, the thermally safe and power-positive attitude for the spacecraft is essentially identical with the observation attitude, except that there is no roll angle control requirement. Safehold must null all body angular rates (including roll about the Sun line) and point the +X axis of the spacecraft within 15 deg of the Sun to meet requirements.

The Safehold mode uses CSSs and the RWA tachometers for sensors and RWAs for actuators. In addition, the IRU is used during periods of eclipse to prevent excessive drift due to unobservability of the rates. Safehold is a PD controller, where the proportional law is dependent on the Sun angle measured by the CSSs, the Y and Z axis rates are derived from the CSS, and the X rate is derived from the RWA tachometers. See [9] for a detailed discussion of the Safehold design. The Safehold control law is

$$
T = K_p \left(\hat{s} \times \hat{s}_d \right) + K_d \left(K_{perp} \begin{bmatrix} 0 & 0 & 0 \\ 0 & 1 & 0 \\ 0 & 0 & 1 \end{bmatrix} \left(\dot{\hat{s}} \times \hat{s} \right) + K_{sunline} \cdot \tilde{\omega}_x \begin{bmatrix} 1 \\ 0 \\ 0 \end{bmatrix} \right)
\tag{7}
$$

where $\dot{\hat{s}}$ is the rate of the CSS-derived Sun unit vector in the body frame. Particular to Safehold is the different treatment of the rates about and perpendicular to the Sun line: K_{perp} is the rate gain perpendicular to the Sun line, $K_{sunline}$ is the rate gain along the Sun line and $\tilde{\omega}_x$ is the spacecraft rate about the X axis calculated from the RWAs.

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During eclipse, the Safehold controller becomes rate-only, with the rate provided by the IRU, and is identical to the SunAcq eclipse controller. This dependence on IRUs for rate measurement in Safehold was forced by the need to prevent drift during eclipse. The thermal/electrical design of SDO allows electrostatic discharge if the arrays are shaded from the Sun and then are exposed to the Sun suddenly, which can happen if the eclipse-exit Sun angle is greater than 45 degrees. Figure 9 shows a high-fidelity simulation of Safehold mode performance. The scenario starts with the spacecraft pointed directly away from the Sun. Safehold acquires the Sun to much better than the 15-deg requirement. During eclipse, the IRU-based rate control keeps the drift to well below the 45-deg requirement.

Figure 9: Safehold mode simulation showing acquisition followed by eclipse and then post-eclipse reacquisition. Horizontal dashed lines show Sun angle performance requirements.

CONCLUSION

The SDO ACS design is ready for implementation. The Critical Design Review process for the SDO mission has revealed no severe flaws or requirements gaps, and minor details are being addressed on schedule with related mission activities. The SDO ACS analysis team is currently shifting its attention to refining the performance of the controllers as spacecraft properties become more and more well defined and to supporting the testing of the flight software that will implement the control algorithms.

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