MANEUVER RECOVERY ANALYSIS FOR THE MAGNETOSPHERIC MULTISCALE MISSION

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Abstract

The use of spacecraft formations creates new and more demanding requirements for orbit determination accuracy. In addition to absolute navigation requirements, there are typically relative navigation requirements that are based on the size or shape of the formation. The difficulty in meeting these requirements is related to the relative dynamics of the spacecraft orbits and the frequency of the formation maintenance maneuvers. This paper examines the effects of bi-weekly formation maintenance maneuvers on the absolute and relative orbit determination accuracy for the four-spacecraft Magnetospheric Multiscale (MMS) formation. Results are presented from high fidelity simulations that include the effects of realistic orbit determination errors in the maneuver planning process. Solutions are determined using a high accuracy extended Kalman filter designed for onboard navigation. Three different solutions are examined, considering the effects of process noise and measurement rate on the solutions.

1 Introduction

The Magnetospheric Multiscale (MMS) mission is designed to study the interactions of the Earth's magnetosphere with high-energy solar plasma. The nominal MMS mission, which has an operational duration of two years, consists of four spin-stabilized spacecraft flying in a tetrahedral formation. The MMS mission will be conducted in two distinct phases, with each phase studying a different region of the Earth's magnetosphere. This study examines the Phase 1 orbit, which is a 1.2 x 12 Earth Radii (R_E) orbit at a 28-degree inclination with a period of approximately one day. During Phase 1, the inter-satellite separations will be adjusted between 10 to 160 kilometers near apogee, where the science measurements will be made.

Figure 1 illustrates the baseline space/ground operations configuration. Each formation member estimates the absolute and relative state vectors for all formation members. Global Positioning System (GPS) measurements for all formation members and crosslink measurements between all formation members are used. Each member transfers its GPS and crosslink measurements via an intersatellite communications link to every other formation member. The estimated state vectors are downlinked to the MMS operations center. The formation maintenance maneuver commands, which are generated on the ground using state predictions derived from onboard solutions, are uplinked to the spacecraft. Each spacecraft will fly an accelerometer to aid in accurate maneuver execution, modeling of the maneuvers in the onboard estimation process, and maneuver calibration for future maneuver planning.

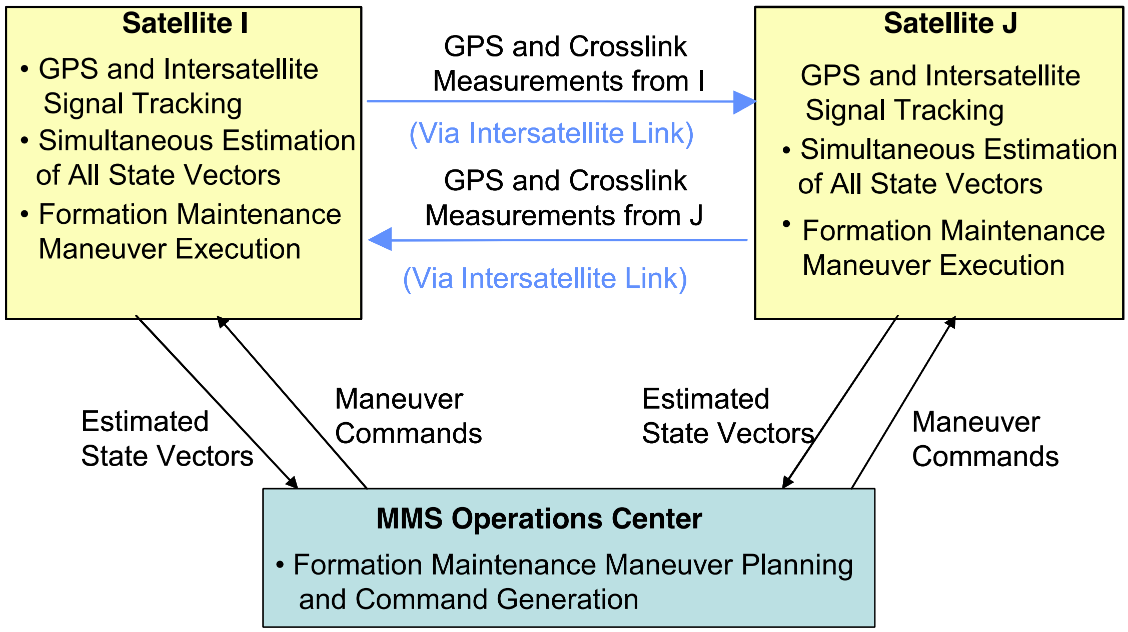

Figure 1: Space/Ground Navigation Configuration

The current formation maintenance concept for Phase 1 consists of executing maneuver pairs every 14 days, the first occurring at a True Anomaly (TA) of 202° and the second at a TA of 158°. These locations reduce any impact on science data acquisition, which occurs around apogee. This strategy provides a 10-hour window between maneuvers, during which the second maneuver is planned using the estimated states following the first maneuver. In addition, this 10-hour window occurs near perigee where the best GPS visibility occurs.

Each spacecraft will fly the GSFC-developed Inter-spacecraft Ranging and Alarm System (IRAS), which consists of the *Navigator* GPS receiver integrated with a crosslink transceiver and a high quality frequency reference (i.e. an ultra-stable oscillator (USO)). The tracking loops in the Navigator receiver are tuned to acquire low strength GPS signals to increase the number of GPS Space Vehicles (SVs) that can be acquired at high altitudes. This receiver has been demonstrated to reduce the acquisition threshold below 25 dB-Hertz as compared with a threshold of 35 dB-Hertz that is typical for GPS receivers designed for low Earth orbiting satellites.[i] Each IRAS will also acquire and transmit one-way crosslink range measurements from the other formation members at 4-minute intervals. The GPS pseudorange (PR) and crosslink range measurements and associated state vectors for each of the formation members will be provided as data via the intersatellite link.

