Monte Carlo Simulations of the Formation Flying Dynamics for the Magnetospheric Multiscale (MMS) Mission

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Abstract: *The MMS mission is an ambitious space physics mission that will fly 4 spacecraft in a tetrahedron formation in a series of highly elliptical orbits in order to study magnetic reconnection in the Earth's magnetosphere. The mission design is comprised of a combination of deterministic orbit adjust and random maintenance maneuvers distributed over the 2.5 year mission life. Formal verification of the requirements is achieved by analysis through the use of the End-to-End (ETE) code, which is a modular simulation of the maneuver operations over the entire mission duration. Error models for navigation accuracy (knowledge) and maneuver execution (control) are incorporated to realistically simulate the possible maneuver scenarios that might be realized. These error models, coupled with the complex formation flying physics, lead to non-trivial effects that must be taken into account by the ETE automation. Using the ETE code, the MMS Flight Dynamics team was able to demonstrate that the current mission design satisfies the mission requirements.*

Keywords: MMS, Simulations, Formation, Automation

1. Introduction

The Magnetospheric Multiscale mission (MMS) is an ambitious space physics mission that is designed to fly four identical spin-stabilized spacecraft in a tetrahedral formation in a set of highly elliptical orbits. The aim of the mission is to increase our understanding of the phenomena of magnetic reconnection in the Earth's magnetosphere and to determine some of the underlying physical mechanisms that bring it about.

Magnetic reconnection is a plasma process that converts magnetic energy into kinetic energy through the acceleration of charged particles. It occurs when magnetic field lines in a plasma approach each other close enough that they can be broken and rejoined. The resulting change in the field topology gives rise to unbalanced $\vec{j} \times \vec{B}$ forces that cause a conversion of magnetic energy to kinetic energy [1]. Since plasmas make up the majority of the observed matter in the universe, an understanding of magnetic reconnection has implications for our understanding of physical processes that occur in stellar evolution, the production of solar flares, laboratory fusion research, and, in particular for MMS, the interaction of the Earth's magnetosphere with the magnetic field and solar wind from the Sun.

The plasma processes that take place in the Earth's magnetosphere have a variety of effects on human life. They are responsible for the auroras, can disrupt power grids and communication networks, and can pose serious threats to spacecraft electronics. Data from spacecraft have revealed a complex and highly dynamic set of plasma behaviors within the Earth's magnetic field. The structures that develop during reconnection can move at speeds ranging upwards of thousands of kilometers per second relative to the Earth. These high speeds make precise in situ measurements from a single spacecraft challenging because there is very little information to separate temporal and spatial variations.

As a result, the trend in space physics missions in the past two decades has been towards simultaneous measurements from several spacecraft. The THEMIS mission [2], launched in February of 2007, flew five spacecraft in a set of co-planar, co-aligned highly elliptical orbits, with apogee altitude differing by many Earth radii (Re) and periods designed to have the constellation come into phase every four days. The orbital separations allowed different spacecraft to bracket a region where reconnection occurs and to observe how the structures move as they flow outward. These observations have been instrumental in determining large scale structure in the magnetosphere but cannot resolve what is happening locally as reconnection occurs. To address this need, spacecraft need to be flown in formation with spacing comparable to the size of the plasma processes of interest. The European Space Agency's Cluster Mission [3], launched in 2000, is comprised of four spacecraft flying in formation about a highly elliptical orbit with a 4x19.6 Re size (i.e., perigee/apogee radius of 4/19.6 Re). The formation is designed so that the spacecraft approximately fall on the vertices of a regular tetrahedron at apogee with a characteristic spacing between them, called the formation scale size, of about 100 km at apogee.

The MMS Mission, which is slated for launch in the late summer to early fall of 2014, aims to improve on the Cluster mission is several ways. Like Cluster, MMS will deploy a formation of four spacecraft moving in a close tetrahedral formation about a highly elliptical orbit with each of the spacecraft carrying a suite of instruments for in situ measurements of electric and magnetic fields and charged particle composition. The orbit selection and relative spacing are quite different with the mission intending to fly two distinct science phases each with multiple formation scale sizes. In the first phase, the formation flies with a relative spacing ranging from 10-160 km in a 1.2x12 Re orbit. Primary science is taken when the tetrahedron is at distances greater than 9 Re from the Earth and is within 30 degrees of the Earth-Sun line, on the sunward side. In the second phase science, the relative spacing is 30-400 km in a 1.2x25 Re orbit. Primary science is taken when the tetrahedron is at distances greater than 15 Re from the Earth and is within 30-40 degrees of the Earth-Sun line on the night-ward side. A transfer phase connects the two science phases by the execution of a series of maneuvers to raise each spacecraft's apogee while keeping the formation from drifting too far apart. The variable distances from Earth and inter-spacecraft spacing allows the MMS formation to act as a science instrument (see Fig. 1), with the inter-spacecraft distances being chosen to match the scale sizes of the physics occurring at different locations in the Earth's magnetosphere.

Execution of the basic mission design requires that both deterministic and random maneuvers be performed in operations. The deterministic maneuvers are used to change the overall orbital characteristics of the formation and the random maneuvers arise from the need to maintain the formation against the relative drift that builds up over time. The number, size, and direction of these maintenance maneuvers are unknown a priori and are very dependent on the realization of a number of error sources (natural and man-made) in the system. Earth's J2 term in the geopotential causes differential evolution of the orbit states of each spacecraft and is the largest natural perturbation. Lunar gravitational affects also play a significant role in Phase 2, because of the large apogee, causing both relative drift and, on occasion, a secular lowering of the perigee altitude below acceptable limits. Taken as a whole, these natural perturbations are well understood and predictable, depending deterministically on the initial orbital state and its subsequent evolution. The formation lifetime, defined as the time between maintenance maneuvers, in the presence of these natural perturbations, is on the order of 40-70 days. In contrast, the man-made perturbations due to knowledge and execution errors in the maneuver process generically dominate the natural perturbations. Taken together, this combination of knowledge and execution error causes a post-maneuver evolution that is different from the one desired. The situation is further complicated by the fact that each spacecraft experiences its own realizations of these errors with no correlation between them. The man-made error sources associated with maneuver operations generally lower the formation lifetime to a range of 5-20 days.

