

Gimbal Control Algorithms for the Global Precipitation Measurement Core Observatory

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There are two gimballed systems on the Global Precipitation Measurement Core Observatory: two single-degree-of-freedom solar arrays (SAs) and one two-degree-of-freedom high gain antenna (HGA). The guidance, navigation, and control analysis team was presented with the following challenges regarding SA orientation control during periods of normal mission science: (1) maximize solar flux on the SAs during orbit day, subject to battery charging limits, (2) minimize atmospheric drag during orbit night to reduce frequency of orbit maintenance thruster usage, (3) minimize atmospheric drag during orbits for which solar flux is nearly independent of SA orientation, and (4) keep array-induced spacecraft attitude disturbances within allocated tolerances. The team was presented with the following challenges regarding HGA control during mission science periods: (1) while tracking a ground-selected Tracking Data and Relay Satellite (TDRS), keep HGA control error below about 4', (2) keep array-induced spacecraft attitude disturbances small, and (3) minimize transition time between TDRSs subject to constraints imposed by item 2. This paper describes the control algorithms developed to achieve these goals and certain analysis done as part of that work.

Introduction

The Global Precipitation Measurement (GPM) Mission is an international partnership to understand global precipitation and its impacts. The GPM Core Observatory (GPMCO), to be launched in 2014, is being built as a partnership between NASA and the Japanese Aerospace Exploration Agency (JAXA); it will be a primary member of the GPM fleet. Carrying both a dual frequency radar instrument and a passive microwave radiometer, the Core Observatory will serve as a calibration standard for the other members of the GPM constellation.

The GPMCO guidance, navigation, and control (GN&C) system will be responsible for aligning the spacecraft attitude to desired targets on the ground or in inertial space, executing instrument calibrations maneuvers, and providing commands to point the solar arrays (SAs) and the high gain antenna (HGA). The purpose of this paper is to describe the SA and HGA pointing control algorithms, used primarily in the observatory's Mission Science Mode.

Figure 1 provides an image of the satellite. The spacecraft coordinate axes (X, Y, Z) are indicated. Of principal interest for the current paper are the two SAs, located on the +Y and -Y sides

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of the observatory, and the HGA, located on a boom on the $-Z$ side. The dual frequency radar (not shown) is located on the $+Z$ side of the spacecraft; the microwave imager (the red-boxed cylinder with green reflector in the figure) is on the $-Z$ side directed towards $+X$ and down.

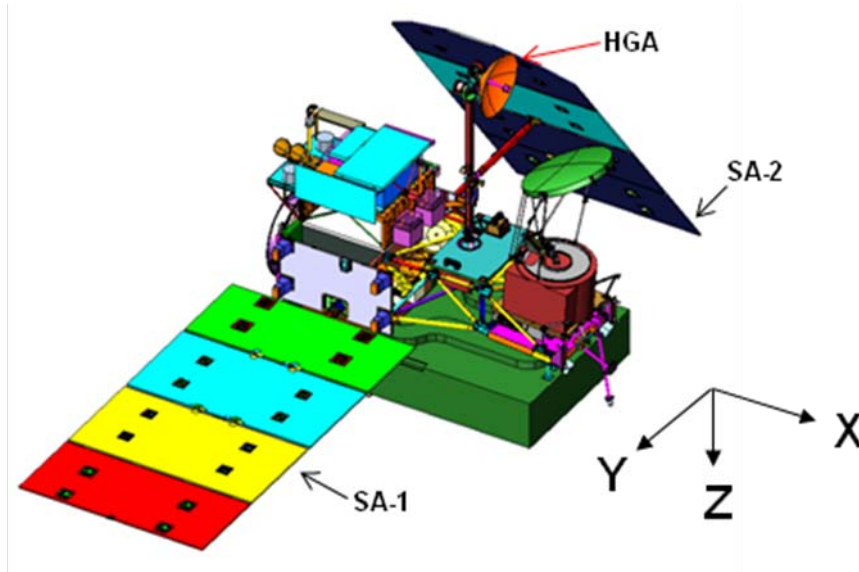


Figure 1: GPM Core Observatory

The GPMCO will fly in a nearly circular, low Earth orbit at a mean altitude of ~ 400 km and an inclination of $\sim 65^\circ$. During many engineering operations, the GPMCO will be flying with the velocity vector either parallel or anti-parallel to the X-axis. During science operations, the front end ($+X$) will be tilted up (towards the zenith) by 4° , but otherwise the spacecraft will still be flying essentially either forwards or backwards (with a few exceptions for calibration purposes).

The $+Y$ side of the spacecraft is the “cold” side; the Sun will typically be kept on the $-Y$ side for proper thermal control. Orbital precession due to Earth oblateness will cause the solar beta angle (β , the angle between the orbit plane and the Sun direction) to vary between $\sim -90^\circ$ and $+90^\circ$, with β defined to be positive when the angle between orbit normal (zenith $\times$ velocity) and the Sun direction is less than 90° . If the spacecraft were to fly with $+X$ forward always, the Sun would spend equal time on the positive and negative Y sides of the spacecraft. To avoid heating the $+Y$ side, the spacecraft will be commanded to fly backwards ($-X$ more-or-less aligned with velocity) when β changes from positive to negative, and then to fly forwards again when β changes from negative to positive. This change-over will happen roughly once per month.

The SAs are mounted as single-degree-of-freedom rotators, each driven by a gimbaled stepper motor. SA-1 rotation is defined as positive about the $+Y$ body axis; SA-2 rotation is defined as positive about the $-Y$ body axis. Each SA gimbal has a range of motion over $[-125^\circ, +125^\circ]$. In Figure 1, both gimbals are at 0° . SA-2 is tilted relative to its rotation axis by 53° ; this is required so that adequate flux can fall on the array over the whole range of solar beta angles. Note that maximum possible flux on that array varies as $\sim \cos(|\beta| - 53^\circ)$. The HGA is a two-degree-of-freedom device, operated using two stepper-motor gimbals. It can be pointed anywhere within the upper ($-Z$) hemisphere, with each of its gimbals having a range of motion over $[-90^\circ, +90^\circ]$.

During normal mission operations, the GN&C system will use a star tracker (ST^{*}) and gyroscopes[†] as the sensors for attitude estimation, a global positioning system (GPS) receiver[‡] for real-time orbit position and velocity estimation and resultant target attitude determination, and reaction wheels[§] as the actuators for attitude control. Thermal flexing of the spacecraft between night and day can cause changes in relative alignment between components (e.g., ST to gyro package, ST to science instrument) by $\pm 0.2'$ (arcminutes) (Reference 1), implying that the concept of “spacecraft attitude” becomes blurred for GPMCO at that level. During mission science operations, GPMCO attitude target generation is required to be good to $1.2'$ (3σ); we estimate an accuracy better than $0.9'$ (sum of “ 3σ ”-errors associated with neglect of nutation, UTC vs. UT1 time difference, and targeting errors based on GPS position and velocity errors and FSW propagation of GPS-provided position and velocity). Note: target attitude error is really only a concern for science instrument pointing, not SA and HGA pointing. Attitude estimation accuracy is required to be $2.8'$ (3σ); we estimate our post-Kalman-filter-convergence attitude estimation accuracy good to $\sim 0.18'$ (3σ), worst axis (including the effects of ST-to-gyro thermal flexing) (Reference 1). Attitude control accuracy (the difference between estimated and target attitude) for GPMCO mission science mode is required to be $2.0'$ (3σ). High-fidelity simulator runs that neglect thermal flexing show control accuracy of $\sim 0.15'$ (“ 3σ ”, i.e., typical maximum error). This neglects the effect of SA-induced disturbance, to be discussed below.