To perform on-board orbit determination, the IRAS hosts the GPS Enhanced Onboard Navigation System (GEONS) flight software [1]. GEONS is a flight software package developed by NASA to provide onboard orbit determination for a wide range of orbit types. GEONS is capable of using GPS measurements and intersatellite crosslink measurements to simultaneously estimate absolute and relative orbital states. GEONS employs an extended Kalman Filter (EKF) augmented with physically representative models for gravity, atmospheric drag, solar radiation pressure, clock bias and drift to provide accurate state estimation and a realistic state error covariance. GEONS' high-fidelity state dynamics model reduces sensitivity to

[i] Values obtained from "Navigator/IRAS Measurement Noise Study", W. Bamford, Emergent Space Technology, September 2006

measurement errors and provides high-accuracy velocity estimates for accurate state prediction during measurement outages and periods with minimal GPS visibility and for science and maneuver planning.

This paper presents the results from a realistic simulation of the current formation maintenance strategy to identify approaches that will provide the navigation accuracy needed to meet the science objectives and the flight dynamics requirements for the MMS mission. The navigation accuracy requirements derived from the MMS science objectives and flight dynamics requirements are (1) definitive knowledge of the absolute spacecraft position to within a maximum of 100 kilometers, (2) definitive knowledge of the inter-spacecraft distances to within a maximum of 1% of the actual separation, (3) a maximum predicted relative position error growth rate to within 1% of the relative separation per day, and (4) a maximum absolute clock error less than 25 microseconds for the definitive and predictive periods[ii].

The software used in this simulation consists of three different programs. The truth trajectories for the MMS Phase 1 orbit are propagated using the Goddard Trajectory Determination System (GTDS), which is used for high accuracy operational orbit determination for a wide range of missions in GSFC's Flight Dynamics Facility. Measurements are simulated using the Measurement Data Simulation (DatSim) program [2] and processed using the GEONS flight software executed in a ground emulation test environment.

Section 2 describes the characteristics of MMS Phase 1 formation used in this study and Section 3 provides a high-level description of the simulation process. Section 4 presents the definitive and predictive navigation accuracy results for the navigation scenarios studied. Section 5 lists the major conclusions.

2 MMS Phase 1 Formation

This section describes the truth trajectory generation, inter-satellite separation, and GPS visibility for the Phase 1 formation spacecraft with a 60-kilometer separation. The truth trajectories are generated using GTDS with a truth force model including drag and solar radiation pressure (SRP) forces with $C_R = 1.4$, $C_D = 2.2$, Mass = 477 kg, Area = 2.5 m^2; Joint Gravity Model 2 50x50 geo-potential model; and point-mass gravity due to the Sun, Moon, Mars, Jupiter, Saturn, and Venus. The truth trajectories are generated using state vector data developed by S. Hughes[iii]. Table 1 lists the initial orbital elements at apogee for this MMS formation.

Table 1: Keplerian Elements for MMS Phase 1 60 Kilometer Separation Formation

Keplerian Elements	MMS Satellite 1	MMS Satellite 2	MMS Satellite 3	MMS Satellite 4
Semimajor Axis (km)	42095.7	42095.7000043072	42095.7000019023	42095.7000026211
Eccentricity	0.81818	0.81719081297	0.81749305346	0.81750706118
Inclination (deg)	27.8	27.80025587911	27.8052023372	27.7935905533
Argument of Perigee (deg)	15.000001	15.018466049	15.0026369333	14.904268095
Right Ascension of the Ascending Node (deg)	0	0.00076380491	359.94611	0.060163692
True Anomaly (deg)	180	179.9921269275	180.018888558	180.017909534

[ii] Russell Carpenter, GSFC, "Flight Dynamics Requirements for IRAS", September 2, 2003.

[iii] Steven P. Hughes, GSFC, "Formation Initial Conditions for Phase I, IIb, and III of the Magnetosphere Multiscale Mission (MMS)", November 2, 2005.

Figure 2 shows the variation of the inter-satellite separation for each pair of satellites in the formation over one complete orbit. Near apogee, where the science data is acquired, the inter-satellite distances are generally about 60 kilometers, but near perigee, the spacecraft separations vary widely. The separations can be as large as 350 kilometers and as small as 10 kilometers. A 10 kilometer separation at perigee is particularly challenging, because the relative position error must be less than 1% of the spacecraft separation. This indicates that the relative position accuracy must be better than 100 meters at perigee for one satellite pair (Satellites 3 and 4).

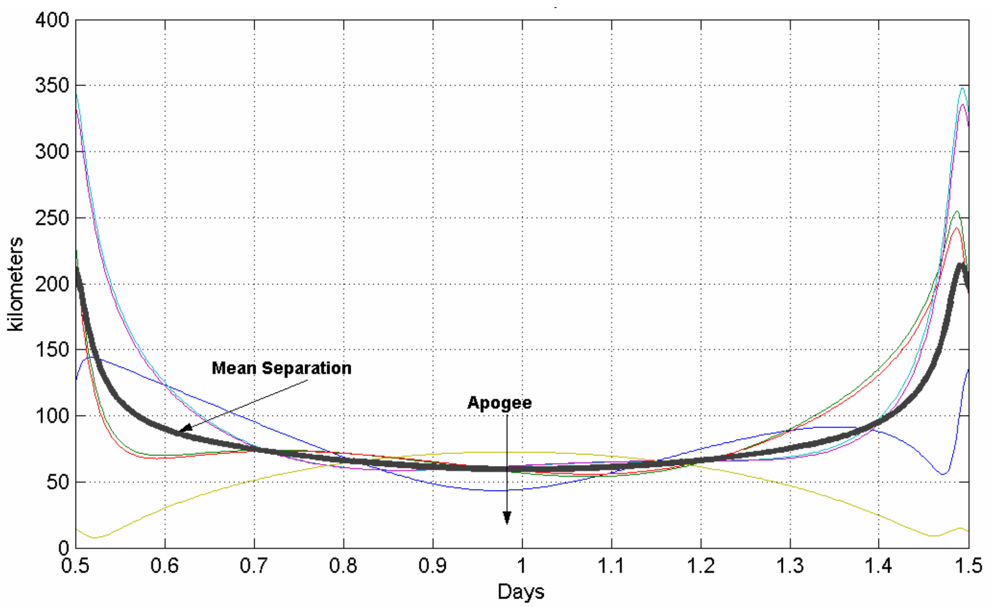

Figure 2: The Inter-Satellite Separation Distance for MMS Phase 1 60-Kilometer Formation

Figure 3 shows the GPS visibility for the spacecraft in the formation over two orbits assuming an acquisition threshold of 25 dB-Hertz. Near perigee, the maximum of 12 GPS SV's are visible (a 12-channel receiver is assumed) and, near apogee, fewer than three GPS SV's are visible. Due to the eccentricity of the orbits, the spacecraft spend approximately 16 hours of each orbit above the GPS constellation. There are only a few hours around perigee when a large number of GPS measurements are available.