The variable nature in the number of and type of maneuvers coupled with the 'cooperative effect' of the errors makes pre-launch planning and operational support of the MMS formation flying problem very difficult. Questions associated with the stability of the design, such as how much fuel to budget, how maneuvers will be performed, and when to schedule ground assets to support the maneuvers, cannot be answered with simple analytical models. It was to address this need that the MMS flight dynamics group developed the End-to-End (ETE) code. The ETE code is designed to perform Monte Carlo simulations of the entire MMS operations phase from launch until the end of Phase 2 science. It combines high-fidelity orbit propagation, realistic models of the propulsion system, and models for navigation knowledge and maneuver execution errors into an event-driven framework that allows it to produce different maneuver scenarios in response to the different errors it encounters. Statistical reduction of the resulting data is then used to allocate resources for the actual launch campaign.

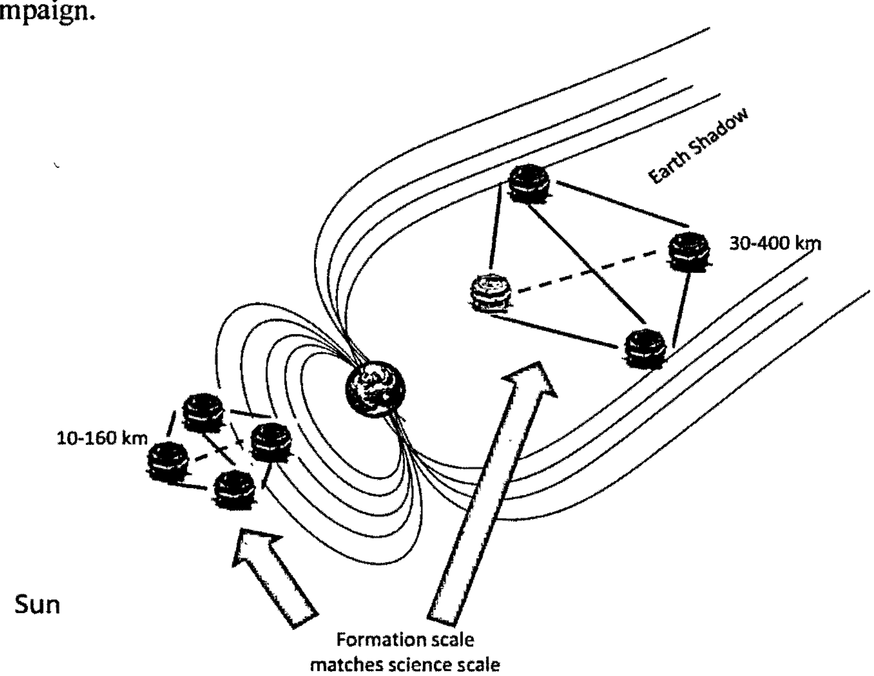


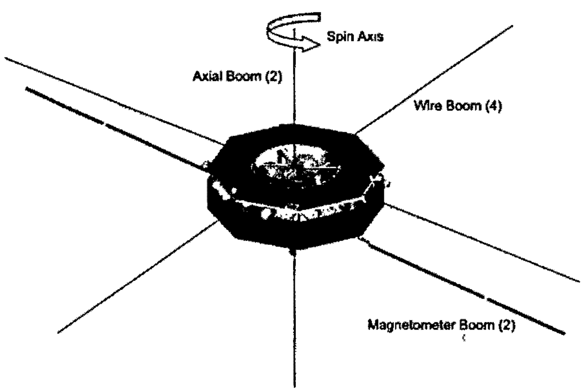
Figure 1: Schematic of the MMS formation as a science instrument concept.

Section 2 of this paper presents the operations concept for MMS, including a brief discussion of the spacecraft hardware, the requirements that impact the mission design, the mission phases and timeline, the quality factor that defines acceptable formations, how maneuvers are to be executed and how contacts with Tracking, Telemetry, and Command (TT&C) assets are used in their support. Section 3 describes how the End-to-End code simulates the MMS operations. Included are the architecture of the code, how the orbital propagation and requirement checking is performed, how the maneuver targets are selected and how the maneuver planning and execution are modeled, and how knowledge and execution errors are applied. The results from a recent set on Monte Carlo trials are presented in Section 4 along with the implications to the mission as a whole. Section 5 discusses the remaining work and future direction of the End-to-End code as MMS transitions from the design phase to implementation.

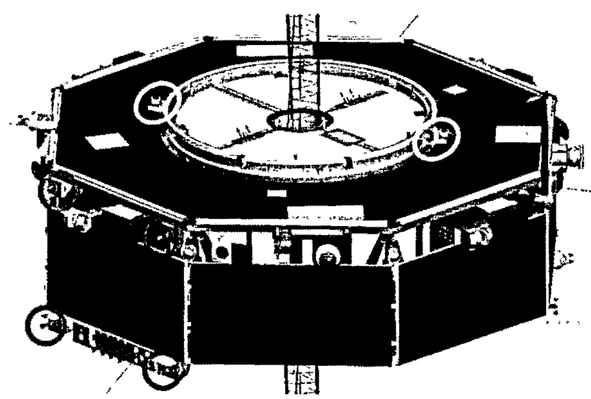
2. MMS Operations Concepts

Each MMS spacecraft is spin-stabilized, with the nominal spin rate at 3.0 ± 0.2 rpm. Each spacecraft deploys 8 booms: 4 spin-plane double probe (SDP) instruments on wire booms in the spin-plane for measuring the electric fields; 2 magnetometer instruments, also on booms in the spin-plane, for measuring the magnetic field; and 2 axial double probe (ADP) instruments on rigid booms parallel to the spin-axis for measuring electric fields. The propulsion system for each spacecraft consists of 8 radial 4-lbf thrusters oriented parallel to the spin plane, with sets of 4 on opposite sides of the spacecraft, and 4 axial 1-lbf thrusters. Each spacecraft is also equipped with a suite of guidance, navigation, and control sensors including a digital sun sensor, a star camera, an accelerometer, and a Navigator GPS receiver with GEONS navigation software built-in. Onboard controllers process data from this sensor suite to actuate the thrusters to the desired accuracy in the orbit and attitude maneuvers while ensuring the stability and safety of the deployed booms. Figure 2 shows a diagram of the spacecraft showing the deployed booms and a schematic of the propulsion system.