Solar Array Control

SA Control Overview

The GN&C team was presented with the following challenges regarding SA orientation control during periods of normal mission science: (1) maximize solar flux on the SAs during orbit day, subject to battery charging limits, (2) minimize atmospheric drag during orbit night to reduce frequency of orbit maintenance thruster usage, (3) minimize atmospheric drag during orbits for which solar flux is nearly independent of SA orientation, and (4) keep array-induced spacecraft attitude disturbances within allocated tolerances.

Automated GPMCO SA control using the GN&C flight software is done via closed loop control. Input is: the measured SA gimbal angles each control cycle, the Earth Centered Inertial (ECI) Sun direction based on the Astronomical Almanac low-precision Solar ephemeris model (good to $\sim 0.01^\circ$), spacecraft position and velocity based on GPS data, spacecraft “in eclipse” status (derived from the Sun direction, the spacecraft position, and a spherical model of the Earth), solar β angle (derived from Sun direction and spacecraft position and velocity), and the spacecraft attitude (either the target attitude derived from spacecraft position and velocity, or the estimated attitude based on star tracker and gyroscope data). Output is a commanded rate at which each gimbal is to rotate. Figure 2 shows the basic, high-level, logic flow, up to an intermediate point (A).

^{*} A-STR Star Tracker, developed by SALEX Galileo

[†] Scalable Space Inertial Reference Unit, developed by Northrop Grumman

[‡] The NASA Navigator GPS receiver, developed by GSFC

[§] Five NASA Demiseable Integrated Reaction Wheel Assembly units, developed by GSFC

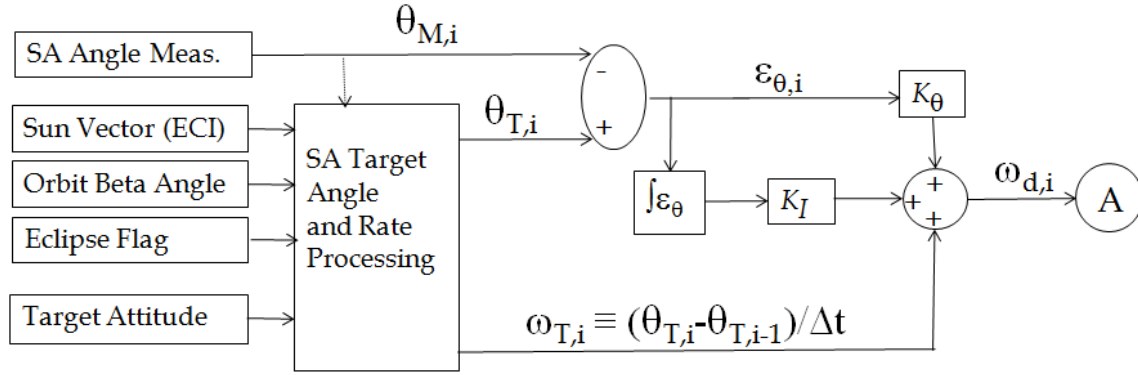


Figure 2: GPMCO SA Control High-Level Logic Flow

Figure 2 pertains for SA-1 and SA-2, individually. $\theta_{M,i}$ is the measured gimbal angle at time t_i . $\theta_{T,i}$ is the target gimbal position (with more information on its computation to be provided below). $\omega_{T,i}$ is the target gimbal rate, computed as the finite difference derivative of $\theta_{T,i}$. $\epsilon_{\theta,i}$ is the position error: $\epsilon_{\theta,i} = \theta_{T,i} - \theta_{M,i}$. $\omega_{d,i}$ is the immediately desired rate, an intermediate estimate for what will become the commanded gimbal rate for the current cycle. It is computed as

$$\omega_{d,i} = \omega_{T,i} + K_{\theta} \epsilon_{\theta,i} + K_I \int \epsilon_{\theta} dt \quad (1)$$

where K_{θ} and K_I are user-selectable gain factors – i.e., our control law is a proportional/integral (PI) control law with rate feed-forward. The challenging part is in the determination of $\theta_{T,i}$ in the “SA Target Angle and Rate Processing” box, which will be discussed in the following sections.

SA Angles for Maximum Solar Flux

If the Earth were a point mass (no shadowing), maximum possible flux were always desired (no offset pointing to reduce input current), and the gimbals were not hard-stop constrained, then $\theta_{T,i}$ would be given by

$$\theta_{1,T,i} = \theta_{1,MaxFlux} = + \tan(-S_{X,i}, -S_{Z,i}) \quad (2a)$$

$$\theta_{2,T,i} = \theta_{2,MaxFlux} = - \tan(-S_{X,i}, -S_{Z,i}) \quad (2b)$$

where equations 2a and 2b pertain for SA-1 and SA-2, respectively, and (S_X, S_Z) are the X and Z coordinates of the Solar direction vector in the body reference frame.

SA Feathering During Eclipse – Part 1

Given that the Earth is extended, and that the spacecraft is in low Earth orbit, the spacecraft regularly passes through the Earth’s shadow (although when $|\beta| > \sim 70^\circ$, the orbit is in full Sun). When in eclipse (i.e., when solar power is unavailable), the SA pointing control algorithm switches to a secondary goal, i.e., minimizing atmospheric drag on the spacecraft. In principal, this would be achieved by orienting the plane of each array to be parallel to the wind velocity (i.e., “feathering” the arrays), taking into consideration the rotation of the Earth and the atmosphere with it. For simplicity, we approximate the spacecraft velocity relative to the atmosphere as equal to the ECI spacecraft velocity. If the spacecraft were flying with its +X axis parallel or anti-parallel to the velocity, the SA angle that would place the velocity vector in the array plane

would be 0. With the spacecraft pitched up by 4° during normal science operations, the feathering angles must be adjusted to compensate. During orbit night, $\theta_{T,i}$ becomes:

$$\theta_{1,T,i} = \theta_{1,Feather} \approx -4^\circ \quad (3a)$$

$$\theta_{2,T,i} = \theta_{2,Feather} \approx +4^\circ \quad (3b)$$

Well, we could have done this, except for the fact that a large discontinuity in $\theta_{T,i}$ would result in large changes in $\varepsilon_{\theta,i}$ and $\int \varepsilon_{\theta} dt$, and therefore rapid accelerations of the arrays, ultimately driving large overshoots in array pointing as the control law seeks a new equilibrium, as well as relatively large reverse disturbance torques on the spacecraft. To reduce these effects, the SA control algorithm includes a number of features to smooth out sudden transitions. First, downstream from point (A) in Figure 2, there is imposition of magnitude limits on SA rate and acceleration magnitude. Second, upon finding a large jump between $\theta_{M,i}$ and the ideal $\theta_{T,i}$ (i.e., as computed without consideration of the actual array location), the algorithm switches to using a smooth profile for computing $\theta_{T,i}$. And third, while the use of the smooth profile is in effect, the algorithm sets the integral term to zero. (Actually, we have so far found that using $K_I = 0$ provides good performance; we may tune this following launch.)