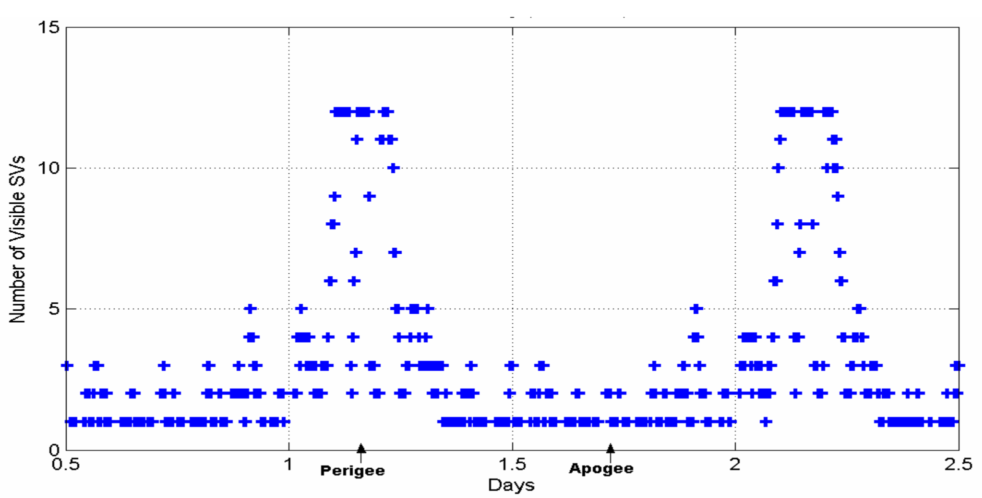

Figure 3: GPS Visibility with GPS Signal Acquisition Threshold of 25 dB-Hz for MMS Phase 1 60-Kilometer Formation

3 Simulation Methodology

The primary goal of this study is to determine whether the current navigation strategy can meet all MMS science and flight dynamics requirements, including the time periods immediately following each formation maintenance maneuver. To achieve this goal, realistic navigation and maneuver planning errors must be included throughout the simulation. The following three-step iterative simulation procedure was designed so that the estimated state errors are included in the maneuver planning process:

1. **Calculate a realistic maneuver for ΔV_1**
    a. Propagate the initial truth state using the truth model and simulate measurement data.
    b. Process the measurement data using the GEONS filter.
    c. Propagate the state estimate at 1 day prior to the first maneuver time, t_{P_1}, to the maneuver time, $t_{\Delta V_1}$, and use to compute ΔV_1.
2. **Calculate a realistic maneuver for ΔV_2**
    a. Add ΔV_1 to the truth state and propagate to the time of the second maneuver, $t_{\Delta V_2}$, using the truth model. Generate measurement data over this time span.
    b. Process the measurement data using the GEONS filter.
    c. Propagate the state estimate at two hours prior to the maneuver, t_{P_2}, to $t_{\Delta V_2}$ and use to compute ΔV_2.
3. **Generate post-maneuver data and perform end-to-end filter run.**
    a. Add ΔV_2 to the truth state and propagate for 14 days using the truth model. Generate measurement data over the first 7 days of this time span.
    b. Process measurement data using the GEONS filter, applying ΔV_1 and ΔV_2 and using a maneuver covariance consistent with a 1% error in each ΔV (consistent with the use of an accelerometer).
    c. Propagate the final estimated state for an additional 7 days.

The first two steps provide the truth trajectories and the simulated measurements that are used in the end-to-end filter runs. The formation maintenance maneuvers are modeled as impulsive ΔVs, which are computed using the Lambert Targeting algorithm in GEONS. The target state for the maneuver sequence was determined by propagating the initial tetrahedron back from apogee to a true anomaly of 158° using the truth model. Figure 4 shows the timeline of events in the simulation process. The timeline starts 14 days before the next formation maintenance maneuvers, using an initial filter state with errors consistent with steady-state filter performance.

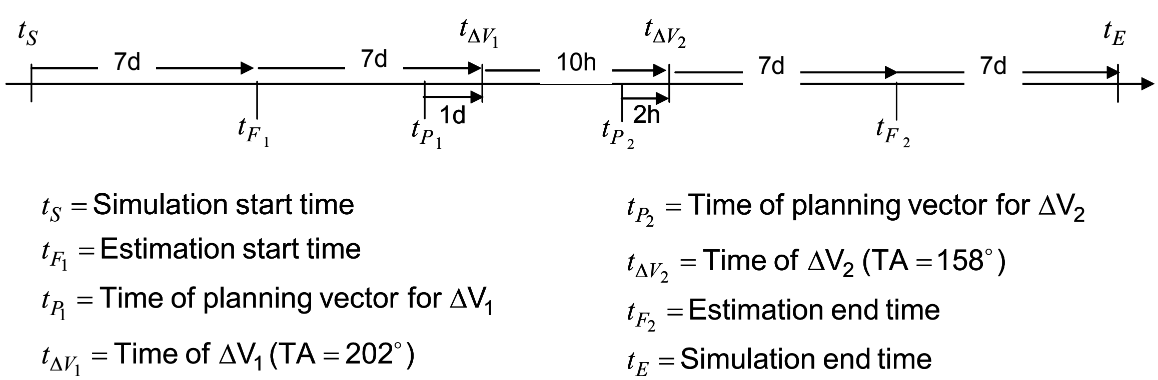

Figure 4: Simulation Timeline

The following simulated measurement errors were applied: GPS clock and ephemeris errors, GPS measurement noise, crosslink measurement noise, ionospheric delays, and receiver clock errors. Table 2 lists error sources and other simulation parameters and their values. All values are consistent with either the MMS operational scenario or the measured performance of the Navigator/IRAS.