From the flight dynamics perspective, the MMS operations concept is best understood by dividing the driving requirements into two categories. The first category, called the baseline, holds all of the requirements related to how the formation as a whole moves with respect to the magnetosphere. The second category, called formation flying, holds all of the requirements related to the relative orbital evolution of the formation.



(a)



(b)

Figure 2: (a) MMS spacecraft showing the 8 booms for the electric and magnetic fields deployed and (b) MMS spacecraft with propulsion system with the location of the 4-lbf radial thrusters shown in red and the location of the 1-lbf axial thruster shown in yellow (with symmetric placement of the other 6 thrusters on the opposite faces).

2.1 Baseline Requirements

Figure 3 shows a visual summary, in geocentric solar ecliptic (GSE) coordinates, of the baseline requirements, starting just after launch into the Commissioning Phase (Phase 0) and ending just after completion of the Phase 2b science campaign. The GSE coordinate system is a rotating coordinate frame whose z-axis is parallel to the ecliptic north pole and whose x-axis is parallel to the instantaneous line joining the Earth to the Sun pointing sunward. In Fig. 3, the GSE z-axis comes out of the page and the x-axis runs along the horizontal from right-to-left. In this frame, inertially-fixed orbits rotate clockwise and the evolving relative geometry between the orbit and the Earth-Sun line is apparent. The four cardinal directions along the x- and y-axes are labeled by the GSE times 00:00, 18:00, 12:00, and 06:00 corresponding to the –x-axis, +y-axis, +x-axis, and -y-axis, respectively. Many of the MMS baseline requirements, most notably the transitions between mission phases, are expressed in terms of the apogee GSE time. The apogee GSE time is determined by projecting the apogee vector, defined as the line from the center of the Earth to the orbit apogee, onto the GSE x-y plane and then assigning it the time corresponding to where it falls relative to the time convention used to label the x- and y-axes.

The Commissioning Phase (Phase 0) starts with the launch of the four MMS spacecraft aboard an Atlas V 421 launch vehicle – the 421 designation meaning that the rocket is equipped with the 4.2 meter fairing, two additional strap-on solid boosters, and only one RL10 engine in the Centaur upper stage. The inclination of the orbit is 28.5 degrees and the apogee distance is 12 Re. The initial perigee altitude is 240 km, corresponding to a perigee radius of 1.04 Re. The launch and coast times are chosen so that the apogee GSE time at mission start is approximately 02:00 and that no MMS spacecraft enters a shadow longer than 1 hour for the first 2 weeks after launch. The MMS Flight Dynamics team, working with United Launch Alliance (ULA), designed the separation event so that the spacecraft safely clear the launch vehicle and continue to drift apart from each other during the first 20 days. During the Commissioning Phase, each spacecraft must execute a set of perigee raising maneuvers to increase perigee distance to 1.2 Re, precess their spin axis to mission attitude, deploy science instruments on booms, check-out and calibrate the instrument suite, and certify the onboard controllers for formation flying. At 98 days after launch, the spacecraft will be put into a formation with a scale size of 160 km. Approximately 22 days later the first science phase, Phase 1, begins with an apogee GSE time of approximately 18:00.

