SDO DELTA H MODE DESIGN AND ANALYSIS

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ABSTRACT

While on orbit, disturbance torques on a three axis stabilized spacecraft tend to increase the system momentum, which is stored in the reaction wheels. Upon reaching the predefined momentum capacity (or maximum wheel speed) of the reaction wheel, an external torque must be used to unload the momentum. The purpose of the Delta H mode is to manage the system momentum. This is accomplished by driving the reaction wheels to a target momentum state while the attitude thrusters, which provide an external torque, are used to maintain the attitude. The Delta H mode is designed to meet the mission requirements and implement the momentum management plan. Changes in the requirements or the momentum management plan can lead to design changes in the mode.

The momentum management plan defines the expected momentum buildup trend, the desired momentum state and how often the system is driven to the desired momentum state (unloaded). The desired momentum state is chosen based on wheel capacity, wheel configuration, thruster layout and thruster sizing. For the Solar Dynamics Observatory mission, the predefined wheel momentum capacity is a function of the jitter requirements, power, and maximum momentum capacity. Changes in jitter requirements or power limits can lead to changes in the desired momentum state. These changes propagate into the changes in the momentum management plan and therefore the Delta H mode design.

This paper presents the analysis and design performed for the Solar Dynamics Observatory Delta H mode. In particular, the mode logic and processing needed to meet requirements is described along with the momentum distribution formulation. The Delta H mode design is validated using the Solar Dynamics Observatory High Fidelity simulator. Finally, a summary of the design is provided along with concluding remarks.

INTRODUCTION

Part of NASA's mission statement is to advance and communicate scientific knowledge and understanding of the Earth, the Solar System, and the Universe. One of NASA's many programs that seek to accomplish this objective is the Living With the Star (LWS) program. The LWS program seeks to provide a better understanding of the Sun, space weather and its effects on Earth. The first mission in this program, which examines the Sun and space weather, is the Solar Dynamics Observatory (SDO) mission. The SDO spacecraft will carry three Sun-observing instruments to geosynchronous (GEO) 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.

The SDO is a large, 2894 kg, Sun-pointing, geosynchronous satellite. 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 located in White Sands, New Mexico. These goals, which evolved into mission and spacecraft requirements, 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. It has been designed and is currently being 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 the instrument payload toward the Sun accurately and to keep the roll about the Sun vector correctly positioned with respect to the solar North Pole. This roll control is especially important for the magnetic field imaging of HMI.

The next section provides a brief overview of the ACS modes. A more detailed description of the Delta H mode design and implementation follows this overview. A brief description of the momentum distribution analysis is provided in the next section. To verify the performance of the Delta H mode for various maneuvers, the SDO High Fidelity (HiFi) simulator is used to simulate two nominal maneuvers. Finally, a summary of the results and concluding remarks are provided in the conclusion section.

ACS MODE OVERVIEW

There are six ACS operational modes: Inertial, Science, Delta V, Delta H, Sun Acquisition, and Safehold [1]. The first five modes are implemented in the Mongoose CPU, while the Safehold mode resides in the Attitude Control Electronics (ACE) CPU. All modes can autonomously transition to Safehold. There are also autonomous transitions from the thruster modes to Inertial, Sun Acq or Safehold. Autonomous transition to Science mode from Inertial mode can be enabled via a command. Transitions into a thruster mode are by command only. This paper focuses on describing the design of the Delta H thruster mode.

Delta V mode uses the ACS thrusters and the main engine to provide the needed ΔV for orbit transfer maneuver and the orbit StationKeeping maneuver. This mode is only entered by ground command from Inertial mode, and normally exits autonomously back to Inertial mode. This mode uses a gyro to measure the rates and propagate attitude.

Delta H mode is a thruster-based mode, which uses the gyro for attitude and rates. The main purpose of Delta H is to unload excess angular momentum. Delta H can also be used to null excess rates due to tip off or some failure which causes the spacecraft to spin up. Delta H is normally entered from Inertial mode or Sun Acquisition Mode and autonomously transitions back to the mode it came from after exiting Delta H mode (Inertial or SunAcq). This autonomous exit occurs after the exit criteria have been met.

DELTA H MODE

The purpose of the Delta H mode is to manage the angular momentum using thrusters to provide the needed external torques. The desired momentum state is defined based on the ACS and science requirement along with the momentum management plan. The SDO Delta H mode design is based on TRIANA heritage. The body rates, and the gyro propagated quaternion along with the commanded attitude are used to determine the control errors. These errors are used in the PD controller, which produces the commanded control torque needed to hold the spacecraft inertially fixed, while the wheels are commanded to a desired wheel speed. SDO Delta H mode uses the entry attitude as the target/commanded attitude (during vehicle separation, the target/commanded attitude is [0.0 0.0 0.0]). Furthermore, SDO uses ground commanded momentums to determine the target wheel speeds. If the ground does not provide a commanded target momentum, the default momentum ([0.0 0.0 0.0]] Nms) is used. The ground must also compute the bias for the Wheel Momentum Distribution Law (WMDL). The mode controller is a PD controller with Pulse Width Modulator (Quantizer) and the wheel control law is a simple proportional controller. The Delta H PD controller is designed to maintain the target attitude to within 5 degrees (3 sigma) while driving the momentum to within 2 Nms (3 axis) of the desired target momentum. In addition, the momentum unload maneuver must be completed within 15 minutes.

The first time that the Delta H mode may be used is at Separation. At Separation (Tip-Off : max body rates of [1,2,2] deg/sec), Delta H may be entered from Sun Acquisition to null excessive body rates. After nulling the rates, the ACS will autonomously transition back to Sun Acquisition, which orients the spacecraft into a power positive attitude. While on orbit or during the transfer orbit, Delta H is used to manage the system momentum to within some defined momentum (Nms) resolution.

