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Abstract

The NASA Goddard Space Flight Center's Global Precipitation Measurement (GPM) mission must meet the challenge of measuring worldwide precipitation every three hours. The GPM core spacecraft, part of a constellation, will be required to maintain a circular orbit in a high drag environment at a near-critical inclination. Analysis shows that a mean orbit altitude of 407 km is necessary to prevent ground track repeating. Combined with goals to minimize maneuver operation impacts to science data collection and to enable reasonable long-term orbit predictions, the GPM project has decided to fly the GSFC autonomous maneuver system, AutoCon™. This system is a follow-up version of the highly successful New Millennium Program technology flown onboard the Earth Observing-1 formation flying mission.

This paper presents the driving science requirements and goals of the GPM mission and shows how they will be met. Selection of the mean semi-major axis, eccentricity, and the ΔV budget for several ballistic properties are presented. The architecture of the autonomous maneuvering system to meet the goals and requirements is presented along with simulations using GPM parameters. Additionally, the use of the GPM autonomous system to mitigate possible collision avoidance and to aid other spacecraft systems during navigation outages is explored.

INTRODUCTION

The Global Precipitation Measurement (GPM) mission is a follow on to the Tropical Rainfall Measuring Mission (TRMM) with the objective of covering a region of the Earth up to 65 degrees in latitude while providing sufficient measurement data for measuring short-term rainfall accumulations.[1] The GPM plan for achieving this increased coverage is to launch a constellation of satellites, each carrying radiometers. The first spacecraft of the GPM constellation, it is being built as a Goddard Space Flight Center (GSFC) in-house mission in

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partnership with the National Space Development Agency (NASDA), the space agency of Japan. The GPM spacecraft is currently scheduled for launch in 2008 aboard an H2A-202 launch vehicle. Its mission orbit will have at an inclination of 65 degrees and a circular altitude near 400 km. In science mode, the spacecraft will be 3-axis stabilized and earth pointing. The spacecraft will carry two radiometer science instruments and the GPM Microwave Imager (GMI) and will serve to calibrate and verify the standards to be used for the constellation. The orbit is chosen based on a series of trade studies, the results of which are presented in the following sections.

ORBIT TRADE SPACE

The goal of the orbit trade analysis is to determine the sensitivity to changes in altitude and inclination for the three onboard instruments: the radiometer, the Ku band radar and the Ka band radar (see Table 1). For this orbit trade, altitude is defined as the height above the Earth's mean equatorial radius of 6378.13 km. This analysis specified a better choice of a nominal altitude as well as the optimal size of the maintenance box (± 0.5km, ± 1km, or ± 5 km)[2]. We use two different figures-of-merit: (1) the time required to cover 100% of the ±65° latitude band and (2) the coverage obtained for a given propagation time (7 days and 30 days). The first figure-of-merit is used for the radiometer as it has a sensor cone half-angle between 3 and 5 times larger than the radars. Thus, it was anticipated that for this instrument the period of the orbit (i.e. altitude) will be the main driver and that the 100% coverage value will be reached within less than a week. The second figure-of-merit is used for the radar instruments as they have small sensor cone half-angle and will, in some cases, never reach the 100% coverage threshold.

Table 1.
Onboard Sensor Properties using a Simple Cone Model (Nadir Pointing)

	Radiometer	Ku Band Radar	Ka Band Radar
Reference Altitude (km)	407	407	407
Swath Width (km)	920	245	125
Half Swath Width (°)	4.1	1.1	0.6
Half-Cone Angle (°)	48	17	9

To complete the coverage analysis, GPM's orbit was integrated over a short time interval and the sensor coverage as a function of the elapsed time was recorded. The propagation step size was set to 60 seconds for the radiometer and 10 seconds for both radars so that the sensors' swaths were continuous. The propagation was stopped when the maximum coverage value (100% of the ±65° latitude band) or the maximum propagation time (30 days) was reached. The force model used in this simulation was limited to the J_2 gravitational perturbation. Eccentricity was held to 0.001 for the analysis. The coverage was restricted to the ±65° latitude band, which translates to a uniform distribution of 4566 points with an area of about 320 km x 320 km/point. Coverage achieved for the radiometer and radars at small variations about the nominal 400 km (+/-10 km) altitude and 65^0 (+/-2^0) inclination is shown in Figures 1 and 2. Results for the radiometer show that an increase in altitude above 400 km altitude yields 100% coverage at 2.5 days. For the Ku-band radar, note that two coverage maxima giving near 100% coverage are attained at two altitude and inclination ranges and that a minimum is found at approximately 395 km. The coverage decrease indicates that a ground-track repeat pattern has been reached and that global coverage may be impossible. A higher orbit altitude of 407 km is recommended for the mission altitude since it represents a less dense atmosphere than the other maxima located at 395 km, allowing for a decrease in maneuver frequency. This altitude is then converted into a mean semi major axis of 6777.14 km that is used in the following orbit maintenance scenario. Results for the Ka-band radar are similar to that of the Ku-band radar.