Smooth SA angle Profiles

Use of a smooth profile is triggered if $\Delta\theta_{MT} \equiv |\theta_M - \theta_{T,ideal}|$ exceeds a user-specified value ($\sim 10^\circ$). The algorithm computes the duration, τ_p , for which the profile will be in effect:

$$\tau_p = \text{Max}(0.5 \pi \Delta\theta_{MT} / \omega_{Max}, \pi (0.5 \Delta\theta_{MT} / \alpha_{Max})^{1/2}) \quad (4)$$

where ω_{Max} and α_{Max} are the maximum permitted angular rate and acceleration, respectively. This profile time limit is based on assuming that the profile will follow the form

$$\Delta\theta(t) = \Delta\theta_{MT} [1 - \cos(\pi(t-t_s) / \tau_p)] / 2 \quad (5)$$

where t is current time, and t_s is the time at which use of the profile begins. Having set the value for τ_p , the profile is actually computed per equations 6-8:

$$w_S = [1 + \cos(\pi(t-t_s) / \tau_p)] / 2 \quad (6)$$

$$w_E = [1 - \cos(\pi(t-t_s) / \tau_p)] / 2 \quad (7)$$

$$\theta_T(t_i) = w_S \theta_{T,S} + w_E \theta_{T,ideal}(t_i) \quad (8)$$

where $\theta_{T,S}$ is the target angle at $t = t_s$, and $\theta_{T,ideal}(t_i)$ is the ideal target angle at t_i (i.e., the target angle without use of the smooth profile).

SA Feathering During Eclipse – Part 2

With this smooth transition profile available, we return to the moment of entering eclipse. The SA controller computes $\Delta\theta_{MT}$ and τ_p using $\theta_{T,ideal}$ for feathering (i.e., $\pm 4^\circ$). The controller also computes the SA angle that will be appropriate at orbit dawn. (To a first approximation, neglecting various daytime offsets to be discussed in the next paragraph, $\theta_{T,ideal}(t_{Dawn})$ is just $-\theta_{MaxFlux}(t_{Dusk})$. Correcting the SA dawn angles for the daytime offsets is straight-forward.) If τ_p is sufficiently short that feathering can be achieved before midnight, the SA controller computes a τ_p for a transition from $\theta_{Feather}$ to θ_{Dawn} , as well as a trigger time for starting that smooth transition

to θ_{Dawn} , i.e., $t_{\text{DawnTrigger}} = t_{\text{Dawn}} - \tau_{\text{P,Feather-To-Dawn}}$. If there is insufficient time to reach feathering before midnight, the controller computes profile parameters appropriate for a direct transition to θ_{Dawn} . Finally, the controller initiates the SA slew, either to θ_{Feather} with a delay before continuing to θ_{Dawn} , or directly to θ_{Dawn} .

SA Angle Offsets During Orbit Day

In specifying the use of the maximum solar flux angles in equations 2a and 2b, we noted the simplifying assumption that maximum possible flux is always desired. There are two possible reasons why this might not be the case: (a) input current to the battery would exceed a specified maximum limit, so offsetting the arrays to reduce current is necessary, or (b) more-than-necessary power is available, so one can intermix the goal of providing power with the secondary goal of reducing drag. To support these possibilities, the SA control algorithm allows four options for offset specification for each array: no offset, a leading offset, a lagging offset, or an offset always towards feathered.

For the leading or lagging offsets, the algorithm computes θ_T as θ_{MaxFlux} plus or minus a user-specified value $\Delta\theta$, applied in the direction so that the given SA always leads or always lags the Sun direction as the spacecraft rotates. (This always leading or always lagging the Sun produces an asymmetry so that $\theta_{T,\text{ideal}}(t_{\text{Dawn}}) \neq -\theta_{T,\text{ideal}}(t_{\text{Dusk}})$; but rather when $\theta_{T,\text{ideal}}(t_{\text{Dusk}})$ is given by $\theta_{\text{MaxFlux}}(t_{\text{Dusk}}) + \Delta\theta$, $\theta_{T,\text{ideal}}(t_{\text{Dawn}})$ will be $-\theta_{\text{MaxFlux}}(t_{\text{Dusk}}) + \Delta\theta$.) Combining a leading offset on one array and a lagging offset on the other allows for a reduction in the maximum total rate of charge. One must use a combination of leading and lagging offsets to reduce the peak current because otherwise, although the current would be reduced over most of the day time part of the orbit, the peak current would nonetheless still occur if the SA hard-stops are encountered, at dawn for lagging SAs or at dusk for leading SAs.

If an offset towards feathered is used, the SA cross section relative to the atmospheric wind is reduced relative to what it would have been – except when $\theta_{\text{MaxFlux}} = \theta_{\text{Feather}}$. When an offset towards feathering is commanded, the array will be stopped at θ_{Feather} whenever $|\theta_{\text{MaxFlux}} - \theta_{\text{Feather}}| < |\Delta\theta|$, which occurs around orbit noon. Peak power will be produced near orbit noon, implying this type of offset is appropriate only if peak current is not problematic.

SA Angle Range Limits

If $\theta_{T,\text{ideal}}$, after application of offsets, would exceed user-specified range boundaries $[\theta_{\text{Min}}, \theta_{\text{Max}}]$, the SA control algorithm limits θ_T to the appropriate boundary. However, control law inertia in the system (the fact that we're using a PI control law, and have an acceleration limit α_{Max}) will result in a bit of overshoot for the actual arrays. For our selection of parameters, we find an overshoot of $\sim 0.1^\circ$ before the array settles back to the boundary value. Given that the $[\theta_{\text{Min}}, \theta_{\text{Max}}]$ boundaries are slightly mushy, we are careful during tuning to set those values comfortably tighter than the actual limits we're willing to accept (including comfortably tighter than the actual hard-stop limits.)

