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SAFEHOLD ATTITUDE DETERMINATION APPROACH FOR GPM

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Spacecraft safing designs generally have minimal goals with loose pointing requirements. Safe pointing orientations for three-axis stabilized spacecraft are usually chosen to put the spacecraft into a thermally safe and power-positive orientation. In addition, safe mode designs are required to be simple and reliable. This simplicity lends itself to the usage of analog sun sensors, because digital sun sensors will add unwanted complexity to the safe hold mode.

The Global Precipitation Measurement (GPM) Mission Core Observatory will launch into lower earth orbit (LEO) at an inclination of 65 degrees. The GPM instrument suite consists of an active radar system and a passive microwave imager to provide the next-generation global observations of rain and snow. The complexity and precision of these instruments along with the operational constraints of the mission result in tight pointing requirements during all phases of the mission. To ensure the instruments are not damaged during spacecraft safing, thermal constraints dictate that the solar pointing orientation must be maintained to better than 6.5 degrees. This requirement is outside the capabilities of a typical analog sun sensor suite, primarily due to the effects of Earth's albedo. To ensure mission success, a new analog sensor, along with the appropriate algorithms, is needed.

This paper discusses the design issues involving albedo effects on spacecraft pointing and the development of a simple, low-cost analog sensor and algorithm that will address the needs of the GPM mission. In addition, the algorithms are designed to be easily integrated into the existing attitude determination software by using common interfaces. The sensor design is based on a heritage, commercial off-the-shelf analog sun sensors with a limited field-of-view to reduce the effects of Earth's albedo. High fidelity simulation results are presented that demonstrate the efficacy of the design.

INTRODUCTION

The Global Precipitation Measurement Core Observatory (GPMCO), scheduled to launch in early 2014, is part of an international network of satellites that provide the next-generation global observations of rain and snow. The GPMCO is a joint NASA/JAXA spacecraft that is carrying both a Dual-frequency (Ku/Ka) Precipitation Radar (DPR) and a spinning passive GPM Microwave Imager (GMI). The spacecraft is designed with a 3 year mission requirement and a 5 year

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goal. It will be launched into a near circular orbit with an altitude of 400km and an inclination of 65 degrees.

The spacecraft will deploy two large solar array wings and a high-gain antenna once on orbit (Figure 1). This makes the GPM Core Observatory a large spacecraft with nearly 3600 kg of launch mass and deployed dimensions of roughly 2.5m in length, 2m in height, and 6m tip-to-tip.

The attitude control system (ACS) consists of: redundant star trackers, inertial rate unit, and GPS receivers for mission science pointing, magnetometers and magnetic torque bars for momentum management, and an array of coarse sun sensors for safehold. Reaction wheels are used in most modes for controlling spacecraft attitude and the spacecraft also has forward and aft thrusters that are nominally used for orbit maintenance.

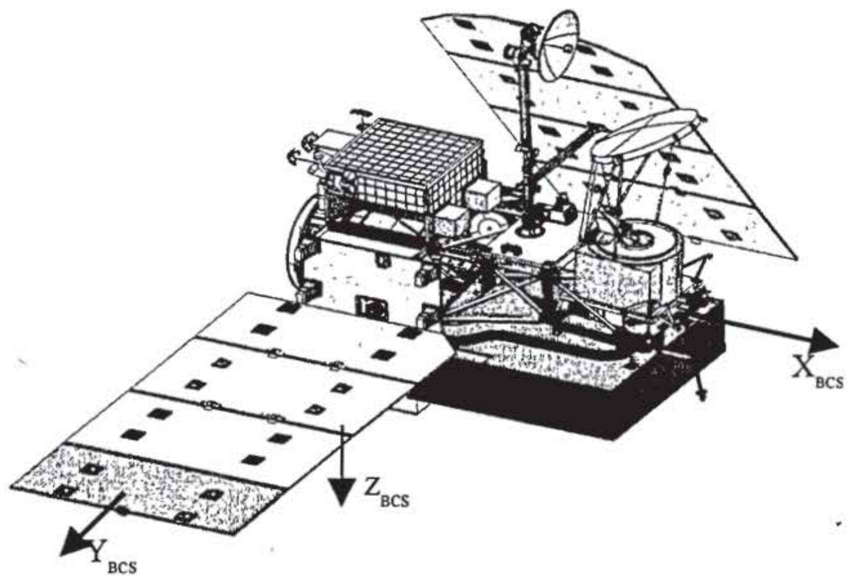

Figure 1. GPM Core Observatory Layout.