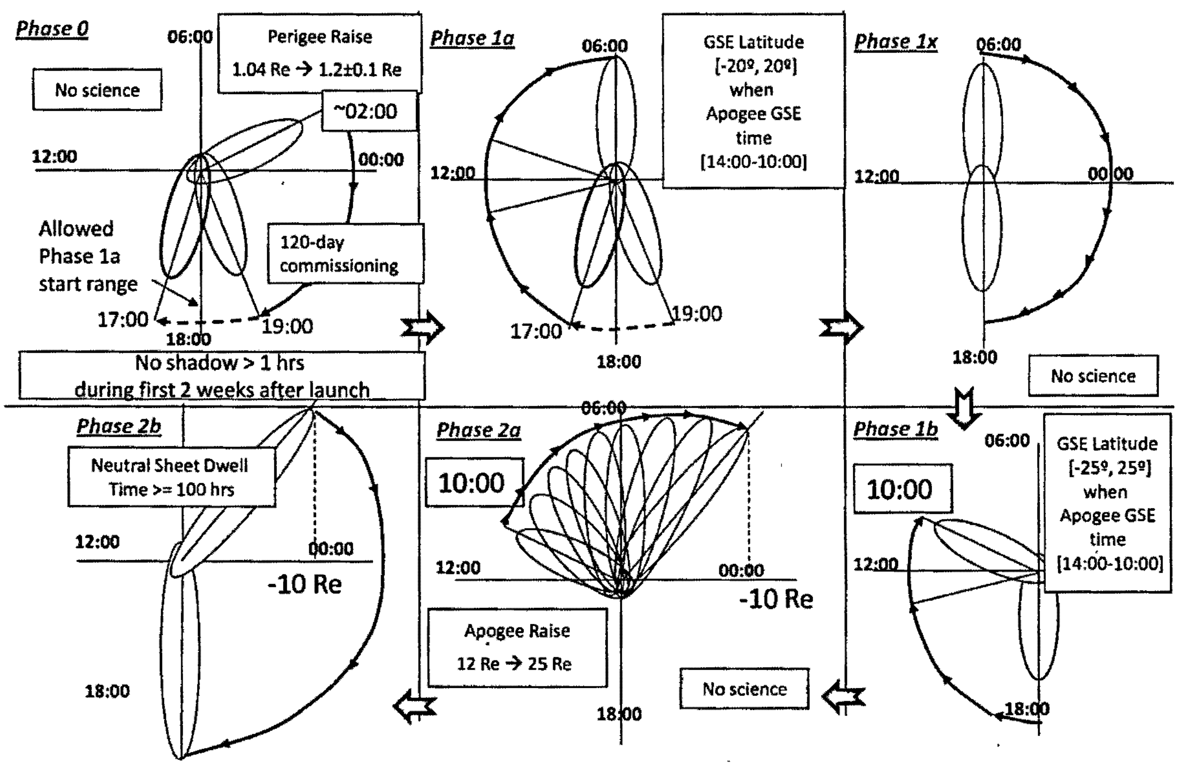


Figure 3: A visual summary of the baseline MMS mission design, in GSE coordinates, and the high level baseline flight dynamics requirements.

Phase 1 consists of three segments, only two of which have formation flying science requirements. The orbital period of each orbit in Phase 1 is approximately one sidereal day. Phase 1a starts at 18:00 and lasts until 06:00 apogee GSE time. The only baseline science requirement is that the GSE latitude of apogee (defined in the usual way) should be between ± 20 degrees during the time when the apogee is day-ward (GSE time between 14:00 and 10:00). Phase 1x begins when the apogee GSE time reaches 06:00 and lasts until 18:00. During this night-ward passage of the orbit apogee through the Earth's geomagnetic tail there are no baseline science requirements. Phase 1b begins at apogee GSE time of 18:00 and lasts until 10:00. The only baseline science requirement is that the

GSE latitude of apogee should be between ±25 degrees when the apogee GSE time is between 14:00 and 10:00.

Phase 2 consists of two segments, the first being Phase 2a, the apogee raising campaign, and the second being Phase 2b, the prime night-ward science collection. Phase 2a begins at an apogee GSE time of 10:00 and ends when the projection of the apogee vector onto the GSE x-axis is no greater than –15 Re. The current design, allowing for margin, sets the end to occur when the projection of the apogee vector onto the GSE –x-axis is equal to –10 Re. This design consideration is reflected in Figure 3. During this phase, each spacecraft performs a set of 8 maneuvers in a coordinated fashion that raises the apogee of the entire formation in fuel-efficient way without allowing the spacecraft to drift too far apart that reforming the formation becomes difficult. The paper by Roberts, et. al. [4] presents details of the approach. Phase 2b then begins the orbit after Phase 2a ends. The orbital period of a Phase 2b orbit is approximately 2.85 civilian days. This phase lasts approximately 60 days as the apogee passes through the night-side region. The baseline science requirement for this phase is that the dwell time in the neutral sheet [5] be at least 100 hours when the spacecraft are not in shadow. The Fairfield model [6] is used for all simulations to compute the neutral sheet dwell time in all pre-launch verifications by simulation.

Three additional baseline requirements apply to all phases: 1) the maximum duration of the combined umbra shadow of the Earth and Moon for any 20-hour window must be less than 216 minutes, 2) the maximum duration of the combined shadow (umbra + 50% of penumbra) of the Earth and Moon for any 20-hour window must be less than 231 minutes, and 3) the minimum perigee height in any phase must be greater than 900 km. The first two requirements are for power considerations and the last one is for the protection of the instrument suite from the effects of atomic oxygen in the upper atmosphere.

2.2 Formation Flying Requirements

The goodness of a formation at any time is determined by the quality factor $Q(t)$, which depends on the instantaneous relative positions of the MMS formation through the product of a function $Q_v(t)$ that depends on the volume of the tetrahedron and $Q_s(t)$ that depends its size. Figure 4 shows a schematic snapshot of one such configuration with the side lengths, s_i, between the six pairs of spacecraft. Defining the average side length as

$$L = \frac{1}{6}\sum_{i=1}^{6} s_i, \qquad (1)$$

the volume function $Q_v(t)$ is then defined by

$$Q_v = \frac{\sqrt{2}}{6L^3}\left|\vec{s}_1 \cdot \left(\vec{s}_2 \times \vec{s}_3\right)\right|, \qquad (2)$$

which is the ratio of the volume of the formation to the volume of a regular tetrahedron of side length L (and where the $\vec{s}_i$ are the vectors corresponding to the appropriate side lengths). Likewise, the size function $Q_s(t)$ is defined as

$$Q_s(L) = \begin{cases} 0, L < \ell_1 \\ (L-\ell_1)^2(L+\ell_1-2\ell_2)^2/(\ell_2-\ell_1)^4, \ell_1 < L < \ell_2 \\ 1, \ell_2 < L < \ell_3 \\ (L-\ell_3)^2(L+\ell_4-2\ell_3)^2/(\ell_4-\ell_3)^4, \ell_3 < L < \ell_4 \\ 0, L > \ell_4 \end{cases} \quad (3)$$

where the scale parameters are given in Tab. 1. The product of Eqs. 2 and 3, guarantee that $Q(t)$ takes on values between 0 and 1. Details of the derivations and the formation design considerations can be found in [7].

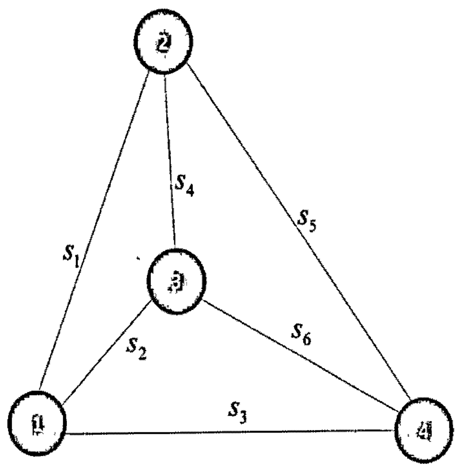


Figure 4: Schematic representation of an instantaneous MMS Formation

Table 1: Scale parameters as a function of formation scale size for Eq. 3.