Table 2: Baseline Measurement Simulation Input Data for Phase 1 60-km Formation

Simulation Parameter	Nominal Values
GPS PR Measurement Rate	1 Measurement set every 4 minutes for each formation member with measurements from all visible GPS SVs
1-way Cross-Link Range Measurement Rate	1 Measurement set every 4 minutes for each formation member with measurements from all other formation members
GPS Ephemeris and Clock Errors	2.0 meters
GPS Acquisition Threshold	25 dB-Hz
1-sigma GPS PR Noise[iv]	4.4 meters above 38 dB-Hz 6.1 meters for (30-38) dB-Hz 8.8 meters for (25-30) dB-Hz
1-sigma Cross-Link Range Noise[iv]	10.2 meters regardless of separation distance
Receiver Clock	USO error model
Ionospheric Delay Model	GPS Ionospheric Model
Minimum Height of Ray Path Altitude	1000 km (eliminates measurements with largest ionospheric delays)

Table 3 summarizes the GEONS filter tuning parameters used in this simulation. These tuning parameters are used for all four satellites, and include the radial (R), intrack (I), and crosstrack (C) velocity process noise parameters and measurement standard deviations. With the exception of the maneuver velocity process noise values, these tuning parameters are based on previous MMS navigation analyses [3, 4].

Table 3: GEONS Measurement Standard Deviations and Velocity Process Noise

Parameter	Value
Steady-State Velocity Process Noise Rate (RIC)	(10^{-14}, 10^{-13}, 10^{-13}) meters^2/second^3
ΔV_1 Velocity Process Noise Variance (RIC)	(5×10^{-5}, 5×10^{-5}, 5×10^{-5}) meters^2/second^2
ΔV_2 Velocity Process Noise Variance (RIC)	(5×10^{-4}, 5×10^{-4}, 5×10^{-4}) meters^2/second^2
Clock Bias Process Noise Rate	10^{-5} meters^2/second
Clock Drift Process Noise Rate	10^{-5} meters^2/second^3
GPS PR Noise Standard Deviation	40.0 meters
Cross-link Range Noise Standard Deviation	500.0 meters

[iv] Values obtained from "Navigator/IRAS Measurement Noise Study", W. Bamford, Emergent Space Technology, September 2006.

Table 4 lists the Mean of J2000 Cartesian coordinates for ΔV_1 and ΔV_2 values computed for each satellite in Steps 1 and 2. These maneuvers are applied to the truth trajectories that are used for all three solutions. Both ΔV 's are large, with ΔV_1 being almost 40 m/sec and ΔV_2 around 35 m/sec. This is due to the fact that the formation maintenance maneuvers were planned using an absolute target state corresponding to the reference tetrahedron from more than 14 days earlier. This results in maneuvers that rotate the line of nodes for the orbits back two weeks. The magnitude of these ΔV 's can be reduced by a factor of almost 100 if a locally constructed reference tetrahedron is used as the target state, which is the planned operational procedure. However, for the purpose of this analysis, using overly large maneuvers is beneficial. It serves to make the overall analysis more conservative, since the navigation system is forced to recover from a larger perturbation. Therefore, even though the formation maintenance maneuvers are larger than would be obtained using the planned operational procedure, they are retained for this analysis.

Table 4: Truth Formation Maintenance Maneuvers (ΔV_1 and ΔV_2)

	Satellite	Mean of J2000 Components (meters/sec)		
		X	Y	Z
ΔV_1	Sat1	26.6016	3.9805	-28.9699
	Sat2	26.3107	3.9460	-28.6924
	Sat3	26.3668	3.9160	-28.8264
	Sat4	26.3608	3.9205	-28.8244
ΔV_2	Sat1	16.3785	10.5516	29.8082
	Sat2	16.1740	10.4533	29.5234
	Sat3	16.2708	10.4574	29.5694
	Sat4	16.3133	10.4617	29.6040

4 Navigation Solutions

The measurement data simulated in Steps 1-3 were processed in three end-to-end GEONS filter solutions and compared with the associated truth trajectories. Table 5 lists the differences between the processing parameters used in the three solutions. Solutions 1 and 2 are consistent with the current MMS operational scenario, while Solution 3 uses a higher GPS measurement rate.

Table 5: End-to-End GEONS Filter Solutions

Solution	Characteristics
Solution 1	Baseline measurement rates ΔV_1 velocity process noise$^+$ applied for 2 hours starting from $t_{\Delta V_1}$ ΔV_2 velocity process noise^{++} applied for 2 hours starting from $t_{\Delta V_2}$
Solution 2	Baseline measurement rates ΔV_1 velocity process noise$^+$ applied for 2 hours starting from $t_{\Delta V_1}$ ΔV_2 velocity process noise^{++} applied for 3 hours starting from one hour before $t_{\Delta V_2}$
Solution 3	GPS PR measurement rate increased to one set every minute for 16 hours following $t_{\Delta V_2}$ ΔV_1 velocity process noise$^+$ applied for 2 hours starting from $t_{\Delta V_1}$ ΔV_2 velocity process noise^{++} applied for 3 hours starting from one hour before $t_{\Delta V_2}$

$^+$ from Table 3, line 2
$^{++}$ from Table 3, line 3

Figure 5 compares the behaviors of the root-sum-square (RSS) position errors (equal to the difference of the estimated and truth trajectories) and the estimated root-variances from Solutions 1, 2, and 3 around the two maneuvers. All three solutions are identical up to $t_{\Delta V_1}$. Between $t_{\Delta V_1}$ and $t_{\Delta V_2}$, there are only minor differences in the solutions. The perturbation due to ΔV_1 appears to settle down quickly because of the large number of GPS PR measurements available near perigee. In this case, the filter reconverges to steady-state performance within 6 hours, in time to successfully plan ΔV_2. Even during the transient period, the errors are small enough to satisfy the absolute and relative position error requirements. The perturbation due to ΔV_2 requires longer to reconverge and also causes more significant difference among the different types of solutions examined here. This is primarily due to the poor GPS visibility following $t_{\Delta V_2}$. When the GPS visibility starts increasing again near the next perigee, the filter reconverges to steady-state performance that is equivalent to the pre-maneuver steady-state solutions. Following ΔV_2, it takes approximately 16 hours to reconverge to steady state performance.

