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Aerothermal Analysis and Design of the Gravity Recovery and Climate Experiment (GRACE) Spacecraft

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April 2000

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Abstract

The Gravity Recovery and Climate Experiment (GRACE) primary mission will be performed by making measurements of the inter-satellite range change between two co-planar, low altitude, near-polar orbiting satellites. Understanding the uncertainties in the disturbance environment, particularly the aerodynamic drag and torques, is critical in several mission areas. These include an accurate estimate of the spacecraft orbital lifetime, evaluation of spacecraft attitude control requirements, and estimation of the orbital maintenance maneuver frequency necessitated by differences in the drag forces acting on both satellites. The FREEMOL simulation software has been developed and utilized to analyze and suggest design modifications to the GRACE spacecraft. Aerodynamic accommodation bounding analyses were performed and worst-case envelopes were obtained for the aerodynamic torques and the differential ballistic coefficients between the leading and trailing GRACE spacecraft. These analyses demonstrate how spacecraft aerodynamic design and analysis can benefit from a better understanding of spacecraft surface accommodation properties, and the implications for mission design constraints such as formation spacing control.

Introduction / Mission Synopsis

This paper describes aerothermal analyses and design recommendations for the Gravity Recovery and Climate Experiment (GRACE) spacecraft. The following paragraphs provide an introduction to the GRACE mission and it's relevance to the GRACE spacecraft design criteria.

The GRACE mission is a joint project between the National Aeronautics and Space Administration (NASA) and the Deutsches Zentrum für Luft und Raumfahrt (DLR). Overall responsibility for implementation of GRACE rests with Prof. Byron Tapley, Principal Investigator, of the University of Texas at Austin, Center for Space Research (UTCSR), and Prof. Christoph Reigber of the GeoForschungsZentrum (GFZ) Potsdam, Co-Principal Investigator. GRACE is sponsored by the Jet Propulsion Laboratory (JPL), and involves an international U.S.-German design and development team. The primary product of the GRACE mission is a new model of the Earth's gravity field with unprecedented accuracy every 15 to 30 days for a period of five years.

The gravity field of the Earth is variable in both space and time. The primary objective of the GRACE mission is to obtain accurate global models for the mean and time variable components of the Earth's gravity field [1]. This objective will be achieved by making accurate measurements (micron-level precision) of the inter-satellite range change between two co-planar, low altitude, near-polar orbiting satellites, using a K-Band microwave tracking system. In addition, each satellite will carry a geodetic quality Global Positioning System (GPS) receiver and a high accuracy accelerometer to enable accurate orbit determination, spatial registration of gravity data, and the estimation of gravity field models.

The Earth gravity field estimates obtained from data gathered by the GRACE mission will provide, with unprecedented accuracy, integral constraints on the global mass distribution and its temporal variations within the Earth system. These improved estimates, in conjunction with other in-situ and satellite data, as well as geophysical models, will provide impetus for advances in a wide variety of Earth System Science disciplines such as oceanography, continental hydrology, glaciology, and solid Earth sciences and geodesy.

The GRACE satellites will be launched on-board a ROCKOT launch vehicle from Plesetsk, Russia on June 23, 2001. The third stage re-ignitable BREEZE will place both the satellites in the same nominal circular orbit of approximately 500 km at an inclination of 87 deg. The GRACE mission will be launched just after the solar flux maximum of cycle 23. Following Launch and Early Orbit Phase (LEOP) operations, the orbits of the two satellites will evolve naturally for the remainder of the mission. During the science data collection, the two GRACE satellites (nominally Earth oriented) will point their K-Band feed horns towards each other to a high precision.

Over the mission lifetime the two satellites will remain in coplanar orbits. Due to differential drag force, the along-track separation will vary, and station-keeping maneuvers will be required to keep the two satellites within 170 to 270 km of each other. Due to the interruption of science data collection during the maneuvers, it is desirable to perform these maneuvers as infrequently as possible. The GRACE mission goal for station-keeping maneuver frequency is 60 days. To ensure uniform environmental exposure and aging of the two satellites, the leading and trailing satellites will nominally exchange positions once during the mission. Additionally, certain in-flight calibration maneuvers may require thruster system activation, and the satellites may be subject to reboost maneuvers if deemed necessary for increasing orbital lifetime.

There are several key design areas that are benefited tremendously by an accurate understanding of the disturbance environment in which the GRACE spacecraft will be operating. These include an accurate estimate of the spacecraft orbital lifetime, evaluation of spacecraft attitude control requirements, and thruster propellant consumption for attitude control, as well as orbital maintenance maneuvers necessitated by differences in the drag forces acting on the two co-orbiting satellites.

The two spacecraft are designed to be physically identical to minimize the cost of design and manufacturing. The K-Band horns, with their centerline aligned with the X-axis of each spacecraft, must be pointed towards one another to make the scientific measurements. This requires that the leading satellite orbit with the aft end facing into the velocity direction. Moreover, it implies that both spacecraft be pitched slightly (about -1 degree) to achieve the proper line-of-sight for the measurements. This results in more of the upper solar array surfaces exposed to the free molecular flow environment on the trailing satellite and more of the lower radiator surface exposed on the leading satellite. Due to the uncertainties in various aspects of modeling the disturbance environment, it is very important to accurately characterize the aerodynamic properties of the spacecraft and be able to bound the uncertainties to assure mission success.

Satellite Configuration Description

The GRACE satellites are derived from the German CHAMP satellite design, which is primarily a magnetic measurement mission with similar orbit characteristics scheduled for launch in the spring of 2000. The overall dimensions of each GRACE satellite are approximately 3.1 x 1.9 x 0.7 m (length x width x height) with a mass of 425 kg. Each satellite is controlled by a cold gas nitrogen thruster system, which is supplemented by magnetic torque rods. Each GRACE spacecraft utilizes three dual winding 30 Amp-m^2 magnetic torque rods to supplement

the cold gas reaction control system (one rod aligned with each spacecraft body axis). Twelve 10 mN thrusters are located to nominally provide coupled attitude control torques. The attitude control thrusters are capable of providing 0.029 N of torque about the Y and Z-body axes (pitch and yaw) and 0.006 N about the X-body axis (roll). The attitude control thrusters can also operate in an uncoupled mode, which provides half the control torque about each axis. Two 40 mN orbit maintenance thrusters are located on the aft face of each satellite, oriented such that the thrust direction is through the spacecraft center-of-mass. Each satellite contains 32 kg of gaseous nitrogen propellant (GN_2) to provide all propulsive capability during the five-year mission.

Three major surface types dominate the external surface area of the spacecraft. These surfaces and several key parts of the GRACE satellite are shown in Figs. 1 and 2. The upper surface consists primarily of solar arrays, and the lower surface is a radiator with a Teflon coating. Included on the zenith solar array panel is additional hardware including the GPS Precision Orbit Determination (POD) antenna. The forward and aft surfaces are machined from Carbon-fiber Reinforced Plastic (CFRP) sandwich panels. These panels will be covered with a protective Kapton foil, as will the K-Band horn aperture. The additional GPS antennas located on the aft panel will not be covered; they provide small geometrical and surface differences with respect to the forward panel. The above three surfaces, along with the corrugated star camera baffles and various aluminum surfaces, make up the major portions of the GRACE spacecraft. For the analyses presented in this paper, the surface elements were grouped into these five major surface types. The remaining surfaces are minor contributors to the surface area and were assumed to have the same surface properties as the aluminum surfaces.

FREEMOL Software

This section provides a brief overview of the free molecular analysis software, FREEMOL [2]. This code was utilized for all GRACE aerothermal analyses performed to date. FREEMOL was developed by the NASA Langley Research Center Spacecraft and Sensors Branch and Analytical Mechanics Associates, Inc. More detailed descriptions of the implementation methodology are provided in subsequent sections.

The FREEMOL software allows the user to simulate a single 3-Dimensional geometric spacecraft in a low Earth orbiting, free molecular flow environment for multi-year mission duration. The high fidelity aerodynamics are formulated using normal and tangential accommodation coefficients, which are mathematically modeled as empirical functions of the energy of the impacting molecules and the angle of incidence between the surface elements and the relative velocity vector. The aerodynamic analysis provides for air molecule accommodation, re-emmittance, reflection, blockage and shadowing with respect to the relative wind, and incorporates finite speed ratio effects. Multiple reflections of the molecules are neglected. The forces and torques acting on the spacecraft can be evaluated more accurately in this manner compared to projected area methods with scaling using drag coefficients.

A high fidelity solar radiation pressure model is also incorporated which determines blockage and shadowing with respect to the Sun, along with diffuse and specular reflection and absorption.

FREEMOL allows the user to select one of several atmospheric density models, which incorporate day of year, altitude, latitude, longitude, seasonal and solar hour angle dependence, diurnal bulge, and rotating atmosphere effects.

A global horizontal wind model is included which calculates typical wind conditions as well as transient geomagnetic storm conditions. Statistical estimates of the solar flux and

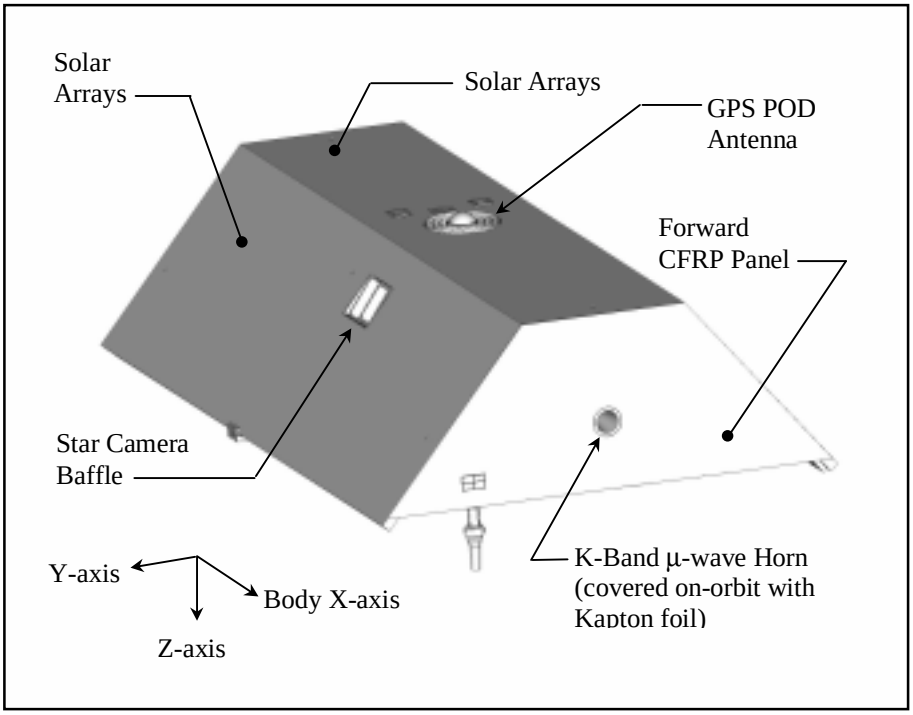

Figure 1. Forward/Zenith View of the GRACE Satellite.

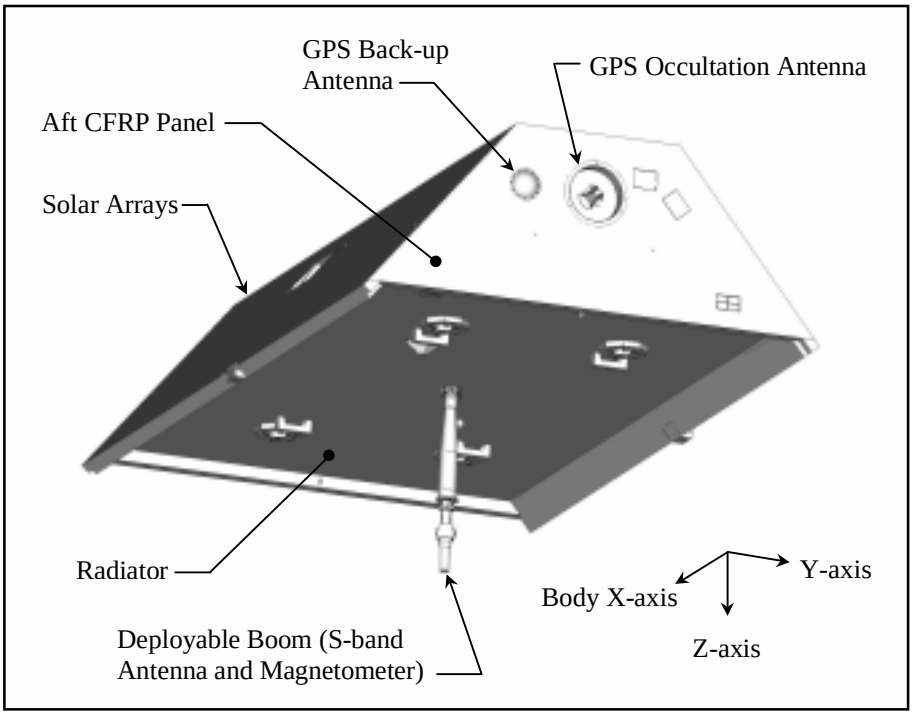

Figure 2. Aft/Nadir View of the GRACE Satellite.

geomagnetic index values are used to evaluate the atmospheric and global wind conditions, and are updated continuously throughout the simulation.

FREEMOL uses a high fidelity 8th order magnetic field model, together with magnetic hysteresis models for simulating angular rate damping, as well as magnetic torque rod models. The GRACE mission attitude control system consists of cold gas thrusters and magnetic torque rods. Analysts specify initial values of spacecraft orientation and angular velocity relative to a Local Vertical-Local Horizontal (LVLH) reference frame.

The analysis software accounts for time-varying orbital motion such as altitude decay, nodal regression due to Earth oblateness, seasonal solar geometry, and Earth occultation and rotation.

FREEMOL provides time histories of spacecraft attitude, altitude decay profile, acceleration data, external and control torques, propellant consumption, angular velocity, atmospheric conditions, and orbital parameters. FREEMOL was originally written to predict and numerically confirm the feasibility of passive aero-stablization [3,4] for the Passively Aerodynamically Stabilized Magnetically Damped Satellite (PAMS). The PAMS experiment provided the first flight validation of the FREEMOL software.

Disturbance Environment

Understanding the disturbance environment in which the GRACE satellites will operate is critical to the design and analysis of the satellite system. Analysts must obtain an accurate estimate of the spacecraft’s orbital lifetime, evaluate spacecraft attitude control requirements in terms of control authority and propellant consumption, and estimate station-keeping requirements in terms of maneuver frequency and propellant consumption.

The drag force acting on the GRACE satellite is dominated by aerodynamics. Solar pressure is not a significant contributor to the drag force acting on the GRACE configuration. Aerodynamic torques are a major contributor to the overall disturbance torques acting on the satellite, as are gravity gradient torques when the pitch angle of the spacecraft changes appreciably from zero, i.e., from LVLH. However, the solar torques can also be significant at the higher altitudes and especially during periods of low solar activity, when solar pressure can dominate the aerodynamic pressure. Residual spacecraft magnetic moments were assumed to be negligible and hence were not analyzed. The GRACE magnetic torque rods can effectively handle the pitch axis gravity gradient, aerodynamic, and solar torques. The aerodynamic drag forces, and the X and Z-body aerodynamic torques acting on the GRACE satellites are key drivers in the design of the GRACE attitude and orbit control system (AOCS) due to the limited propellant budget and requirements for attitude control and dual satellite station-keeping.

There are several important factors that affect the aerodynamic forces and torques acting on the satellites. The atmospheric density is an important contributor to the aerodynamic drag. The Earth's thermosphere (90 - 500 km altitude range) is driven by energy received from the Sun. It expands and contracts during the 11-year solar cycle as well as during the much shorter duration fluctuations in solar activity (i.e., solar flares). Atmospheric density models rely primarily on two measured solar activity indices, the 10.7 cm wavelength solar flux ($F_{10.7}$), and the geomagnetic index (A_p). These quantities are measured terrestrially and directly relate to the Sun's extreme ultraviolet energy output levels. Since the atmospheric density decreases exponentially with the altitude above the Earth's surface, the operating orbit is extremely important for evaluating the aerodynamic conditions. Additionally, the relative wind direction is constantly changing due to the rotating Earth's atmosphere and the global thermospheric wind variations that dominate the polar regions of the atmosphere. Fig. 3 shows

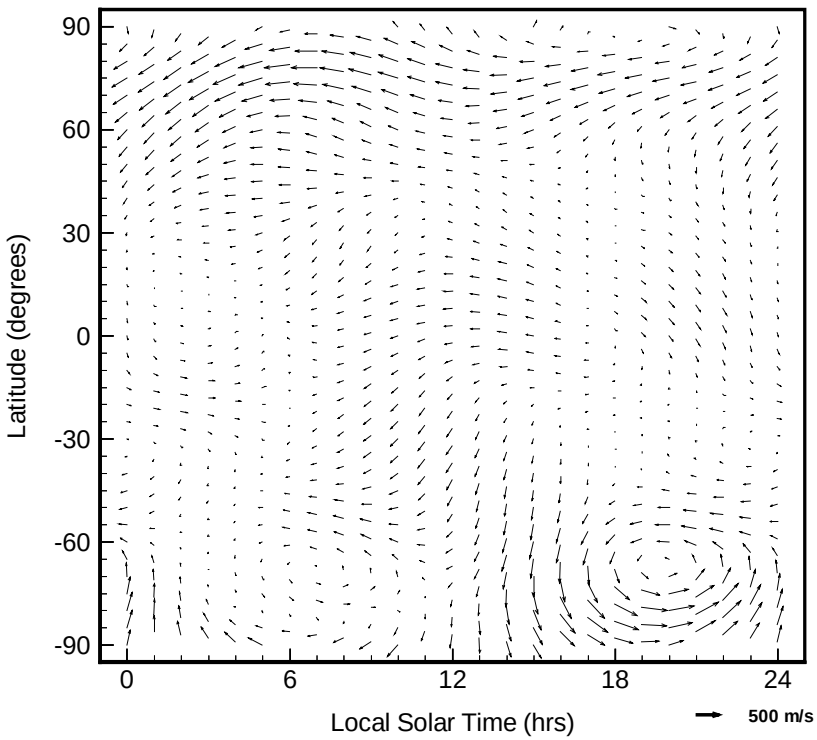

Figure 3. Example of Global Horizontal Winds.

an example of the global winds, which are modeled in FREEMOL using the model developed by Al Hedin (NASA Goddard Space Flight Center)[5]. Near the poles, these winds can reach speeds in excess of 1000 m/s during periods of intense solar activity. With a GRACE orbital velocity of approximately 7700 m/s, the global winds can significantly alter the relative wind direction, and hence the direction which the atmospheric molecules collide with the satellites.