Full Sun (High Beta) Orbits

If $|\beta|$ is large ($> \sim 70^\circ$), the spacecraft never enters orbit night. For sufficiently large $|\beta|$, SA-1 is shadowed by the body of the spacecraft, and therefore useless as a power source. In such circumstances, SA-1 is moved to its feathered position. When $|\beta| = 90^\circ$, the solar flux on SA-2 becomes constant, independent of solar array angle. For somewhat smaller values, the flux on SA-2 is very nearly constant. Therefore for some range of $|\beta|$ near 90° , setting SA-2 to its feathered position provides near maximum power while minimizing drag. The SA control algorithm feathers an array when $|\beta| > \beta_k$, using different β_k for each array.

If $|\beta|$ is large ($|\beta| > \sim 70^\circ$), but not so large as to force feathering ($|\beta| < \beta_k$), θ_T will be driven to a range boundary until θ_{MaxFlux} approaches its transition value of $\pm 180^\circ$. (Upon reaching $+180^\circ$ from below, θ_{MaxFlux} flips to -180° and continues to increase, and vice-versa.) For such transitions, the SA will unwind through $\sim 240^\circ$ at near maximum rate. This large angle change triggers the use of the smooth profile for the transition. In order to better center the transition around the $\pm 180^\circ$ position, the SA control algorithm triggers the transition at a user-specified angle (currently set to 15°) before the $\pm 180^\circ$ point is reached.

Fading Memory Filtering

A relatively large change in target rate (ω_T) (such as can occur when θ_T encounters a range limit, or at day/night or night/day transitions) can translate into a relatively large disturbance torque through the gimbal into the spacecraft. To avoid this disturbance, we have added some extra smoothing to the computation of θ_T . Let $\theta_T(t_{i-1})$ be θ_T as used in the prior cycle, and $\theta_{T0}(t_i)$ be an initial computation of θ_T for the current cycle as described in the preceding paragraphs. A filtered value for $\theta_T(t_i)$ is then computed as

$$\theta_T(t_i) = (1 - (\Delta t / \tau_F)) \theta_T(t_{i-1}) + (\Delta t / \tau_F) \theta_{T0}(t_i) \quad (9)$$

where Δt is the control cycle time step, and τ_F is a user-specified “fading memory” time scale. This smooths out high-frequency changes in $\theta_T(t)$, such as occur upon encountering the boundary points. Using $\tau_F = \Delta t = 0.1$ s, a lower limit, eliminates use of this fading-memory smoothing. We find that setting $\tau_F = 10$ s works well in suppressing attitude disturbances while not significantly compromising SA pointing.

Suppression of SA-Induced Attitude Jitter

The SA gimbal motors are stepper motors. This means that when the flight software sends a commanded rate (ω_C) to the gimbal control electronics (GCE), the GCE will send out a stream of pulse step commands to the hardware. The step frequency is related to the commanded rate by

$$f = \omega_C / \Delta\theta_{\text{Step}} \quad (10)$$

where $\Delta\theta_{\text{Step}}$ is the gimbal step size (0.0075°). Finite element analysis of the GPMCO design has shown that spacecraft structural resonances in the range $[0.5, 1.5]$ Hz can be excited if an array is commanded to move with a nearly constant pulse frequency near one of the resonances (Reference 2).

The primary design feature used to suppress this phenomenon is gimbal microstepping. Microstepping uses continuous powering of the motor windings to smoothly move the rotor from detent to detent. As implemented for GPMCO, microstepping is achieved by breaking the cardinal step into a sequence of smaller steps. The SA drive uses 16 substeps per cardinal step, with a duty cycle of 100%, in other words the windings are powered continuously. A simulation using 12 substeps per cardinal step (slightly less effective than the current 16-substep design) shows reduction in SA induced jitter from 0.73' for cardinal stepping down to 0.2' for microstepping, a 72% decrease. The improvement is not by a factor of twelve because the rotor is offset from its equilibrium point through the step. As the rotor initially moves away from the starting detent, it lags the electrical equilibrium point, i.e., the point at which the torque from the three winding phases would balance, effectively the instantaneously commanded angle. A lag persists until the rotor reaches the torque peak between the two detents, at which point the rotor “pops through” and thereafter leads the commanded angle until it reaches the next detent.

As an additional risk reduction step, we have implemented two optional schemes in the SA control algorithm: (a) jitter zone rapid passage, and (b) jitter zone dithering. For each, the user specifies upper and lower target rate limits $[\omega_{TL}, \omega_{TU}]$ for a jitter zone, actually two zones placed symmetrically around zero. For jitter zone rapid passage, if the to-be-commanded rate ω_d (i.e., at the end-point (A) of Figure 2) enters the jitter zone, ω_d is held on the edge on which it entered until the pre-(A) calculation places ω_d beyond the mid-point. When the pre-(A)-calculated ω_d passes the mid-point of the jitter zone, the control algorithm resets it to the far end (subject to the maximum rate and acceleration constraints). Using a separate Matlab-based jitter analysis, we estimate that proper tuning of the jitter zone rapid passage algorithm will reduce jitter by over $\sim 90\%$. For jitter zone dithering, if ω_d enters the jitter zone, the control algorithm superimposes a rapid, small amplitude, saw-tooth, rate dither on ω_d to prevent the rate from dwelling too long near any particular frequency. Faster rate transition through a problematic observatory mode can achieve a reduction of 40-60% in jitter.

Note that the optimal slew profile, from a jitter perspective, is a linear ramp in commanded rate (constant acceleration). This crosses any problematic modes with a constant rate. A cosine profile, in contrast, dwells for a longer time on modes near the start and end frequencies of the slew. The higher start and end torques for the linear ramp, as compared to the cosine, are a smaller contributor to overall jitter than the ringing induced as the step rate crosses a mode. In the GPM design, the rapid passage and dither algorithms will allow any problematic regions in the cosine profile (cf. equations 3-8) to be accommodated.

SA Control Example and Quality

Figures 3-5 present an example of SA motion and the effect on spacecraft attitude from a high-fidelity simulator run. The example is for an orbit β angle of 0, i.e., maximum eclipse duration. Figure 3 shows SA angles over one orbit; it calls out periods of Sun tracking with offset towards feathering during orbit day, SA feathering during eclipse, and array feathering near orbit noon. Note that the transition out of eclipse has the arrays arriving at Sun tracking a little early.