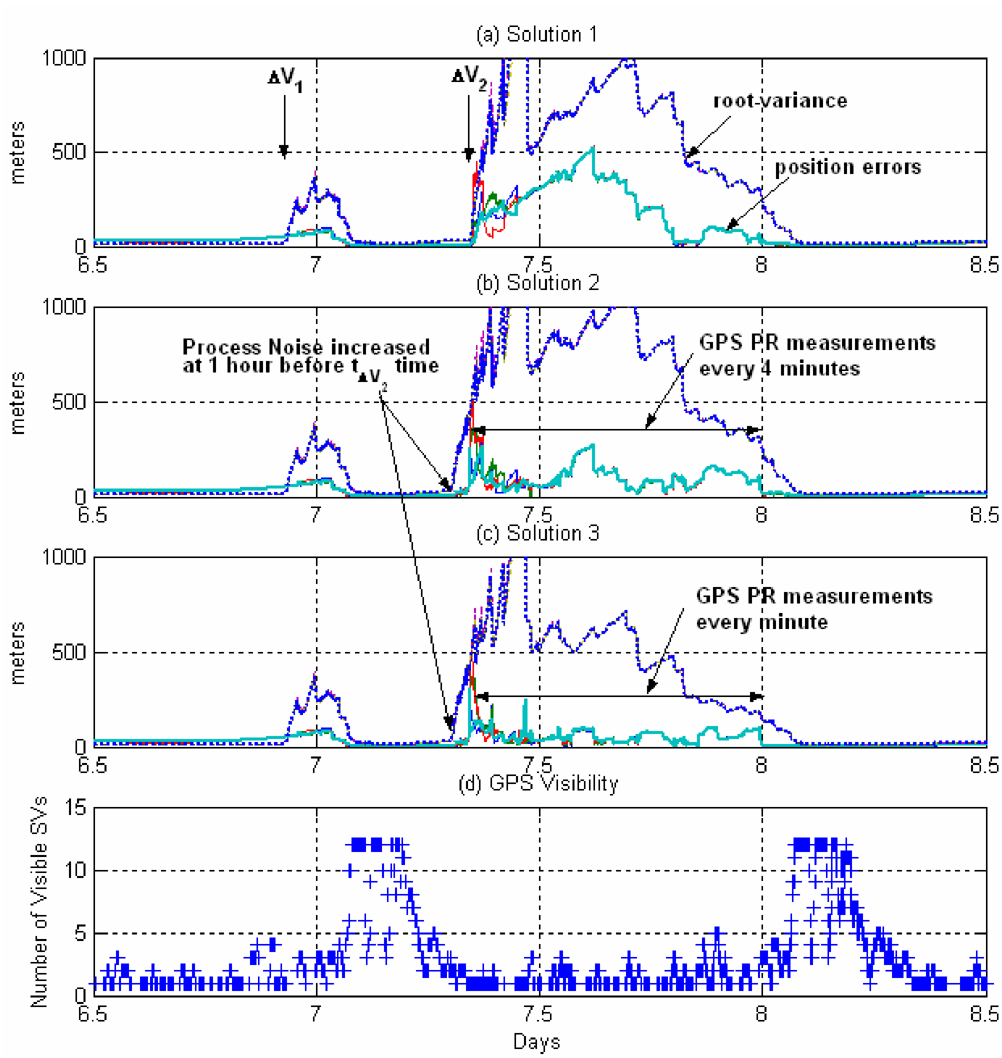

Figure 5: Position Error Behaviors Around the Maneuvers

With the exception of the time immediately following $t_{\Delta V_2}$, the navigation errors of all three solutions are very similar. This is due to the fact that the filter tuning parameters are identical, except for the time around the maneuvers. Solution 2 is discussed in detail for the remainder of this section, since it is representative of all three solutions.

Table 6 summarizes the root-mean-square (RMS) and maximum errors for the pre- and post-maneuver steady state time-periods for Solution 2. The RMS absolute position errors for the post-maneuver steady state period are somewhat larger than those of pre-maneuver steady-state solutions. The maximum relative position errors for the post-maneuver steady state period are somewhat larger than those of pre-maneuver steady-state solutions. Both the absolute and relative errors for the steady-state periods are similar for all satellites.

Table 6: Position and Velocity Errors Statistics for Steady State Solutions (Solution 2)

Satellite and Satellite pairs	Pre-Maneuver Steady State Period		Post-Maneuver Steady State period	
	RMS	MAX	RMS	MAX
Absolute Position Errors (Meters)				
Sat1	34.9254	99.8390	44.7740	107.4585
Sat2	33.4603	98.5086	43.3657	104.2583
Sat3	34.1002	97.3192	43.7402	104.1961
Sat4	34.1612	102.6456	43.7640	104.8381
Relative Position Errors (Meters)				
Sat2-Sat1	2.5288	4.5911	2.6229	11.6226
Sat3-Sat1	1.8597	4.5227	2.5290	10.7691
Sat4-Sat1	3.2278	6.2587	2.7853	13.7577
Sat3-Sat2	2.3130	6.8293	2.3099	10.5461
Sat4-Sat2	2.1363	5.4859	2.4951	14.4267
Sat4-Sat3	3.5453	7.8890	1.8380	6.9770
Absolute Velocity Errors (Millimeters/Second)				
Sat1	2.6198	11.3308	2.8689	11.2113
Sat2	2.5448	11.1583	2.7198	11.0411
Sat3	2.5644	11.1073	2.7683	10.9557
Sat4	2.5946	11.7123	2.7483	11.0310

Figure 6 shows the absolute RSS position errors and the estimated root-variances for Solution 2. The absolute estimated root variances are smaller than the true absolute position errors. Figure 7 shows the relative position errors for Solution 2 between the "local" satellite (satellite 1) and the "remote" satellites (satellite 2-4) and between the remote satellites. The relative root variances are larger than the true relative position errors. Estimating the relative states directly has subsequently been shown to provide more reasonable relative root variances. However, even though the variances are sub-optimal, all three solutions easily meet the absolute navigation accuracy requirement of less than 100 km absolute state errors.