	Formation Scale Size (km)					
Scale Parameters	**10**	**25**	**40**	**60**	**160**	**400**
ℓ_1	4	15	25	45	135	250
ℓ_2	6	20	30	50	140	300
ℓ_3	18	35	55	75	190	550
ℓ_4	24	40	65	80	210	600

Basics orbital mechanics precludes the formation from holding a constant shape, similar to the one shown in Fig. 4, for the entire orbit. In fact, the formation has to 'invert' twice an orbit with spacecraft that are closer to the Earth exchanging with those starting further and the same for spacecraft starting above and below with respect to the Earth's equator . To account for this physics, the MMS quality factor requirement is written in terms of a related parameter called $T_Q$which is a weighted average of the quality factor over the science region-of-interest (ROI). To evaluate T_Q, imagine each spacecraft's trajectory represented by a set of ephemeris points listed at times t_i. Selecting the number of points, N_{ROI}, that fall into the science of region of interest, T_Q takes on the form

$$T_Q = \frac{100}{N_{ROI}} \sum_{i=1}^{N_{ROI}} M_i \text{ where } \begin{cases} M_i = 1 \text{ if } Q(t_i) \geq 0.7 \\ M_i = 0 \text{ if } Q(t_i) < 0.7 \end{cases}. \quad (4)$$

The MMS design threshold is to have T_Q be greater than 80 at all times formation science is being collected in the science ROI, which is defined as the portion of the orbit where the spacecraft are greater than 9 Re and 15 Re from the Earth for Phases 1 and 2b, respectively.

During the course of the mission, the formation will be resized as a method to determine the scale-size that best matches the science objectives. In Phase 1, the initial formation starts at 160 km and decreases once every 15 days to 60 km, 25 km, and 10 km. In Phase 2b, the initial formation starts at 400 km and decreases once every 15 days to 160 km, 60 km, and 30 km. After reaching the minimum scale-size in either phase, the formation will then be resized to an intermediate value based on analysis of the science data seen.

Establishing a formation, either initially for a mission phase or subsequently re-establishing it with a new scale size requires the execution of what is termed a resize maneuver. In the absence of perturbations (natural or man-made), after a formation is established with a resize maneuver it will repeat itself periodically, and the value of T_Q in one orbit will be the same for all subsequent orbits. Once any perturbations are introduced, the value of T_Q generally will degrade over time and occasionally, the perturbations will also cause the formation, or some subset of the spacecraft, to begin to come too close. At this point a maintenance maneuver is required to reestablish the desired formation behavior. For MMS, the same type of maneuver is used for both resizing and maintaining a formation.

A formation resize or maintenance maneuver is performed in pairs of orbital maneuvers for each spacecraft on the inbound leg and the subsequent outbound leg on a given orbit. Since each spacecraft has the same frequency and communications are required during all burns, maneuvers must be done one at a time in a process called staggering. Usually, one spacecraft is chosen as the reference and the other three adjust their trajectories to get into formation around it. The targets for the maneuver are obtained at a time boundary (set at or after the final maneuver) from the Formation Design Algorithm (FDA). The FDA selects the orbital conditions for each spacecraft based on one of several criteria, including those that maximize the time that T_Q stays above the 80 threshold [7,8] or those that minimize the delta-V required to establish the formation [9]. Regardless of exactly which flavor of the FDA is employed, once the targets are obtained a Lambert solver determines the maneuver magnitudes and directions that bring the spacecraft back into formation. A downlink of the onboard navigation solution just after perigee provides the insight into maneuver 1 execution that is needed for replanning maneuver 2.

Figure 5 shows a representation of the maneuver events for the formation during a Phase 1 orbit. The timeline starts on the inbound leg with a degraded formation that is trending to either go below the T_Q threshold of 80 in the next orbit or to bring two spacecraft into too close an approach, or both. The instrument suite (IS) is safed and a maneuver is performed by at least three of the spacecraft during the segment of the orbit labeled 'Maneuver #1'. These maneuvers adjust the trajectories to achieve the desired positions at maneuver 2. As the formation comes through perigee, the onboard navigation system picks up 4 GPS space vehicles allowing it to have a highly accurate post-maneuver estimated trajectory. The onboard solution is downlinked at the segment of the orbit labeled 'R/T GEONS TLM'. These updated orbital states allow the ground to replan maneuver 2. Due to operational considerations in Phase 1, maneuver 2 is performed around apogee to allow the ground sufficient time to ensure that the updated command load is ready for execution.

The situation for Phase 2b is very similar with the only differences being: 1) the region-of-interest is defined as orbital distances greater than 15 Re; 2) maneuver 2 is performed before entry to the ROI; and 3) the onboard navigation solution is not available during the entire orbit. These variations change only the location in true anomaly of the maneuvers and the accuracy of the downlinked onboard navigation solution.

The final requirement, pertaining to formation maintenance and resizing, limits the time between maneuvers (i.e. from maneuver 2 of one to maneuver 1 of the following) to be on average 14 days to allow for reasonable operations tempo and more science collection time.