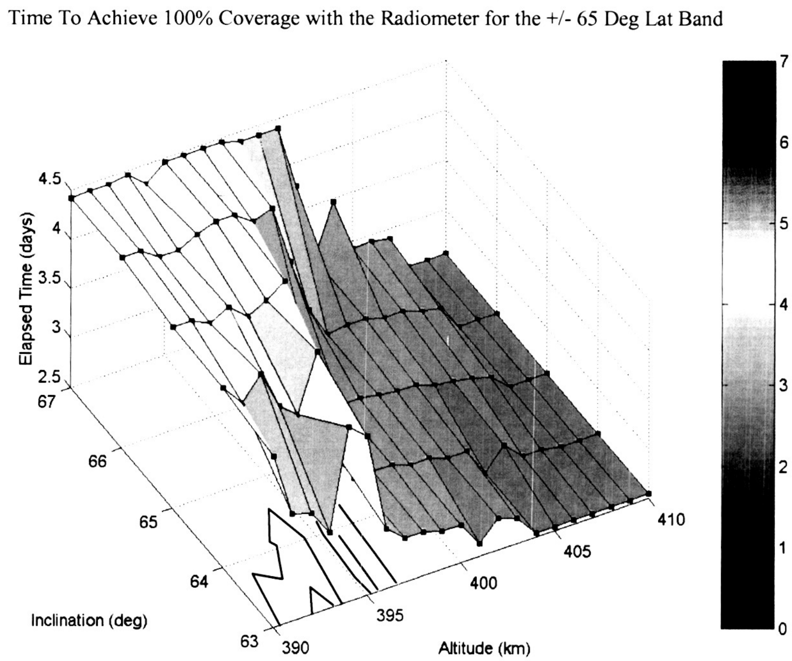

Figure 1. Coverage for Radiometer

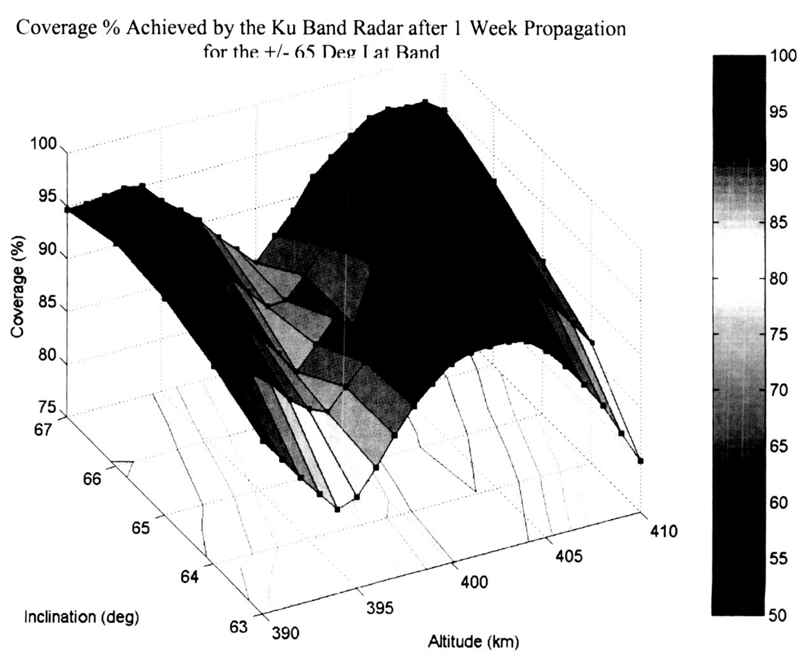

Figure 2. Coverage for Ku-band Radar

Frozen Orbit and Altitude Repeatability

The GPM orbit definition thus far is a circular orbit with an altitude of 407 km. This altitude was converted into a mean semi-major axis by simple addition to the Earth's mean equatorial radius to facilitate the determination of Keplerian elements. The inclination of 65^0 selected by the project was chosen in order to allow precipitation measurements at higher latitudes than the TRMM spacecraft, which has an inclination of 35^0. The right ascension of the node will be fixed by the selected launch window and is not reported herein. Other constraints such as instrument resolution and calibration, which are very sensitive to altitude variations, are used to define the remaining orbital elements of eccentricity and argument of periapsis. A well-known solution for minimizing the altitude variation for a given latitude is to use a frozen orbit[3]. For a frozen orbit, the mean eccentricity and mean argument of perigee are chosen such as to "freeze" the line of apsides motion under the Earth potential perturbations (mainly J_2 and J_3). A mean argument of perigee of 90° (or 270°) is usually required to reach such an orbit except when the spacecraft is at a critical inclination (63.5° or 116.5°). The combination of the Earth's oblateness and the eccentricity yields a predictable altitude variation over an orbit. However, the GPM scientists are not only interested in minimizing the altitude variation over a given latitude but would also like to minimize the amplitude of the altitude over one orbit. Since the specified inclination is near the frozen orbit critical inclination of 63.5^0, the mean argument of periapsis and mean eccentricity do not follow the typical closed evolution circular motion. The GPM inclination of 65^0 increases the time to complete a one period revolution to infinity. The eccentricity therefore becomes independent of the inclination and argument of periapsis terms and any value can be chosen. Figures 3 and 4 present the altitude variation over the orbit for a given eccentricity and the orbital parameter information vs. inclinations respectively. Figure 3 shows the orbit altitude variation at given latitudes over a 365-day duration. This data was generated from a simulation, which included stationkeeping maneuvers. The maneuvers maintained the mean semi-major axis to 1km tolerance and the mean eccentricity to the target of 1e-4. This mean eccentricity yields an overall amplitude variation of roughly 18km, with a repeatable band of approximately +/- 2 km. This band should not be confused with the maintenance of the mean semi-major axis to 6777.14 km +/- 0.5 km as the amplitude variation width is a function of eccentricity growth due to perturbations. Figure 4 shows how a selected eccentricity changes the amplitude of the GPM orbit altitude. For the GPM mission an eccentricity of 1e-4 with a tolerance of 0.5e-4 is recommended. The argument of periapsis can be allowed to float, as it does not have a significant impact on the altitude variation as shown in Figure 4.

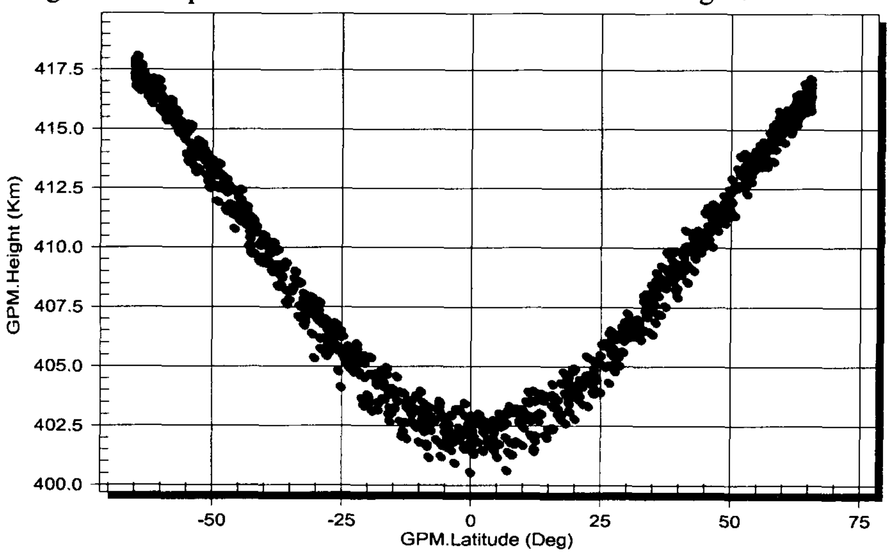

Figure 3. Latitude vs. Altitude for a Selected 407 km Orbit with e=0.0001 and ω = 0 degrees

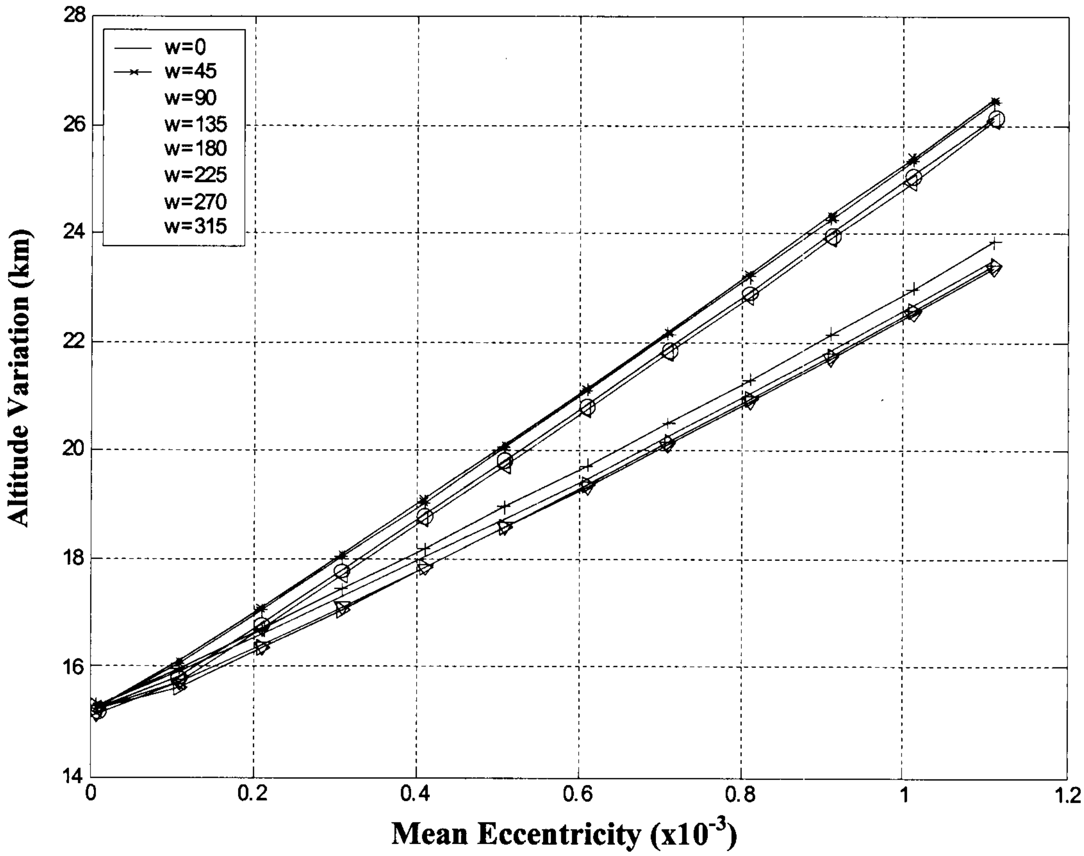

Figure 4. Altitude Amplitude Variation for a given Mean Eccentricity and Argument of Latitude