SAFEHOLD DESIGN REQUIREMENTS

To protect the spacecraft against hardware faults and to have a nominal starting mode upon power-on, a Safehold mode has been implemented. This mode is intended to put the spacecraft into a power-positive and thermally safe attitude. Best practices dictate that this design be simple and practical, employing only the minimum set of hardware needed to meet the requirements.[1] The design must also be reliable and should limit the use of complex algorithms and onboard models.

For the GPMCO the thermal requirements drive the Safehold design. The instruments and their heaters are not powered during this mode. The spacecraft must be positioned to allow even heating across the two radars. However, there are certain areas of the DPR and the GMI that are exclusion zones for prolonged exposure to sunlight which also must be considered. These restrictions result in relatively tight requirements for the Safehold design.

To meet the thermal requirements, the spacecraft attitude in Safehold is required to be controlled such that the Sun is positioned at a positive pitch angle of 24 degrees from the spacecraft X-Y plane (i.e. the Sun is in the +X /+Z quadrant). This orientation is to be held to within 5 degrees (3σ) indefinitely. Spacecraft pointing knowledge accuracy is also required to be better than 3 degrees, which is much tighter than previous missions. As will be demonstrated later in this paper, this knowledge requirements prevent the usage of typical coarse sun sensors. In addition, due to the scan boresight of the GMI, there are limited locations on the spacecraft where additional sensors can be located.

CENTER OF IRRADIANCE

The coarse sun sensors (CSS) used on many spacecraft as part of the ACS are essentially cosine response sensors with a near hemispherical field-of-view (FOV). The sensor output is approximately proportional to the cosine of the angle between the vector normal of the sensor and the vector from the sensor to the sun. The sensor produces a current proportional to the irradiance (energy flux) incident on its surface. Generally, the ACS attempts to orient the spacecraft such that the maximum flux is aligned along a specific axis, thus minimizing the angle. Sources other than the sun, which lie within the field-of-view of the sensor, can also introduce energy to the sensor's solar cell. For an Earth-orbiting spacecraft, those sources could be other orbital bodies (such as the Moon) or reflections from the spacecraft itself. Generally, however, the largest source (other than the Sun itself), is Earth's albedo.

Albedo is measured as a percentage of the incident light that is reflected. In low Earth orbit, the mean albedo over the surface of the Earth is approximately 30%, of which the gross majority is diffuse. The energy from the diffuse albedo and the Sun are additive in the sensor's response. Thus, the sensor output is actually proportional to the center of irradiance of all sources.

The effect on attitude determination from the sensors can be determined by summing the dot products of the sources. A simplified approximation using only the Sun and Earth albedo yields:

$$I_n = \underline{n} \cdot \underline{I}_A + \underline{n} \cdot \underline{I}_S \tag{1}$$

The angle between the vectors of the flux due to the sun ($\underline{I}_S$) and the sensor normal ($\underline{n}$) is defined as alpha (α). The angle between the vectors of the flux due to albedo ($\underline{I}_A$) and the sensor normal is defined as beta (β). The total flux at the sensor face (I_n) is equal to the sum of these two dot products. Another useful quantity in this analysis is the angle between ($\underline{I}_S$) and ($\underline{I}_A$), which is defined as gamma (γ) (illustrated in Figure 2, bottom left). Continuing the derivation of Eq. (1), along with the assumption that albedo is 30% of the solar irradiance:

$$I_n = I_A \cos\beta + I_S \cos\alpha = I_S(0.3\cos\beta + \cos\alpha) \tag{2}$$

For a given angle between the sun and albedo vectors (γ), plotting the variation in solar flux versus solar incident angle can give insight into the effect of albedo on attitude determination (Figure 2). As previously stated, the spacecraft ACS attempts to align a desired axis with the maximum flux. If no albedo existed, the theoretical pointing angle to the Sun would be zero. However, as shown in this figure, the influence of albedo moves the maximum flux off of the zero sun angle. The dotted line links the flux maxima.

Looking another way at this result, the maxima can be plotted as the Sun-albedo angle (γ) versus the Sun angle (α) (Figure 3). This plot visually shows the errors due to albedo on attitude determination. From this figure, the maximum error due to 30% albedo is around 17.5 degrees. It can also be determined from this figure that limiting (γ) would effectively drive down the errors. To achieve the the required pointing of 5 degrees, the albedo-Sun angle must be limited to 21.9 degrees. For sun sensors, the most effect way to limit (γ) is to limit the total sensor FOV.