Controller

The Delta H mode consists of several computational and logical components. The first step in the Delta H mode is the error calculation. Next, these errors are used in the controller to produce the appropriate body frame control torques that will reduce the attitude and rate errors. In order to remove the design dependence of the spacecraft inertia, the controller is premultipled by the estimated inertia. Therefore, major changes in inertia (fuel usage) will not call for a total redesign of the control gains. It simply requires an update of the estimated inertia. In addition, the body frame control torque is passed through a structural filter, which removes the effects/dynamics associated with high frequency jitter. The output of the controller is a filtered body frame control torque. The output of the controller is then converted to a commanded fire time in the body frame. This fire time is then processed by the selection logic to produce fire times in the thruster frame.

Thrusters Selection Logic

The selection logic utilizes a table loaded selection matrix. The selection matrix is based on the geometry and location of the thruster layout provided in Figure 1. A more detailed description of the thruster layout can be found in Reference 2.

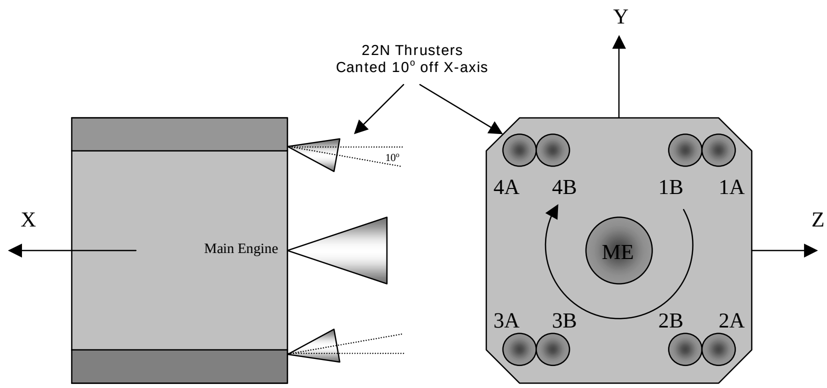

Figure 1 SDO propulsion Thruster Numbering and Identification

In order to switch from the primary thruster set (Side A) to the backup set (Side B), a table load is required. This matrix defines the thrusters used to produce a positive (or negative) roll, pitch or yaw torque. A mathematical description of the transformation from the body frame fire times, *Time_BcsF_cmd*, to the thruster space fire times, *Time_ThrF_Cmd*, is provided in equation 1

$$Time\_ThrF\_Cmd = \begin{matrix} Time_{BcsF\_cmd\_Roll} * \vec{R}_{pos} \\ or \\ Time_{BcsF\_cmd\_Roll} * \vec{R}_{neg} \end{matrix} + \begin{matrix} Time_{BcsF\_cmd\_Pitch} * \vec{P}_{pos} \\ or \\ Time_{BcsF\_cmd\_Pitch} * \vec{P}_{neg} \end{matrix} + \begin{matrix} Time_{BcsF\_cmd\_Pitch} * \vec{Y}_{pos} \\ or \\ Time_{BcsF\_cmd\_Pitch} * \vec{Y}_{neg} \end{matrix} \quad \textbf{Eq (1)}$$

Where $R_{\text{pos or neg}}$ is the set of thrusters (selection vector) in the selection matrix used to produce a positive or negative roll torque. This is also true for the pitch and yaw selection vector. To avoid ambiguity, the selection matrix consists of only 1's and 0's. In addition, the calculation of the fire time in the thruster domain does not contain any division. By using simple mathematical logic, singularities and ambiguities are avoided. This philosophy is also followed in the Three Thruster Firing logic, described next.

Three Thrusters Fire Logic

The thruster frame commanded fire times, which are produced from the selection logic, should maintain thruster control torque direction with minimal ΔV. For example, if all primary (Side A) thrusters are fired at the same level at the same time, only ΔV is produced along with residual torques. Therefore, the desired control torque is not exerted on the spacecraft. In order to ensure that only three thrusters fire at any given instance, the thruster fire time is passed through the Only Three Thrusters Fire logic. A graphical description of this logic is provided in Figure 2

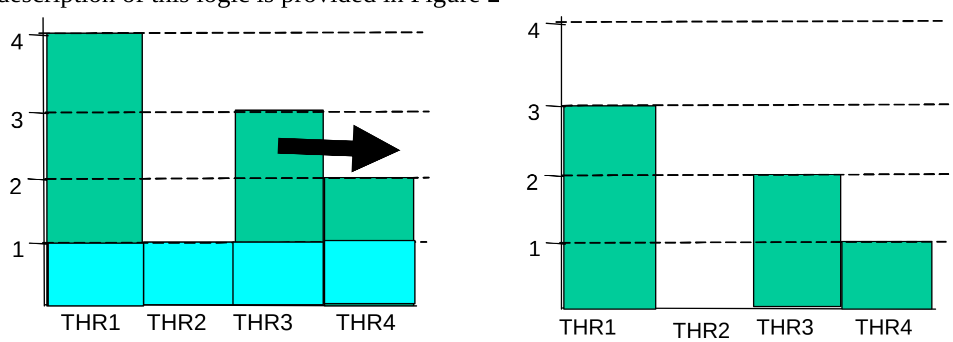

Figure 2 Only Three Thrusters Fire Logic

In the first graph, the green portion will produce an effective torque. The light blue level is the minimum level out of the four thrusters. This level of commanded pulses produces an unwanted ΔV. To minimize the amount of ΔV produced, the ΔV levels (light blue) are subtracted. The resulting fire times produce an effective torque with minimal ΔV. Since the fire times are all positive, there is no numerical/logical ambiguity. When all thrusters are fired at the same level, logic produces an equivalent gain reduction on the commanded control effort. However, this logic does not ensure that the commanded fire time remains within the physical limits of the system. The logic that performs this task is the Duration Scaling.