Orbit Control Box Trade

The goal of the orbit control box study is to determine which control box size is best suited for the GPM orbit parameters discussed previously. Considered are various metrics ranging from the science and instrument requirements to the spacecraft operations constraints. As before, a minimal variation in altitude is desired. Figures 5-7 show the altitude variation for latitudes of 0 deg., 40 deg. and 65 deg. respectively for three different orbit maintenance scenarios. The first scenario is labeled "Reference Orbit" or 'No Drag' as its force model only includes the Earth Geopotential (JGM2 4x4). The two remaining scenarios include the drag perturbation (Jacchia-Robert) in addition to the Earth potential perturbation defined for the reference orbit case. Their orbit is maintained to a +/- 0.5 km and +/- 5 km control box respectively about the reference orbit semi-major axis. With every maneuver modeled to restore the semi-major axis to the beginning of its box, the maneuver magnitude and direction is also varied such that the spacecraft eccentricity is reset to its recommended value. This value is slightly different from the reference orbit. Recall that the eccentricity value was chosen such that the altitude variation over an orbit is minimized. A duration of 65 days is chosen to coincide with a maneuver frequency for the +/-5km scenario.

Table 2 summaries the information in the figures[4]. The worst-case scenario is the +/- 5 km control where the altitude variation is about 8 km regardless of the latitude considered. Both the +/- 0.5 km control box scenario and the reference orbit exhibit small altitude variations (about 2 km) for a given latitude crossing. However, their performances differ depending on the latitude crossing considered. The +/- 0.5 km maintenance strategy not only maintains the spacecraft in a tight box about the optimum semi-major axis but also about the recommended eccentricity. This eccentricity is different from the reference orbit where no maneuvers are performed to correct its evolution under the Earth Geopotential perturbation. Note that for the +/- 0.5 km scenario, the altitude variation at all three latitudes considered is about 2 km. However, for the reference orbit, the altitude variation is more sensitive to the latitude crossings ranging from 3.707 km at the Equator to 0.507 km at a latitude of 65°.

Table 2.
Summary of Control Box Trade

ase #	+/- Box (km)	Total Box (km)	Altitude Variation Lat = 0° (km)	Max. Altitude Lat =0°. (km)	Min. Altitude Lat =0° (km)	Altitude Variation Lat = 40° (km)	Max. Altitude Lat =40°. (km)	Min. Altitude Lat =40° (km)	Altitude Variation Lat = 65° (km)	Max. Altitude Lat =65°. (km)	Min. Altitude Lat =65° (km)
1	Reference	N/A	3.707	405.294	401.586	2.351	411.734	409.383	0.507	417.658	417.151
2	+/- .0.5	1	2.842	404.049	401.207	2.351	411.610	409.259	1.582	418.788	417.205
3	+/- 5.0	10	7.048	406.971	399.923	7.683	414.679	406.995	9.568	422.529	412.961

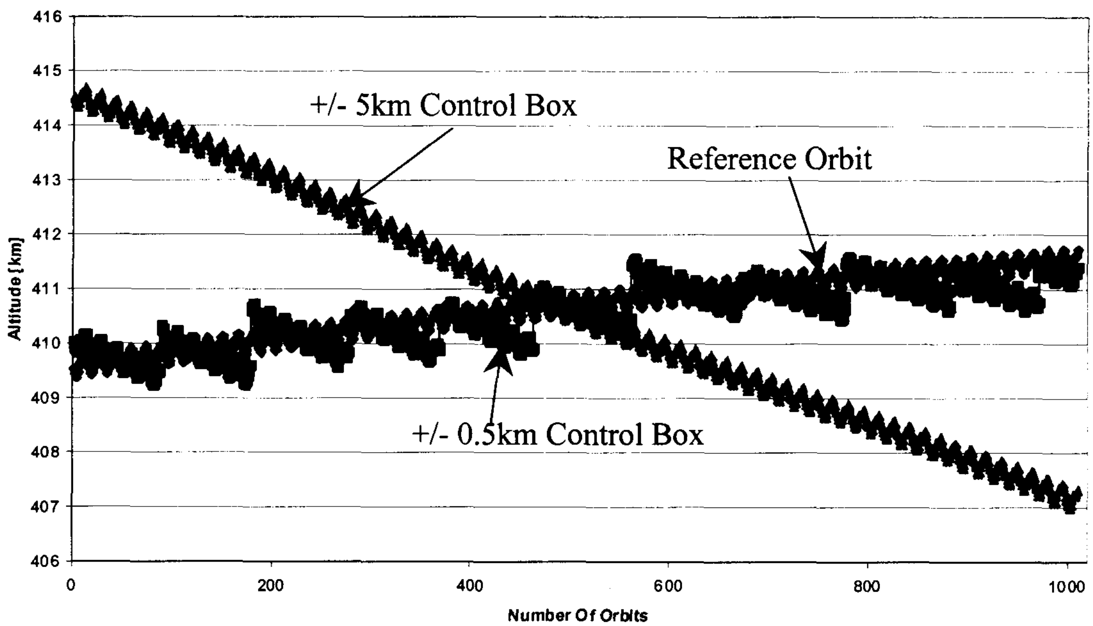

Figure 5. GPM Altitude Variation at the Ascending Node Crossing (Equator)

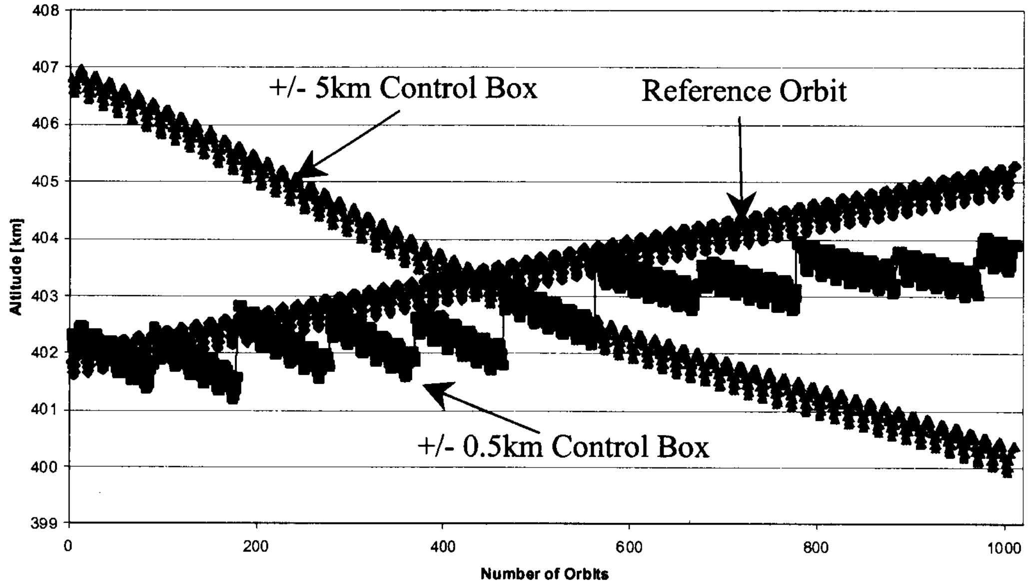

Figure 6. GPM Altitude Variation at Latitude = 40 deg.