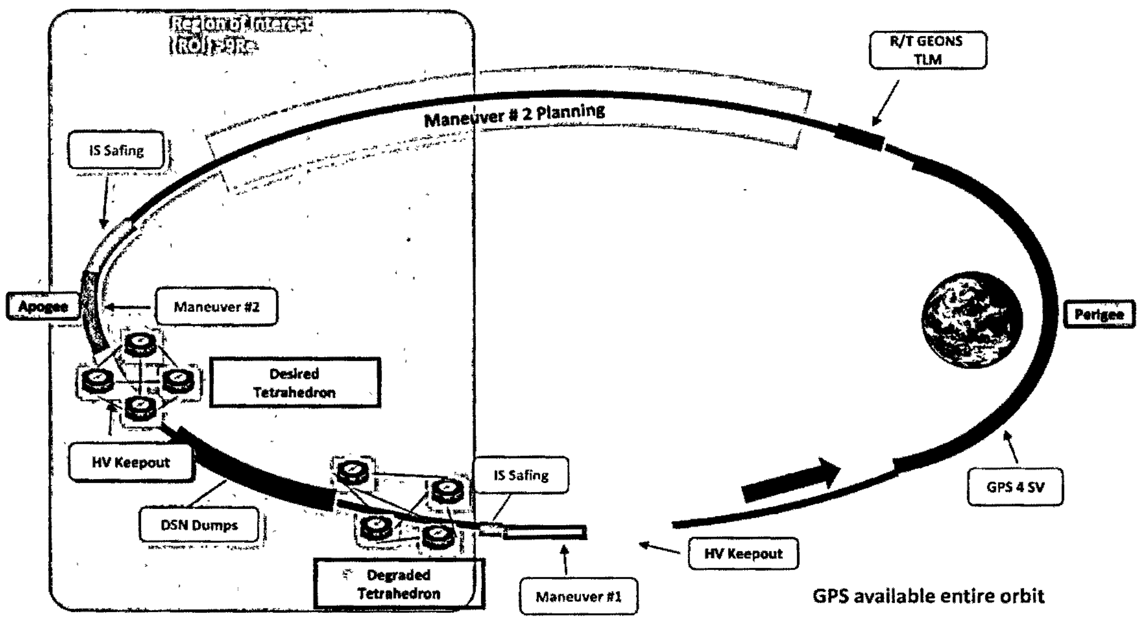


Figure 5: A day in the life of MMS with formation maintenance maneuvers for Phase 1.

3. End-to-End Code Simulation Approach

Standard best practices at Goddard Space Flight Center call for an accurate estimate of the fuel required to meet a missions operations concept taking into account a variety of 3-sigma error sources, including launch vehicle dispersions, maneuver execution errors based on propulsion and attitude control system, and related flight dynamics error sources. The complexity of the MMS mission design results in an operational support that will produce different maneuver scenarios in response to the different realization of these error sources. The number of formation maintenance maneuvers, each one dependent on the maneuvers that came before, can only be determined by a detailed model of how operations would respond to a given set of outcomes.

For this reason, the MMS Flight Dynamics group developed the End-to-End (ETE) code as a simulation that mimics, as best as possible, what will happen in over two years of operations. The ETE is built on a multi-component architecture that is designed to allow testing and changes on a given component without affecting the others. All the components are hosted in either FreeFlyer or GMAT and use a high-fidelity force model including the JGM-2 Earth geopotential, Sun and Moon point-mass gravity, solar radiation pressure, atmospheric drag, and a blow-down model of the onboard propulsion system. The components are run in succession with a Matlab driver coordinating the exchange of data between the components via an HDF5 file data repository. Common propulsion system and error models, for fuel usage, launch vehicle dispersions, maneuver execution errors, and navigation accuracy, are shared across the system to facilitate Monte Carlo (MC) analyses. Customization of each component is done to balance execution time against modeling fidelity. For example, the Phase 0 component models maneuver execution and navigation errors but omits the detailed finite maneuver modeling since the perigee raising maneuvers are performed at orbit apogee where the impulsive approximation is sufficient. In contrast, the apogee raising campaign, simulated in the Phase 2a component, models the maneuvers as finite burns since they are executed around perigee, where gravity-loss and finite burn effects are expected to be significant.

Reproducibility of a given MC trial is guaranteed by having a pre-determined list of random numbers that is managed by the Matlab driver. Initial conditions for a specific launch date and time and the corresponding modeling parameters are obtained from the Reference Orbit, which is a lower-fidelity realization of the trajectory of the fictitious center of the formation. The ETE code propagates these initial conditions based on a deterministic schedule of known events such as launch and the transition between mission phases based on absolute or GSE times or maneuver events that include the perigee raising in Phase 0, the apogee raising in Phase 2a, and the cadence of formation re-sizing maneuvers in Phases 1 and 2b. As the random error sources from earlier events are realized, additional events are required in the form of unscheduled formation maintenance maneuvers. To be able to respond to the dynamic scheduling present in each Monte Carlo trial, the ETE code has an event-driven framework where it decides how and when to perform maneuvers to best meet mission requirements. Figure 6 shows the End-to-End architecture.

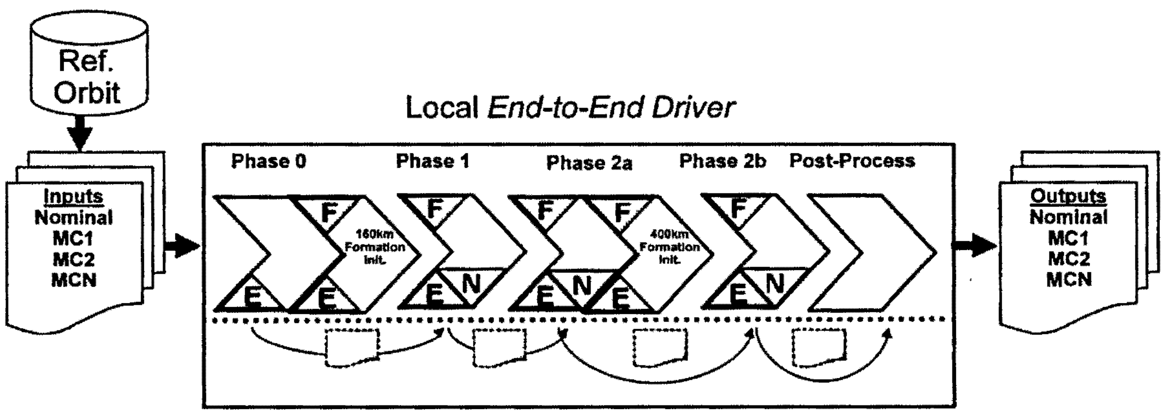


Figure 6: The End-to-End architecture. Separate components are denoted by the 'chevron shape'. Letter code: 'F' – finite burn capable, 'E' – maneuver execution errors, 'N' – Navigation uncertainty.