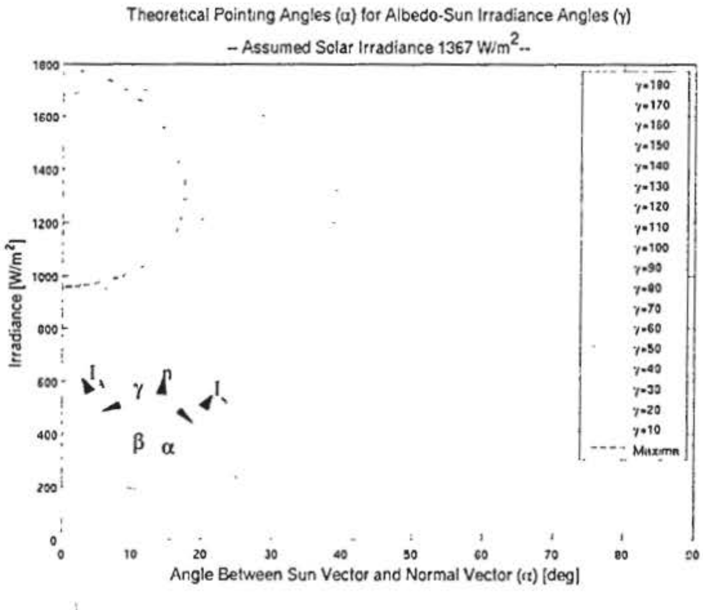

Figure 2. Albedo Influence on Pointing.

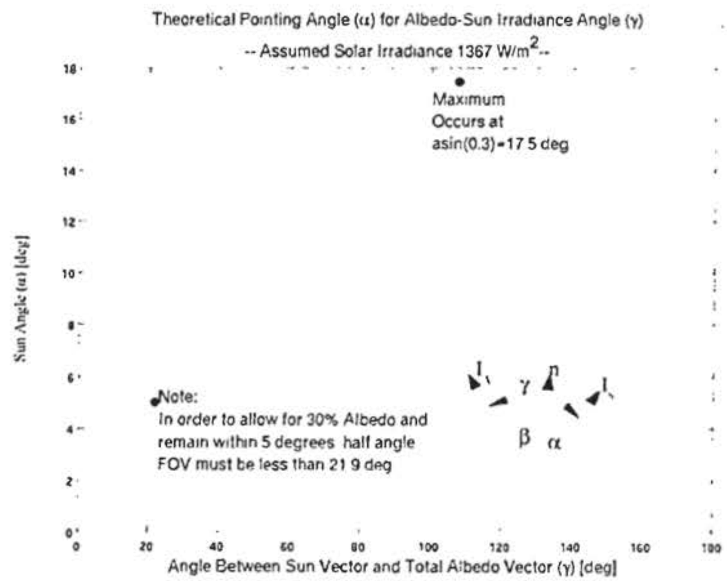

Figure 3. Attitude Determination Errors due to Albedo.

SOLUTION FOR GPM

CSS are hemispherical, relatively low accuracy, and single axis. Also, the design of the devices does not lend itself to easily restricting the FOV. For the GPMCO, the need was for a low cost, mid-level accuracy, two axis analog sun sensor with high reliability and flight heritage. In addition, the new strict FOV requirement had to be met. A survey of current technology yielded no COTS hardware that could meet all requirements. Thus, a modification to a heritage design was needed.

The Adcole Coarse Analog Sun Sensor (CASS) is a single-axis, analog sun sensor with an accuracy of 0.75 degrees. Two units at right angles were used to get the required two measurement degree-of-freedom. Baffles were added to the sides of the sensors to restrict the FOV to 17.5 degrees. The combined sensor suite has been dubbed the GPMCO Medium Sun Sensor (MSS) designed my Adcole (Figure 4).