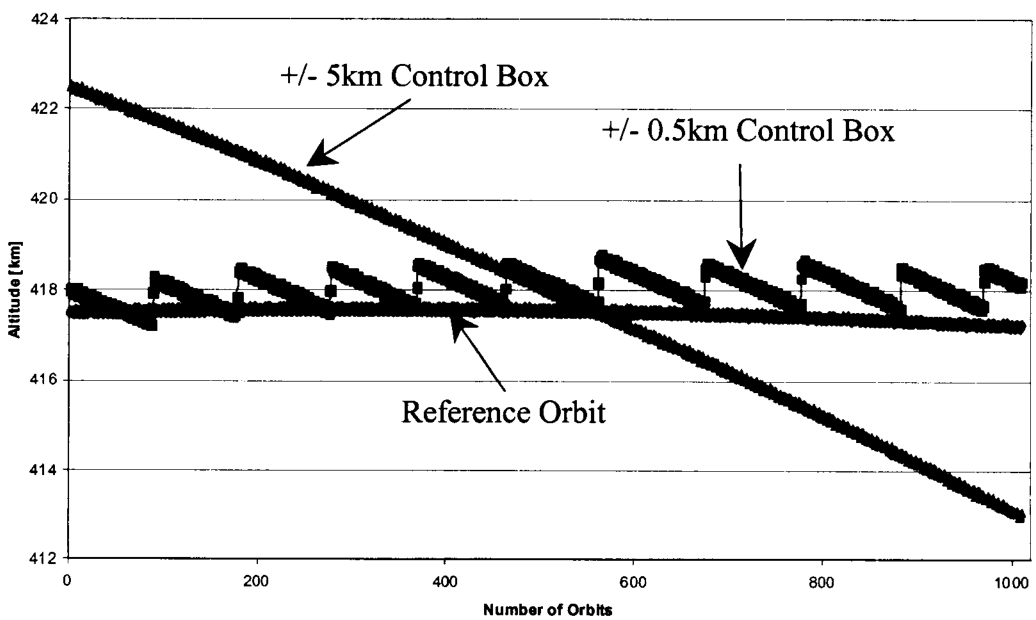

Figure 7. GPM Altitude Variation at Latitude = 65 deg.

MANEUVER ANALYSIS

To maintain the mission orbit parameters discussed above, a maneuver analysis is performed that spans the length of the mission.[5] A mission lifetime of three years with an extension of one year and a mission lifetime of five years with an extension of one year were analyzed. Maneuver analysis for the GPM spacecraft is dependent upon several external inputs and spacecraft characteristics. It is dependent upon the Schatten solar flux profiles, which are released several times per year as well as the ballistic properties of the GPM spacecraft.

A variety of launch dates further complicates the analysis. To bound the solar flux contribution, a profile of 'flying over the solar flux peak' was followed which centered the mission duration at the peak solar flux point. This profile allows a maximum ΔV scenario to be studied without regard to an exact launch date. Consideration is given for impulsive and finite maneuver models for a monopropellant hydrazine system. A sample of the stationkeeping analysis is provided in table 3 for a control range of 1km below the reference mean semi-major axis of 6777.14 km. The lower limit of the semi-major axis is 6776.15 km. Table 3 presents altitude, area -to-mass ratios, initial mass, total ΔV, total number of maneuvers, and the minimum and maximum time between maneuvers. Note that during the maximum solar flux periods maneuvers may occur as frequently as every other day. The atmospheric density used for the analysis is the +2sigma and mean density curves from the November 2002 release. The peak solar flux occurs on January 2011 with a f10.7cm value of 205.

The ΔVs are shown to be approximately 118 m/s for a GPM area to mass ratio of 0.003 m^2/kg. The time between maneuvers is a function of the solar flux and is a maximum at the beginning and end of the mission. Maneuvers at this area to mass ratio can occur as frequently as every three days or less if the solar flux has a spike lasting several days. The total number of maneuvers over a three-year duration is expected to be on the order of 220. The orbit was maintained to the Keplerian elements derived above. All maneuvers in this table were performed as Hohmann transfers using a differential corrector scheme.

Table 3.
GPM ΔV Analysis

Profile	Altitude (km)	Area-to-Mass (m^2/kg)	Mass Start -kg	Area m^2	Launch Epoch	Total ΔV (m/s)	# of maneuvers	average DV/man. (m/s)	min. time (days)	max. time (days)	Est. Fuel mass (kg) Approx
DMR#1	407	0.003	3000	9	Jun-09	118	216	0.546	3.1	11.3	160
peak +/- 18 months	407	0.005	3000	15	Jun-09	201	367	0.548	1.8	6.9	268
+2σ Solar Flux	407	0.007	3000	21	Jun-09	290	518	0.560	1.4	5	378
	407	0.009	3000	27	Jun-09	382	672	0.568	1	3.9	487
DMR#1	407	0.003	3000	9	Jun-09	146	268	0.545	3.1	11.3	197
peak +/- 18 months	407	0.005	3000	15	Jun-09	250	456	0.548	1.8	6.9	329
Plus 1-yr extended	407	0.007	3000	21	Jun-09	362	650	0.557	1.4	5	464
+2σ Solar Flux	407	0.009	3000	27	Jun-09	478	846	0.565	1	3.9	598
DMR#2	407	0.003	3000	9	Jun-08	96	175	0.549	5.9	38.5	131
peak +/- 30 months	407	0.005	3000	15	Jun-08	160	294	0.544	3.1	24.5	215
MEAN Solar Flux	407	0.007	3000	21	Jun-08	227	416	0.546	2.2	17.8	301
	407	0.009	3000	27	Jun-08	299	541	0.553	1.8	14.8	389
DMR#2	407	0.003	3000	9	Jun-08	109	198	0.551	5.9	38.5	148
peak +/- 30 months	407	0.005	3000	15	Jun-08	182	335	0.543	3.1	24.5	244
Plus 1-yr extended	407	0.007	3000	21	Jun-08	259	474	0.546	2.2	17.8	340
MEAN Solar Flux	407	0.009	3000	27	Jun-08	341	619	0.551	1.8	14.8	439

Minimizing ΔV Magnitudes

Moving the locations of stationkeeping maneuvers has an effect on the total ΔV magnitude for the mission duration.[4] By moving the locations of the maneuvers to the ascending and descending nodes, a minimum in the total ΔV budget was found. Figure 8 shows both the ΔV variation for a three-year mission lifetime and the total number of maneuvers as the argument of latitude is incremented about the orbit. The increase in total ΔV is a function of attaining the Keplerian orbit parameters and the inefficiency of maneuvers not performed near apoapsis and periapsis. This trade space analysis was run for mean semi-major axis control of 6778.14, +/- 1 km, but the general trend is not expected to change for the recommended orbit.