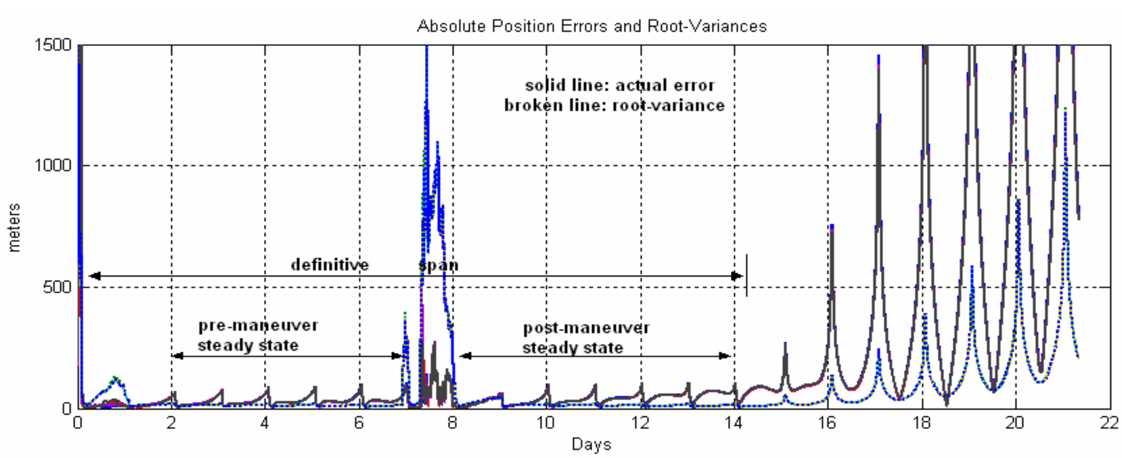

Figure 6: Absolute Position Errors for Solution 2

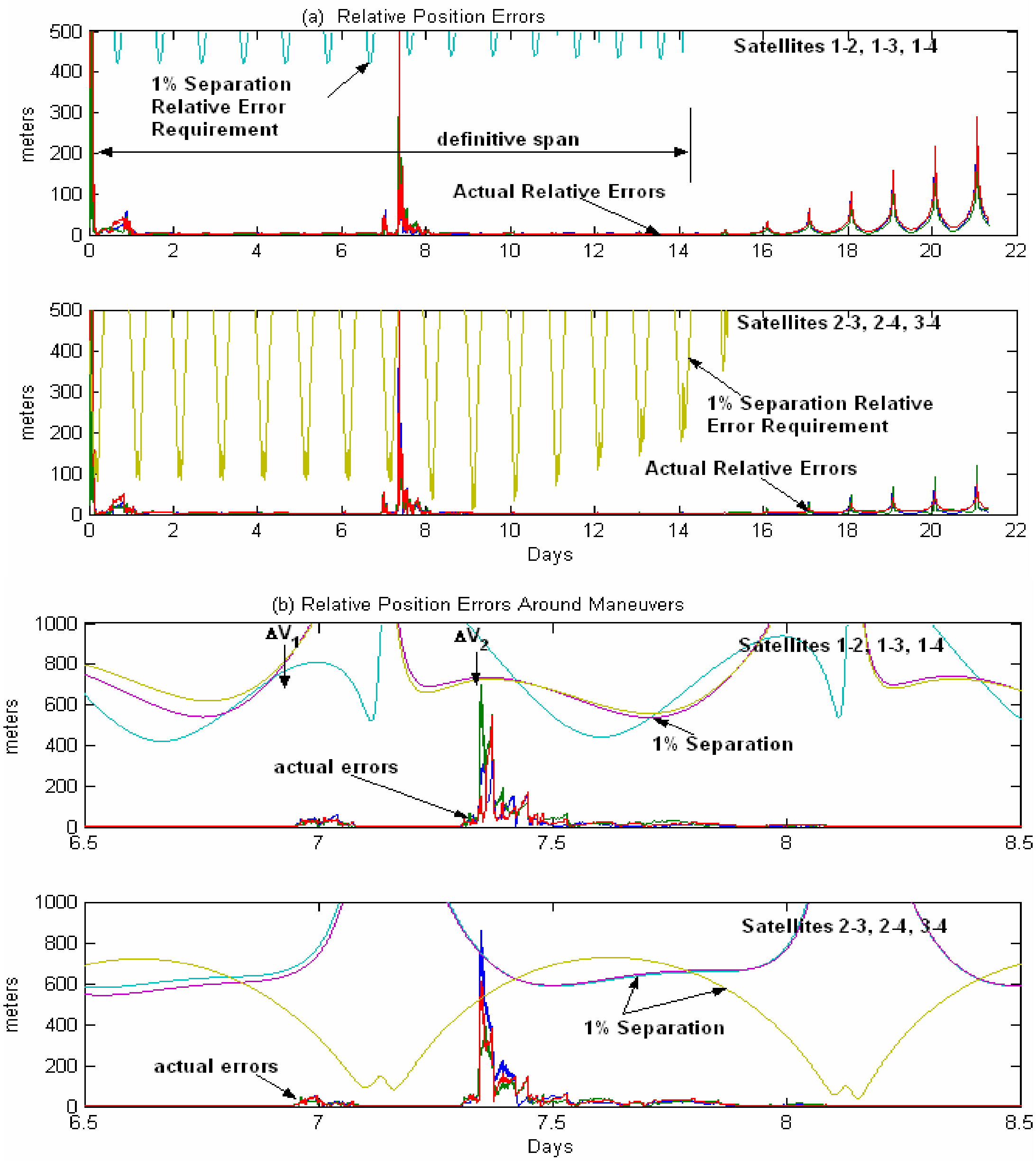

Figure 7: Relative Position Errors for Solution 2

The relative position requirements are met except for a brief period (about an hour) immediately following $t_{\Delta V_2}$. Satellites 3 and 4 have small separations near perigee as shown by the yellow lines in Figure 1 and the bottom plot of Figure 7(b). Even for this pair of satellites, the relative position errors are well below the prescribed requirements except for about 1 hour immediately following $t_{\Delta V_2}$.