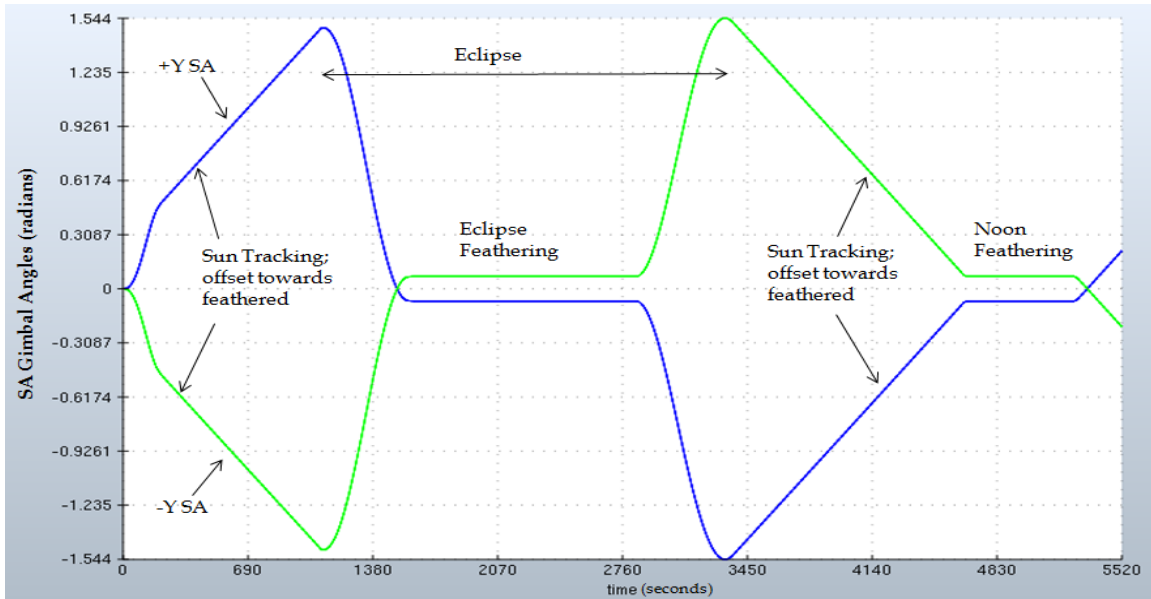


Figure 3: SA Gimbal Angles over one orbit; $|\beta| = 0$; 20° Offset Towards Feathering

Figure 4 shows SA angle rates over the orbit. Note the relatively sharp changes in rate at the start of eclipse, on reaching maximum permitted rate while transitioning to feathering during eclipse, upon reaching the feathered angle, on starting the transition to dawn, on resuming Sun tracking, and at the start and end of noon feathering. Figure 5 shows the Mission Science Mode attitude control errors over the same period. Note that at each of the events mentioned in the preceding sentence, an attitude disturbance occurs, with the strongest effect seen in pitch – not surprising given that the SAs rotate about the pitch axis.

The spikes shown in Figure 5 are primarily pitch disturbances*, and primarily due to SA-2 motion; its moment of inertia about Y is about five times greater than is that for SA-1 because of the tilted mounting of SA-2. The combined Y moment of inertia for the two SAs is roughly 1% of the spacecraft Y momentum of inertia. The maximum pitch disturbance is $\sim 0.3'$, well below the $4.8'$ attitude estimation and control requirement. The background noise in the simulation is due primarily to a combination of residual GMI off-balance rotation and ST noise (“residual” as in disturbances getting through the control system). The simulation that produced Figure 5 does not include gimbal-pulse-induced attitude disturbance; as previously discussed, this effect can contribute up to $\sim 0.2'$ from SA gimbal pulsing in the range [0.5, 1.5] Hz, assuming use of gimbal microstepping (Reference 2). Pointing accuracy of the arrays during periods of Sun tracking is good to $\sim 0.3^\circ$ (3σ), subject to the caveat that our simulation treats the SA gimbals as continuous rotators rather than stepper motors, and neglects any torsional twisting of the arrays. (Note, the high rate errors near the start of Figure 5 are just a consequence of simulation start up and settling.)

* Figure 5 shows spacecraft attitude disturbances in all three axes overlaid. For readers with a color copy, the color code for the figure is roll: blue, pitch: green, yaw: red.

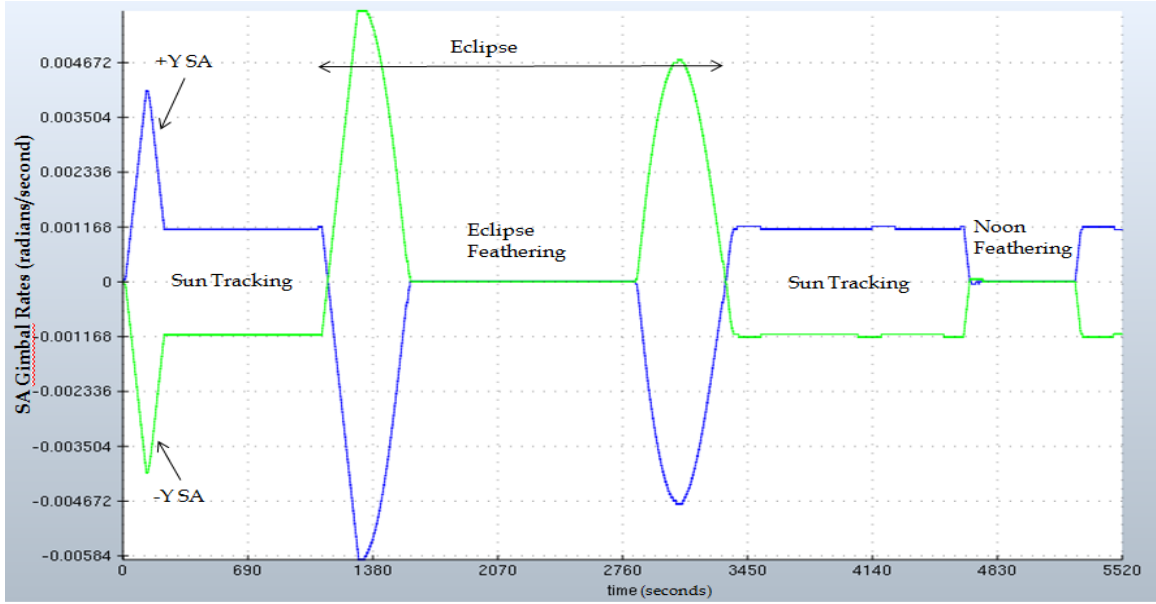


Figure 4: SA Gimbal Angles Rates over one orbit; $|\beta| = 0$;