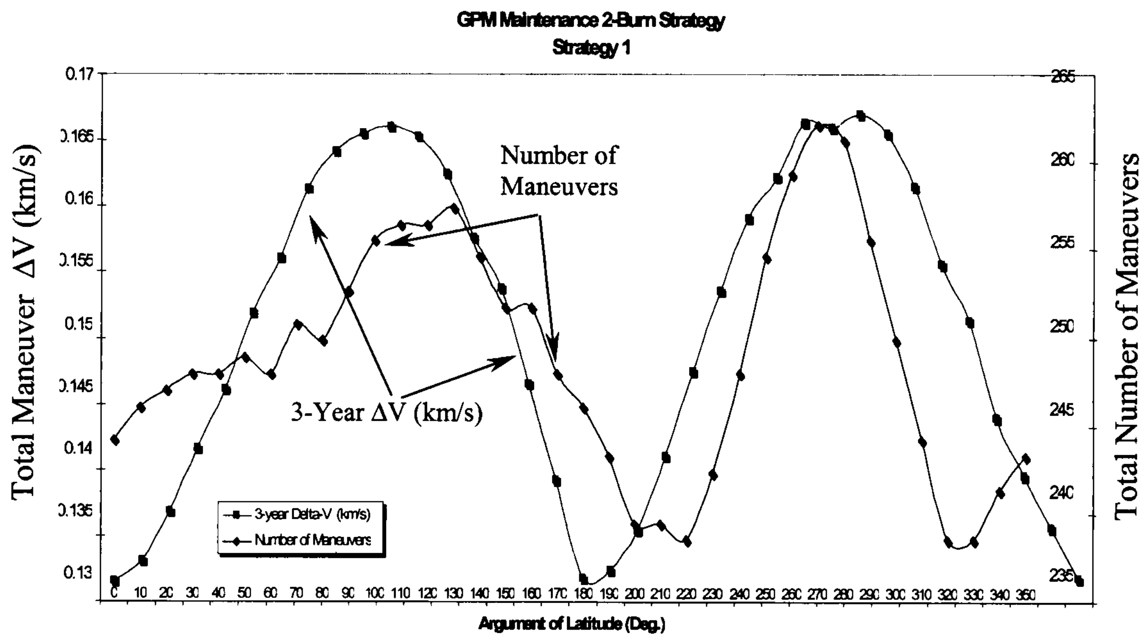

Figure 8. Variation in Total Maneuver ΔV and Number

APPLICATION OF NMP TECHNOLOGY - AUTOCON

Maintaining GPM precisely to the reference orbit while allowing frequent non-intrusive maneuver operations is one of the goals of the New Millennium Program (NMP). Maneuver operations frequently result in the loss of or in the reduction of science data collection. Combined with a low thrust propulsion system that minimizes attitude perturbations during maneuvers, a proven technology will allow continuous science data collection. For example, without the need to turn instruments off approximately 2 orbits out of every 28 (a two day maneuver frequency) during maneuver maintenance yields a 7% increase in data collection.

An orbit maintenance strategy is chosen that allows maneuver operations to be performed independent from ground intervention. To enable this strategy, the GPM mission will use an autonomous system, called AutoCon™.[6] AutoCon™ is a technology that was successfully demonstrated for a year onboard the NMP Earth Observing-1 (EO-1) spacecraft to meet formation flying requirements. The AutoCon™ system can be easily adapted to GPM altitude maintenance as the basic components of orbit prediction and maneuver planning remain the same as that on EO-1. Furthermore, the formation flying requirements for EO-1 are much more stringent than that of GPM orbit altitude maintenance. The maneuver maintenance algorithm used in the AutoCon™ system is the Folta-Quinn (FQ) universal 3-D algorithm that will maintain both the altitude, eccentricity and argument of periapsis if necessary, while allowing operations and environmental constraints to be considered.[7,8]

AutoCon Background

AutoCon™, a ground-based mission-planning tool, was originally developed to satisfy automation needs of the mission analyst. Ground-based AutoCon™ includes a user-friendly graphical interface, plots and report generation. It uses fuzzy logic to resolve multiple conflicting constraints and plan maneuvers. Fuzzy logic can be used to control mission planning through a rule-based scheme by combining constraints such as orbit true anomaly and shadow events. Mission and instrument constraints can also be incorporated into a flexible maneuver-planning scenario. The on-board flight version of AutoCon™ consists of a subset of the ground based AutoCon™ plus a flight software interface. Table 4 shows the functionality included in the flight and ground versions of AutoCon™. The flight interface connects directly with the Command and Data Handling (C&DH) system to retrieve all required data, including GPS position information, and to create command loads for computed burn times and durations. Only the objects and methods needed to support GPM are incorporated in the AutoCon™ system, conserving on-board resources. AutoCon™ inherits from its ground-based counterpart its object-oriented C++ design. Scaling the existing ground software for on-board use not only saves money in porting, but also saves in testing, since the development path automatically provides a ground reference system. AutoCon™ is designed to be flexible and extendable. It provides the architecture to facilitate interchangeable maneuver control algorithms. AutoCon™ is built around a structure called an *event*. Events can be added to AutoCon™ as necessary to support new algorithms or capabilities, thus providing expendability. To be flexible, AutoCon™ uses natural language scripting. The scripting provides the flow control. A new algorithm can be defined by events that are scripted to represent the algorithmic process. As long as all the necessary events exist, a new algorithm can be uploaded and executed on-board without changing the flight software.

Table 4. AutoCon™ Functions

Function	Ground	Flight
Multiple Spacecraft States	Yes	Yes
Time Conversion	Yes	UTC-TAI
Event Calculations	>100	10
Integrators	6	1
Coordinate Conversion	Multiple	EO-1
Maneuver Decision	Fuzzy Logic	EO-1 Specific Fuzz Logic
Force Models	Multiple	EO-1
C&DH Interface	None	EO-1

Onboard Development

The GPM autonomous system is modeled after the successfully flight demonstrated formation flying technology autonomous system. It was used to perform over a dozen autonomous maneuvers for EO-1 to maintain its formation with Landsat-7. Figure 9 presents the functional architecture of the proposed AutoCon™ system for the GPM mission. GPM is slated to fly the Position Velocity Time (PiVoT) GPS receiver. PiVoT includes a filter so the GPM version of AutoCon™ will not need the smoother that was flown onboard EO-1. The GPS data will be ingested and a simple trigger for maneuver planning based on mean semi-major axis will be performed. This computation will use the fuzzy logic system of AutoCon™ to kick off the maneuver planning when GPM approaches the lower mean semi-major axis.

It is anticipated that this computation would be performed every 12 hours, similar to the formation control box onboard EO-1. At that time, AutoCon™ will check for altitude violation and if necessary plan a maneuver that targets the mean semi-major axis, and eccentricity. If a maneuver is necessary, AutoCon™ will invoke the FQ algorithm using the high fidelity propagator to plan a maneuver that meets operational constraints such as shadow, sun angle, or other timing requirements. After the computation of the maneuver is completed, a maneuver command that is identical to a command load from the ground will be placed into the command data processor via the C&DH system.