The satellite's geometric shape and mass properties greatly influence the aerodynamic and solar forces and torques acting on the spacecraft, as well as the gravity gradient torques. The GRACE configuration referred to as GRACE DSS-A was used for all the analysis described in this paper. The satellite's center-of-mass will be calibrated on the ground and will be adjusted on-orbit with a maximum error of only ±400 μm along any axis direction with respect to a target location coincident with the center-of-mass of a high precision accelerometer proof-mass located on each satellite. Coincidence of the two centers of mass is critical since the accelerometer will ultimately measure the non-gravitational forces acting on the GRACE satellite, thus enabling the extraction of the gravity field measurements. A ±400 μm uncertainty in the center-of-mass location translates to about a ±1% uncertainty in the aerodynamic torques for the GRACE satellite, which is negligible.

Finally, gas-surface interaction for the external surfaces of the GRACE satellite has a critical effect on the aerothermal characteristics of the satellites. There currently exists no extensive on-orbit experimental data that provides a great deal of certainty about the specifics of these gas-surface interactions. These interactions are modeled in FREEMOL as a set of surface accommodation coefficients, which are a function of surface properties, incidence angle and gas energy.

Aerodynamic Force and Torque Modeling

The altitude decay of the two spacecraft, and thus the long-term orbital evolution, is determined principally by the drag accelerations acting on the satellites. This is particularly important due to the relatively low operating altitude and the launch of the satellites near the time of the next solar maximum in 2001.

The drag environment acting on the GRACE satellite is evaluated using a pre-processor code of the FREEMOL software that accounts for shadowing and finite speed ratio effects [6]. The surface of the GRACE satellite is designed in a Computer Aided Design (CAD) environment and then discretized into many small triangular facets. In order to calculate the force on each facet, accommodation coefficients (i.e., normal (σ_n) and tangential (σ_t) accommodation coefficients as functions of gas energy and incidence angle) must be evaluated.

Nominal accommodation coefficient values were obtained from experimental results [7] by Knechtel and Pitts at NASA Ames Research Center. These experiments were conducted for nitrogen ions impacting on aluminum between ion energies of 9 and 40 eV, and incidence angles of 15 to 75 deg. For the altitude of interest for GRACE, atomic oxygen is the major contributor to the atmosphere with an activation energy of approximately 5 eV. Knechtel and Pitts also provide extrapolations at these lower energies and incidence angles between 0 and 90 deg.

Empirical formulae resulting from these experiments show that the coefficient of drag for a spherical shaped satellite is about 1.68, which is much lower than the "conventional" value of 2.2. It should also be noted that several density models are based upon satellite drag measurements assuming a coefficient of drag of 2.2. Therefore, if the coefficient of drag of a spherical satellite were only 1.68, then the density estimated from drag measurements would be under-predicted by approximately 31%. To be consistent with the atmospheric density models being used, the tangential accommodation coefficient profile as a function of incidence angle was modified to yield a coefficient of drag of 2.2 for a spherical satellite. The rationale for changing the tangential accommodation coefficients rather than the normal accommodation coefficients are:

- Trends of normal accommodation results from Knechtel and Pitts are easier to corroborate via classical collision models than the tangential accommodation coefficients.

- It appears that the experimental apparatus is better suited to evaluate the normal accommodation coefficients rather than the tangential accommodation coefficients at small incidence angles.

Fig. 4 shows the experimental accommodation coefficients predicted by Knechtel and Pitts (K&P) based on empirical formulae obtained after curve fitting the experimental results, together with the calibrated coefficients.

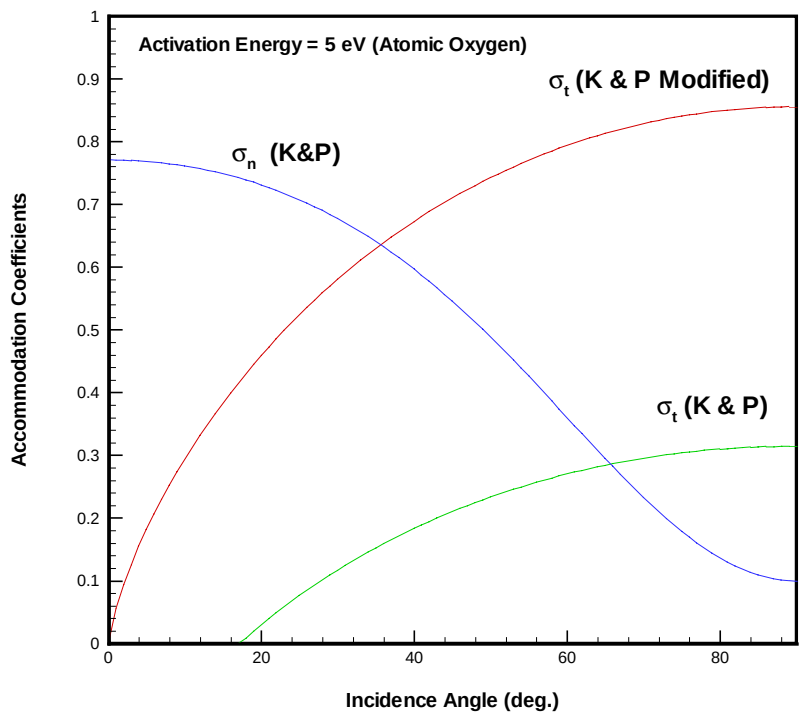

Figure 4. Nominal Normal and Tangential Accommodation Coefficients Based on Knetchtel and Pitts (K&P) Experimental Results (Ref. 6) and Tangential Curve Modified to Yield a C_D = 2.2 for a Spherical Satellite.

Figs. 5 and 6 show a comparison of the altitude decay of two satellites: the German GFZ-1 (GeoForschungsZentrum) satellite and NASA's LDEF (Long Duration Exposure Facility), respectively. These simulations utilized the atmospheric densities predicted by the Marshall Engineering Thermosphere (MET) model [8] and daily 10.7 cm wavelength solar flux ($F_{10.7}$) and geomagnetic index (A_p) data provided by the University of Texas at Austin, Center for Space Research (UTCSR). The curves shown are based on the experimental and calibrated accommodation coefficients, along with actual orbital tracking data. Fig. 5 shows that the calibrated coefficients match the decay profile much better than the experimental coefficients. Additionally, since GFZ-1 is spherical in shape, scaling the density by a factor of 1.31 yields the same results as the calibrated coefficients. However, for a cylindrically shaped satellite like LDEF, Fig. 6 shows that the calibrated coefficients predict the actual decay more closely than simply scaling the atmospheric density. The calibrated accommodation coefficients shown in Fig. 4 are termed "nominal" for all analyses discussed in this paper.

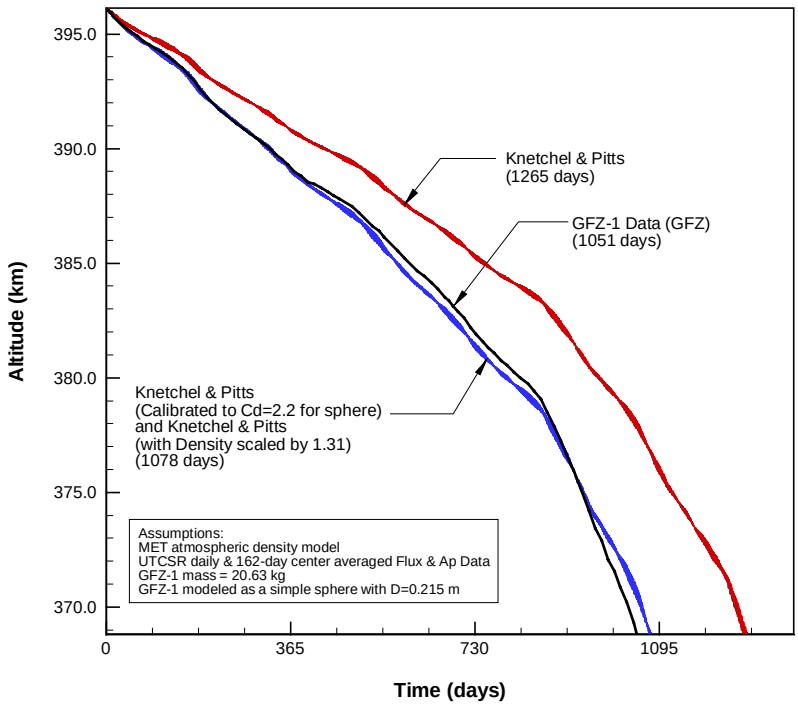

Figure 5. Orbital Lifetime Comparison of Accommodation Coefficient Assumptions and Density Scaling with GFZ-1 Altitude Decay Profile.

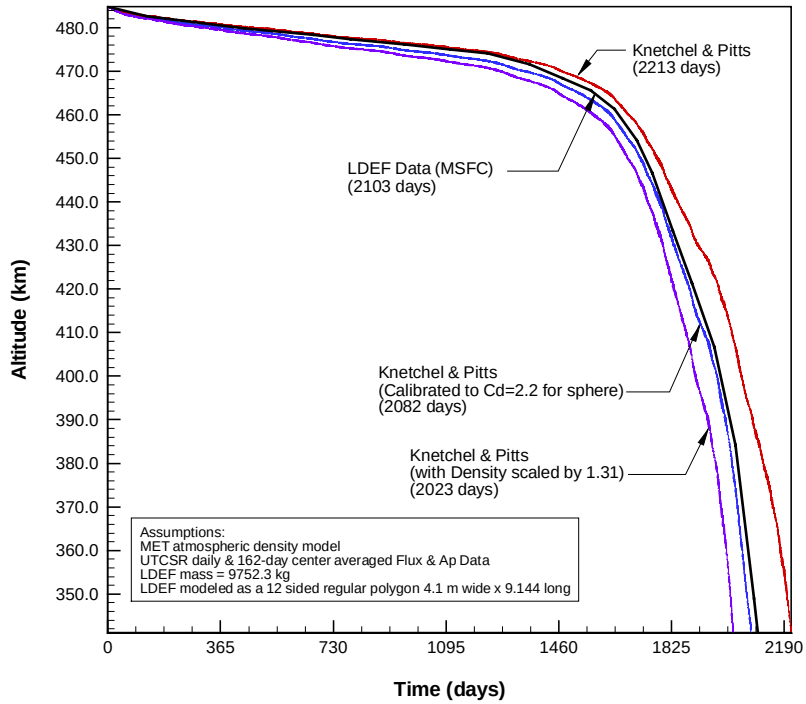

Figure 6. Orbital Lifetime Comparison of Accommodation Coefficient Assumptions and Density Scaling with LDEF Altitude Decay Profile.

The aerodynamic forces and torques acting on the GRACE satellites are simulated utilizing free molecular flow assumptions, where the normal and tangential force components acting on the satellite are defined (Eqs. (1) and (2)) for each elemental area and summed for all the surface elements of the spacecraft.

The normal and tangential accommodation coefficients (Eqs. (3) and (4)) are modeled as empirical functions of the activation energy of the impacting molecules, and the angle of incidence between the surface elements and the relative velocity vector (inward). For each element, the aerodynamic torques are equal to the vector cross product of the position vector (area centroid with respect to the GRACE center-of-mass) and the corresponding elemental force.

Eqs. (1)-(4) are incorporated in the FREEMOL pre-processor software, which provides the forces and torques on the GRACE satellite from any relative velocity direction (inward). Since these forces and torques are functions of the dynamic pressure, only forces and torques normalized by the dynamic pressure are stored, and during the simulation, these are multiplied by the dynamic pressure at a given altitude.

$$d\vec{\mathbf{F}}_n = \hat{\boldsymbol{n}}\ dA\left[\rho V^2 \cos^2\alpha + \sigma_n \rho V\ V_B \cos\alpha\right] + \hat{\boldsymbol{n}}\ dA\left[(1-\sigma_n)\rho V^2 \cos^2\alpha\right] \quad (1)$$

$$d\vec{\mathbf{F}}_t = \hat{\mathbf{t}}\ dA\left[\sigma_t \rho V^2 \cos\alpha \sin\alpha\right] \quad (2)$$

$$\sigma_n = a_n - b_n \cdot e^{\left(-c_n E \cos^2\alpha\right)} \quad (3)$$

$$\sigma_t = a_t - b_t \cdot e^{\left(-c_t E \sin^{\frac{3}{4}}\alpha\right)} \quad (4)$$

where:

$d\vec{\mathbf{F}}_n$	normal elemental aerodynamic force
$d\vec{\mathbf{F}}_t$	tangential elemental aerodynamic force
$\hat{\mathbf{n}}$	elemental inward surface normal unit vector
$\hat{\mathbf{t}}$	elemental surface tangent unit vector
σ_n	normal accommodation coefficient
σ_t	tangential accommodation coefficient
ρ	atmospheric density
V	velocity magnitude of incoming molecules
α	angle between incoming velocity direction (inward) and elemental surface normal
dA	elemental surface area
V_B	mean velocity of the diffusely re-emitted molecules
E	collision activation energy
a_n, b_n, c_n	normal accommodation curve fit parameters – nominally 1.0, 0.9, 0.28, respectively
a_t, b_t, c_t	tangential accommodation curve fit parameters – nominally 1.67, 1.67, 0.147, respectively

Solar Force and Torque Modeling

The effects of solar radiation pressure are simulated by modeling the momentum flux of solar photons, which is transferred to the satellite upon incidence. Some of the photons that hit the satellite are completely absorbed upon incidence. A coefficient σ_a is used to model the percentage of photons that are absorbed by the satellite. This absorption

produces a net force in the Sun-Spacecraft direction. Of the fraction that is not absorbed, $(1-\sigma_a)$, a coefficient σ is used to represent that fraction that is reflected diffusely. Depending on the surface material and finish, σ may vary from 0 to 1. The diffuse reflection produces a net force along the inward normal direction of the surface. The remaining radiation is specularly reflected.

The elemental force acting on an unshadowed elemental area due to solar radiation pressure is given by Eqs. (5)-(7) and summed for all the surface elements of the spacecraft. For each element, the solar torques are equal to the vector cross product of the position vector (area centroid with respect to the GRACE center-of-mass) and the corresponding elemental solar force. The resultant solar forces and torques normalized by half the solar pressure constant are calculated by the FREEMOL pre-processor for various Sun-Spacecraft line orientations. During simulation, the actual forces and torques are obtained based on the actual solar pressure constant and instantaneous Sun-Spacecraft line.

$$d\vec{\mathbf{S}} = pdA\cos\alpha\left(\sigma_a + \sigma_{rD}\right)\hat{\boldsymbol{S}} + pdA\cos\alpha\left(\frac{2}{3}\sigma_{rD} + 2\sigma_{rS}\cos\alpha\right)\hat{\boldsymbol{n}} \quad (5)$$

$$\sigma_{rD} = \sigma\left(1-\sigma_a\right) \quad (6)$$

$$\sigma_{rS} = \left(1-\sigma\right)\left(1-\sigma_a\right) \quad (7)$$

where:

$d\vec{\mathbf{S}}$	elemental solar force vector
$\hat{\mathbf{S}}$	unit vector along Sun-Spacecraft line of sight
$\hat{\mathbf{n}}$	elemental inward surface normal unit vector
σ_a	ratio of photons absorbed to incoming photons
σ	ratio of photons reflected diffusely to total photons reflected
p	solar pressure constant ($4.5605\cdot10^{-6}$ N/m²)
α	angle between incoming photons and elemental surface normal
σ_{rD}	ratio of photons that are reflected diffusely
σ_{rS}	ratio of photons that are reflected specularly
dA	elemental surface area

Gravity Gradient Torque Modeling

The central term of the Earth's gravitational potential is simulated adopting GM = $3.986005\cdot10^{14}$ m^3/s^2 (gravitational constant times mass of the Earth) and R = 6378.136 km (semi-major axis). The disturbing potential is modeled taking into account the dynamic flattening $J_2 = 1.083\cdot10^{-3}$ for orbit propagation. However, for the calculation of gravity gradient torques acting on GRACE, a simple spherical Earth model (GM/r^3) is assumed with no J_2 effects modeled.

Atmospheric Modeling

In order to provide AOCS and orbital lifetime analysis for the GRACE mission, one must utilize a uniform set of assumptions and parametric values with regards to solar activity and atmospheric density. The methodology for establishing these assumptions and values is described in this section. There are two basic considerations to be given to modeling the atmospheric density in which the GRACE satellites will be operating. The first consideration is the input parameters that are required by the atmospheric density models used in the analyses, and the second is the models themselves. The uncertainties in the models and solar activity predictions warrant a certain degree of conservatism in their selection to assure mission success.

Solar Activity Predictions

The two most important parameters associated with atmospheric density estimation are the 10.7 cm wavelength solar flux ($F_{10.7}$) and the geomagnetic index (A_p). These quantities are measured terrestrially, and directly relate to the Sun's extreme ultraviolet energy (EUV) output levels, which cannot be measured on the ground. It is primarily the EUV radiation that heats the atmosphere, causing it to expand. Currently, the NASA Marshall Space Flight Center (MSFC) provides monthly estimates of these values calculated by smoothing 13-month intervals of data [9]. MSFC provides statistical estimates of the solar activity based on the lowest, nominal, and highest activity that can be expected. The values are referred to as the MSFC 5%, 50% and 95% predictions, respectively. The Space Environment Center (SEC) at the National Oceanic and Atmospheric Administration also provides predictions of future solar activity. Fig. 7 shows a comparison of the MSFC and SEC estimates for solar cycles 23 and 24. As can be seen in the plot, there is some variation in the predictions. For the period following the GRACE launch, the MSFC predictions are somewhat higher than the SEC predictions when comparing the MSFC 95% values with the SEC "Upper" estimate. This results in the MSFC numbers being more conservative than the SEC values for orbital lifetime predictions. Additionally, the MSFC 50% predictions are slightly higher than the SEC "Predict" estimates during the period of the GRACE mission when the satellites will be at the lower end of the operational altitude range.