Duration Scaling

Duration Scaling logic is used to ensure that the maximum thruster fire time, T_{THR}, does not exceed the control cycle time (0.2 sec). The intent of this logic is to maintain the direction of the intended torque but reduce its magnitude to an acceptable value. The mathematical description of this logic is provided in equation 2.

$$if \ \ (Max(T_{THR}) > Max\_THR\_Lim) \quad then\left\{\tilde{T}_{THR} = T_{THR} * \frac{Max\_THR\_Lim}{Max(T_{THR})}\right\} \quad else\{T_{THR} = T_{THR}\} \qquad \textbf{Eq (2)}$$

If any component of the thruster fire time vector is larger than the maximum fire time, then normalize the fire time vector by the maximum fire time and multiply by the maximum allowed fire time. This logic produces a gain reduction when the demanded control efforts exceed the physical limits of the system. Since the *Max _THR _Lim* is never zero, there will never be a divide by zero event. At this point the commanded fire times can be processed (Pulse width modulation) and sent to the actuators.

Thruster Processing

The thruster processing quantizes the fire times, T_{fire}, into counts while accounting for dithering. In order to avoid dithering, if the maximum quantized fire time is equal to a full control cycle, then it is replaced by a fire time that is 125 % of a full control cycle. This logic ensures that the thrusters will not cycle on and off at the end/beginning of a new control cycle. When a full control cycle of firing is commanded, the dithering logic keeps the thruster on until there is a new command is executed. The thruster processing logic can be described by the following logic expression.

$$If\left\{\begin{array}{c} T_{fire} < 0 \\ 0 \le T_{fire} < 0.05 \\ 0.05 \le T_{fire} < .1 \\ 0.1 \le T_{fire} < .15 \\ 0.15 \le T_{fire} < .2 \\ T_{fire} = .2 \end{array}\right. \quad then \quad \left.\begin{array}{c} Cnt\_Cmd = 0 \\ Cnt\_Cmd = 0 \\ Cnt\_Cmd = 1 \\ Cnt\_Cmd = 2 \\ Cnt\_Cmd = 3 \\ Cnt\_Cmd = 5 \end{array}\right\} \qquad \textbf{Eq (3)}$$

The placement of the equal sign in the expression above ensures that one command can not produce two different commanded thruster commands, *Cnt_Cmd*. In addition, the simplicity and conciseness of the logic helps to ensure unique outcomes that can be analyzed or debugged. After passing through the thruster processing, these fire commands are sent to the Electronic Valve Driver (EVD), which fires the thrusters.

Wheel Control Law

During an Unload maneuver, a simple proportional wheel controller is used to drive the wheels to their desired wheel speed. This controller utilizes the wheels tachometer (ω_w) and the transformation matrices, A_{bw}, from body to wheel (and wheel to body) to calculate the appropriate wheel torque commands in the wheel frame. The mathematical description of this controller is provided below.

$$e_H = A_{bw}\left(h_{BcsF\_cmd} - A_{wb}\left(I_w \omega_w\right)\right) \qquad T_{RwaF\_cmd} = K_p e_H \qquad \textbf{Eq (4)}$$

Where h_{BcsF_cmd} is the commanded momentum in the body frame. I_w is the wheel inertia. Once the wheel torques commands are calculated in the wheel frame, they are passed through torque scaling[2] before being sent to the wheels.

Exit Criterion

The Delta H exit criterion can be mathematically expressed by the following:

$$\begin{array}{lll} h_{Sys\_Cal\_BcsF} = I_{sc}\omega_b + A_{wb}h_w & & h_e = \left(h_{Sys\_Cmd} - h_{Sys\_Cal}\right) \\ If \quad \alpha > Mag\,\{h_e\} & & Exit \end{array} \qquad \textbf{Eq (5)}$$

If the magnitude of system momentum error (h_e) is less than a desired threshold (α), then exit. One of the benefits of this criterion is that it does not over-constrain the wheels to reach an exact wheel speed and the body rates do not have to be below an exact threshold to exit. The exit criterion is satisfied when the combination of wheel speeds and body rates produce a momentum state below the desired threshold, which allows for a quicker exit from the mode.

1 Hz Equivalent Logic

During a Delta H maneuver simulation, the angular accelerations produced by the thrusters were too large for the gyro to track. During a thruster maneuver simulation, the gyro rates would lag the true rates. This resulted in false exits, which produced wheel speeds in Inertial and Science mode that were above the expected values. Since the wheel speed range is very small due to jitter concerns, logic was needed to allow the measured gyro rates to catch-up to the true rates. This logic, which is referred to as the 1 Hz equivalent Logic, allows the thruster to fire only in the first 0.2 seconds of a 1 Hz cycle. The exit criterion is then evaluated at the 4th or 5th cycle in the time interval of one second. The implementation of this logic requires a counter that counts at 5 Hz and repeats every second. The addition of this logic improves the accuracy of the control mode and the reliability of the momentum management plan. However, it will increase the Delta H maneuver times.

MOMENTUM MANANGEMENT PLAN

Momentum plan

The goal of the SDO momentum management plan is to provide the scientist with the maximum amount of uninterrupted observation time. Using the disturbances associated with the SDO GEO orbit, mechanical models, and known magnetic properties of the spacecraft, an analysis was performed to determine the amount of momentum buildup over different durations of time. Employing an early jitter analysis and the wheel momentum capacity, it was determined that SDO could store four weeks (32 Nms to 40 Nms) of momentum with margin and unload to a zero momentum state during the unload maneuver. Therefore, the momentum management plan was to allow the momentum to build up from a zero momentum state for four weeks. At the end of this four week time period, a Delta H unload maneuver would have been performed to drive the wheel speeds (and therefore the momentum) towards zero.