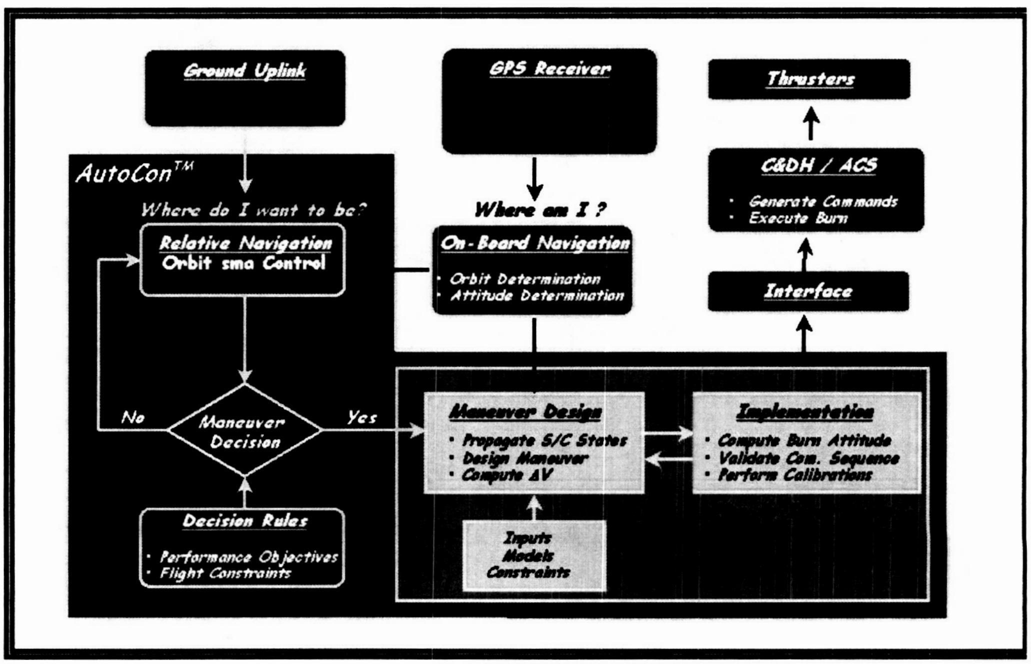

Figure 9. AutoCon Functional Diagram

The AutoCon™ flight control system is designed to be compatible with various onboard navigation systems (i.e. GPS, uploaded ground-based ephemeris, etc). It can backup the PIVOT system by using uploaded state vectors. One interface to the C&DH system is employed to obtain all onboard attitude and propulsion system data. This C&DH interface is used to ingest GPS state information, and command tables, and output telemetry and maneuver commands. The maneuver algorithm input data are provided internally though propagation of the initial states. Autonomous control requires that a known control regime be established consistent with mission parameters. Control data was provided once to the spacecraft as a set of relative altitude limits.

When orbital differential perturbations carry the spacecraft close to the established boundaries, the spacecraft reacts (via maneuver) to maintain itself within its altitude error box. The system is currently set to check the tolerance requirements every 12 hours. From this point, AutoCon™ propagates the states forward (a commandable setting) and will execute a maneuver plan if needed.

Several updates are required for the GPM version of AutoCon™. The work items for the changes from EO-1 technology experiment to GPM mission operations are list in Table 5. New items are listed in boldface.

Table 5. Changes to NMP EO-1 Technology for GPM

Work Items	GPM Assumptions
Interface Definition	Similar to EO-1
AutoCon Table Interface	Similar to EO-1
Building AC-F	GNU type compiler used, supported
AutoCon Users guide	Depends on FOT involvement
Benchmark testing	**Additional test for GPM scenarios**
Integration into FSW code base	Similar to EO-1
STOL Proc	**More Procs necessary**
Telemetry	**More telemetry definition**
Table create utility	**New tables formats**
S/C Testing	Similar to EO-1
Ops procedures	**More proc necessary**
Sys Engineering support	Similar to EO-1 during calibration

Maneuver Control Algorithm Description

The GSFC FQ algorithm for formation flying solves the formation maintenance problem by combining a modified Lambert problem and Battin's 'C*' matrix.[9] The algorithm enables the spacecraft to autonomously execute complex three-axis orbital maneuvers.[8] For EO-1, the maneuvers were restricted to in-plane components. EO-1 monitors the Landsat-7 position and performs maneuvers designed to maintain the relative position imposed by the formation requirements. The FQ algorithm plans maneuvers by determining a Keplerian path which will transfer the EO-1 spacecraft from some initial state, (r_0, v_0), at a given time, t_0, to a target state, (r_t, v_t), at a later time, t_t so as to maintain the formation. A desired state is also computed by back propagating the target state to find the non-maneuvered initial state (r_d, v_d) that EO-1 would need at time t_0. These states give rise to differenced states, δr and δv. The FQ algorithm computes state transition matrices for calculation of the maneuver ΔVs. Selecting initial conditions prescribed at a time t_0 a state transition matrix, $\Phi(t_1,t_0)$ can be constructed such that it will be a function of both t and t_0 and satisfy matrix differential equation relationships. Having partitioned the state transition matrix, $\Phi(t_1,t_0)$ for time $t_0 < t_1$ we use symplectic properties in equation 1 (assuming a reversible Keplerian path) to find the inverse where the matrix $\Phi(t_0,t_1)$ is based on a propagation forward in time from t_0 to t_1 and is sometimes referred to as the navigation matrix, and $\Phi(t_1,t_0)$ is based on a propagation backward in time from t_1 to t_0 and is sometimes referred to as the guidance matrix. When the fundamental matrix C* is defined as $C^* \equiv V^* R^{*-1}$, (see equation 2), it can be found using the submatrices that $C^*\delta r = \delta v_0$ becomes the velocity deviation required at time t_0 as a function of the measured position error δr at time t_0 if the spacecraft is to arrive at the reference position. With parameters derived from the Gaussian problem of planar motion, the target and desired states, and the F and G series using universal variables, R and V are defined. From sub-matrices the C* matrix is then computed and the expression for the impulsive maneuver generated, (see equation 3). For EO-1's orbit a long, iterative window requiring many small burns is not necessary, and ΔV maneuvers resemble a Hohmann transfer performed over 1½ revolution. This algorithm and strategy can be used for GPM without modification.

$$\phi^{-1}(t_1,t_0) \equiv \phi(t_0,t_1) \equiv \begin{bmatrix} \phi_1(t_0,t_1) & \phi_2(t_0,t_1) \\ \phi_3(t_0,t_1) & \phi_4(t_0,t_1) \end{bmatrix} \qquad \phi^{-1}(t_1,t_0) \equiv \begin{bmatrix} \phi^T{}_4(t_1,t_0) & \phi^T{}_2(t_1,t_0) \\ \phi^T{}_3(t_1,t_0) & \phi^T{}_1(t_1,t_0) \end{bmatrix} \tag{1}$$

$$C^*(t_0) = V^*(t_0)\left[R^*(t_0)\right]^{-1} \tag{2}$$

$$\Delta V = \left[C^*(t_0)\right]\delta r_0 - \delta v_0 \tag{3}$$

GPM Maneuver Safety

One of the major concerns of any mission is to make certain that the autonomous maneuver system is as safe as possible. There is always a considerable concern that an autonomous system would enable a command that would result in an extremely long maneuver duration and jeopardize the mission. Several safeguards are created to alleviate such concerns, including a standard of 48(tbd) hours notice before any planned maneuver (the time length is