The MSFC flux predictions, along with long term geomagnetic index predictions, are updated and distributed on a monthly basis so that changes can be easily factored into future GRACE analyses. Figs. 8 and 9 show a comparison of the MSFC 13-month smoothed solar flux and A_p estimates, respectively. It should be noted that the actual 13-month smoothed values have historically been at the 95% level for significant periods of time during previous solar cycles. This leads to the general satellite design practice of using the 95% predictions to assure a mission with adequate orbital lifetime. Due to these considerations, the

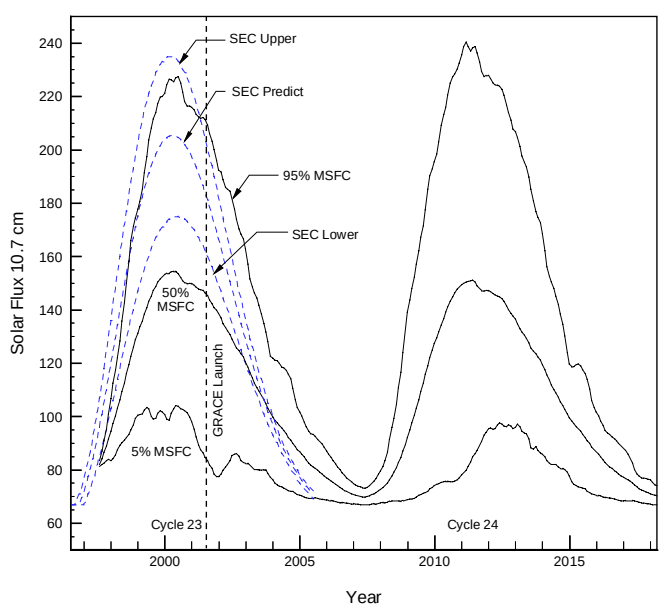

Figure 7. Comparison of MSFC and SEC Solar Activity Predictions for Cycles 23 and 24.

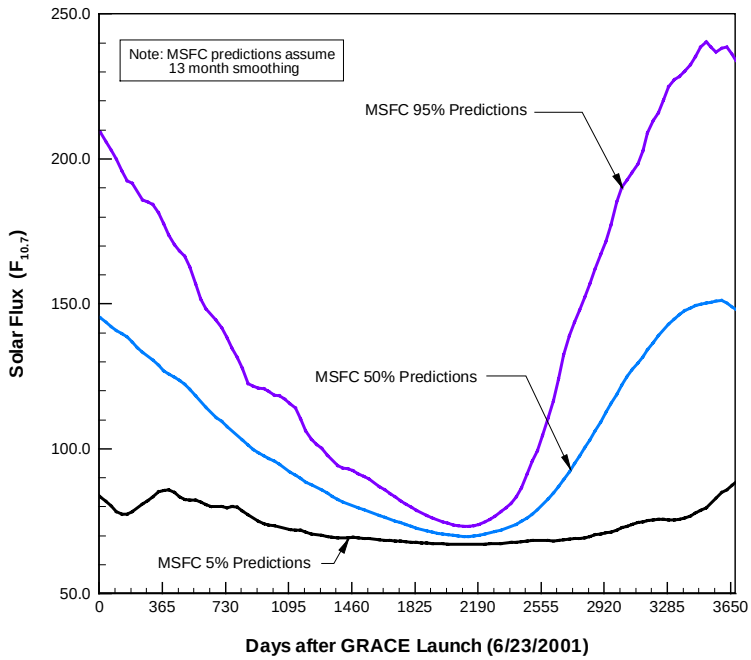

Figure 8. Comparison of MSFC 5%, 50%, and 95% Solar Flux ($F_{10.7}$) Predictions.

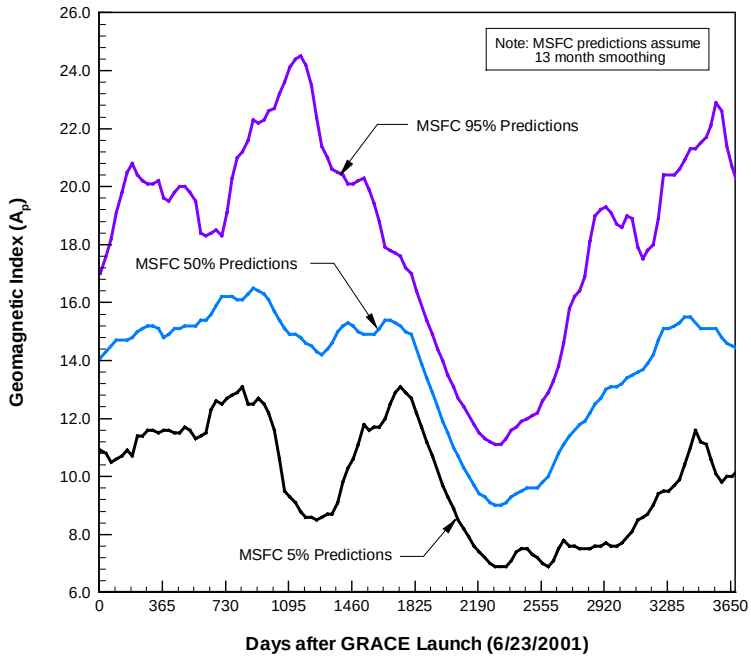

Figure 9. Comparison of MSFC 5%, 50%, and 95% Geomagnetic Index (A_p) Predictions.

MSFC 95% predictions are used for GRACE orbital lifetime and the 50% values are used for evaluating "nominal" AOCS design and nitrogen cold gas consumption.

Accurately predicting the actual daily solar flux and geomagnetic index values, which can deviate significantly from the mean values, is currently not possible. However, the mean values are well understood and commonly used in spacecraft design. The 13-month smoothed values are very reasonable (and slightly conservative in comparison with daily values) for orbital lifetime calculations, but for periods shorter than 90 days, they do not furnish accurate results when used with the atmospheric density models. For systems that are sensitive to thermospheric effects and variations over time periods of a few days or less (e.g. control and pointing systems), different values must be used. The daily solar flux numbers can be substantially higher than the mean values, and the 3-hour and daily A_p values can reflect the atmospheric activity associated with geomagnetic storms that can last hours or days. Estimates of the short term variations, provided by the University of Texas at Austin, Center for Space Research (UTCSR), are shown in Figs. 10 and 11, respectively. During the GRACE mission, the solar activity will occasionally reach fairly extreme values of solar flux and geomagnetic index, especially during the peak of the solar cycle.

Although the GRACE satellite should rarely experience intense atmospheric conditions, it is important that the satellite AOCS be able to handle these conditions and be able to minimize their impact on mission objectives. These conditions form what is termed a "Maximum Atmosphere" which is only used as a worst case point design condition. It is not used for any long-term analyses such as orbital lifetime. It should also be noted that the Maximum Atmosphere combines the maximum solar activity with the minimum mission altitude. Since the GRACE mission is planned to be launched shortly after the solar cycle peak, the atmospheric density resulting from these conditions should not be experienced by the GRACE satellites. However, these upper values provide valuable insight and understanding into the sensitivities of the AOCS to these conditions. Conversely, the concept of a "Minimum Atmosphere" is also introduced to provide a lower bound on the atmospheric density to characterize the minimum aerodynamic forces and torques that can expected during the mission. These values correspond to a minimum solar activity and a maximum mission altitude.

Atmospheric Density Models

Many atmospheric density models exist, and are based on various data such as satellite drag analyses and direct mass spectrometry measurements. Each of these models has certain strengths and weaknesses when predicting the atmospheric density for given set of conditions.

Two models are used for the GRACE analysis. The first is the Marshall Engineering Thermosphere (MET) model, which is the standard neutral atmospheric density model used for control and lifetime studies for most NASA spacecraft projects [8]. It is based on the Jacchia family of models, but contains several improvements. The second is the Mass Spectrometer Incoherent Scatter Thermosphere Model (MSIS) model, which is a much more "responsive" model under Maximum Atmosphere conditions. In addition, the MSIS model shows many higher density harmonics for higher inclination orbits compared to the MET model, which may be important to GRACE from both an AOCS and science viewpoint. However, the MSIS model is slightly more computationally expensive than the MET model, and does not appear to predict orbital lifetime with any more accuracy than the MET model. Based on the above observations, inspection of the model results, and the recommendations of various NASA atmospheric experts, it was determined that the MSIS model would be used for short period analyses and the NASA standard MET be used for orbital lifetime analyses.

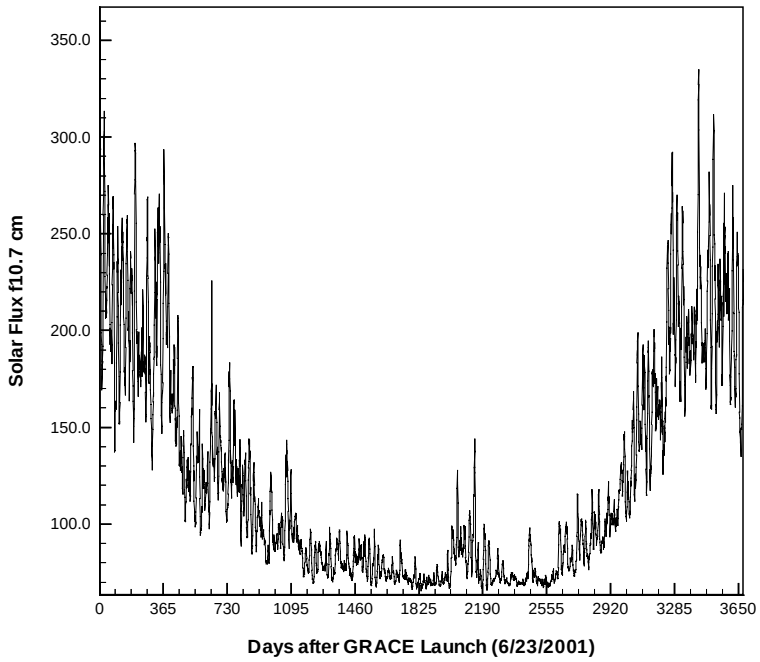

Figure 10. Representative Daily Solar Flux ($F_{10.7}$) Values (Courtesy of UTCSR).

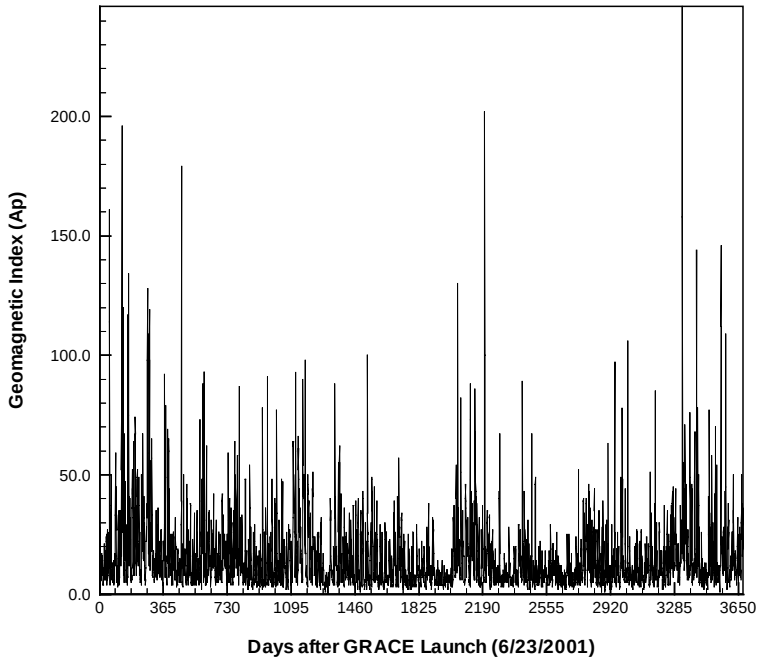

Figure 11. Representative Daily Geomagnetic Index (A_p) Values (Courtesy of UTCSR).

Assumptions for Analyses

Based on the preceding discussions of solar activity predictions and atmospheric density model selection, a set of assumptions for performing orbital lifetime, attitude control system design and evaluation, and propellant consumption estimates for the AOCS have been established.

Orbital Lifetime Conditions

The following assumptions provide reasonable guidelines for the GRACE orbital lifetime analysis:

- Marshall Engineering Thermosphere (MET) atmospheric density model.
- MSFC predicted 95 percentile solar flux ($F_{10.7}$) and geomagnetic index values (A_p).
- Initial Altitude based on nominal injection altitude with maximum and minimum based on injection uncertainty.
- Launch Date: June 23, 2001.

AOCS Design Conditions

For analyzing "nominal" AOCS design conditions during the GRACE mission, discrete altitudes from 500-300 km are evaluated every 50 km along the orbital lifetime mission profile for GRACE. Since the goal of the GRACE mission is to provide at least a five-year lifetime, AOCS design conditions for the altitudes were obtained in the following manner:

- Starting at an altitude of 497 km and assuming the 95% MSFC solar activity predictions, a five-year mission lifetime down to 300 km was obtained as shown in Fig. 12. The GRACE lifetime curve is a plot of altitude as a function of the number days after launch. Note that starting at an altitude of 467 km would only provide approximately two years of mission lifetime under these atmospheric conditions.
- Pick the dates (values of X) when this decay curve crosses the altitudes of interest (values of Y). Note that for the 500km altitude the launch day is arbitrarily picked as the date for determining solar flux and geomagnetic index values.
- For a given altitude and date, nominal solar flux and geomagnetic index values are used with the MSIS atmospheric density model, and average density and velocity along the local horizontal are obtained. These values are then used to calculate the nominal dynamic pressures for several altitudes of interest. Table 1 lists average orbital values of atmospheric density and wind velocity utilized for nominal AOCS design and propellant consumption. Moreover, the maximum and minimum atmospheres are also enumerated.

During the actual flight, the solar flux and geomagnetic index will typically be smaller than the 95% predictions and the resulting decay profile will be "above" the 95% MSFC prediction lifetime curve. This is demonstrated in Fig. 13, which shows a mission lifetime of approximately ten years using the 50% MSFC predictions with an initial altitude of 497km and approximately five years starting at 467 km. This higher profile (497 km curve), coupled with the fact that GRACE will be launched at the solar cycle maximum, guarantees that the solar flux ($F_{10.7}$) and A_p values, at the selected dates, are conservative for a *five-year mission*. For example, if the flux and A_p are smaller than the 95% predicted value, the particular altitude of interest would only be reached at a later date. Since the flux is decreasing during the next five years (approximately 11-year solar cycle), the density at this altitude would only be lower during the five-year mission lifetime.

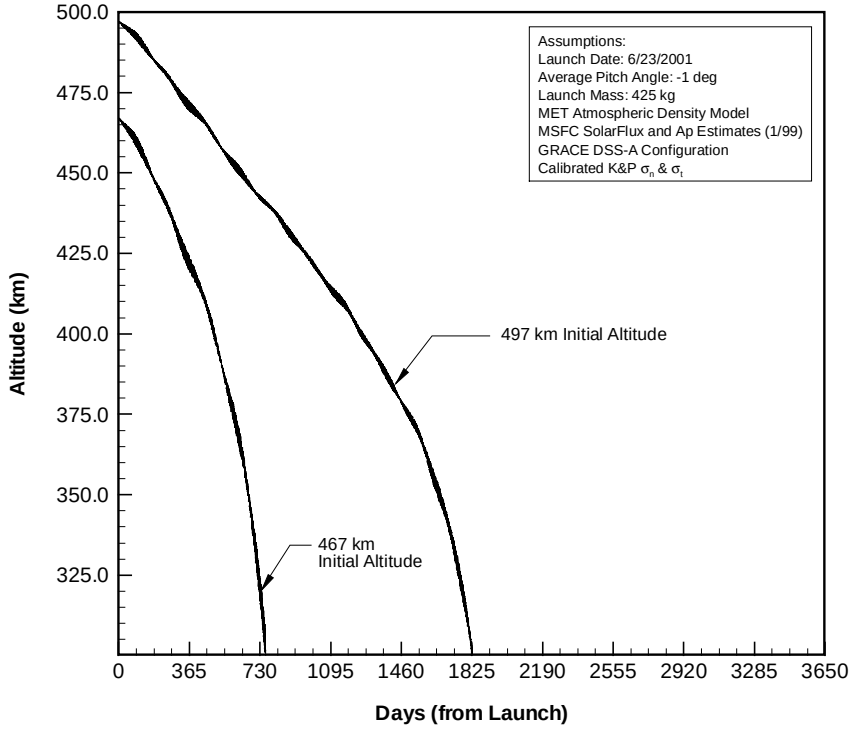

Figure 12. Mission Profile for GRACE Satellites Assuming 95% MSFC Solar Activity Predictions.

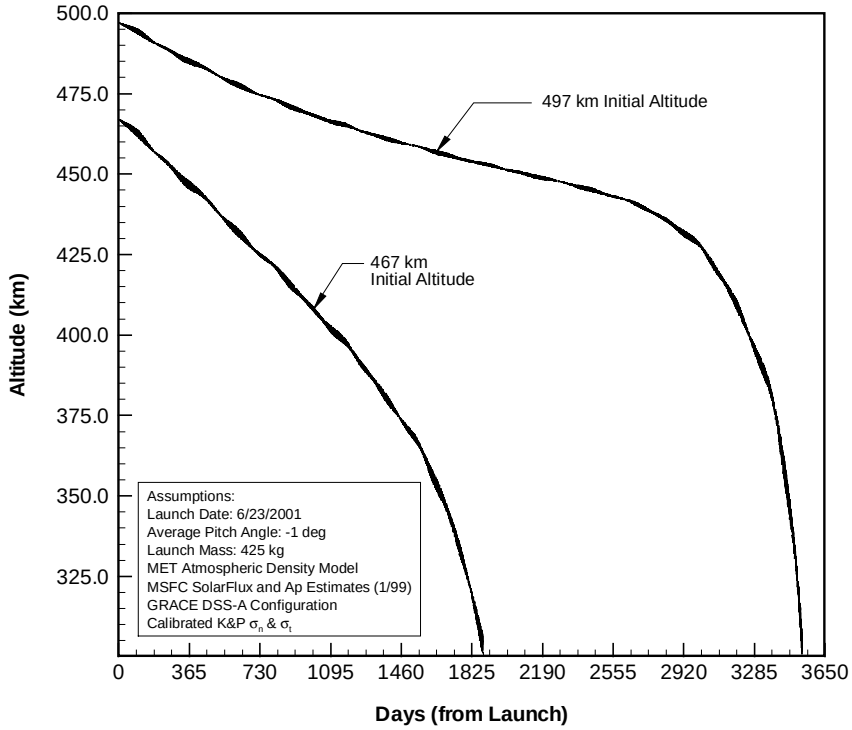

Figure 13. Mission Profile for GRACE Satellites Assuming 50% MSFC Solar Activity Predictions.