The Delta H momentum exit criterion, which was described in the Delta H processing section, was designed based on the results of the initial jitter analysis and the momentum initial management plan. Using this exit criterion, momentum management and the initial HiFi simulator, it was determined that SDO could unload with a resolution of 3 Nms. After obtaining more wheel test data, the jitter analysis was revisited. This analysis predicted that the wheel speeds associated with the four week momentum buildup levels would violate the jitter requirements. Therefore, the SDO momentum management plan was revised to allow for lower wheel speeds without reducing the observation time between momentum unload maneuvers (four week). Since most of the build up occurs in the Y axis of the body, the current momentum management plan allows the momentum to build up to [h_X 15 h_Y] Nms and unload to [h_X -15 h_Y] Nms. The drawback of this plan is that zero wheel speed crossing will occur during science operations. These zero crossings can result in arcsecond level attitude errors. By driving the wheels through zero, these attitude errors and the time near or at zero wheel speed can be reduce and therefore cause significant degradation of the science. Due to space constraints, zero crossings are not examined, only the nominal Delta H simulation results are presented.

The revised jitter analysis showed that a 3.0 Nms momentum magnitude error exit criterion would absorb much of the allocated wheel speed range. Therefore, additional tuning of the mode was performed to reduce the exit criterion to 2.0 Nms. This reduction was verified using the Monte Carlo analysis with Higher Fidelity (HiFi) simulation. After the autonomous exit from Delta H to the Inertial mode, the wheels will remove any residual rates and slew the spacecraft to a desired target. The final wheel speed after the slew maneuver will be a function of the residual rates, the slew target and the momentum bias. It should be noted that both the Inertial and Science mode utilize the Wheel Momentum Distribution Law (WMDL) to reduce the wheel speeds. If the momentum bias associated with WMDL is incorrect, the wheel speeds will deviate from the desired wheel speeds. This could potentially reduce the jitter constrained momentum capacity of the spacecraft. The next section describes the momentum distribution Law and formulation of the bias predictor.

Wheel Momentum Distribution Law

The Wheel Momentum Distribution Law is an extension of the average wheel steering law. It was first developed for Tropical Rainfall Measurement Mission (TRMM) and was flown on X-Ray Time Explorer (XTE). The purpose of the WMDL is to reduce the absolute wheel speed without inducing an unwanted torque on the spacecraft. This law is only applicable for a configuration with four or more wheels. This work focuses on only the four wheel pyramid configuration. The WMDL utilizes the null space properties of the non-orthogonal wheel transformation between the body reference frame and the 4 dimensional wheel space. This transformation has a non-unique set of wheel speeds that can produce the same wheel momentum in the body frame. The set of wheel speeds that produce the same total wheel momentum are related by the null vector. By modifying the magnitude of the null vector one can produce different wheel momenta in the wheel space that will yield the same total wheel momentum vector in the body frame.

The first step in the development of the WMDL is to define the transformation and the related unit vectors. The R^3 wheel speed direction unit vectors in the body frame momentum direction are e_i. This set of vectors can be used to construct the transformation from the wheel space to the body frame $A_{wb}=[e_1\ e_2\ e_3\ e_4]_{3x4}$. This transformation matrix is sometimes refered to as the steering matrix. The transformation from three dimensional body space to four dimensional wheel space is not unique in R^3 and has a null space defined as $A_{wb}X=0$. X is a null vector that satisfies the expression. Therefore, the set of vectors e_i are linearly dependent.

$$A_{wb} X_{null} = a_1 e_1 + a_2 e_2 + a_3 e_3 + a_4 e_4 = 0 \qquad \textbf{Eq (6)}$$

Where a_i are a set of normalized coefficients. Since the set of vectors are linearly dependent, then the transformation from body frame to wheel space is determined by the pseudo inverse.

$$A_{bw} = pinv\left(A_{wb}\right) \qquad \textbf{Eq (7)}$$

Multiplying both sides of Eq 7 by a scalar function, *f*, results in the following

$$f\left(a_1 e_1 + a_2 e_2 + a_3 e_3 + a_4 e_4\right) = 0 \qquad \textbf{Eq (8)}$$

The total wheel momentum is the vector sum of the wheel momentums

$$h_w = A_{wb} H_{rw} = A_{wb}\left(I\vec{\omega}_{4x1}\right) = A_{wb}\left[I_{11}\omega_{rw1} \quad I_{22}\omega_{rw2} \quad I_{33}\omega_{rw3} \quad I_{44}\omega_{rw4}\right] \qquad \textbf{Eq (9)}$$

Where I is the diagonal matrix containing the individual wheel inertia on the diagonal. An equivalent set of wheel momenta having the same total wheel momentum but different wheel speeds can be obtained by subtracting the null vector multiplied by *f*.

$$h_w = A_{wb} H_{rw} - fX_{null} = \left(\frac{H_{rw1}}{a_1} - f\right) a_1 e_1 + \left(\frac{H_{rw2}}{a_2} - f\right) a_2 e_2 + \left(\frac{H_{rw3}}{a_3} - f\right) a_3 e_3 + \left(\frac{H_{rw4}}{a_4} - f\right) a_4 e_4 \qquad \textbf{Eq (10)}$$

Next, the absolute wheel speeds are minimized by driving the wheel momentum (and therefore the wheel speed) down to the lowest equivalent total wheel momentum h_{weq}. The cost function associated with this optimization is

$$J\left(f\right) = \left\|h_{weq}\right\| = \left\|\left[\left(\frac{H_{rw1}}{a_1} - f\right) \quad \left(\frac{H_{rw2}}{a_2} - f\right) \quad \left(\frac{H_{rw3}}{a_3} - f\right) \quad \left(\frac{H_{rw4}}{a_4} - f\right)\right]\right\| \qquad \textbf{Eq (11)}$$

The cost function is minimized

$$f = \left(\frac{\max(h_{weq}) + \min(h_{weq})}{2}\right). \qquad \textbf{Eq (12)}$$