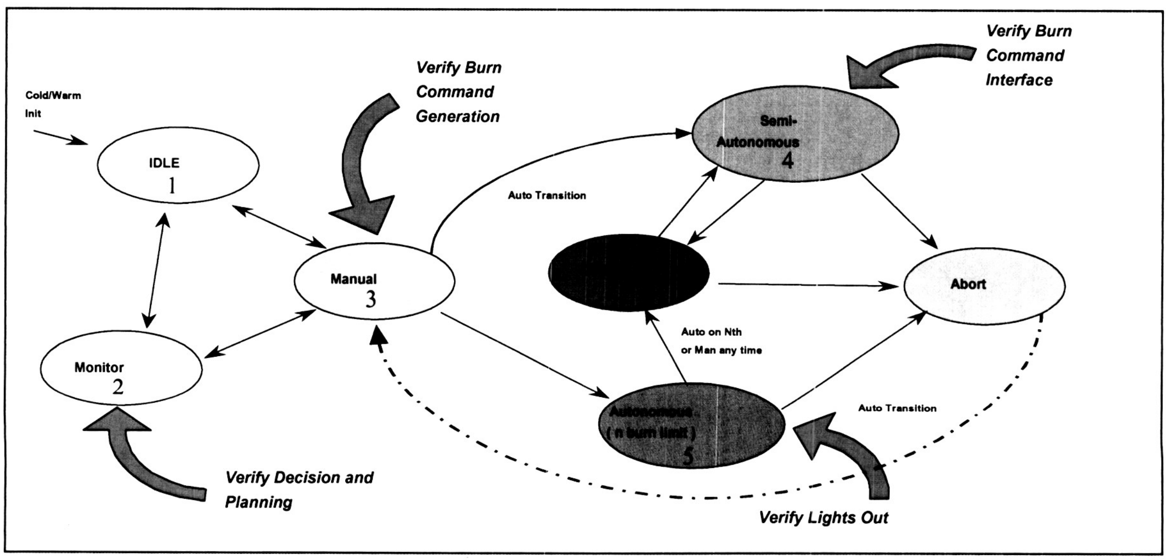

Figure 10. AutoCon Safety Modes

adjustable) and a phased in approach to autonomy. The 48-hour notice gives the ground time to review the planned maneuver before its execution. Safety modes, shown in Figure 10, include a monitor mode, which allows burn plans to be generated and reviewed, a manual mode, which allows maneuvers to be predicted but not implemented and a semi-autonomous mode, which allows burn plans and the resulting command to be verified by the ground before execution. The autonomous mode allows the generation and execution of a maneuver, but the maneuver information is still available two days prior to a maneuver event. The autonomous mode can be interrupted by ground command. Also, the autonomous mode is limited to a specified number of burns before it automatically transitions back to manual mode. In addition to AutoCon™'s built-in safety features, the attitude control system (ACS) limits all burns to 60 seconds or less. The stored command sequence also limits burn duration. Additionally, GPM may have a watchdog timer to make sure no task, such as AutoCon™ exceeds CPU utilization, depriving other critical tasks processing time. Finally, the spacecraft has a safehold mode that can disable AutoCon™, if necessary.

GPM AUTOCON MANUEUVER PLANNING RESULTS

Using the maneuver scripts defined during the EO-1 formation-flying mission, a GPM altitude maintenance maneuver plan was constructed. This plan amounts to changing the input state vectors to those of GPM and a reference and changing the target position for the maneuver-planning algorithm to a point that is 0.5 km above the reference orbit semi-major axis. Other files such as the atmospheric density files using a Jacchia-Roberts model, the Geopotential model inputs using an 8x8 degree and order, and Solar-Lunar-Planetary data remain the same. The change to the target was only a change in the relative position to the reference. For EO-1 the reference was the Landsat-7 spacecraft. For GPM the reference is a fictitious point directly above GPM when the lower altitude boundary is crossed. The simulation was set to plan a maneuver whenever GPM semi-major axis dropped 0.5km below the reference orbit. The targets and the reference input allow AutoCon™ to plan a maneuver that will re-establish the orbit at 0.5km above the reference orbit with the correct eccentricity. No changes were made for inclination. An area to mass ratio of 0.0033 was used.

Figures 11 through 14 presents simulation results that include three maneuvers. Each maneuver planned used the same target conditions of raising the mean orbit semi-major axis by 1km. The eccentricity is controlled through the definition of the reference orbit eccentricity. Figure 11 presents the mean semi-major axis of both GPM and of a reference orbit. The orbit decays to the lower boundary where a maneuver was executed. The maneuver is performed as a Hohmann transfer and places GPM into an orbit that is one km higher. After the maneuver, GPM is propagated until the lower condition is crossed again. As seen, the mean semi-major axis is maintained. For the onboard application, fuzzy logic that is already built into AutoCon™ will be used to further define the times that a maneuver can be planned. Figure 12 presents the evolution of the mean eccentricity, another parameter that needs to be controlled. The reference in this case shows an eccentricity of 1e-4 as specified in the previous analysis. The maneuvers executed maintain the eccentricity to the tolerance of 5e-5, also as specified earlier. Notice that the reference eccentricity trends toward a lower value for this simulation that uses an 8x8 Geopotential model and that the GPM eccentricity also follows this trend. The onboard implementation will take into consideration a constant semi-major axis and eccentricity rather than one determined from the propagation of a reference point.

Figure 13 presents the altitude variation over the orbital latitude reached by the 65 degree inclination. As discussed earlier, the profile that minimized the amplitude and the variation over each latitude is met. The variation in the band is approximately two kilometers, which is near the minimum that can be achieved, even with a zero eccentricity. The duration of the data on the plot is for three maneuvers. And finally, Figure 14 shows how GPM varies in the orbit with respect to a fictitious reference point. The plot shows the difference in the radial position vectors versus the difference in the along-track arc between GPM the reference point. The drift is due to the increased area to mass ratio (ballistic property) of GPM as compared to a fictitious point in the reference orbit. The results of this plot show that a long-term reference ephemeris can be produced for both science and communications planning and that the GPM can be controlled to a small along-track distance. The distance in this simulation amounts to a 60km drift, roughly an eight second difference in time. It is important to point out that this can be further minimized by centering the GPM motion about the reference point and that this benefit can be met without additional maneuvers or planning constraints.