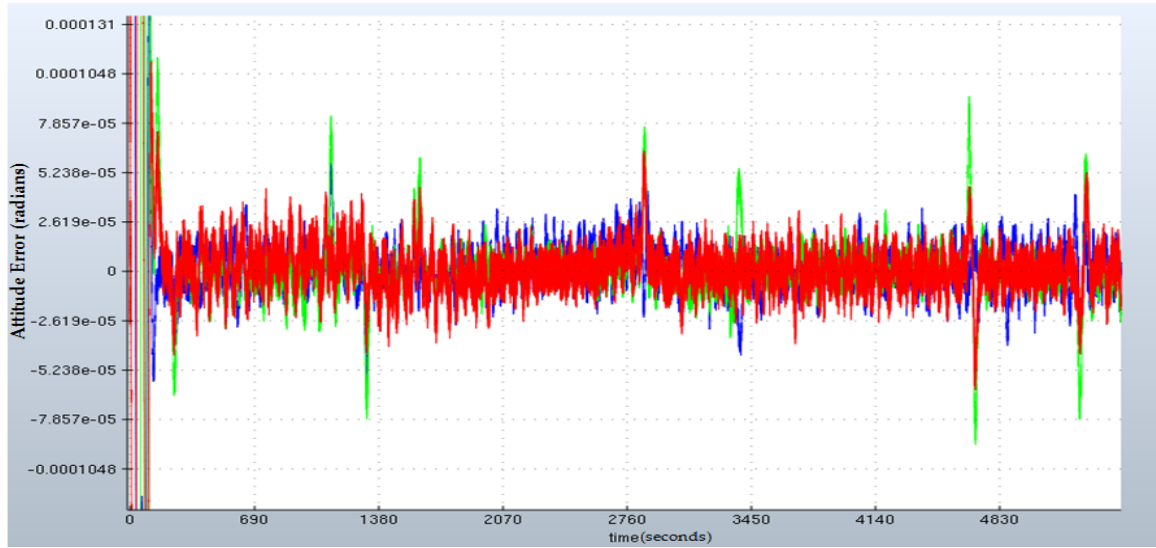


Figure 5: Spacecraft Attitude Error over one orbit; $|\beta| = 0$;

Aerodynamic Drag Reduction Using Feathering

Using detailed numerical simulations, we have estimated the reduction in orbit-mean aerodynamic drag produced by SA feathering for various solar beta angles. For $0^\circ < |\beta| < \sim 30^\circ$ (i.e., long eclipses), drag is reduced by $\sim 12\%$. Over the range $\sim 30^\circ < |\beta| < \sim 63^\circ$, drag reduction drops to zero as eclipse duration drops to zero. For $|\beta| > \sim 70^\circ$, SA-1 becomes shadowed by the spacecraft; feathering SA-1 over the whole orbit reduces drag by $\sim 34\%$. For $|\beta| = 90^\circ$, solar flux on SA-2 becomes insensitive to gimbal angle; feathering both SAs reduces drag by $\sim 57\%$. Depending on parameter tuning, we estimate overall mean drag reduction by ~ 10 to 15% .

High Gain Antenna Control

HGA Control Overview

The GN&C team was presented with the following challenges regarding HGA control during periods of normal mission science: (1) while tracking a ground-selected TDRS, keep algorithm control error below $\sim 4'$, (2) keep HGA-induced spacecraft attitude disturbances small, and (3) minimize transition time between TDRSs subject to item-2 constraints.

Automated GPMCO HGA control using the GN&C flight software is done via closed loop control. Input is: the measured HGA gimbal angles each control cycle, the ECI TDRS direction based on Flight Dynamics Facility (FDF)-provided osculating elements (each element set good over 24 hours to $\sim 8.4'$ [Reference 3]), spacecraft position and velocity based on GPS data, and the spacecraft attitude (either the target attitude derived from spacecraft position and velocity, or the estimated attitude based on star tracker and gyroscope data). Output is a commanded rate at which the gimbal is to rotate. Figure 6 shows the basic, high-level, logic flow, up to an intermediate point (A). Figure 6 pertains to each HGA gimbal individually and is essentially a repeat of Figure 2 for the SA gimbals.

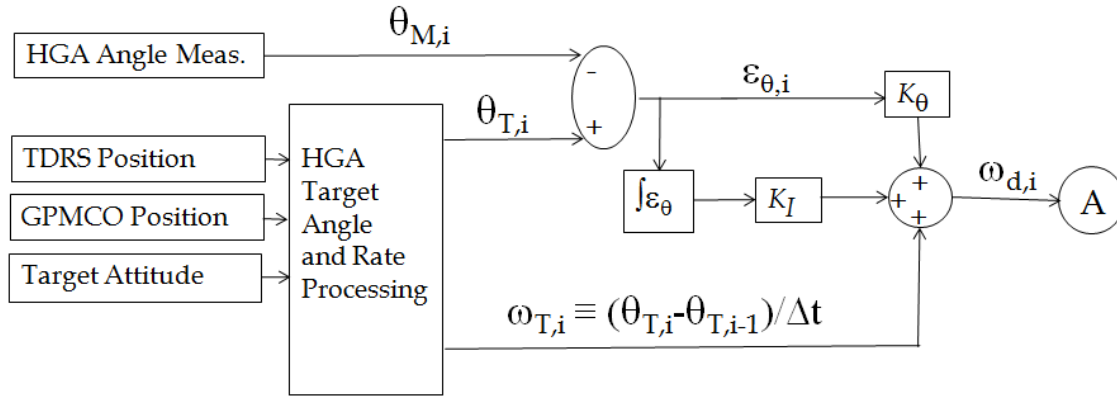


Figure 6: GPMCO HGA Control High-Level Logic Flow

HGA Angles for TDRS Pointing (Reverse Kinematics)

Given a spacecraft-to-TDRS direction vector in the body frame $[T_X, T_Y, T_Z]$, the appropriate inner and outer HGA target gimbal pair angles $(\theta_{T1}, \theta_{T2})$ to point the HGA at TDRS are:

$$\theta_{T1} \approx \text{atan2}(T_Y, -T_Z) \quad \text{if } |T_X| \neq 1 \quad (11a)$$

$$\theta_{T2} \approx \text{asin}(-T_X) \quad (11b)$$

Equations 11a and 11b are only approximate in that they do not take account of possible misalignments of the gimbal rotation axes and the antenna dish central axis. The HGA control algorithm actually solves for $(\theta_{T1}, \theta_{T2})$ given $[T_X, T_Y, T_Z]$ and the true axes alignments for the gimbals and dish (References 4, 5, & 6). Using $(\theta, \phi) \equiv (\theta_{T1}, \theta_{T2})$, and given the following definitions for $(\mathbf{h}_\theta, \mathbf{h}_{\phi 0}, \mathbf{a}_{00}, \mathbf{T})$:

$\mathbf{h}_\theta \equiv \theta$ gimbal axis; $\mathbf{h}_\theta \approx \mathbf{X}$ body frame axis

$\mathbf{h}_{\phi 0} \equiv \phi$ gimbal axis when $\theta = 0$; $\mathbf{h}_{\phi 0} \approx \mathbf{Y}$ body frame axis

$\mathbf{a}_{00} \equiv$ HGA pointing direction where $(\theta, \phi) = (0, 0)$; $\mathbf{a}_{00} \approx -\mathbf{Z}$ body frame axis