Next, the optimization is modified to account for a wheel speed bias. This bias will shift the wheel speed away from zero. To achieve this we utilize an additional null vector with a defined scalar multiple. This null vector will have the following form

$$h_{bias}\left(a_1 e_1 + a_2 e_2 + a_3 e_3 + a_4 e_4\right) = 0 \qquad \textbf{Eq (13)}$$

Adding this additional null vector to the cost function yields

$$J = \left|hweq\right| = \left\|\left[\left(\frac{H_{rw1}}{a_1} + \left(h_{bias} - f\right)\right) \quad \left(\frac{H_{rw2}}{a_2} + \left(h_{bias} - f\right)\right) \quad \left(\frac{H_{rw3}}{a_3} + \left(h_{bias} - f\right)\right) \quad \left(\frac{H_{rw4}}{a_4} + \left(h_{bias} - f\right)\right)\right]\right\| \qquad \textbf{Eq (14)}$$

Thus the cost is now minimized if

$$f = \left(\frac{\max(h_{weq}) + \min(h_{weq})}{2}\right) + h_{bias} \qquad \textbf{Eq (15)}$$

The resulting f from the optimization can now be added to the wheel command torques by $T_{dist}=kf$. Where k is a simple proportional control gain. The full wheel command will have the following form.

$$T_{total\ wheel\ cmd} = T_{ACS} + T_{dist} \qquad \textbf{Eq (16)}$$

The wheel momentum distribution law minimizes the difference in speed between the four wheels and does not introduce momentum into the spacecraft since the vector sum of the torque distribution command is equal to zero. In addition, the minimized wheel speeds will have some minimum value which is a function of h_{bias}. The next step is the determination of f and h_{bias}. Since f is a function of h_{bias}, the section below describes a formulation for determining h_{bias}.

Momentum Distribution prediction

After leaving Delta H, the WNDL is used to reduce the difference between the wheel speeds. To obtain the best results, the f and h_{bias} must be determined before entering the Inertial mode. In this section, a brief formulation of h_{bias} is provided. Since the transformation from body to wheel is not an orthogonal transformation, the null vector for the wheel to body transformation is nonzero. The full transformation can then be described by the following:

$${}^{w}\overline{H}_{w} = A_{bw}\,{}^{b}\overline{H}_{w} + v_n h_{bias} \qquad \textbf{Eq (17)}$$

If h_{bias}. is constrained or defined, then a one-to-one transformation can be achieved. Therefore, if h_{bias} is determined explicitly, then the wheel momentum in the wheel frame can be specified uniquely. Assuming $h_{bias} = v_n^T\,{}^{w}\overline{H}_{w}$, then Eq 17 can be written as:

$${}^{w}\overline{H}_{w} = A_{bw}\,{}^{b}H_{w} - v_n v_n^T\,{}^{w}\overline{H}_{w} \qquad \textbf{Eq (18)}$$

Now, substituting for body frame wheel momentum we obtain

$${}^{RwaF}\,\overline{H}_{w} = A_{bw}A_{wb}\,{}^{w}\overline{H}_{w} - v_n v_n^T\,{}^{w}\overline{H}_{w} \qquad \textbf{Eq (19)}$$

Grouping terms together and removing the wheel momentum yields

$$\left(I - A_{bw}A_{wb}\right) = v_n v_n^T \qquad \textbf{Eq (20)}$$

This expression is only valid, if $v_n = null\left(A_{wb}\right)$. Therefore the assumed form for h_{bias} does provide a unique transformation for the wheel momentum. Once h_{bias} is determined, f can be found using the expression in Eq 15.

RESULTS

The two nominal Delta H maneuvers examined in this section are a Tip-Off and an unload maneuver. During Tip-Off, the ACS will enter into Delta H to null rates which exceed the capabilities of the wheel. This maneuver utilizes the default system momentum command of [0 0 0] Nms. The unload maneuver drives the spacecraft to a desired momentum state, which is defined ground command, while maintaining an inertially fixed attitude. The results section contains an overview of the maneuvers to be simulated, a brief description of the HiFi Simulator and a concise overview of the simulation setup/parameter load utility.

High Fidelity Simulator

The purpose of the SDO HiFi simulator is to provide a simulation that can be used to evaluate the ACS design. The HiFi is developed in the Simulink environment, which is a graphical modeling, simulation and analysis tool for dynamics systems. The current version of HiFi includes all the control modes and the mode manager. In addition, it contains all of the nonlinear dynamics and logic associated with the sensors, actuators and rigid body spacecraft. Due to computational limitations, the HiFi does not currently contain the flexible body dynamics. The orbital dynamics, which drive environmental disturbance models, are also part of the HiFi. The HiFi also produce telemetry and Fault Detection and Correction FDC logic and flags.

Setup/ Parameters

Before each simulation all parameters are loaded into the HiFi via a Matlab script. The master copy of these parameters is housed in the SDO parameter repository. This shared repository ensures that the SDO development and testing team utilizes the same parameters as the ACS team. The HiFi initialization/configuration is also achieved by Matlab scripts. These scripts load the correct mass properties (Beginning of Life, 18% fill fraction and End Of Life) and initializes the HiFi workspace with the appropriate spacecraft configuration (Fully deployed, Fully stowed, deployed with HGA stowed). Scripts are also used to initialize the attitude, rates, wheel speeds, commands, mode transition sequence and failure events for the simulation. All of these scripts are called by one main run script.

Case 1: Nominal Tip-Off:

During Tip-Off simulation, the ACS starts in Sun Acquisition mode with excessive rates. The Sun Acquisition mode attempts to null the rates. Since the rates are too high, the mode manager will transition the ACS into Delta H, which nulls the rates. Unlike the nominal Unload maneuver, the Tip-Off maneuver drives the system momentum to a default momentum command. The default momentum command is [0 0 0] Nms. When the spacecraft is within 2 Nms of the commanded momentum the mode manager exits Delta H and transitions back to Sun Acquisition. The worst case Tip-Off rates used in this maneuver are [1 2 2] deg/sec. The simulation is initialized with Beginning Of Life deployed mass properties are used.