The following sections provide a summary of the parameters used for determining the atmospheric conditions utilized for AOCS analysis and design of the GRACE satellites.

Maximum Atmosphere Conditions

- Altitude = 300 km (Minimum mission design altitude).
- Daily Solar flux ($F_{10.7}$) = 384 (watts/m^2/Hz) (largest recorded solar flux) and 162 day average solar flux ($F_{10.7}$) = 240 (watts/m^2/Hz).
- Daily/3-Hour A_p = 400 (largest possible geomagnetic index during geomagnetic storm conditions and high degree of uncertainty for near polar orbits).

Note that this atmosphere is used for AOCS design, performance analysis, and sensitivity analysis. The solar activity numbers represent the worst case environment that GRACE could experience for short periods of time (approximately 1% of mission or less). It should also be noted that since the GRACE mission will nominally begin it's mission at the peak of the solar cycle at an altitude of 497 km, by the time the two spacecraft decay to 300km, the solar cycle will likely be near a minimum. Therefore, *nominally* the spacecraft should never experience the conditions simulated by the Maximum Atmosphere.

Minimum Atmosphere Conditions

- Altitude = 500 km (Maximum mission design altitude).
- Daily Solar flux ($F_{10.7}$) = 67 (watts/m^2/Hz) and 162 day average solar flux ($F_{10.7}$) = 67 (watts/m^2/Hz).
- Daily/3-Hour A_p = 6.9.

The values above correspond to the minimum NASA MSFC 5% predictions during solar cycles 23/24. The AOCS must demonstrate acceptable performance for the minimum atmosphere in the normal vehicle configuration. A robust AOCS design should also demonstrate acceptable performance even if the net external disturbing torques acting on the GRACE satellites are zero.

Nominal Atmosphere Conditions

Nominal design discrete data points are evaluated every 50 km (500 km to 300 km) to determine the aerodynamic conditions for nominal GRACE operations (see Table 1). Solar flux and A_p index values are taken from the NASA MSFC nominal predictions at altitudes along the mission altitude decay profile. Analysis at each altitude is performed with:

- Average orbital density (1 day average) at given altitude.
- MSIS atmospheric density model with 50% MSFC predictions.

Altitude (km)	Avg. density (kg/m^3)	Avg.wind velocity (m/s)
500	7.0e-13	7600
450	1.3e-12	7625
400	1.8e-12	7650
350	3.5e-12	7675
300	8.5e-12	7700

Maximum Atmosphere:
Altitude = 300 km
Peak density = 2.1e-10 kg/m^3
Peak wind velocity = 8400 m/s
Minimum Atmosphere:
Altitude = 500 km
Lowest Density = 3.0e-14 kg/m^3
Lowest wind velocity = 7470 m/s

Table 1. Sample Density and Wind Velocity Magnitudes for AOCS Design and Propellant Consumption.

Differential Drag Estimates and Effect on Station-keeping

In order for the GRACE satellites to collect science data, the two satellites must maintain a relative distance from each other and accurately point the K-Band horns at each other. Due to small differences in their ballistic coefficients, the satellites will slowly move apart as they orbit the Earth, requiring periodic station-keeping maneuvers. These maneuvers require the firing of the two 40 mN thrusters located on the rear of each satellite. In addition, the leading satellite will have to perform a 180-deg yaw maneuver in order to orient the thruster force along the velocity direction.

The objective of the station-keeping maneuvers is to keep the two GRACE satellites within a nominal separation of 220 km. The desired separation bounds are ±50 km from the nominal, although the science instruments are capable of functioning within a range of 100 km to 500 km separation.

The principal reason for the change in along-track separation between the two satellites is the differential drag acceleration. Assuming nominal accommodation coefficients, the FREEMOL analysis of the GRACE DSS-A satellite indicates that the leading satellite always experiences slightly greater drag than the trailing satellite. The statement above is true for any separation within a range of 100-500 km. The minimum separation of 100 km requires the pitch attitude of each spacecraft to be –0.4 deg in order for the K-Band antennae to be properly oriented, and the maximum separation of 500 km requires a pitch angle of –2.1 deg. As a result, the altitude of leading satellite decays faster than the trailing satellite, causing the former to speed-up relative to the latter, and thus increasing the separation. Once an upper bound of separation is reached, a station-keeping maneuver is initiated. This maneuver raises the semi-major axis of the leading satellite to a value greater than the semi-major axis of the trailing satellite. As a result, the leading satellite begins to lag and the distance between the satellites decreases. However, due to the differential drag, the rate of closure progressively decreases, until the trailing satellite begins to fall behind and the separation increases again.

For each maneuver, the required semi-major axis change for the leading satellite is a function of the mean drag difference between the two satellites. An estimate of this difference, either from drag models, or more likely, from analysis of the satellite tracking data in the period preceding the maneuver, is a pre-requisite.

If the estimate of drag difference is too large, then the maneuver leads to an excessive semi-major increment for the leading satellite. The separation will then tend to decrease below the permissible minimum, possibly requiring another station-keeping maneuver. In this case, the second station-keeping maneuver would be to re-circularize the two orbits to exactly the same semi-major axis once the minimum separation is reached.

If the estimate of drag difference is too small, then the maneuver leads to a deficit in the semi-major axis difference, in which case the separation again increases after too short an interval. In this case, a second increment in the semi-major axis would be required sooner than anticipated. Details of the station-keeping theory, strategy and sample maneuvers can be found in [10].

Simulation Results

The following section describes simulation results for the GRACE DSS-A configuration during the nominal science data acquisition mode. All of the simulations were performed at the lowest operational altitude (300 km) five years after launch (June of 2006). These assumptions represent the largest aerodynamic disturbance conditions that the GRACE satellites would *nominally* be exposed to. Although many simulations have been performed for the GRACE mission, the goal of the analyses presented here was to characterize the effect of

variations in the accommodation coefficients on a per surface basis. As mentioned earlier, the GRACE DSS-A configuration is grouped into five surface types: solar arrays, radiator surface, front and back panels, aluminum and star camera baffles. All other surfaces are assumed to have the same accommodation properties as aluminum. The accommodation coefficients on these surface groups were allowed to vary to understand the effect if one surface, such as the solar arrays, behaved substantially differently than another surface, such as the radiator. The sensitivity of the differential drag (represented in the simulation results as a ballistic coefficient differential percentage, ΔB_c) could be significantly affected by differences in the forward and aft panel surface/geometry characteristics, as well as the fact that the trailing satellite tends to project the solar arrays towards the incoming wind direction, while the leading satellite projects the radiator surface. Additionally, varying the accommodation coefficients affects the spacecraft aerodynamic torques, and subsequently the propellant consumption for attitude control.

The FREEMOL simulations were divided into two sets. The first analysis set allowed the accommodation coefficients on four of the five surface groups (the star camera baffles were assumed to always be absorptive) to vary such that they are either unity or zero for all surface angles. Assuming the full range of accommodation provides an *extremely* conservative approach and clearly represents a theoretical worst case scenario, but at the same time provides valuable insight into the aerodynamic characteristics of the GRACE satellites. The second analysis set assumes a smaller range of accommodation coefficients based on the typical range of drag coefficients observed for Earth orbiting satellites. The rationale for each of these assumptions is described in more detail in the following sections.

For each set of analyses, 259 FREEMOL simulations were performed. The number of accommodation coefficient bounding cases is equal to 256 ($4^{(\text{\# of surface groups})}$), since σ_n and σ_t are both allowed to possess a minimum as well as a maximum value. The three additional simulations were performed assuming all surfaces were either completely reflective, completely absorptive, or possessed nominal accommodation coefficients.

Note that in the differential ballistic coefficient plots, both the trailing and leading GRACE satellites have the same negative pitch angle (since the leading satellite is yawed by 180 deg) for a given separation distance. The nominal separation corresponds to about 220 km and the pitch angle of each satellite is approximately -0.9 deg. When the satellites are close together (the minimum allowed separation is 100 km) the pitch angles are about -0.4 deg each, and when the satellites are farthest away (the maximum allowed separation is 500km) the pitch angles are about -2.1 deg each. The GRACE Science Mission Requirements Document (SMRD) [1] states that the nominal separation is 220±50 km. This corresponds to a separation between 170 km and 270 km, which implies a pitch attitude between -0.71 and -1.15 deg. The preceeding pitch angles are calculated based on the assumption that the K-Band horn is aligned along the X-body axis.

Assuming Full Range of Accommodation (Theoretical)

For the full range of accommodation coefficients, Fig. 14 shows that the difference in ballistic coefficient, ΔB_c, could be as high as 25% at 500 km separation. This is much higher than the SMRD requirement of 0.5%. However, it is to be noted that this represents an absolute worst case scenario, both from an accommodation standpoint, as well as an operating range standpoint. Within the nominal operating separation range, Fig. 15 shows that the ΔB_c could be as high as 13% at 270 km separation. Fig. 15 shows that the nominal and the fully absorptive case (for all surfaces) are within the 0.5% requirement, and the purely reflective case (for all surfaces) is outside the 0.5%, but still in an acceptable range in terms of

station-keeping maneuver frequency and propellant consumption (documented in the "Sample Station-keeping Results" section).

The unacceptable ΔB_c cases arise from the assumption in this analysis set that accommodation coefficients can vary to their extreme theoretical values. Three distinctive curve groupings can be observed at a pitch angle of zero degrees in Fig. 14. At zero degrees, these differences are primarily a result of the dissimilarities between the forward and aft panel surfaces. Since the two spacecraft possess a high degree of symmetry with respect to the Y/Z-body plane (forward compared to aft), the other surfaces contribute almost equally to the aerodynamic drag. Although the panels are composed primarily of Kapton coated CFRP, there are other surfaces that are not identical on both panels (refer to Figs. 1 and 2). These small differences can result in significant ΔB_c values due to the extreme range of accommodation assumed in this analysis set. It should be noted that the projected area of the GPS Occultation Antenna and the GPS Back-up Antenna is approximately 7% of the total aft panel area. Based on this projected area assumption, the ΔB_c could be as high as 7% if these surfaces accommodated the incoming molecules in a reflective manner and the forward panel acted in a fully absorptive manner. Although these large, theoretical ΔB_c differences would never actually be experienced on-orbit, this observation would make it desirable to design the spacecraft with the same surface configuration (i.e., identical hardware or coverings) on both panels. However, due to other design constraints, including cost, this design option would not be feasible for the GRACE mission.

The other factor influencing the spread of the ΔB_c curves arises from cases where the radiator surface accommodation coefficients behave completely opposite from the solar array surfaces. For example, if the radiator has high skin drag (σ_t = 1) while the solar array surfaces have no skin drag (σ_t = 0), or vice versa, ΔB_c will increase. These differences dominate as the pitch angle is increased. This results in the trailing satellite's solar arrays becoming more exposed to the incoming wind direction, while the leading satellite's radiator surface is more visible.

The above inference is best understood by observing Figs. 16 and 17 which are equivalent to Figs. 14 and 15, respectively, with the added assumption that the radiator and solar array surfaces are identical in accommodation. As shown in Figs. 16 and 17, the maximum ΔB_c resulting at the largest pitch angles could be reduced by approximately half if the two surfaces possessed similar accommodation characteristics. Note that the ΔB_c at a zero degree pitch angle is nearly the same as before due to the differences in the forward and aft panels. One design suggestion as a result of this observation is to cover the radiator surface with a glass surface similar to the solar array surfaces. However, the thermal and cost implications have to be addressed carefully. Another suggestion is to tilt the K-Band horn down by 1 deg. This design modification would reduce the operating pitch angle to range of ±0.25 deg, thereby decreasing the ΔB_c by a substantial amount. This can be seen in Fig. 14, where the largest ΔB_c is less than 5% for a pitch angle -0.25 deg. However, this solution adds the physical complexity of mounting the horn by an angle of 1 deg. From a programmatic stand-point, the complexity of assembling the tilted horn appears to outweigh the effort to mitigate the risk of such large ΔB_c.

Figs. 18-20 respectively show the aerodynamic X,Y, and Z-body torques on the GRACE spacecraft during the nominal science mode. From these plots it can be clearly observed that the accommodation coefficient assumptions can also have a significant impact on the aerodynamic torques of a particular spacecraft geometric configuration. For the GRACE satellite, the aerodynamic X-body

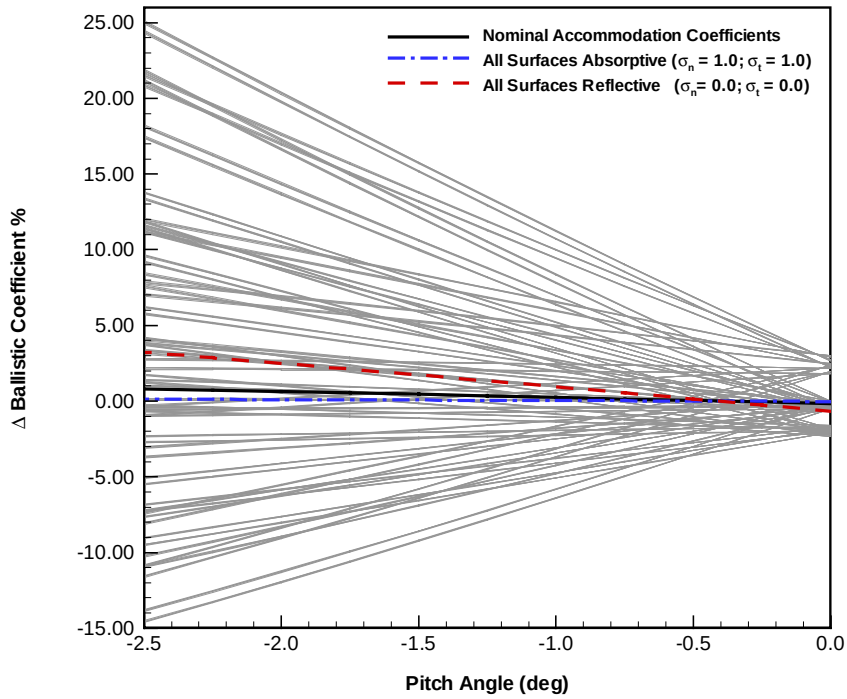

Figure 14. Differential Ballistic Coefficient vs. Pitch Angle for Maximum Separation Range (Full Range of Accommodation).

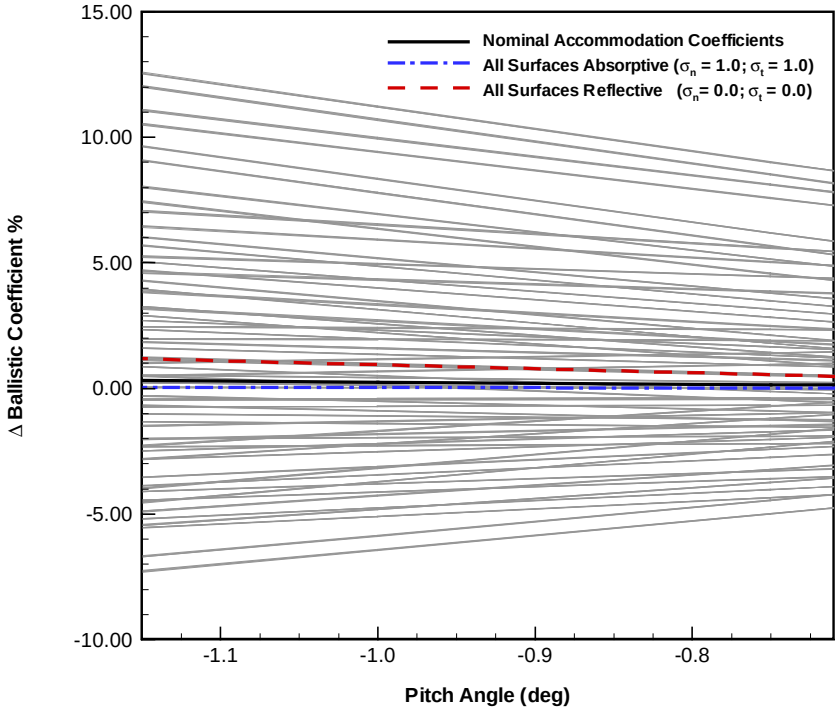

Figure 15. Differential Ballistic Coefficient vs. Pitch Angle for Nominal Separation Range (Full Range of Accommodation).

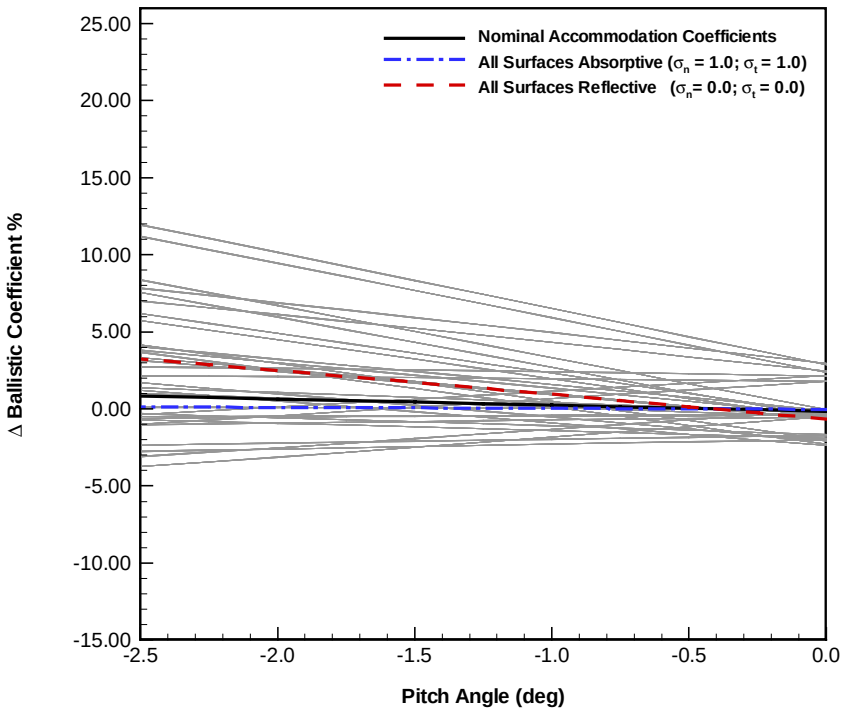

Figure 16. Differential Ballistic Coefficient vs. Pitch Angle for Entire Separation Range (Full Range of Accommodation – Solar Array and Radiator Surfaces Similar).