$\mathbf{T} \equiv$ desired pointing direction in reference frame

the target HGA angles (θ, ϕ) can be computed as:

$$\begin{aligned} k_1 &= (\mathbf{T} \cdot \mathbf{h}_{\phi 0}) - (\mathbf{T} \cdot \mathbf{h}_\theta) (\mathbf{h}_\theta \cdot \mathbf{h}_{\phi 0}) \\ k_2 &= \mathbf{T} \cdot (\mathbf{h}_\theta \times \mathbf{h}_{\phi 0}) \\ k_3 &= (\mathbf{T} \cdot \mathbf{h}_\phi) - (\mathbf{T} \cdot \mathbf{h}_\theta) (\mathbf{h}_\theta \cdot \mathbf{h}_{\phi 0}) \\ \theta &= \text{asin}([k_2 k_3 + k_1 (k_1^2 + k_2^2 - k_3^2)^{1/2}] / (k_1^2 + k_2^2)) \end{aligned} \quad (12a)$$

$$\begin{aligned} \mathbf{h}_\phi &= \mathbf{h}_{\phi 0} \cos(\theta) + (\mathbf{h}_\theta \times \mathbf{h}_{\phi 0}) \sin(\theta) + \mathbf{h}_\theta (\mathbf{h}_\theta \cdot \mathbf{h}_{\phi 0}) (1 - \cos(\theta)) \\ \mathbf{a}_{00} &= \mathbf{a}_{00} \cos(\theta) + (\mathbf{h}_\theta \times \mathbf{h}_{\phi 0}) \sin(\theta) + \mathbf{h}_\theta (\mathbf{h}_\theta \cdot \mathbf{a}_{00}) (1 - \cos(\theta)) \\ b_X &= (\mathbf{T} \cdot \mathbf{a}_{00}) - (\mathbf{h}_\phi \cdot \mathbf{a}_{00})^2 \\ b_Y &= \mathbf{h}_\phi \cdot (\mathbf{a}_{00} \times \mathbf{T}) \\ \phi &= \text{atan2}(b_Y, b_X) \end{aligned} \quad (12b)$$

These equations rely on the HGA field-of-regard being limited to the GPMCO $-Z$ hemisphere, with some padding added to account for possible misalignments. If $\mathbf{T}$ with $T_Z > 0$ is provided, the algorithm will track a projection of $\mathbf{T}$ on the HGA field-of-regard boundary.

HGA Algorithm Simplifications: No Smoothing Profile, No Fading Memory Filter

Operationally, for automated control of HGA pointing, there are three states that the HGA can be in: stopped, tracking a TDRS, and slewing between TDRSs. We will ignore the stopped state as trivial for this discussion. At first glance, tracking a TDRS would seem to map to tracking the Sun for the SA control problem, while slewing between TDRSs would seem to map to any rapid change in SA pointing (e.g., Sun tracking to night feathering). This mapping would suggest that the smooth profile and fading memory filter might be useful tools for the HGA problem as well. As it turns out, studies with our high-fidelity simulator have demonstrated that, because of the relatively low mass of the HGA components, rapid HGA acceleration associated with the start and stop of a TDRS-to-TDRS slew produces negligible disturbance to the spacecraft attitude. We have therefore chosen to keep the HGA control algorithm relatively simple by including neither the smoothing profile nor the fading memory filter. The control law integral term is reset to zero during slews – however, as for SA control, we currently find that setting $K_I = 0$ provides good performance in simulations.

Suppression of HGA-Induced Attitude Jitter

Although “macro” accelerations of the HGA gimbals is a non-issue, this is not the case for high-frequency, stepper-motor pulse-induced disturbances. We have found that the largest HGA-induced spacecraft attitude jitter contributions are from modes in the [0.5, 3.0] Hz range, with the GMI also being susceptible to excitation of modes in the range of [8, 10] Hz (Reference 2). Similar to the SA drive, the HGA drive is provided with microstepping capability to minimize jitter induction. The situation is complicated, however, by thermal issues involved in dissipating heat from the HGA actuators. The inner gimbal was found to be most problematic with respect to jitter-induction around the disturbance modes of interest because its axis of actuation couples more strongly to HGA boom modes. In contrast, the outer gimbal was found to be most problematic with respect to dissipation of heat, which is a greater problem if microstepping is used.

Consequently, the GN&C, HGA, and thermal teams have opted for a compromise whereby when the HGA gimbals are moving slowly (stepper rates below ~ 15 Hz, more-or-less the range used when tracking a TDRS), the inner gimbal uses microstepping and the outer gimbal uses cardinal stepping. Furthermore, in contrast to the continuous power use of SA microstepping, HGA microstepping uses a maximum pulse width of 0.3 second. For stepper rates below 3.3 Hz, the inner gimbal is only powered during part of each step, which reduces heat generation associated with microstepping. Although the inner gimbal can have higher rates when tracking a TDRS, its rates are typically within ± 5 Hz of 0, so the gimbal is often unpowered when tracking. For the high gimbal stepper rates used when the HGA is slewing between TDRSs (~ 50 Hz [selected to avoid body resonances at higher frequency]), there is little power difference between cardinal- and microstepping, so both gimbals use microstepping. Due to the use of cardinal stepping on the outer gimbal, and non-continuous power to the inner gimbal when microstepping for tracking, the jitter reduction due to microstepping is not as great for the HGA as it was for the SA. Using 0.1-second as the maximum pulse width, we found HGA-induced attitude jitter of 0.13', vs. 0.19' with cardinal stepping in both axes. Use of 0.3-second max pulse (a relatively recent change) will improve that slightly.

Although the attitude jitter imposed by the HGA stepper motors while tracking a TDRS is fairly small, the jitter imposed on HGA pointing itself is fairly substantial; we estimate $\sim 25'$ (3σ). As extra risk reduction, we have included the same two jitter-suppression options in the HGA algorithm as used in the SA control algorithm. We plan to evaluate the efficacy of these jitter-suppression options over the next few months and during orbital verification.

HGA Control Example and Quality

Figures 6-8 present an example of HGA motion from the same high-fidelity simulator run used for Figures 3-5. Figure 6 shows HGA angles over one orbit. For the simulation, three “TDRS” targets were used, spaced with separations of 120° around the equator, each with an inclination of 5° . The simulation was set up to begin the slew from one TDRS to the next when the second TDRS is closer to the zenith than is the current TDRS. Over the course of a single orbit, there are four periods of TDRS tracking, with slews at maximum rate in between. Because of the relative geometries between the GPMCO and TDRS orbits, the direction in the body frame to the currently selected TDRS ranges from near the $-Z$ direction to within $\sim 5^\circ$ of the X/Y plane. Similar to the SAs, the HGA does produce some attitude disturbance at moments of relatively large acceleration, i.e., at the start and end of transitions between TDRSs. The disturbance spikes, with peak amplitudes of $\sim 0.016'$, are lost in the background noise in Figure 5. Note, the HGA outer-link moment of inertia is about 1% of that for the SAs, whereas the HGA-imposed attitude disturbance at moments of rapid acceleration is about 5% of the disturbance imposed by the SAs. Including slew profiling and fading memory filtering similar to that used for SA control would probably have gained a factor of ~ 5 in HGA disturbance reduction, but that seemed unneeded given its already small size. As for the SAs, our time-domain simulation does not include gimbal-pulse-induced attitude disturbance. Our detailed jitter analysis shows the HGA gimbal-induced attitude disturbances can be as large as $\sim 0.13'$ from HGA gimbal pulsing in the range [0.5, 3.0] Hz, given our current cardinal- vs. micro-stepping settings.