Figure 3 ACS Mode Flag

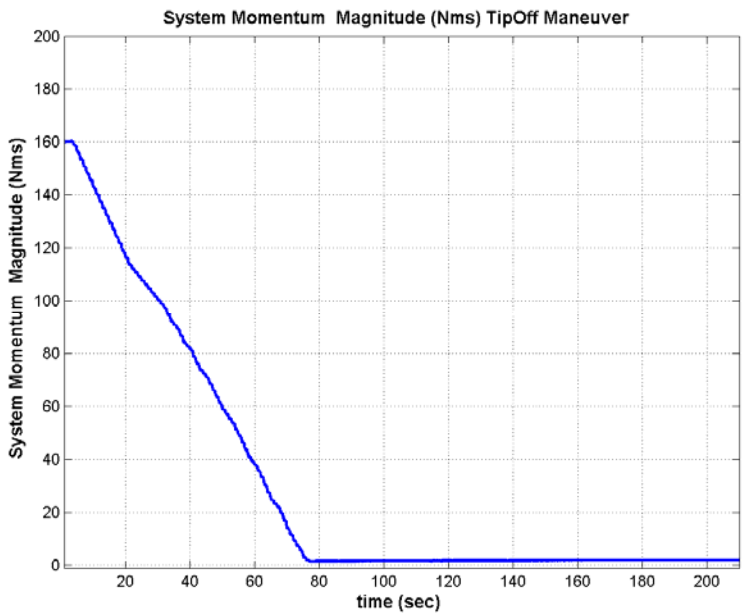

Figure 4 Momentum Magnitude

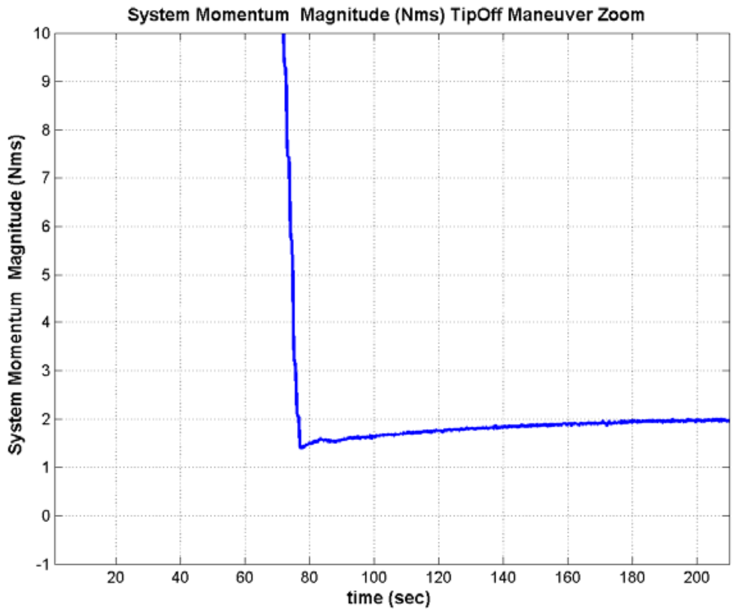

Figure 5 Momentum Magnitude (Zoom)