The variation in the maneuver history between different Monte Carlo trials due to the random errors makes it difficult to find an event-driven framework that makes intelligent decisions in all circumstances. To appreciate some of the complexities that resulted, the application of the error models that were major factors are first presented followed by some of the design choices that were made before the production runs used to support the recent mission Critical Design Review in August of 2010 were performed.

3.1 Application of the Error Models

Three separate error models are used in the End-to-End code to account for 1) launch vehicle dispersion, associated with uncertainty in the burnout state and separation events, 2) the uncertainty in the navigation solution, and 3) the maneuver execution error. Since the launch vehicle error model is used only on the initial state, the following discussion will be confined to the navigation and maneuver execution error models, which are used frequently in the ETE.

The uncertainty in the navigation solution is based on an orbit determination error analysis using the ground based version of the GEONS software used onboard. A run of the End-to-End code with all the errors turned off is used to provide needed ephemeris data. Models of the onboard Navigator receiver, accelerometer, and clock characteristics are used with the ephemeris data to generate simulated tracking measurements that are consistent with visibility of the GPS constellation and Navigator's weak-signal tracking capabilities. These data are then processed by GEONS with the output being the state covariance matrix, expressed in the local RIC (radial, in-track, cross-track) frame, for each spacecraft at key points in the orbit.

The maneuver execution error model is based on the analysis of the onboard control algorithms performed by the MMS Attitude Control System and Guidance, Navigation, and Control groups. The model specifies the magnitude error as a zero mean normally distributed random variable with a standard deviation equal to a percentage of the size of the maneuver plus an absolute offset. Pointing error is specified in terms of the off-pointing, α, which is defined as the angle between the commanded and achieved directions and the azimuth, ϕ, which is the angular location of the achieved direction on the cone of half angle α about the commanded direction. The off-pointing is a zero mean normally distributed random variable with a standard deviation also equal to a percentage of the size of the maneuver plus an absolute offset. The azimuth is a uniform random number between 0 and 2π and negative values of the off-pointing are interpreted as simply moving the value of ϕ by π.

The application of these error models in a Monte Carlo trial requires the use of two sets of MMS spacecraft. One set corresponds to the truth trajectories, which are unknown in operations but available in the simulation, and the other set to the known trajectories, which in operations are comprised of the definitive trajectories estimated from the orbit determination process and the predicted trajectories that come from the maneuver planning process. The method the ETE code uses to apply the error models is best explained by referring to Fig. 7, which illustrates the approach for one spacecraft performing a pair of maneuvers for a formation resize or maintenance. The code starts by assuming a truth trajectory (solid black line). At the point labeled by the '1' an estimated state is derived from the truth by applying the navigation error model. This estimated state is propagated forward (solid red line) to the maneuver 1 start time (point labeled '2') where maneuver 1 is targeted to achieve the desired formation conditions. The truth state is then propagated to the same time where both the maneuver and the maneuver execution error are applied. The realized change in the truth trajectory is different from the planned change due to a combination knowledge error (the planned maneuver was targeted using a different state than the truth) and control error (the achieved maneuver was different from the commanded maneuver). A new navigation update is obtained at '3' and this state is propagated to '4' where the process of planning and execution are repeated for maneuver 2. Finally, a navigation update is taken from the truth at '5' and the corresponding state now represents the best idea that operations will have about the post maneuver state of the spacecraft. This process is applied to every maneuver that the ETE code models in its simulation of approximately 2½ years of MMS operations.

3.2 Design Choices in the End-to-End Code

Combining the error models described in Sec. 3.1 with the MMS mission operations concept, one may be tempted to conclude that the automation of the maneuver planning process should be straightforward. After all, the MMS mission design can be thought of as a series of campaigns executing a set of deterministic maneuvers punctuated by periods of stationkeeping. For a number of reasons, such a conclusion is wrong. The primary reasons for this are that the navigation and maneuver execution errors are uncorrelated across the formation and that, because the maneuver directions for the resize and maintenance maneuvers are arbitrary, even relatively small errors can be combined to give undesired fuel usage and/or maneuver frequency.

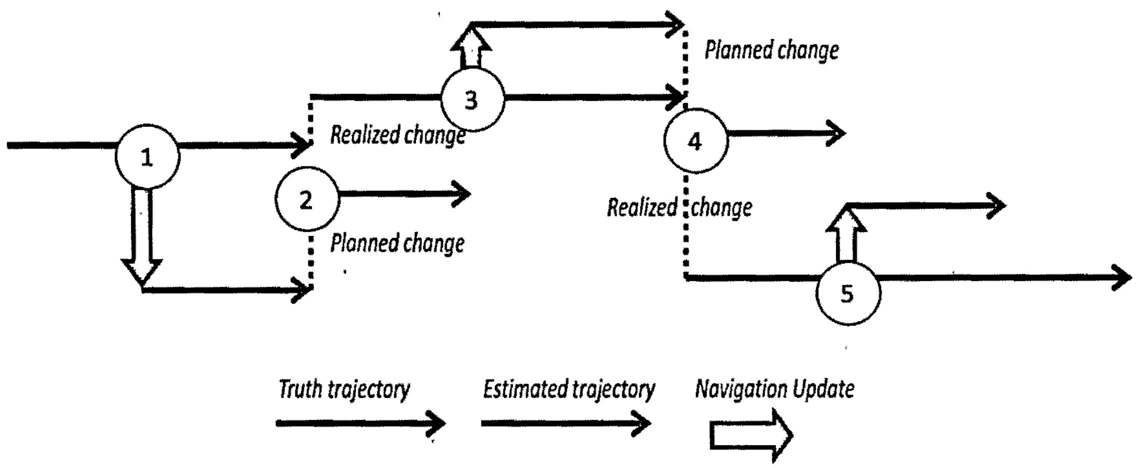


Figure 7: A schematic of the way the navigation (knowledge) and maneuver execution (control) error models are applied in the End-to-End code.