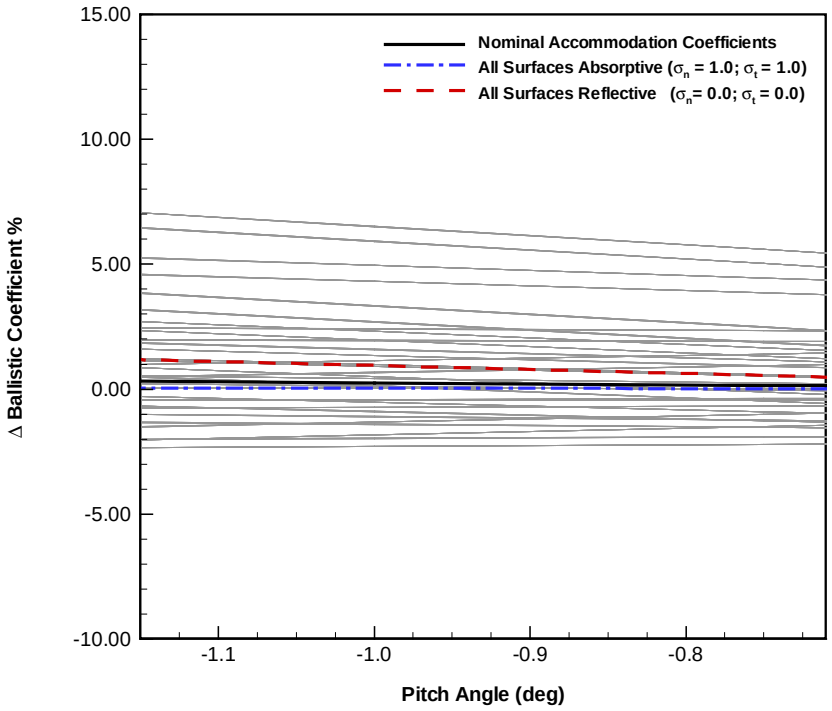

Figure 17. Differential Ballistic Coefficient vs. Pitch Angle for Nominal Separation Range (Full Range of Accommodation – Solar Array and Radiator Surfaces Similar).

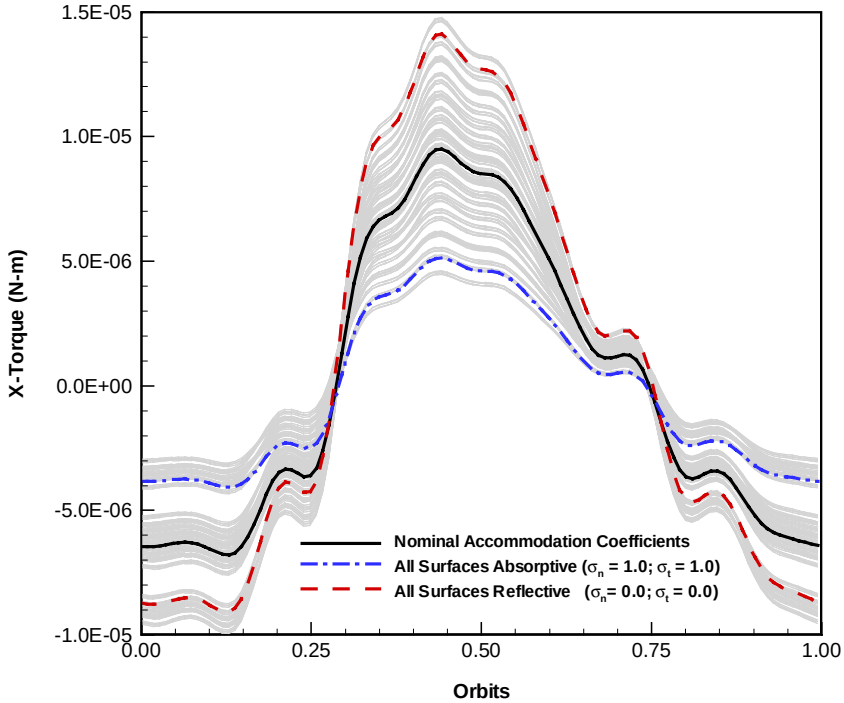

Figure 18. X-Body Aerodynamic Torque vs. Time for One Sample Orbit (Full Range of Accommodation).

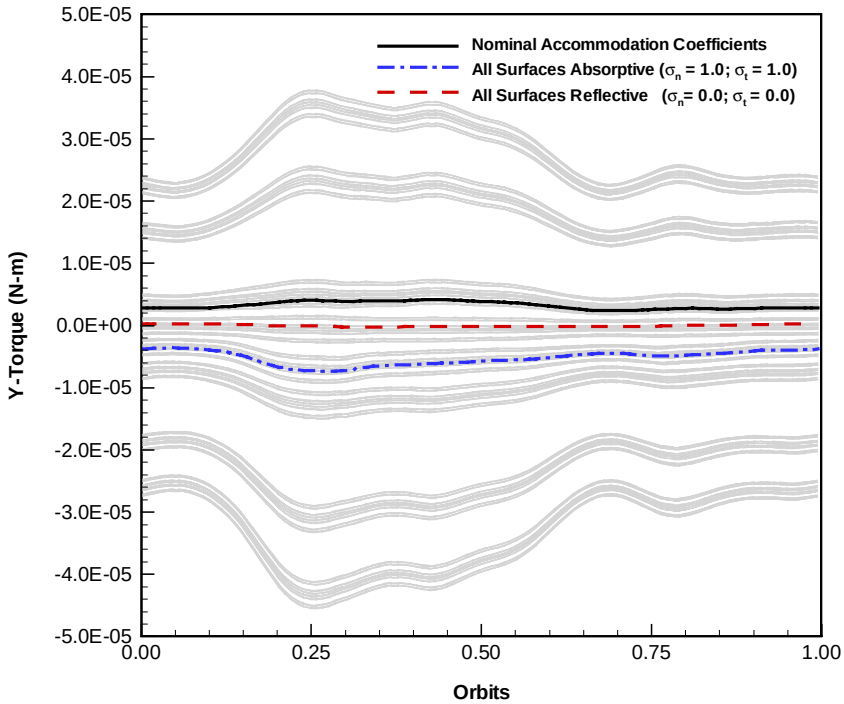

Figure 19. Y-Body Aerodynamic Torque vs. Time for One Sample Orbit (Full Range of Accommodation).

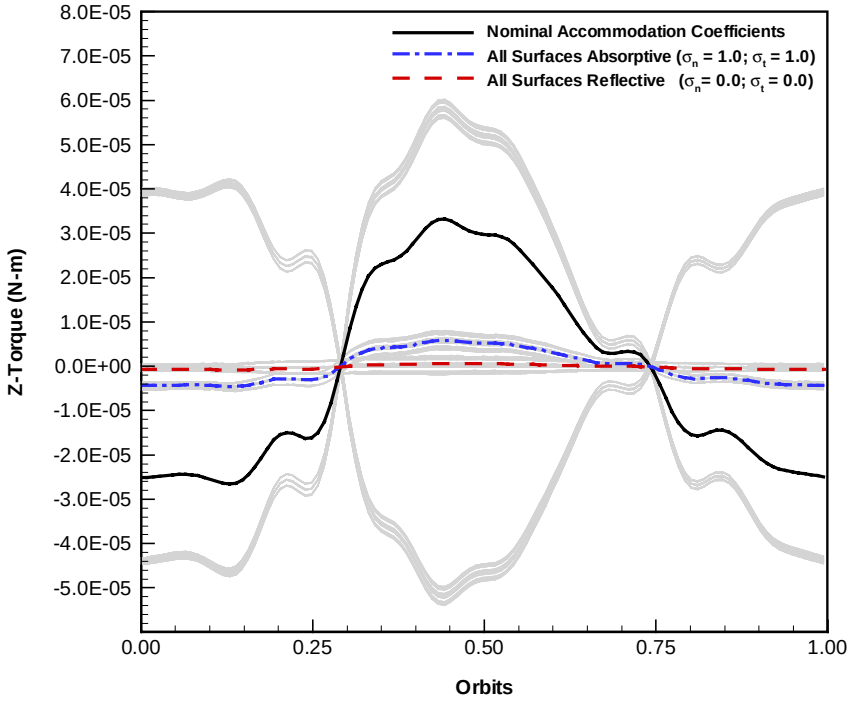

Figure 20. Z-Body Aerodynamic Torque vs. Time for One Sample Orbit (Full Range of Accommodation).

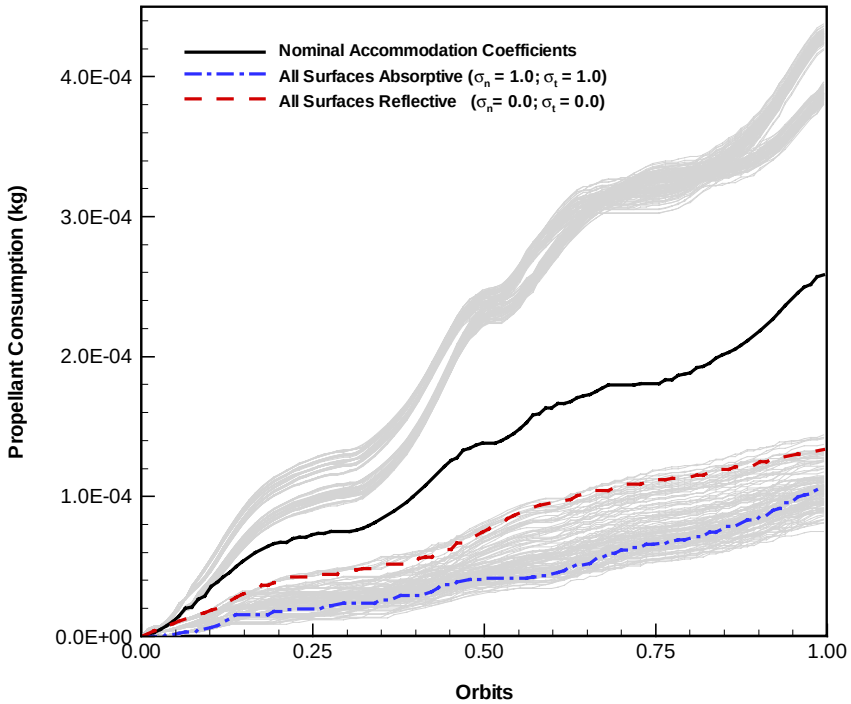

Figure 21. GN_2 Propellant Consumption vs. Time for One Sample Orbit (Full Range of Accommodation).

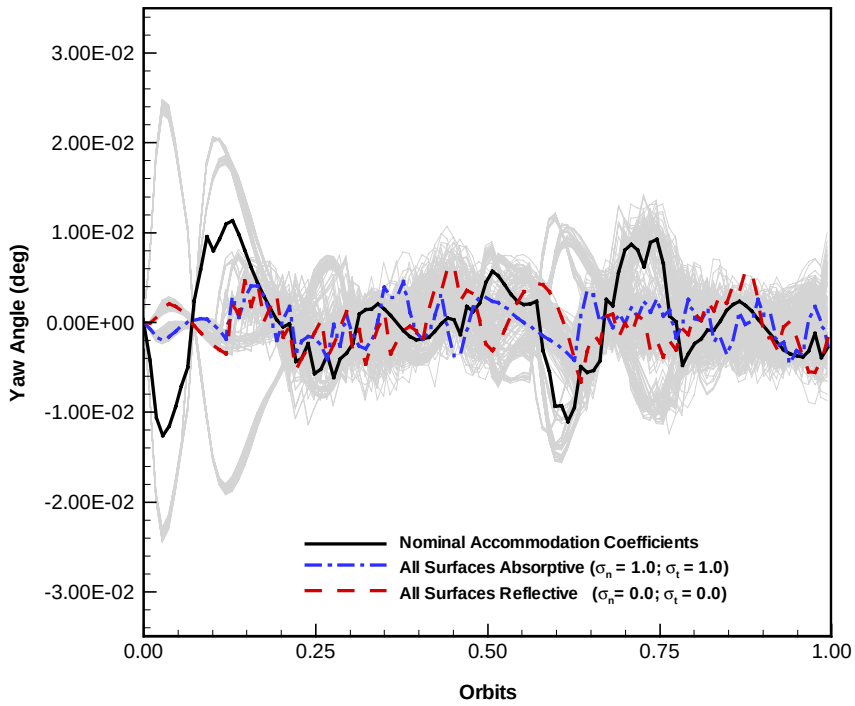

Figure 22. Yaw Angle (3-2-1 Euler Sequence) vs. Time for One Sample Orbit (Full Range of Accommodation).

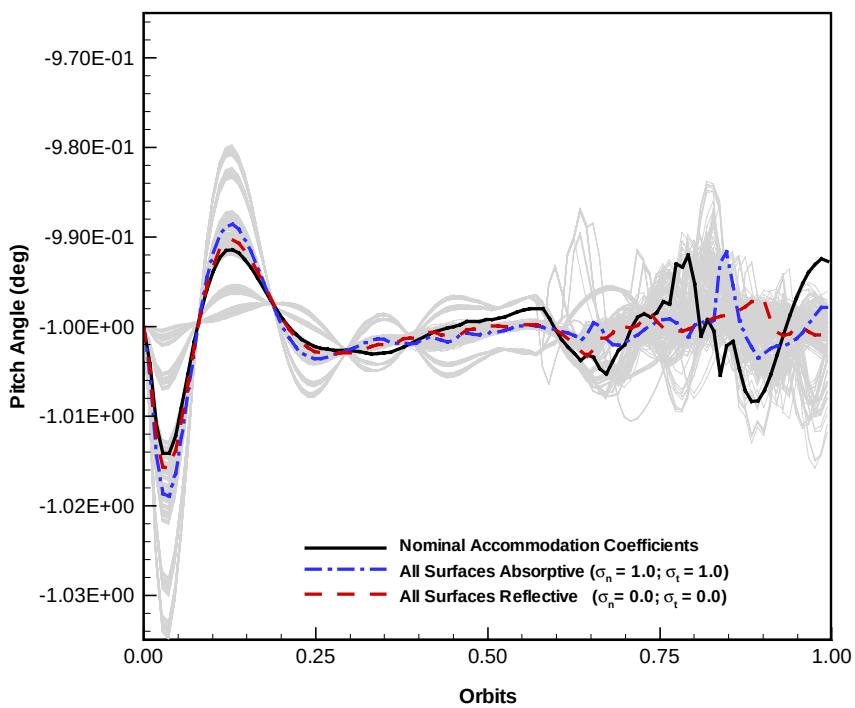

Figure 23. Pitch Angle (3-2-1 Euler Sequence) vs. Time for One Sample Orbit (Full Range of Accommodation).

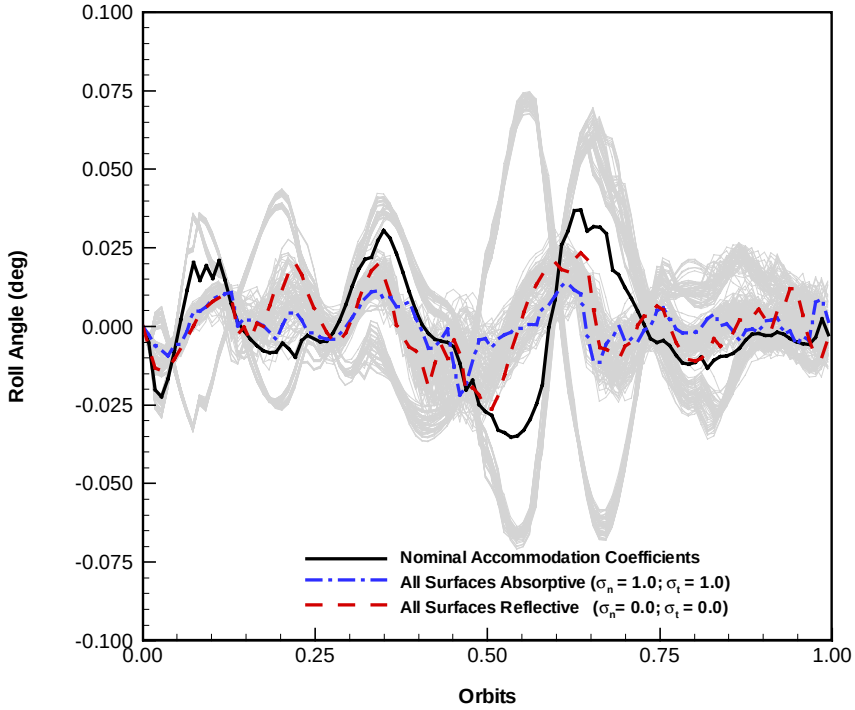

Figure 24. Roll Angle (3-2-1 Euler Sequence) vs. Time for One Sample Orbit (Full Range of Accommodation).