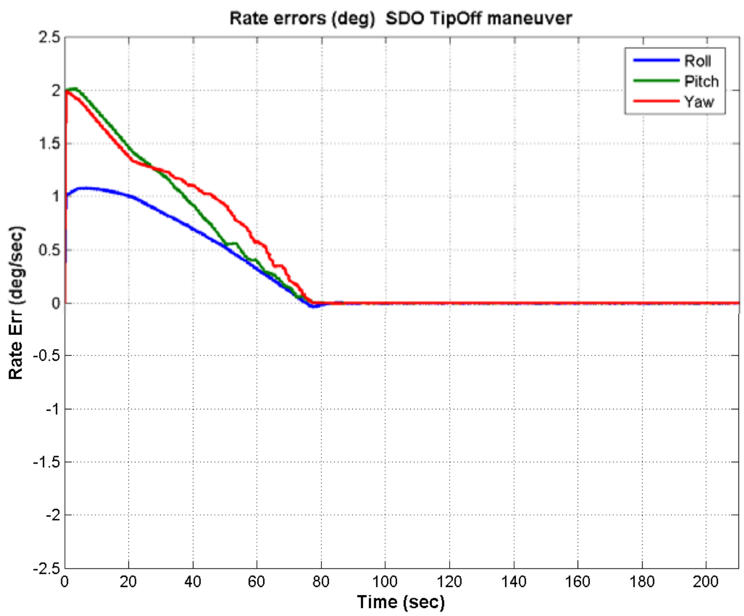

Figure 6 Rate Error

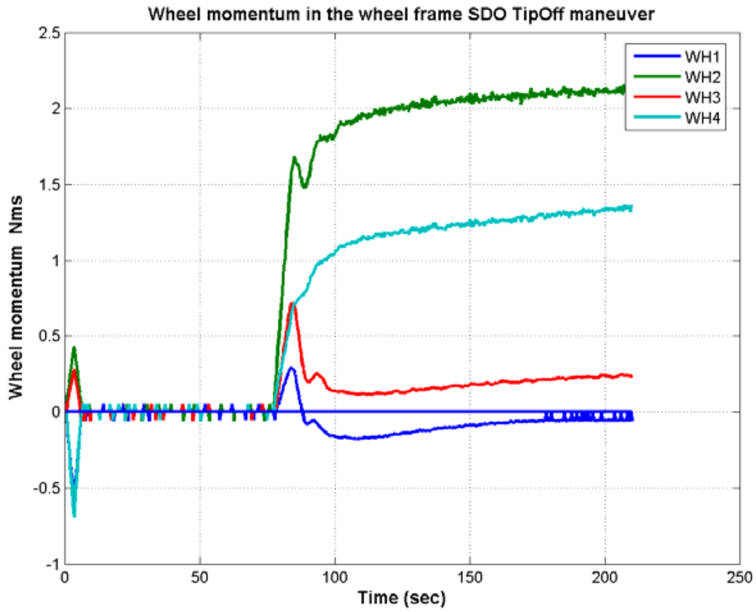

Figure 7 Wheel momentum

The maneuver starts in Sun Acquisition mode and tries to null the rates. After 3 seconds the ACS enters Delta H mode. In Delta H the Tip-Off rates are nulled within 75 seconds Figure 3. This maneuver time is significantly longer than the initial DeltaH mode design because of the equivalent 1Hz logic. At the start of DeltaH system momentum magnitude is 150 Nms. At the exit of Delta H the system momentum is below 2 Nms Figure 5. The rate errors (Figure 6) drop fairly quickly to values below 0.1 rad/sec. During a Tip-Off maneuver the entry attitude can be anything. Therefore, after exiting Delta H the Sun Acquisition mode must slew the spacecraft back to a sun pointing attitude and absorb the residual body rates. Figure 7 shows the wheels spinning up initially during Sun Acquisition. During Delta H the wheels spin down and remain near zero. After Delta H, the wheels spin up to absorb the residual rates and slew the spacecraft back to a sun pointing attitude.

Case 2 Nominal: Momentum Unload:

During a nominal momentum unload maneuver, the ACS starts in Inertial mode and transitioned to Delta H to unload the momentum. In this simulation, the system momentum starts at [0 9.025465 -9.025465] Nms and the ground commanded target momentum is [0 -9.025465 -9.025465] Nms. The calculated momentum bias for this target is 0.0 Nms. After achieving the desired system momentum state, the ACS system transitions back to Inertial mode. The initial attitude and rate errors in Inertial mode are zero. The nominal configuration used is a fully deployed spacecraft with 18% fuel fill fraction.