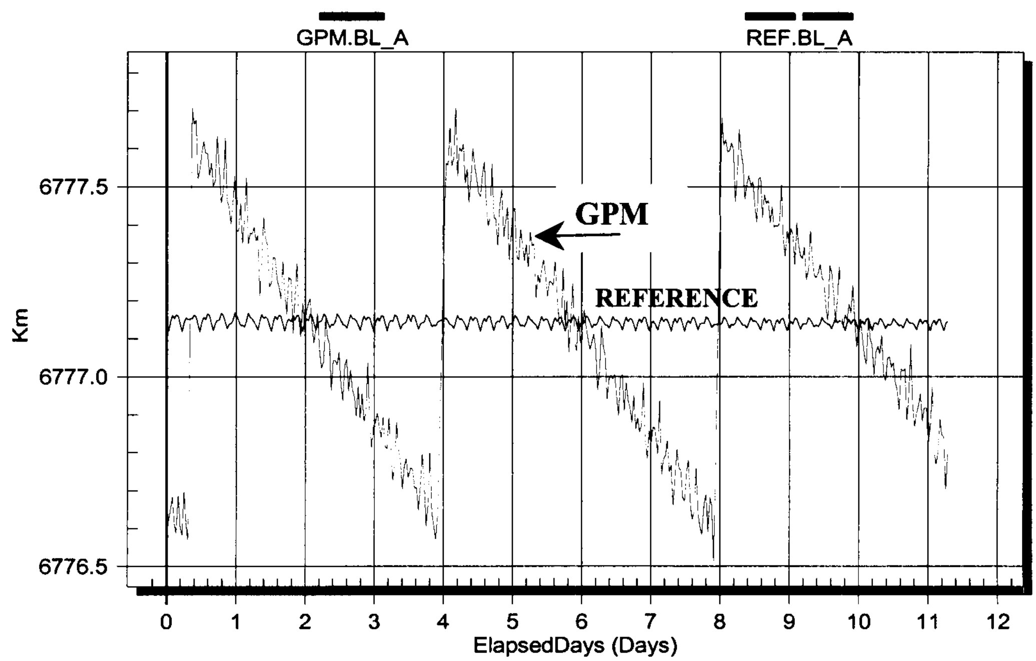

Figure 11. Mean Semi-major axis vs. Time for GPM and Reference

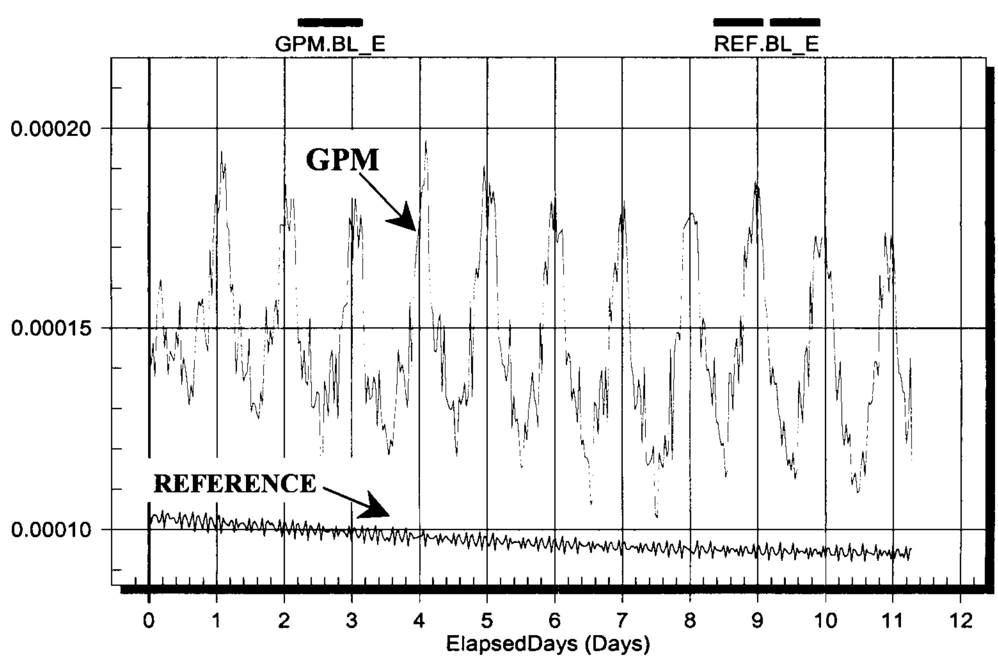

Figure 12. Mean Eccentricity vs. Time for GPM and Reference

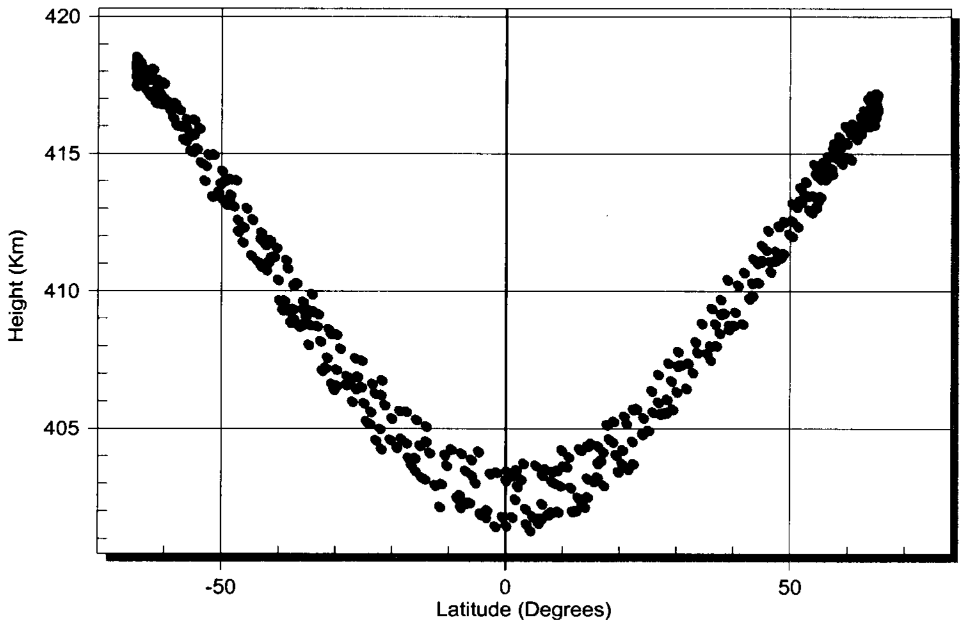

Figure 13. GPM Altitude vs. Latitude for 11 Days

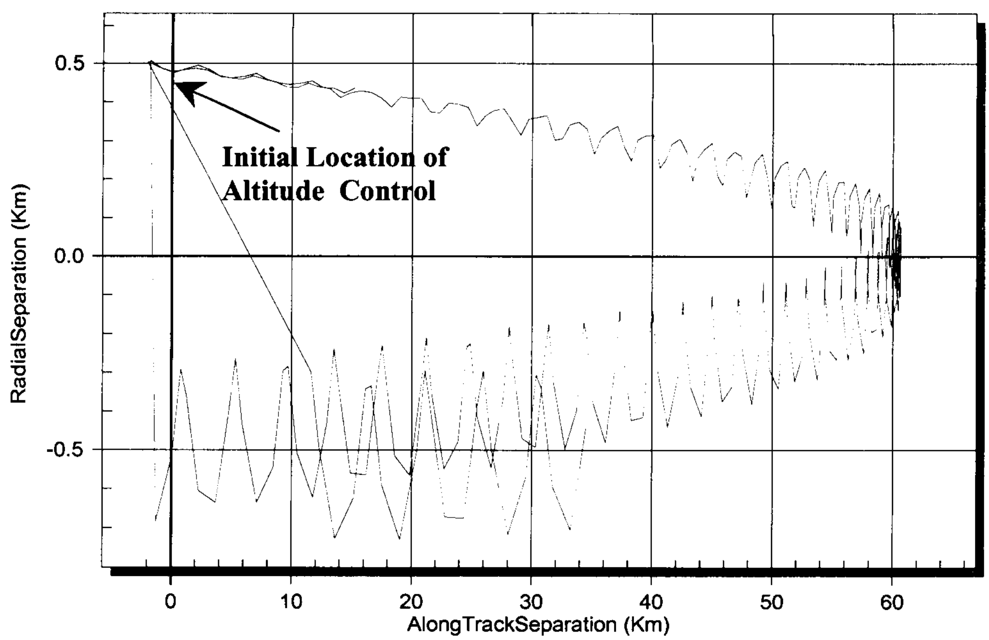

Figure 14. GPM Radial Separation vs. Along-track Separation

AUTOCON BENEFITS

The use of AutoCon™ also introduces other benefits into the GPM mission beyond the obvious orbit maintenance and reduced maneuver operations cost. In terms of ground and spacecraft operations, these benefits are:

Relative to ground operations -

- Eliminate or reduce ground maneuver operations
- Eliminate or reduce ground uplinks for maneuvers
- Eliminates ground post-maneuver calibration assessment
- Eliminates ground generation of post maneuver products
- Promotes continuous science data collection

Relative to spacecraft operations -

- Provides backup propagation for GPS PIVOT outages
- Provides data for HGA antenna pointing computations
- Accepts upload states for propagation and maneuvers
- Provides fuzzy logic to plan around science or orbital constraints
- Supports yaw maneuver planning
- Minimizes ACS propagation code
- Supports collision avoidance with other resources (e.g. ISS)

These benefits to the ground and spacecraft operations result in real cost savings while improving the quality of science data collection and increasing the amount of science data collected. This is accomplished by autonomously maintaining the reference orbit to precise specifications. The benefits for the spacecraft include reduced complexity in the generation of onboard data to meet navigation and communication requirements. Duplication can be eliminated as AutoCon™ can be used for backup ephemeris (GPS) input to the Attitude Control System (ACS), or to the science data processor. Collision avoidance with other NASA resources, such as the ISS, can be easily accomplished as AutoCon™ can ingest state vectors as simple table loads, which are propagated onboard and used by a maneuver planning script. The FQ algorithm already implemented within AutoCon™ can be used to compute the possibility of close approach or collision and plan a maneuver that will avoid such a problem.

CONCLUSIONS

The Mission design of GPM is driven by the need to maintain a close proximity to the reference orbit altitude. Orbit parameters have been recommended and ΔV analysis and AutoCon™ simulations have shown that the GPM orbit parameters can be easily maintained. The NMP EO-1 flight demonstration has proven that GPM can use this proven technology to meet orbit goals while doing so autonomously and reducing cost and errors associated with frequent uploads to a spacecraft. The benefits of autonomous maneuver planning make clear how autonomous technology developed under the NMP can be used to improve or enable the goals of GPM and other future NASA missions.

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