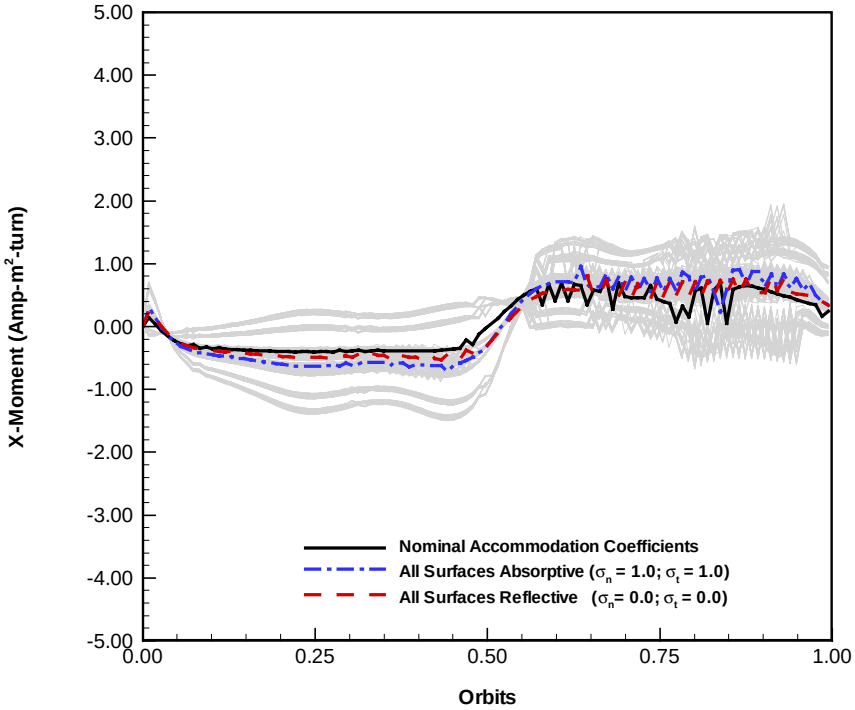

Figure 25. Magnetic Torque Rod X-Moment vs. Time for One Sample Orbit (Full Range of Accommodation).

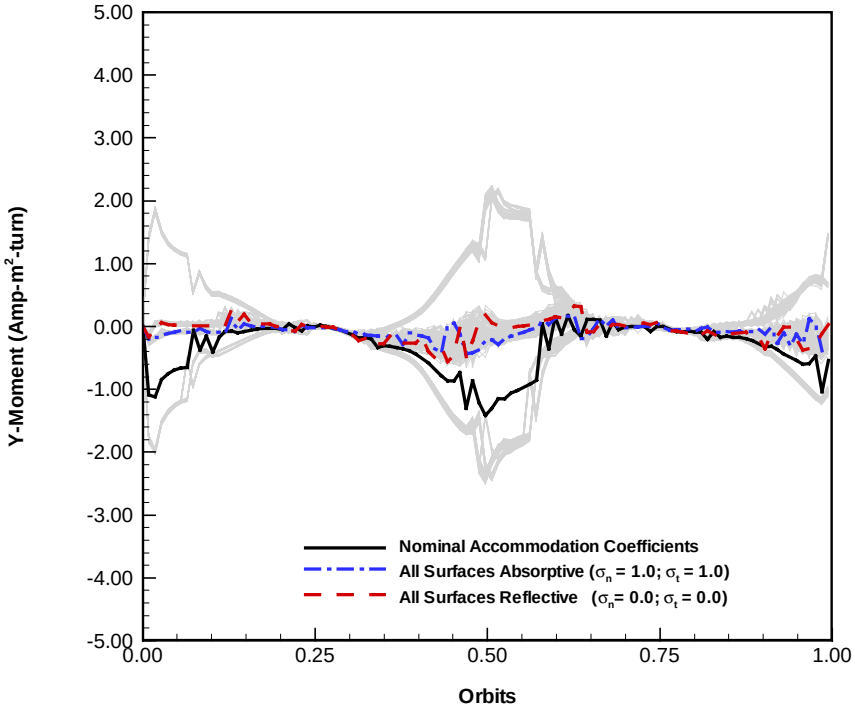

Figure 26. Magnetic Torque Rod Y-Moment vs. Time for One Sample Orbit (Full Range of Accommodation).

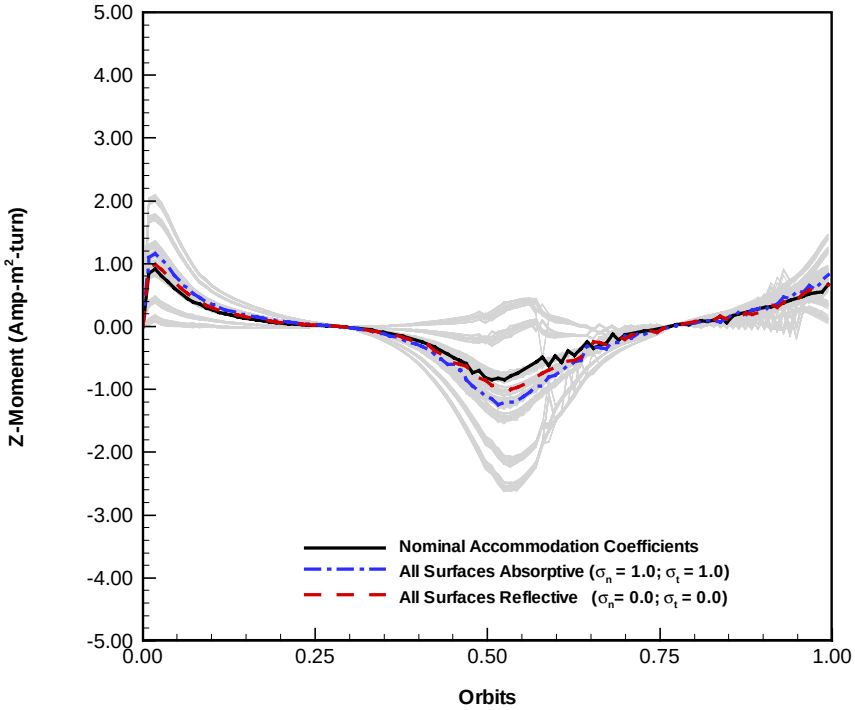

Figure 27. Magnetic Torque Rod Z-Moment vs. Time for One Sample Orbit (Full Range of Accommodation).

torque can be bounded fairly well by the absorptive and reflective simulations, with the nominal coefficient case falling approximately in the middle. However, for some of the various combinations of surface accommodation the Y and Z-body torques (pitch and yaw) can be significantly greater than either of these two limiting cases. For the full range of accommodation, many of the curves resemble the absorptive or reflective results, which is a result of allowing σ_n and σ_t to vary individually between unity and zero.

Fig. 21 shows the propellant consumption over an orbit resulting from the various surface accommodation coefficient combinations. Also, included in these simulations (but not shown) are the other disturbing torques (gravity gradient, solar, gyroscopic) acting on the satellite during science mode. The other torques are nearly the same for all of these simulations, so the increase in propellant consumption is the result of the higher X and Z-body aerodynamic torques for some of the accommodation cases. Note again, the curve groupings around the absorptive and reflective simulations.

Figs. 22-24 show the Euler angles (3-2-1 sequence) over an orbit for the full range of accommodation coefficients. The requirement for the GRACE science mode is for the spacecraft to track the specified trajectory to a deadband of ±0.029 deg (±0.5 mrad) in pitch and yaw, and ±0.573 deg (±10 mrad) in roll. The nominal pitch attitude was commanded at –1.0 deg for all of the simulations, while the yaw and roll angles were commanded to 0.0 deg. The simulation results shown here were performed with a preliminary, non-optimized control algorithm. Other than a small transient excursion in the pitch channel for some of the coefficient bounding cases, the Euler angles are all well within the required attitude deadbands.

Figs. 25-27 show the required magnetic moment that is supplied by the magnetic torque rods during the one-orbit simulations. Each GRACE spacecraft utilizes three dual winding 30 Amp-m^2 magnetic torque rods to supplement the cold gas reaction control system (one rod aligned with each spacecraft body axis). For nominal atmospheric conditions, the simulations show that the magnetic torque moments for each body axis are only a fraction of the torque rod capacity.

A significant observation from these simulations is that even these wide ranges of accommodation for different surfaces do not pose a significant problem in terms of the aerodynamic disturbing torques during the science mode. The worst-case pitch torques are still controllable with the magnetic torque rods of 30 Amp-m^2, assuming nominal atmospheric densities. Additionally, the control torque provided by the uncoupled attitude control thrusters is up to several hundred times greater than the maximum disturbance torques under nominal atmospheric densities. The fact that the GRACE satellite is able to meet the attitude pointing requirement even with worst case accommodation assumptions is a very encouraging result for the GRACE AOCS. For all of these cases, the propellant usage at the lowest operating altitude is reasonable and does not exceed the attitude control allocation (approximately 16 kg) when extrapolated over the five mission.

Assuming Smaller Range of Accommodation

The range of accommodation coefficients suggested in the previous section is extremely conservative. In practice, no surface is completely reflective or absorptive. However, the conservative approach provides a worst case bound for the differential ballistic coefficients, and suggests design modifications which *guarantee* a small ΔB_c regardless of the surface properties.

It is difficult to provide a reasonable range of accommodation coefficients based on experimental data for different surface properties and different gas interactions, as a function of incidence angles. This is due to the lack of sufficient reliable experimental results. An

approach to provide a reasonable bound for the accommodation coefficients is suggested below:

- The selection of nominal accommodation coefficient profiles has been discussed previously in the section on modeling aerodynamic forces and torques. These profiles are shown in Fig. 4.

- To obtain reasonable bounds for the accommodation coefficient profiles that each surface could exhibit, the profiles are varied to a lower and upper bound which is ±20% of the nominal values. The 20% variation was derived as the maximum deviation based on a Monte Carlo analysis that demonstrated that any combination of accommodation coefficients within these bounds yields a variation of coefficient of drag, C_D, between 1.9 and 2.5 for a uniform sphere, which is the typical range for spherical satellites [11]. Fig. 28 shows the allowable range of variation of the accommodation coefficients as a function of incidence angle. A statistically significant number of intermediate accommodation coefficient curves were randomly calculated within these bounds as shown in Fig. 29. These intermediate values were then used to calculate the C_D of a uniform sphere, and the corresponding spread of C_D, based on these various combinations of σ_n and σ_t, is shown in Fig. 30. The bounds shown in Fig. 28 provide the values for the accommodation coefficients used in this analysis set.

For the smaller range of feasible accommodation coefficients, it can be observed from Figs. 31 and 32 that the differences in ballistic coefficients are much smaller. For the nominal separation range of 170 km to 270 km corresponding to pitch angle ranges of -0.71 deg to -1.15 deg, the largest absolute ΔB_c is approximately 3% (compared to 13% for the full range of accommodation assumption). Additionally, the maximum ΔB_c occurring at a pitch angle of zero degrees is only approximately 1% (compared to 3% for the full range of accommodation assumption). The analysis assuming identical accommodation coefficients for the solar array and radiator surfaces was repeated for the smaller range of coefficients with a similar reduction in ΔB_c at the larger pitch angles. The results are depicted in Figs. 33 and 34. Moreover, the aerodynamic torque variations as shown in Figs. 35-37 are also much smaller compared to results shown in Figs. 18-20, as expected. Also note that the curves are now all grouped in bands with the nominal accommodation coefficients curve in the center. This type of spread is what is expected when assuming a ±20% variation in the coefficient profiles from the nominal profile.

The propellant consumption is considerably smaller as shown in Fig. 38 in comparison to Fig. 21. Note that when all surfaces were assumed to be completely absorptive or reflective, the simulations actually result in a propellant consumption less than *any* of the accommodation bounding cases. The nominal propellant consumption is approximately two times the absorptive case, and the largest estimate is more than three times greater than the absorptive case.

Figs. 39-41 show the Euler angles (3-2-1 sequence) over an orbit for the smaller range of accommodation coefficients. The Euler angles for all these simulations are all much closer to the nominal accommodation coefficient curve than the simulations assuming the full range of accommodation (Figs. 22-24), and again are well within the required attitude deadbands.

Figs. 42-44 show the required magnetic moment that is supplied by the magnetic torque rods during the one-orbit simulations. The simulations show that the magnetic torque moments for each body axis are reduced by approximately half from the simulations assuming the full range of accommodation coefficients (Figs. 25-27). Again, the magnetic moments are all well below the capacity of the magnetic torque rods.

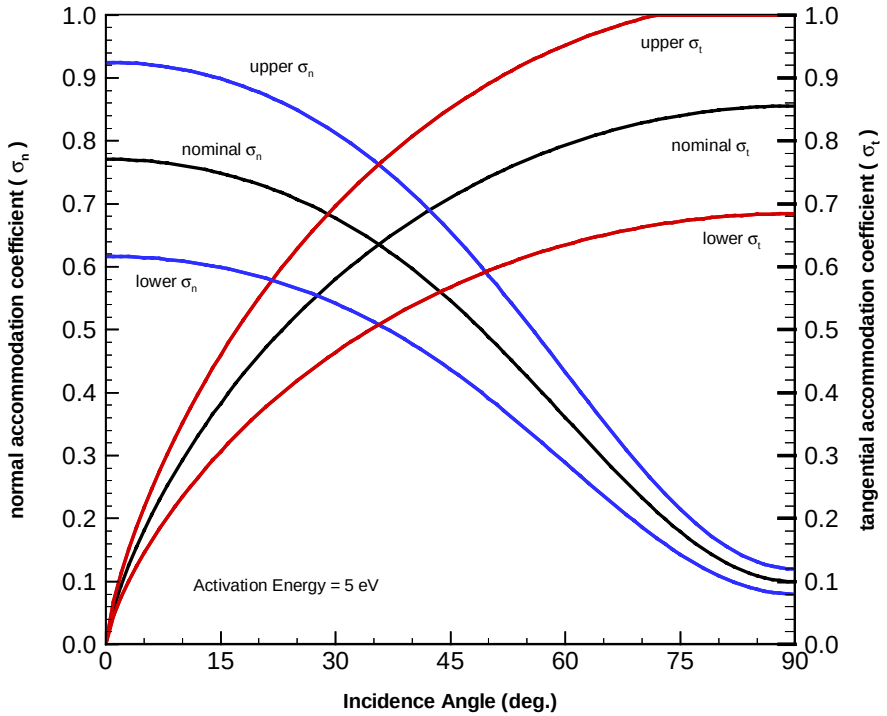

Figure 28. Lower and Upper Bounds for Accommodation Coefficients.

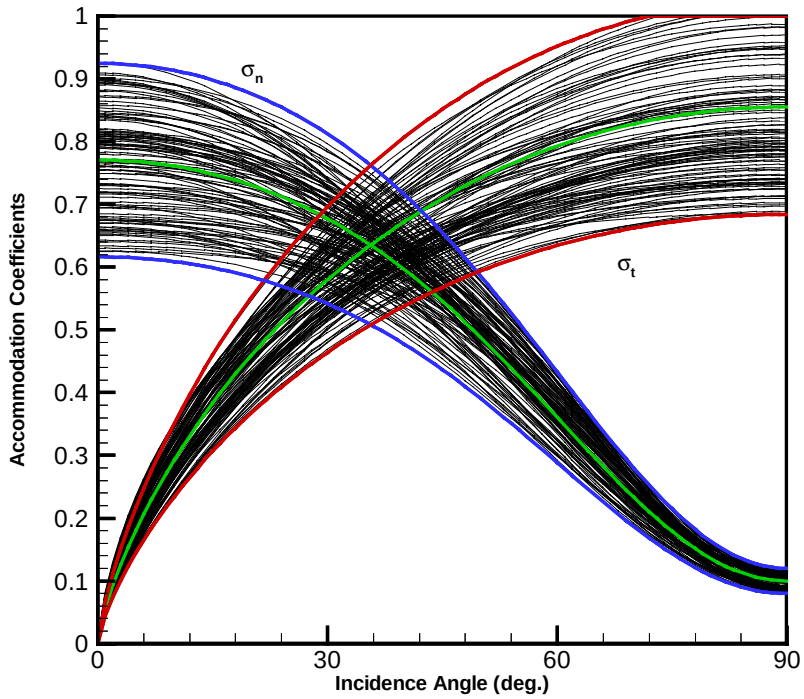

Figure 29. Intermediate Accommodation Coefficient Profiles for Calculating C_D of Uniform Sphere.

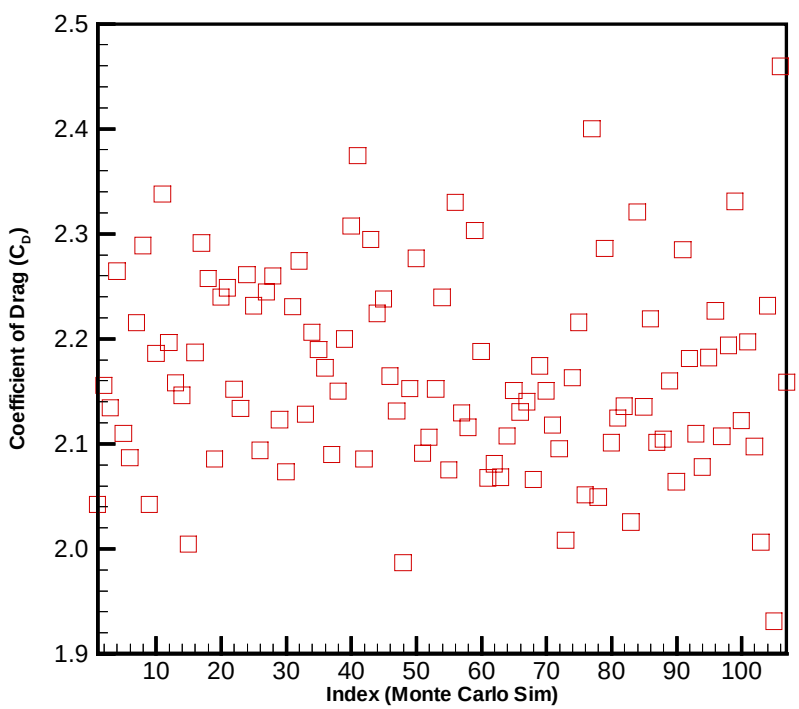

Figure 30. C_D of a Uniform Sphere Using Intermediate Accommodation Coefficient Profiles.