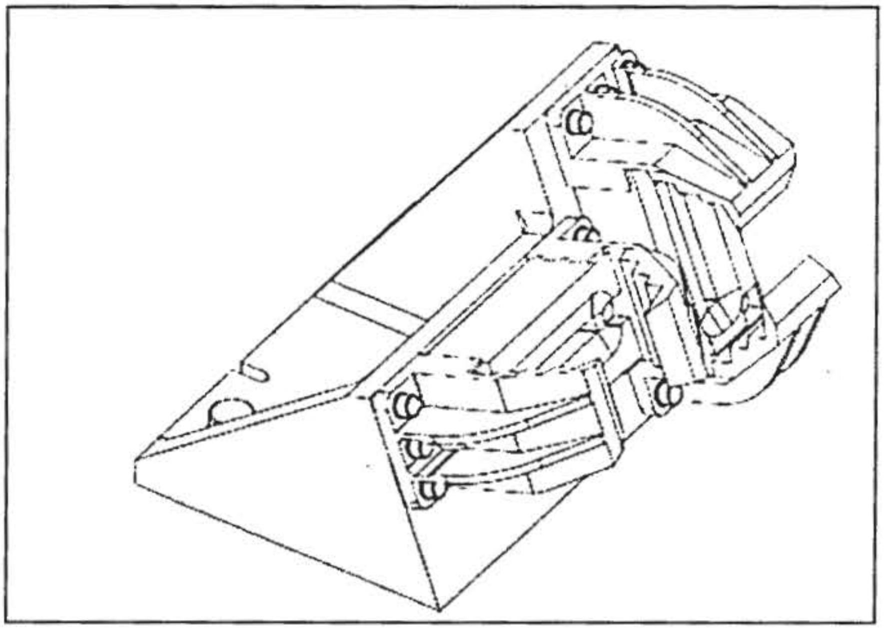

Figure 4. Isometric View of MSS on Mounting Bracket.

Each CASS outputs two analog signals corresponding to the sensitive and insensitive axes of the unit. Since two units are used in one MSS, a total of four analog signals are used for determining the Sun vector orientation. First, the Sun angle from each CASS is determined based on the manufacturer's calibration information. Since the MSS is only used within 17.5 degrees, a small angle approximation is used to determine the sun vector in the MSS coordinate system.

$$\underline{u}_{S,MSS} = \begin{bmatrix} \theta_X & \theta_Y & \sqrt{1-\theta_X^2-\theta_Y^2} \end{bmatrix}^T \qquad (3)$$

This vector is then transformed into the body coordinate system and either filtered to reduce noise or sent to the sensor selection logic (Figure 5). The suite of 12 redundant CSS's on the spacecraft has a field-of-view of 4π steradian. Thus, when the sun is visible to the spacecraft, a CSS derived sun vector is always available. The MSS can only compute a sun vector when both

CASS units have the Sun in their respective FOV. This fact drives the selection logic of the ACS. If the output of the MSS is above a certain threshold for *both* CASS measured angles, the MSS derived sun vector is used. This logic drives the MSS FOV toward the sun using the CSS on the spacecraft and turns the attitude determination over to the MSS for finer pointing.

The remaining controller logic is the same regardless of the sun vector used. The estimated vector is crossed with the desired vector to determine the position error. A proportional gain is applied to this error and added to a derivative gain applied to rates from a gyro to determine desired torques. These torques are filtered and passed to the reaction wheel assembly (RWA) controller.

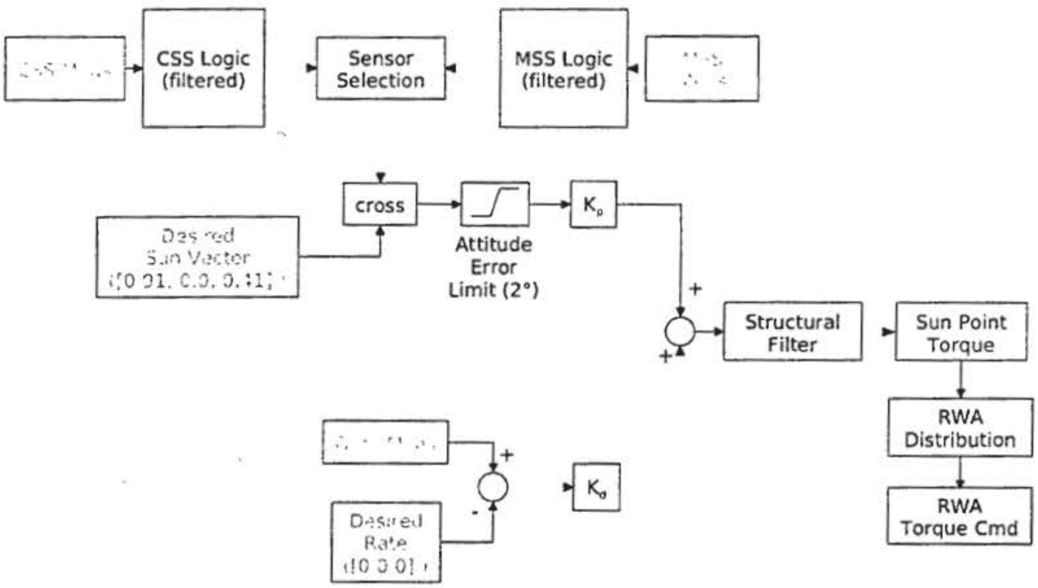

Figure 5. Safehold Block Diagram.