Figure 8 shows the absolute velocity errors for Solution 2. The general error behavior is similar to the absolute position errors in Figure 5. The perturbation due to the application of ΔV_1 is not significant. The post-ΔV_1 velocity errors can be as large as 25 mm/sec, but most of those errors are reduced to the level of steady-state errors within 6 hours, allowing sufficient time to plan the second maneuver (ΔV_2). The second maneuver ΔV_2 causes much larger velocity errors. The RMS and maximum velocity errors for pre- and post-maneuver steady state periods are listed in Table 6. In the case of the velocity errors, the pre- and post-maneuver results are nearly equivalent.

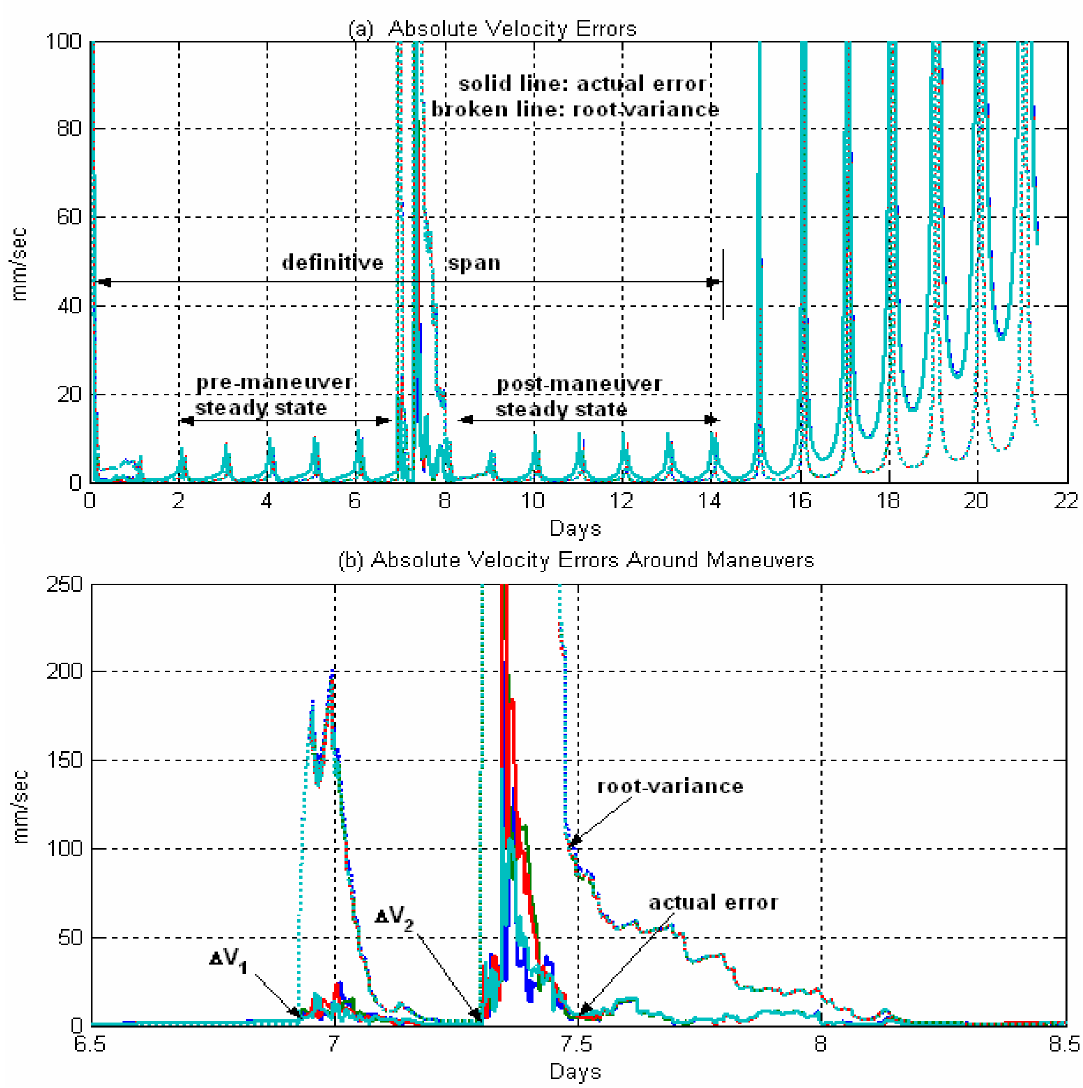

Figure 8: Absolute Velocity Errors for Solution 2

Figure 9 shows the variation of the absolute and relative semi-major axis (SMA) errors over the entire simulation time span. The absolute SMA errors for the steady-state solutions (both pre- and post-maneuver periods) are typically 7 meters near apogee, and show spikes near perigee (with a maximum of about 60 meters). The relative SMA errors are a factor of 5 to 10 smaller than the absolute SMA errors. The SMA errors are good indicators of the stability of the absolute and relative solutions and good predictors of prediction errors based on these solutions.