Figures 7 and 8 show the HGA pointing error relative to the modeled position of TDRS, with 7 highlighting the error during TDRS-to-TDRS slews, and 8 highlighting TDRS tracking periods. Maximum HGA pointing error during tracking relative to the modeled TDRS direction due to the HGA controller is $\sim 0.65'$ when gimbal rates are well removed from zero, and $\sim 2.1'$ around periods of zero rate crossing. These errors are small relative to the total HGA pointing error budget, which is 105'.

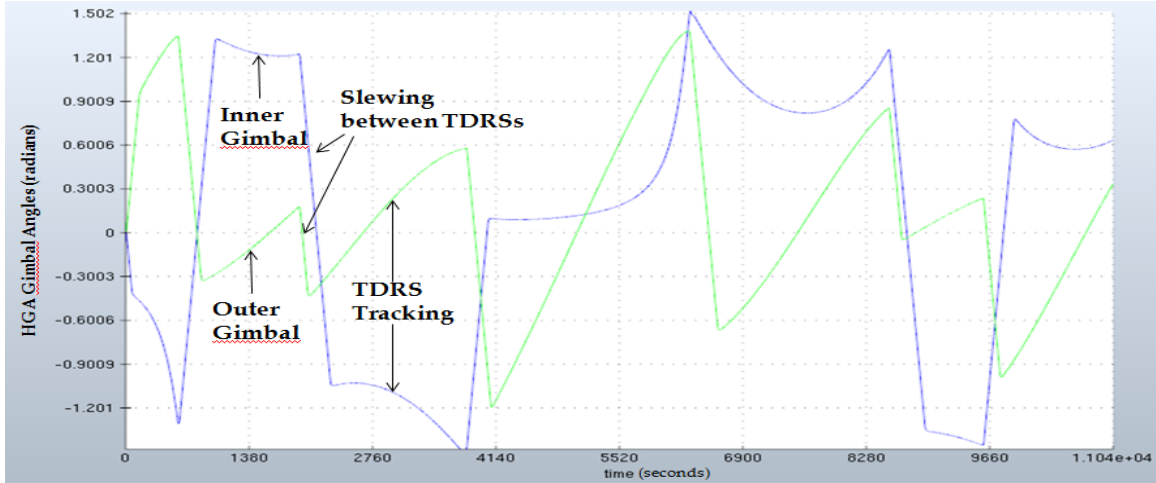


Figure 6: HGA Gimbal Angles over two orbit; $|\beta| = 0$

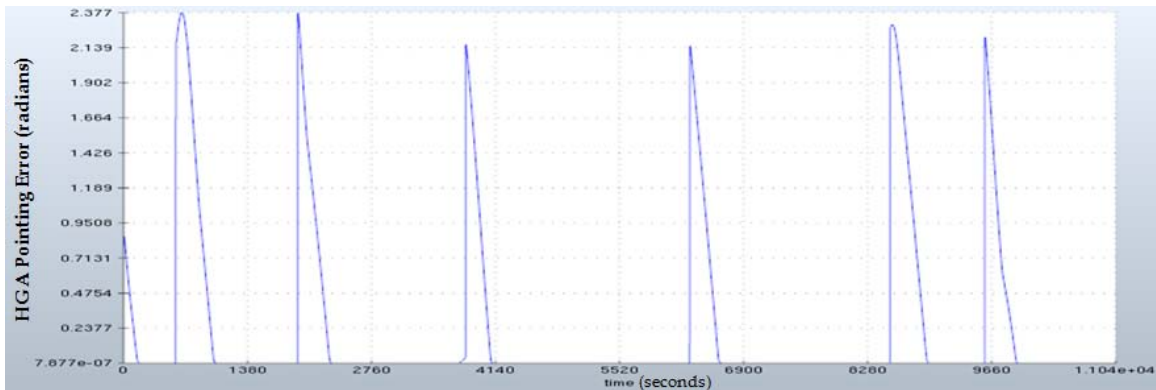


Figure 7: HGA Pointing Error (highlighting TDRS-to-TDRS slews)

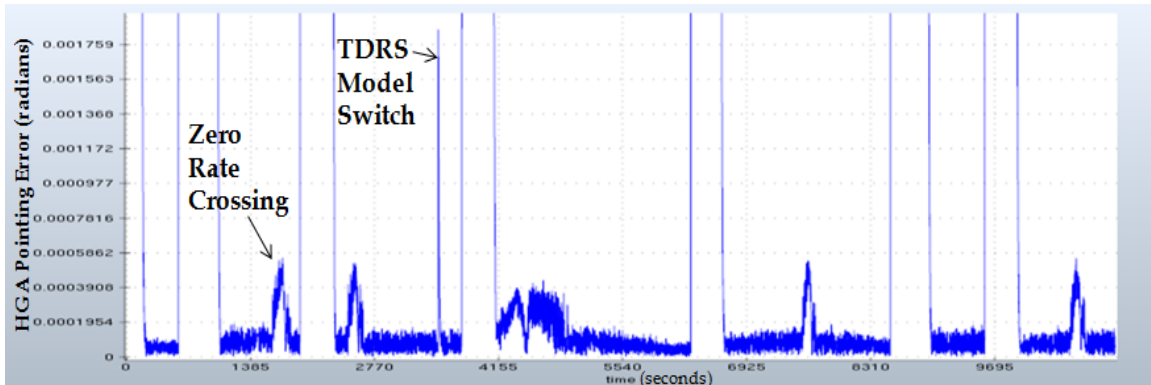


Figure 8: HGA Pointing Error (highlighting TDRS tracking periods)

Summary

We have presented a review of the GPMCO SA and HGA pointing control law algorithms. The SA controller (1) maximizes solar flux on the SAs during orbit day, subject to array offsets to avoid an excess rate of charge, (2) minimizes atmospheric drag during orbit night to reduce frequency of orbit maintenance thruster usage, (3) minimizes atmospheric drag during orbits with high solar beta angle when SA-1 is shadowed by the spacecraft and the solar flux on SA-2 is essentially independent of orientation, and (4) keeps array-induced spacecraft attitude disturbances within allocated tolerances. We estimate mean drag reduction by ~ 10 to 15%, depending on final algorithm tuning. The HGA controller (1) contributes less than 2.1' to HGA pointing error, (2) keeps HGA-controller-induced spacecraft attitude disturbances to within $\sim 0.016'$, and (3) supports adequately fast transitions between TDRSs. Regarding points 1 and 2 for the HGA controller, HGA stepper-induced HGA pointing jitter and spacecraft attitude jitter are estimated to be $\sim 25'$ and $0.13'$, respectively. Jitter suppression options in the HGA controller may allow us to reduce these jitter values, though at the cost of some increased error caused by the controller itself. We'll be evaluating this possibility over the next few months, as well as following launch in 2014.

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