PERFORMANCE

To analytically test the sensor implementation, a high-fidelity simulation was developed. The code of choice in this study is an in-house guidance navigation & control (GN&C) simulation tool-set called *Freespace,* which was developed at NASA Goddard Flight Center. This C-based code takes into account multi-state dynamics of the spacecraft with a variable-step integrator. Spacecraft environmental models are also included. For gravity and geomagnetics, spherical models are used.[2] Solar pressure and aerodynamic torque models use the orientation of the large solar arrays in combination with spacecraft body physical dimensions. All these models are rigorously tested and compared to data collected by various spacecraft that are currently orbiting the earth. These models together result in a realistic depiction of the environment seen by the spacecraft.

The albedo model used in *Freespace* was developed by Thomas W Flatley, in which the Earth is discretized and the algorithm calculates solar radiant energy from the diffused reflectance of the Earth[3]. There are several assumptions used in this model such as including diffuse reflectance

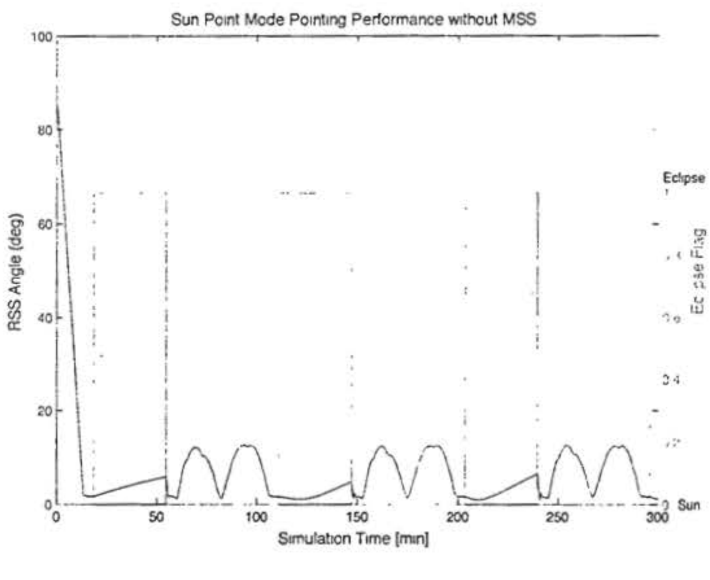

Figure 6. Pointing performance without the use of the MSS.

Figure 7 demonstrates the effect of the limited FOV of the MSS. For comparison reasons we use the same initial conditions for this simulation. We see that the spacecraft starts 90 degrees away from the sun and starts to rotate toward the target. At about 50 minutes into the simulation, the spacecraft is in sunlight and the MSS blocks out the albedo effects and keeps the spacecraft on the Sun. The pointing errors in this simulation are well under 2 degrees.

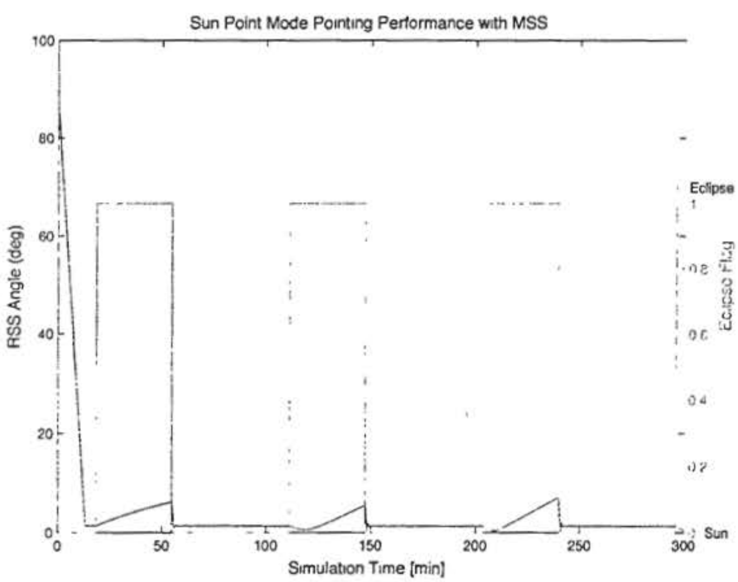

Figure 7. Pointing performance with the use of MSS.