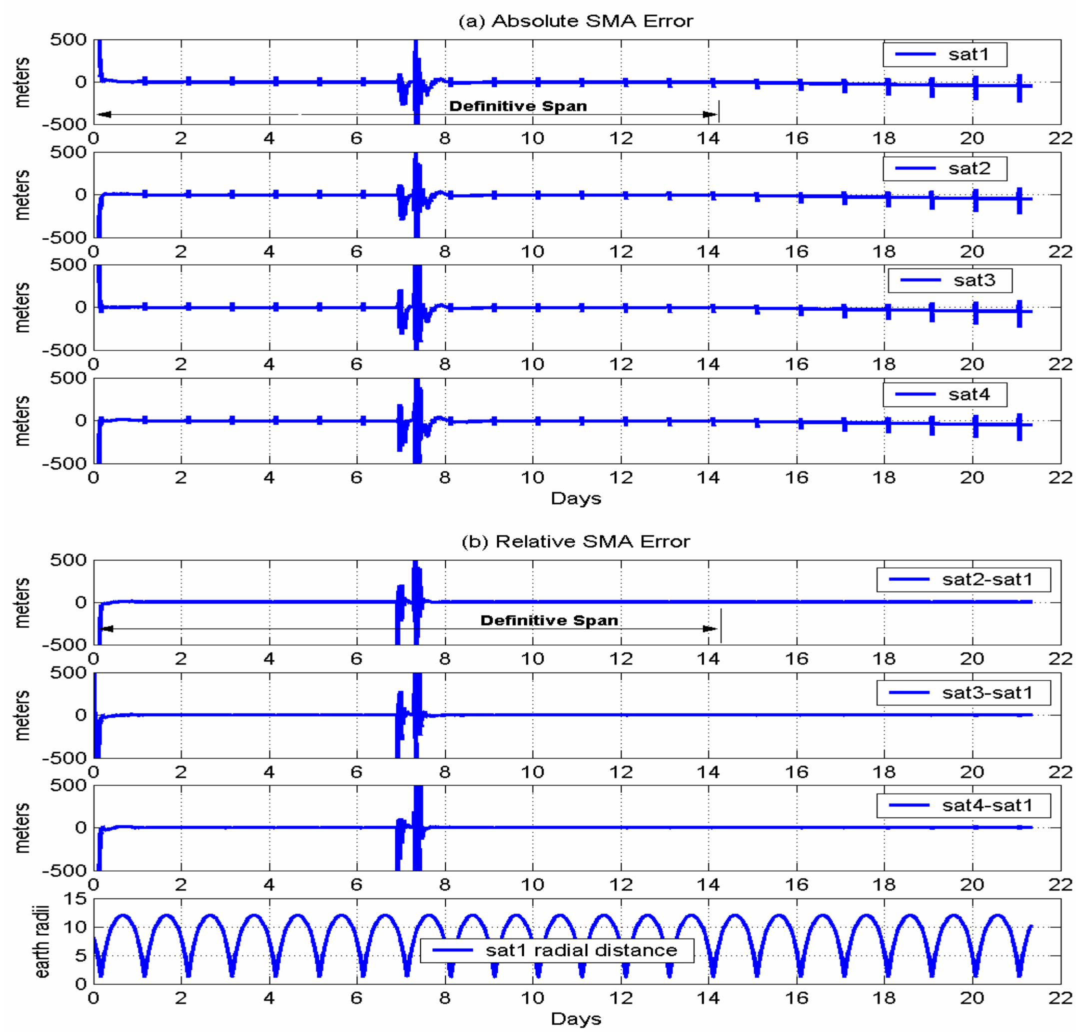

Figure 9: Absolute and Relative Semi-Major Axis Errors for Solution 2

5 Conclusions and Future Work

This paper presents the results of a navigation analysis performed for the MMS mission Phase 1 formation with a 60 kilometer separation, incorporating formation maintenance maneuvers, which occur every 14 days. The tracking measurement types used to evaluate the proposed navigation approaches are GPS PR data with a 25 dB-Hertz receiver signal acquisition threshold and one-way crosslink range measurements, processed at 4-minute intervals. The data rates used are consistent with the current MMS operational scenario, and the measurement noise standard deviations are based on the recent TRL 5 Navigator/IRAS test-bed results.

The analysis was performed in multiple steps: (1) creating a truth trajectory that includes the maneuvers that would be computed using predictions based on onboard state estimates, (2) simulating Navigator/IRAS tracking measurements based on the truth trajectory ending 7 days after the second maneuver and (3) computing filter solutions by processing the simulated Navigator/IRAS measurements for 7 days before and 7 days after the formation maintenance maneuvers, including the tracking measurements collected between the two maneuvers, and propagating for an additional 7 days. This simulation methodology includes realistic navigation errors as well as maneuver planning and execution errors.

This study demonstrates that the current navigation approach for support of the MMS formation maintenance maneuvers is expected to provide the navigation accuracy needed to meet the science objectives and flight dynamics requirements. The major conclusions from this study are as follows:

- All three filter solutions satisfy the definitive requirements for the absolute position errors. This includes the transient periods associated with the first maneuver (ΔV_1 applied at TA = 202°) and the second maneuver (ΔV_2 applied at TA = 158°).
- All three solutions briefly violated the relative position error requirement for a period of about 1 hour just after ΔV_2 . However, this violation does not impact the science requirements for the mission.
- The first maneuver caused only minor perturbation. The post-ΔV_1 filter solutions reached a steady state in approximately 6 hours and can provide the required predicted solutions for planning ΔV_2 .
- The recovery from the second maneuver takes approximately 16 hours for all three solutions. This is essentially the time from ΔV_2 to the time when the GPS visibility starts to increase near the next perigee.
- The re-convergence time after both maneuvers seems to be primarily driven by the availability of GPS data. Changes to the filter tuning did not significantly affect the re-convergence time.
- The errors during the time span immediately following ΔV_2 are sensitive to the filter tuning. By adjusting the level of the velocity process noise around ΔV_2 (and also measurement noise standard deviations), the errors during the transient period can possibly be further reduced. The addition of one-way forward Doppler data available during the space-to-ground link contacts may also improve the solution recovery.
- The maneuvers simulated are larger than what would be expected when using a relative targeting scheme, and therefore the maneuver recovery times may be reduced using more realistic maneuvers. Additionally, the brief violation of the relative navigation requirement might be eliminated with smaller maneuvers.

Future work items include a detailed investigation of navigation approaches for support of maneuvers performed during the commissioning phase and for support of the apogee raising maneuvers later in the mission. Additional end-to-end simulations will be required to verify navigation performance as the formation maintenance maneuver strategy for MMS is refined.

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