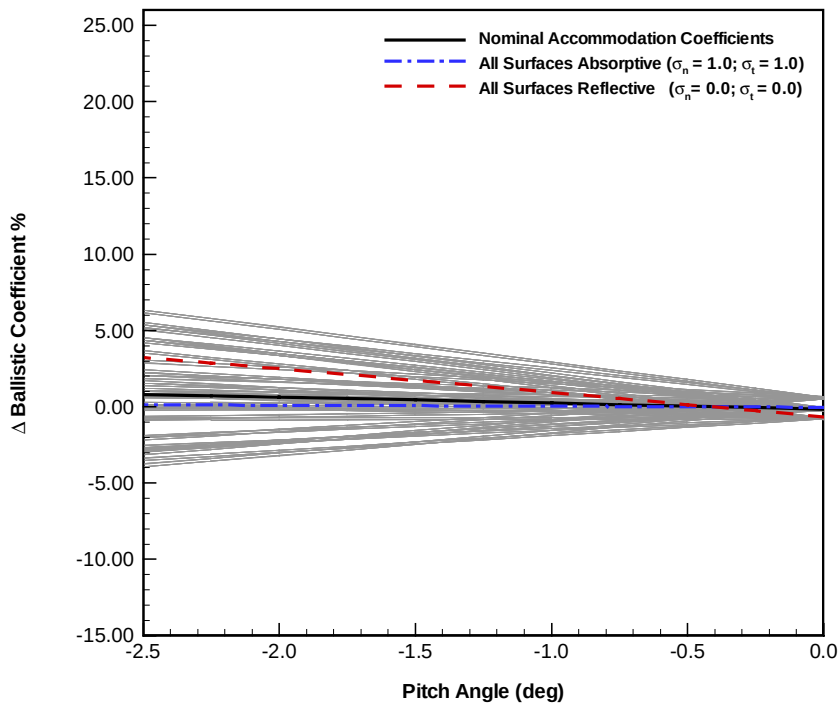

Figure 31. Differential Ballistic Coefficient vs. Pitch Angle for Entire Separation Range (Small Range of Accommodation).

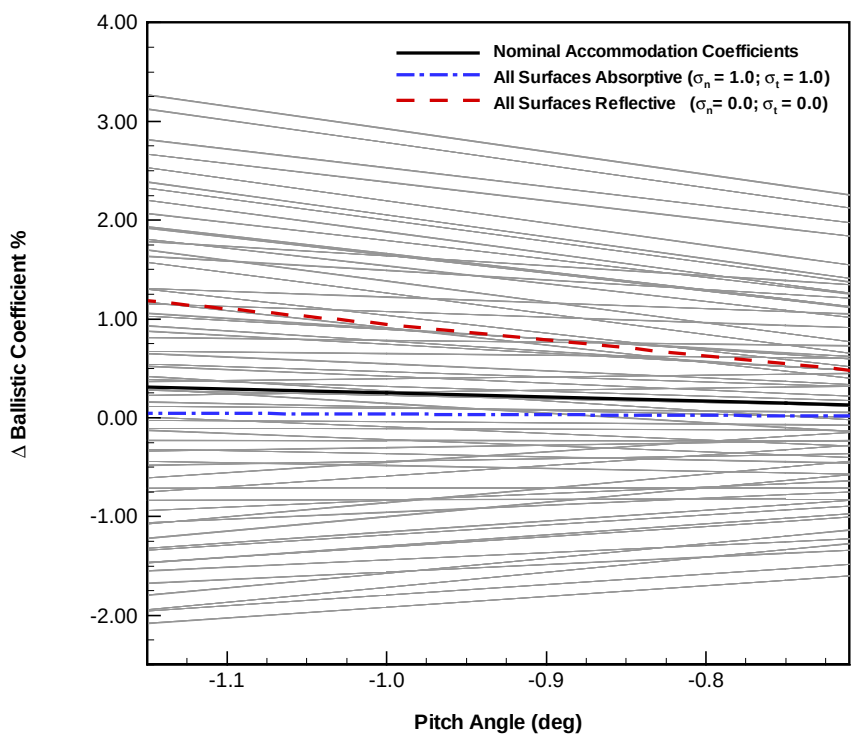

Figure 32. Differential Ballistic Coefficient vs. Pitch Angle for Nominal Separation Range (Small Range of Accommodation).

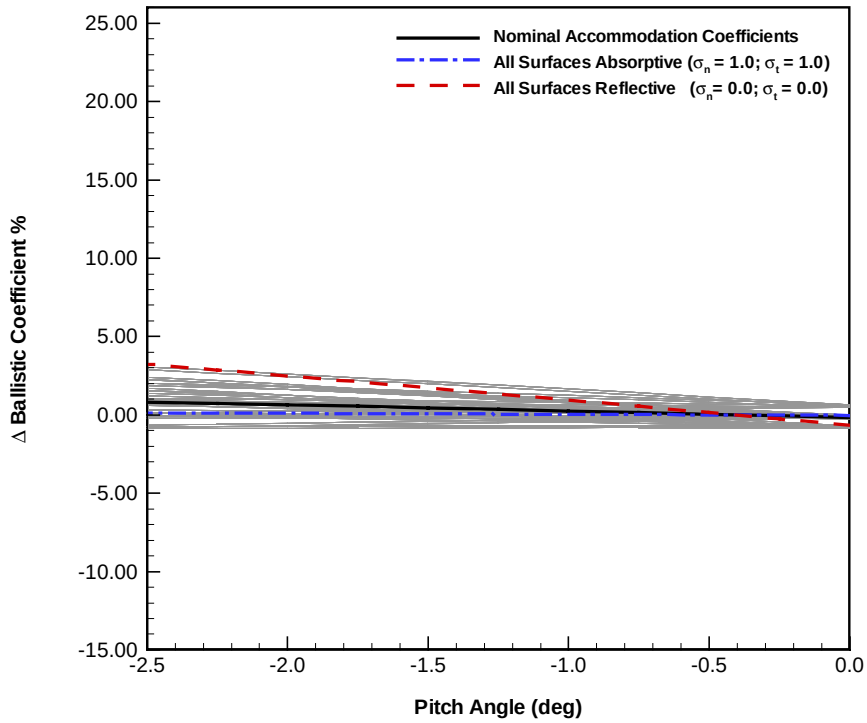

Figure 33. Differential Ballistic Coefficient vs. Pitch Angle for Maximum Separation Range (Small Range of Accommodation – Solar Array and Radiator Surfaces Similar).

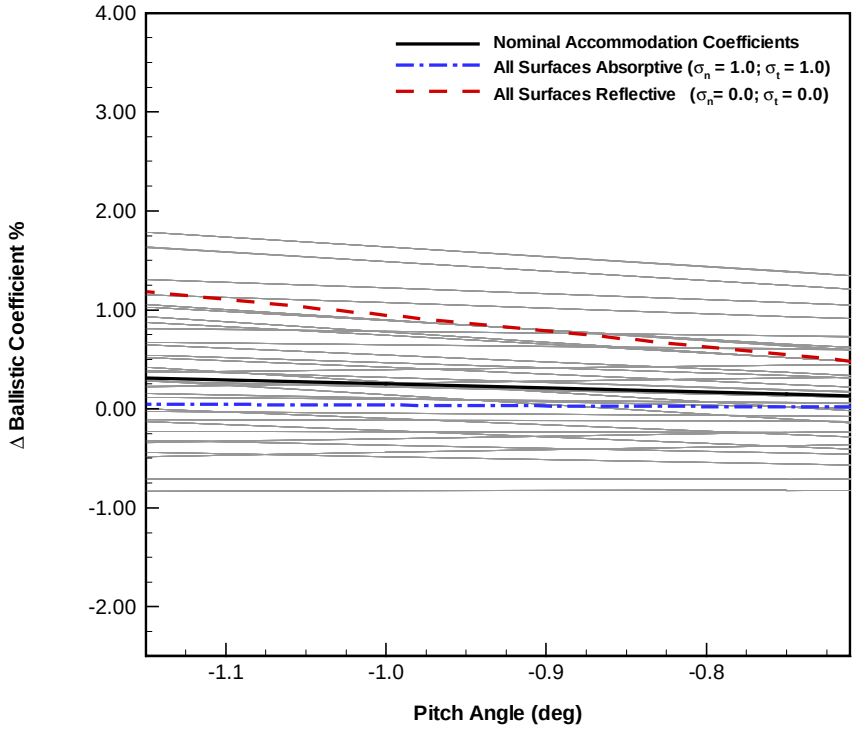

Figure 34. Differential Ballistic Coefficient vs. Pitch Angle for Nominal Separation Range (Small Range of Accommodation – Solar Array and Radiator Surfaces Similar).

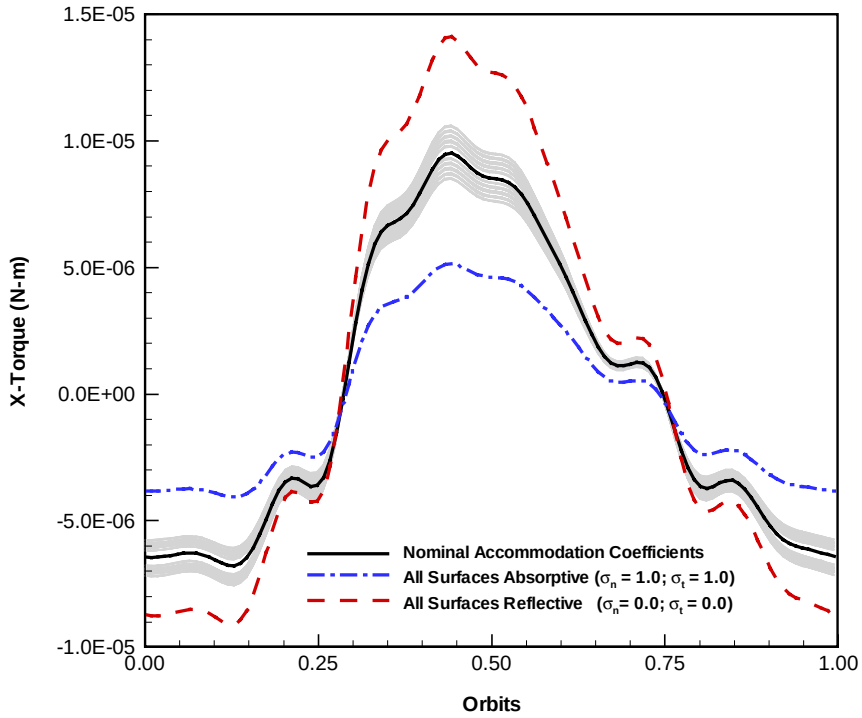

Figure 35. X-Body Aerodynamic Torque vs. Time for One Sample Orbit (Small Range of Accommodation).

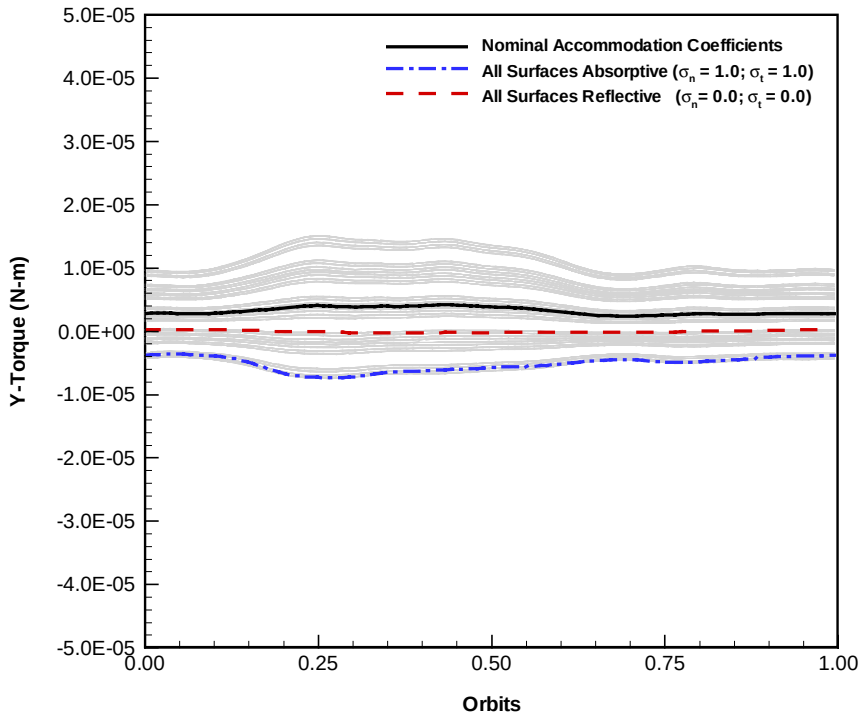

Figure 36. Y-Body Aerodynamic Torque vs. Time for One Sample Orbit (Small Range of Accommodation).

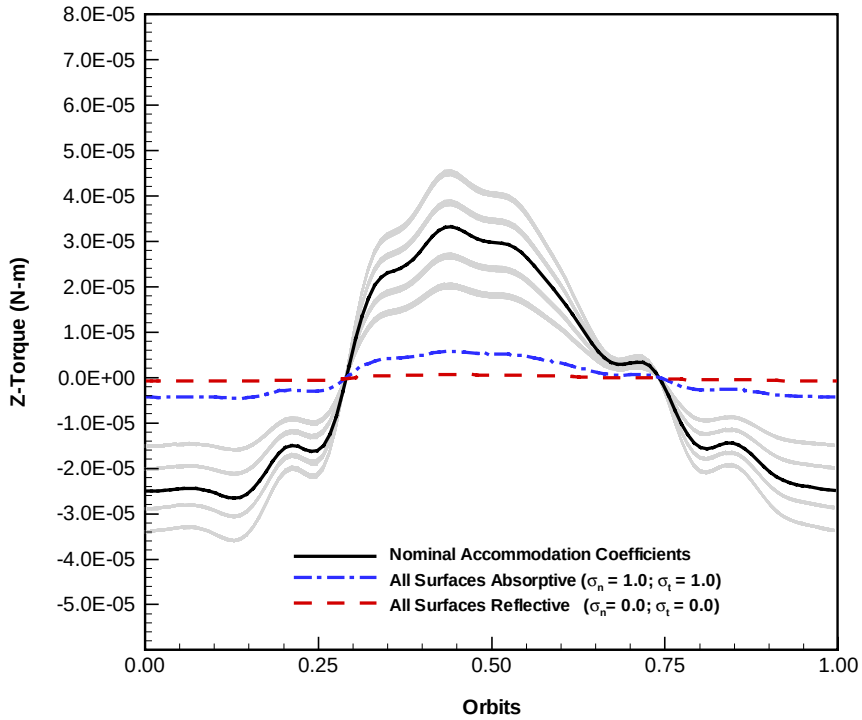

Figure 37. Z-Body Aerodynamic Torque vs. Time for One Sample Orbit (Small Range of Accommodation).

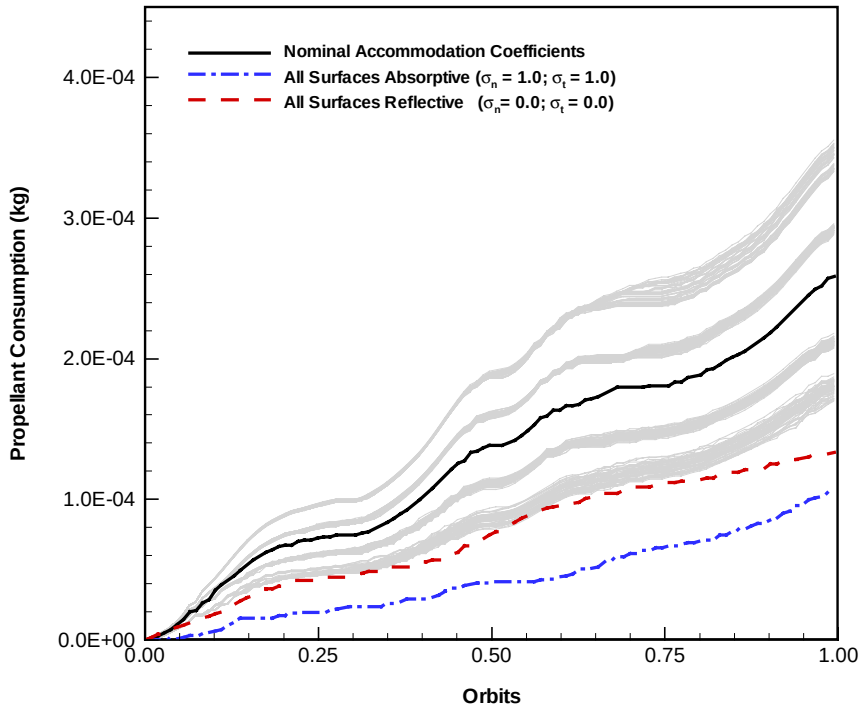

Figure 38. GN_2 Propellant Consumption vs. Time for One Sample Orbit (Small Range of Accommodation).

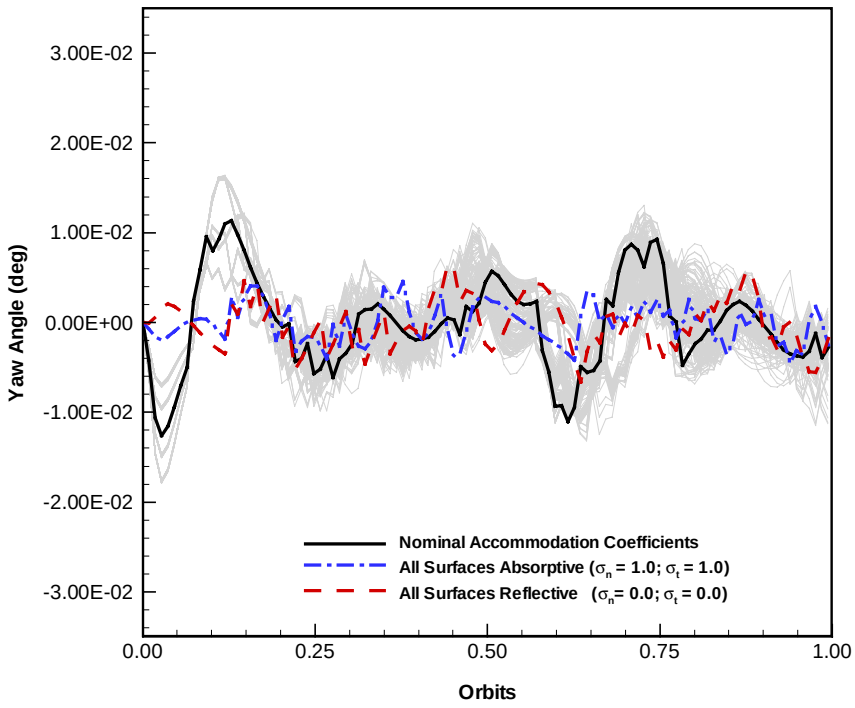

Figure 39. Yaw Angle (3-2-1 Euler Sequence) vs. Time for One Sample Orbit (Small Range of Accommodation).