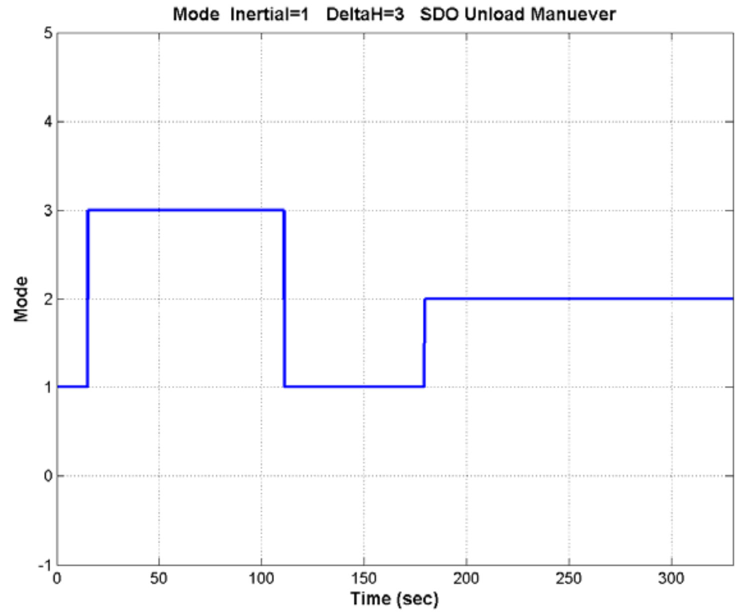

Figure 8 ACS Mode Flag

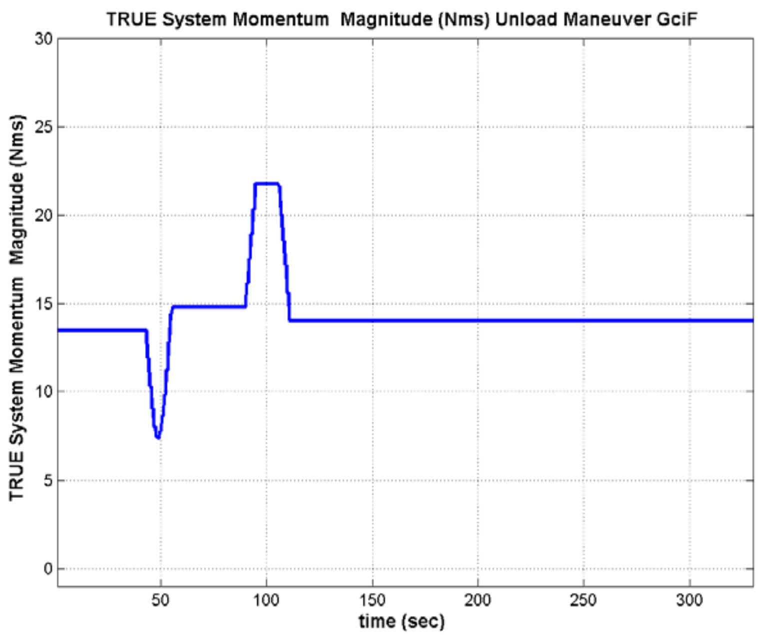

Figure 9 Momentum Magnitude

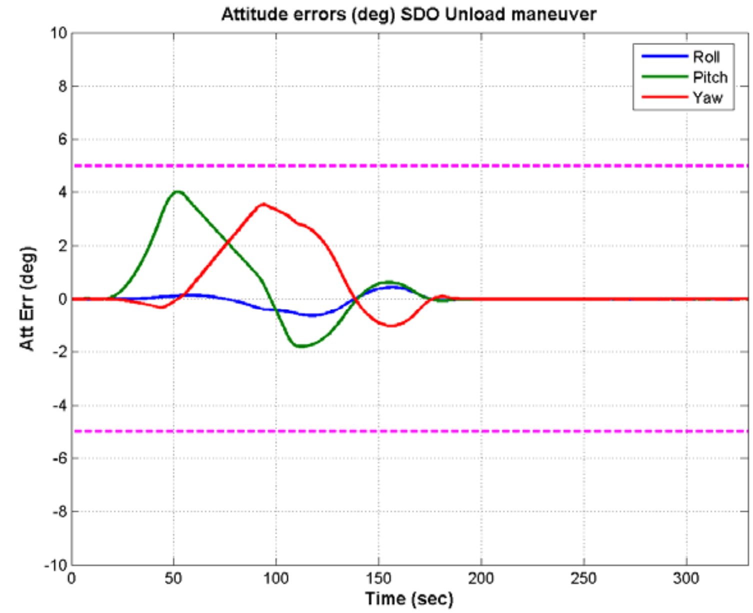

Figure 10 Attitude Error

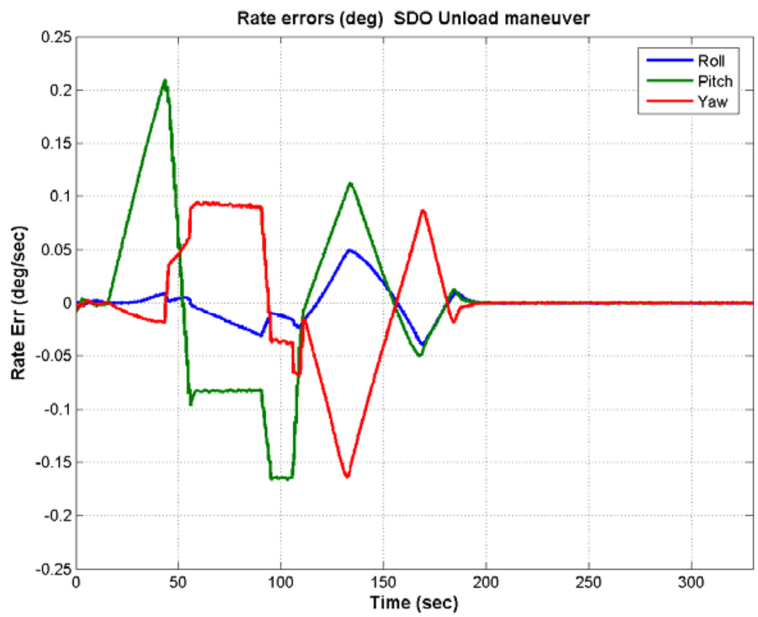

Figure 11 Rate Error

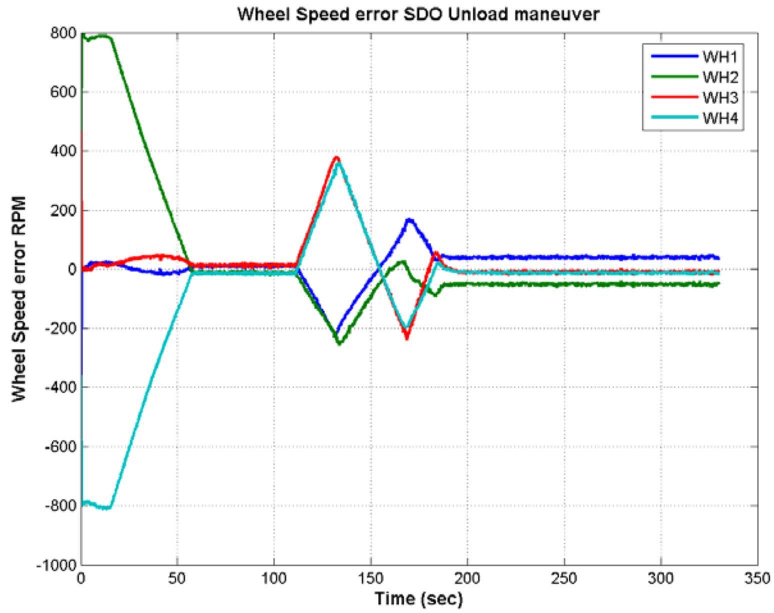

Figure 12 Wheel Speed Errors

The simulation starts off in inertial mode and transitions into Delta H after 91 seconds. The ACS exits Delta H at 190 seconds (Figure 8). Momentum magnitude entering Delta H is 13.0 Nms. After 70 seconds, the momentum is within 2 Nms magnitude of the target momentum (Figure 9). The attitude errors are below 5 degrees Figure10. The After exiting Delta H mode, Inertial mode reduces the residual rates and attitude errors. At approximately 290s, the attitude errors are small enough to transition into science mode (Figure not shown). The wheel speed errors at the end of Delta H are all below 85 RPM (Figure 12). At 120 seconds, while in Inertial mode, the wheels spin-up to remove momentum from the residual rates and attitude errors.

CONCLUSION

This paper describes the SDO Delta H mode design and the formulation of the WMDL. The mode design is evaluated using the HiFi Simulator. Two maneuvers are used to evaluate the performance of the Delta H design. The first maneuver is a Tip-Off maneuver, and second is a momentum Unload maneuver. The Tip-Off maneuver was slower than the initial Delta H design due to the addition of the 1 Hz equivalent logic. In addition, the 2 Nms exit criterion was satisfy. After entering the Sun Acquisition mode the ACS slewed the spacecraft back to the sun. The Unload maneuver simulation also performed as expected. The time in mode was not substantially different from the initial Delta H design since the expected thruster profile/duty cycles is significantly smaller. In addition, the unload maneuver meets the new 2 Nms exit criterion. In summary, the HiFi simulations illustrated that the Delta H mode design does satisfy all the performance requirements.

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