Figure 8 plots the controller error from the simulation. About 10 minutes into the simulation there is a discontinuous jump in the measurement error, when the MSS transition from CSS occurs. We see that the error is very small with spikes in the error as we are exiting out of eclipse. This is due to the spacecraft controller off only using gyros with no position information and then getting knowledge of the sun and correcting for it.

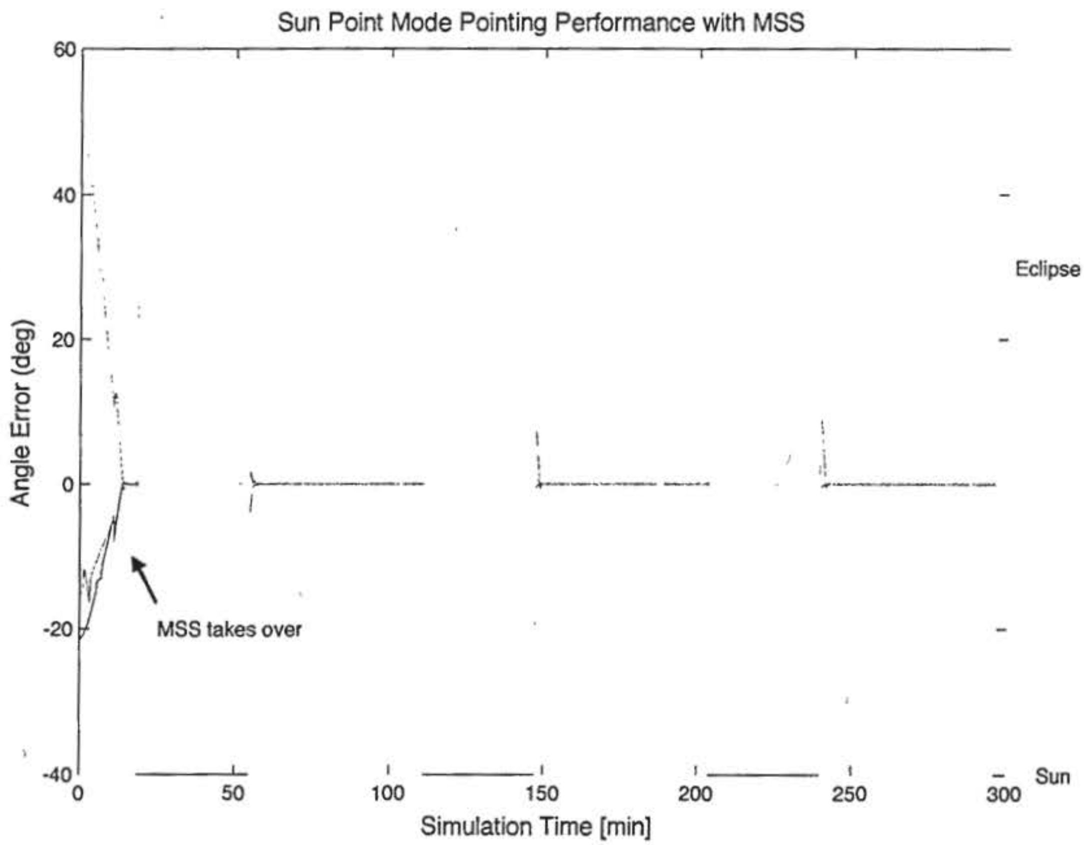

Figure 8. Angular Error when MSS is in use

CONCLUSION

Safe mode design issues involving albedo effects on spacecraft pointing were discussed. To solve these challenges, a detailed solution was presented that implements a simple, low-cost sensor and algorithm design. The implementation of the algorithm easily integrates into the existing attitude determination software design by using common interfaces. The sensor design uses augmented heritage COTS analog sun sensors with a limited field-of-view to reduce the effects of Earth's albedo. A high fidelity simulation was set-up to test the proposed design. Simulation results show an improvement in attitude pointing of over 10 degrees when the new design is implemented. This study demonstrates a simple and effective method for limiting the effects of albedo in spacecraft safe mode solar pointing.

REFERENCES

[1]*Goddard Space Flight Center Rules for the Design, Development, Verification, and Operation of Flight Systemss.*GSFC-STD-1000E, 2009, Rule 1-17.

[2]Wertz, James R. ed. *Spacecraft Attitude Determination and Control*, D. Reidel Publishing Company. Hingham, MA. 1978.