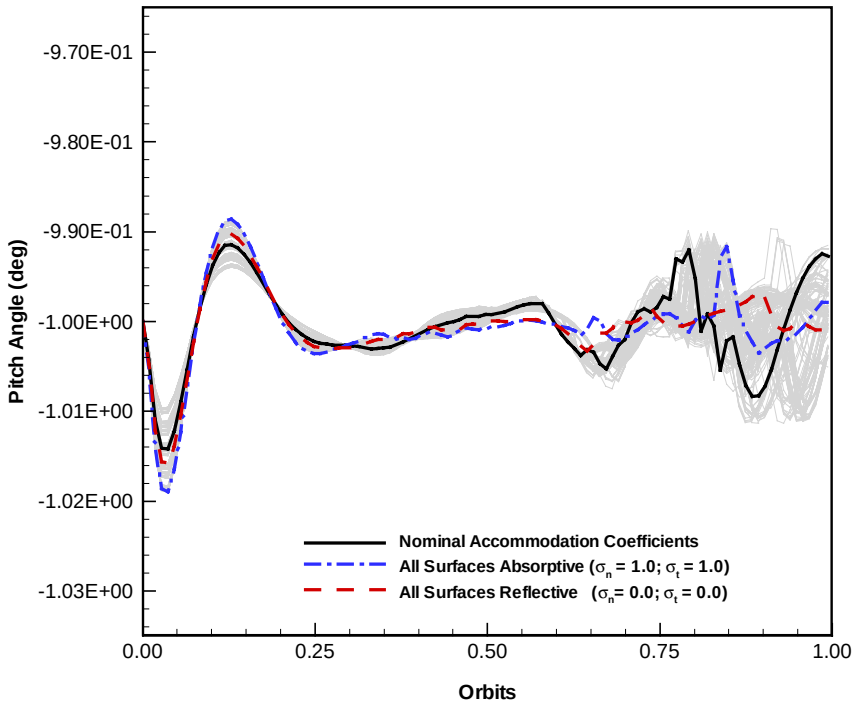

Figure 40. Pitch Angle (3-2-1 Euler Sequence) vs. Time for One Sample Orbit (Small Range of Accommodation).

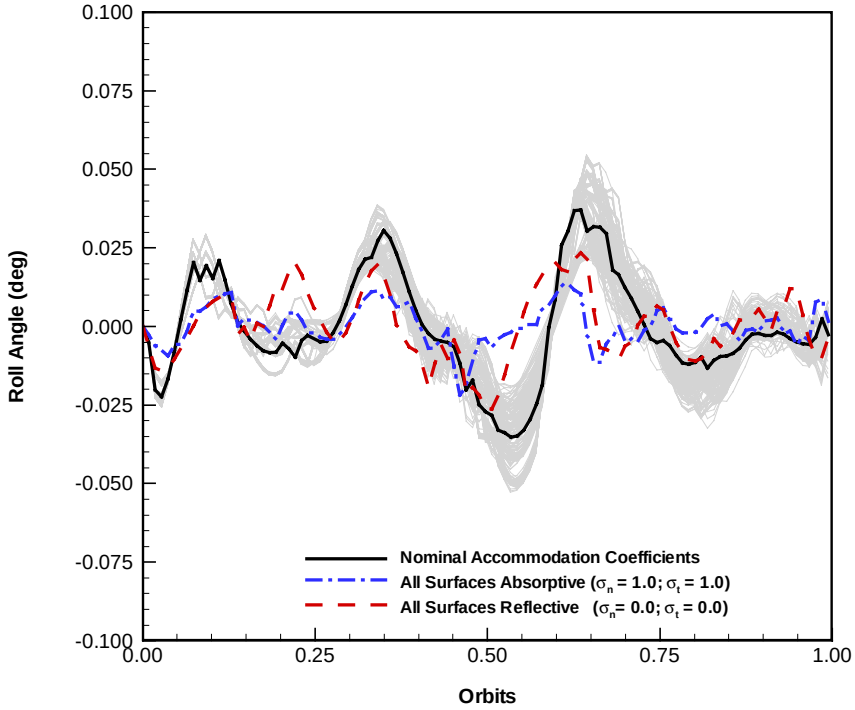

Figure 41. Roll Angle (3-2-1 Euler Sequence) vs. Time for One Sample Orbit (Small Range of Accommodation).

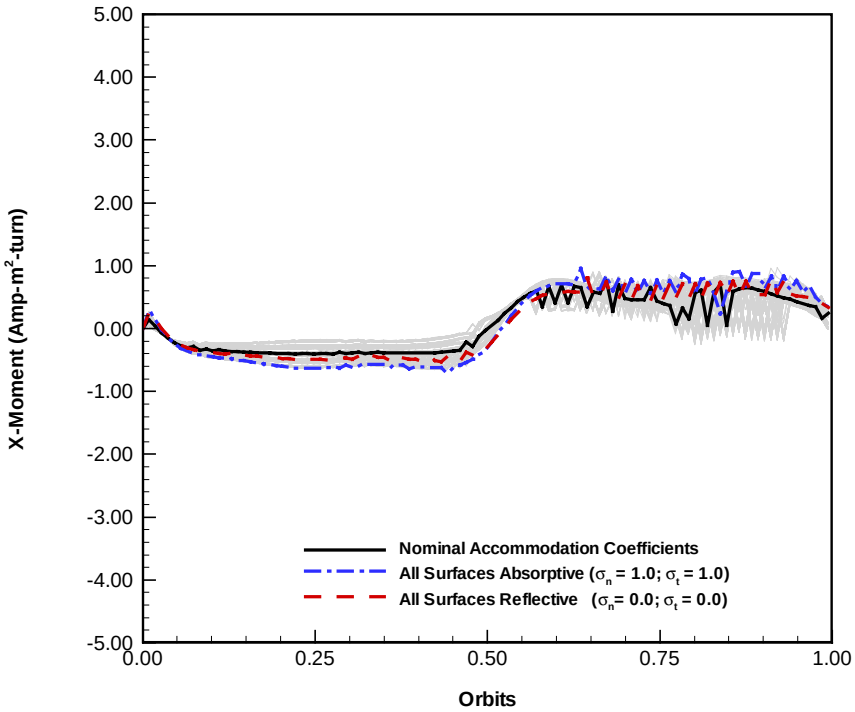

Figure 42. Magnetic Torque Rod X-Moment vs. Time for One Sample Orbit (Small Range of Accommodation).

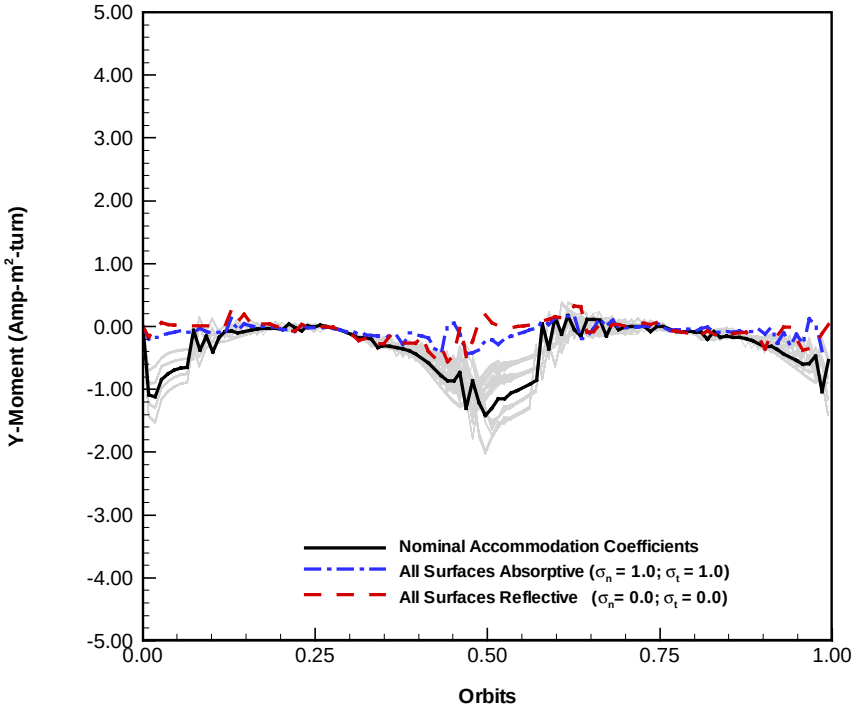

Figure 43. Magnetic Torque Rod Y-Moment vs. Time for One Sample Orbit (Small Range of Accommodation).

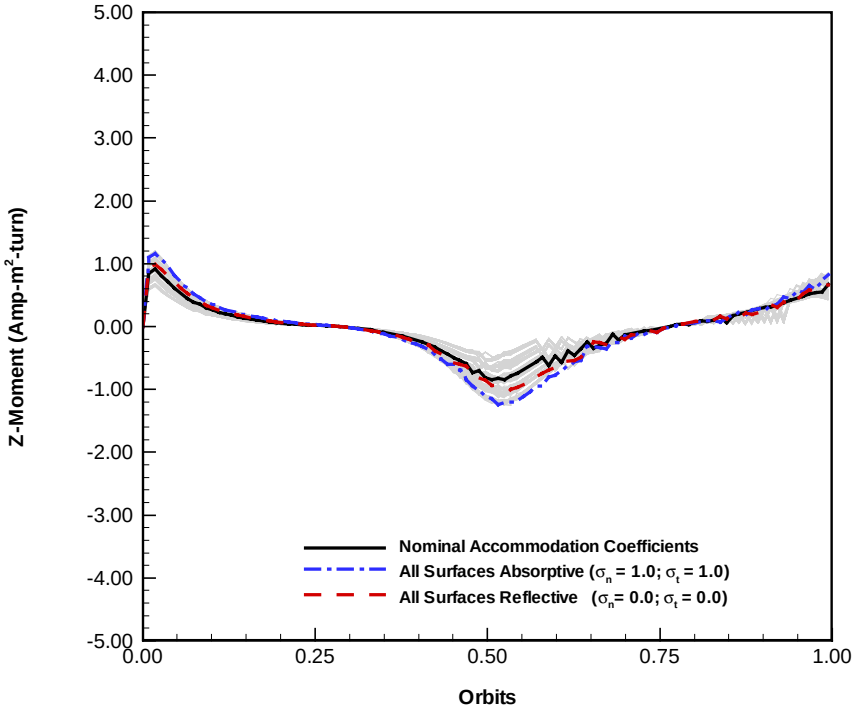

Figure 44. Magnetic Torque Rod Z-Moment vs. Time for One Sample Orbit (Small Range of Accommodation).

Sample Station-keeping Results

Based on nominal (50%) atmospheric predictions, the MSIS atmospheric density model, and conservative day of the year estimates for the altitudes of interest, density values were obtained as shown in Table 1. A station-keeping strategy [10] derived from simple linearized Clohessey-Wiltshire equations provided estimates of propellant consumption and maneuver frequencies as shown in Table 2. The propellant consumption data for each spacecraft (S/C) are for approximately one year. The values in Table 2 are based on the along-track formation spacing of the twin GRACE satellites being maintained between 170 and 220 km.

The maneuver frequency and propellant calculations are obtained using "good" estimates of the differential drag between the two satellites, as described previously in the section titled "Differential Drag Estimates and the Effect on Station-keeping."

The GRACE mission goal for station-keeping maneuver frequency is 60 days. Given the range of ballistic coefficient differential percentages, as can be seen in Table 2, this goal can be met even with a ΔB_c of 3% at the higher

ΔB_c (%)	Altitude (km)	Density (kg/m^3)	Prop. S/C 1 (kg/year)	Prop. S/C 2 (kg/year)	Total Prop. (kg/year)	Maneuver Frequency (days)
0.5	500	7.0e-13	0.007	0.011	0.018	200
1.0	500	7.0e-13	0.026	0.016	0.042	141
2.0	500	7.0e-13	0.037	0.044	0.081	100
3.0	500	7.0e-13	0.072	0.055	0.127	82
0.5	450	1.3e-12	0.025	0.015	0.040	146
1.0	450	1.3e-12	0.035	0.043	0.078	103
2.0	450	1.3e-12	0.081	0.091	0.172	73
3.0	450	1.3e-12	0.136	0.112	0.248	59
0.5	400	1.8e-12	0.029	0.036	0.065	124
1.0	400	1.8e-12	0.067	0.051	0.126	87
2.0	400	1.8e-12	0.131	0.108	0.239	62
3.0	400	1.8e-12	0.160	0.176	0.336	50
0.5	350	3.5e-12	0.066	0.050	0.116	88
1.0	350	3.5e-12	0.129	0.107	0.236	62
2.0	350	3.5e-12	0.233	0.252	0.485	44
3.0	350	3.5e-12	0.347	0.370	0.717	36
0.5	300	8.5e-12	0.142	0.157	0.299	56
1.0	300	8.5e-12	0.312	0.278	0.591	40
2.0	300	8.5e-12	0.600	0.552	1.152	28
3.0	300	8.5e-12	0.832	0.870	1.702	23
0.5	300	2.1e-10*	4.644	4.702	9.346	9
1.0	300	2.1e-10*	9.397	9.434	18.831	6
2.0	300	2.1e-10*	18.873	18.656	37.529	5
3.0	300	2.1e-10*	28.230	28.311	56.542	4

Table 2. Station-keeping Propellant and Maneuver Frequencies
(*Maximum Atmosphere).

altitudes (500-450 km). This is the altitude region where GRACE will spend much of the mission time, assuming nominal atmospheric conditions.

At the lowest altitude for the mission (300 km), the ΔB_c must be below 0.5% in order to meet the station-keeping frequency goal. A ΔB_c of 3% would lead to a maneuver frequency of 23 days at this altitude, assuming 50% atmospheric conditions.

Also included at the bottom of Table 2 are the station-keeping results assuming the Maximum Atmosphere conditions. Although these conditions are not expected to occur for the nominal GRACE mission, the results highlight the extremely rapid spacecraft separation and rapid altitude decay that can occur under these extreme atmospheric conditions.

Concluding Remarks

The FREEMOL software has been developed and used to analyze and suggest design modifications to the GRACE spacecraft. Aerodynamic accommodation bounding analyses were performed, and worst-case envelopes were obtained for the aerodynamic torques on each spacecraft and the differential ballistic coefficients between the leading and the trailing GRACE spacecraft.

Assuming *nominal* atmospheric conditions and theoretical worst-case uncertainty in surface accommodations, it has been shown that the cold gas nitrogen thruster system can control the GRACE satellite using a reasonable amount of GN_2 propellant in the science attitude control mode, provided that the three orthogonal 30 Amp-m^2 magnetic torque rods are utilized to supplement the cold gas thruster system.

The lack of experimental or theoretical data regarding gas-surface interactions in Earth orbit makes it difficult to determine "reasonable or feasible" worst case surface accommodation coefficients, and thus bound the differences in ballistic coefficients. This in turn makes it difficult to determine the station-keeping propellant usage and maneuver frequency. However, based on satellite drag observations a narrower set of feasible surface accommodations has been established. Based on this reduced set, and an assumption of *nominal* atmospheric conditions, analysis indicates that the worst-case differential ballistic coefficient could be as high as 3% for pitch angles corresponding to formation spacing control distances of 220±50 km. This could lead to a maneuver frequency of 23 days at an altitude of 300km.

The analysis results reported in this paper demonstrate how spacecraft aerodynamic design and analysis can benefit from a better understanding of spacecraft material surface accommodation properties. Although laboratory experiments currently provide important insight into the gas-surface interactions for various materials, they are not able to reproduce the environment the spacecraft surfaces are exposed to in Earth orbit. On-orbit experimental data for a variety of spacecraft surfaces over a sufficient period of time would greatly improve the fidelity and confidence of surface accommodation coefficients for spacecraft aerodynamic design and analysis in the future.

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Afterword

Very early in the formulation of the GRACE Mission we recognized the need to quiet the disturbing forces on the satellites - particularly those coming from the attitude control system. The accelerometer used by GRACE to measure the non-gravitational forces on the satellites is sensitive at the level of 1E-10 m-s^{-2}-Hz$^{-1/2}$. Aerodynamic forces are our main concern. Fortunately, NASA had the foresight to conduct the PAMS experiment in May of 1996. For the authors of this TM, this experiment provided an opportunity to develop a capability to model and understand forces and torques on a satellite in the free-molecular-flow environment. The GRACE Project was attracted by the experience of the authors. It was immediately recognized that the FREEMOL code implemented for PAMS produced answers quickly but it could not to model different accommodation coefficients for different facets of the external surface of the satellites. The authors solved this problem without compromising the efficiency of FREEMOL as a design tool. At this point, they solicited a peer review from colleagues at LaRC. Their colleagues challenged the assumption in FREEMOL that the thermal velocity of the molecules was negligible when compared to the satellite's velocity. Once again, the authors solved the problem and have accounted for the "finite speed ratio" without compromising the efficiency of FREEMOL as a design tool. The work represented herein has been important to the attitude control system design, the approach to controlling the separation between the twin GRACE satellites, the design of the mission, and the design of the external configuration of the satellites. The commitment to excellence of the authors and their respective organizations has won the respect of everyone associated with the GRACE Mission. It is the kind of commitment that is necessary in an environment where organizational interdependence is essential to success.

Ab Davis Project
Manager, GRACE Mission
Jet Propulsion Laboratory of the
California Institute of Technology
3 September 1999

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13. ABSTRACT *(Maximum 200 words)*
The Gravity Recovery and Climate Experiment (GRACE) primary mission will be performed by making measurements of the inter-satellite range change between two co-planar, low altitude near-polar orbiting satellites. Understanding the uncertainties in the disturbance environment, particularly the aerodynamic drag and torques, is critical in several mission areas. These include an accurate estimate of the spacecraft orbital lifetime, evaluation of spacecraft attitude control requirements, and estimation of the orbital maintenance maneuver frequency necessitated by differences in the drag forces acting on both satellites. The FREEMOL simulation software has been developed and utilized to analyze and suggest design modifications to the GRACE spacecraft. Aerodynamic accommodation bounding analyses were performed and worst-case envelopes were obtained for the aerodynamic torques and the differential ballistic coefficients between the leading and trailing GRACE spacecraft. These analyses demonstrate how spacecraft aerodynamic design and analysis can benefit from a better understanding of spacecraft surface accommodation properties, and the implications for mission design constraints such as formation spacing control.

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