As an example, consider the formation control. The formation quality is expressed in terms of the parameter T_Q which measures the time between formation maintenance maneuvers. As written, the requirement on formation quality suggests that between the maintenance maneuvers the value of T_Q should be above 80 for every orbit and the planning process takes this into account by targeting individual maneuvers for the three maneuvering spacecraft (recall one is used as a reference) to achieve a formation that starts with T_Q at or near 100 and that slowly degrades over time. However, the combination of navigation and maneuver errors results in a set of maneuvers that fail to give exactly the planned change in the orbital states and the requirement on T_Q must be interpreted statistically with the expectation that a percentage of the orbits will have T_Q values less than 80. The next decision is to determine what value of T_Q triggers a formation maintenance maneuver. Earlier analysis showed that T_Q rapidly degrades when it nears 80 and the expectation was that trigger values below 80 would not be beneficial. However, the results from the ETE code showed that lowered values significantly improved the fuel usage. Figure 8 shows a comparison of the T_Q values for the Phase 1 orbits from a Monte Carlo trial with the T_Q trigger value set at 80 compared with on set at 60. The percentage of orbits with $T_Q < 80$ marginally rises from 6.2 to 8.0 percent with the average value of T_Q over Phase 1 dropping from 93.2 to 91.3. The average fuel usage per spacecraft significantly dropped by 25-30 kg and the time between formation maintenance maneuvers went from an average of 9 days with a minimum of 4 days to an average of 16 days with a minimum of 10 days. Based on this analysis, the MMS science community has agreed to operate the mission with the lower T_Q value.

In addition to the selection of design and operating parameters, there is a degree of intelligence that will be exercised with a human in the loop in operations that can't be trivially encoded to an automated simulation. As an example, consider the possibility of a maintenance maneuver occurring a day or two before a scheduled resize. In operations, a number of options would be considered with possible outcomes including 1) that the maintenance maneuver would be executed and the resize delayed, 2) that the maintenance maneuver would be waived and the resize executed as scheduled, and 3) that the maintenance maneuver would be waived and the resize execution moved up to that time. The ETE code has some simple control logic designed to handle this situation but it

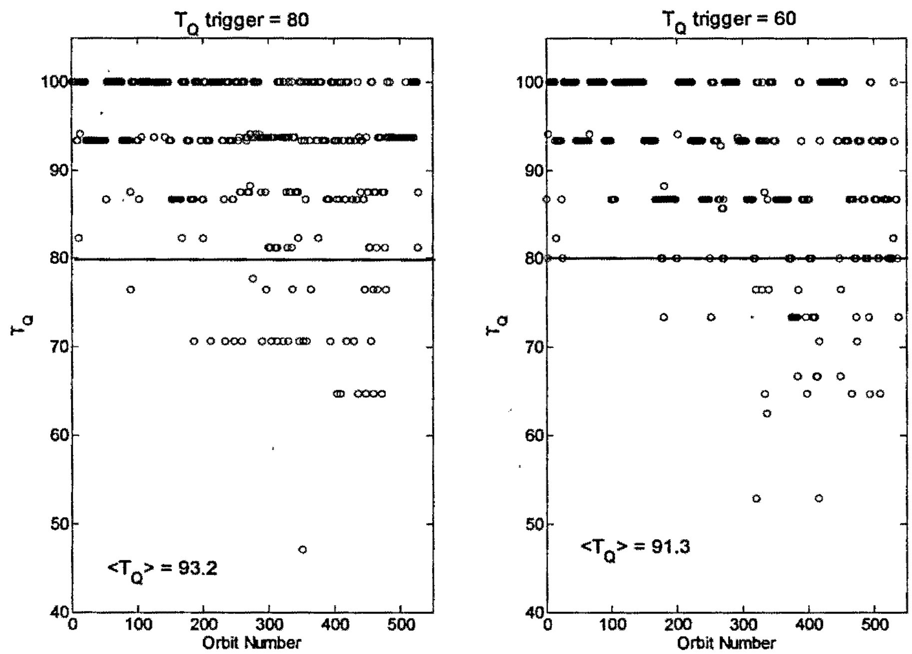


Figure 8: A comparison of T_Q trigger values for two separate ETE Monte Carlo Runs.

is not equipped with a sophisticated decision engine that can mimic human intelligence. As a result, the simulations sometimes show the execution of an unwanted maneuver which must be taken into account when using the simulations results to verify the mission requirements. Additional examples that fall into this category include how the non-maneuver reference is chosen (e.g., the one with the least fuel or the one that results in the smallest fuel consumption over the whole formation) and which flavor of the Formation Design Algorithm should be used for a given resize or maintenance maneuver. Clearly trying to adapt an automation algorithm to be more intelligent in the face of so many choices is a difficult undertaking.

4. ETE Results

Once the operating parameters and design considerations for the End-to-End code had been settled upon, production runs in support of the MMS Mission CDR were made. Each Monte Carlo run takes between 6-8 hours on a modern Windows-based machine, limiting the number of trials that could be reasonably performed. For the results presented here, 102 cases for a November 1st 2014 launch date were logged, with the production runs distributed over 7 machines.

Table 2 shows a comparison of the values for the six baseline requirements between the Reference Orbit ephemeris that was used to provide the initial conditions and the ensemble minima and maximum over the Monte Carlo samples. The comparisons for the GSE latitude requirements are close and none of the cases showed any shadows during the first 2 weeks of the mission. The maximum shadow and the neutral sheet dwell time for the ETE generally were larger than those of the Reference Orbit but in all cases met the requirements with ample margin. Since the largest shadows in the MMS mission generally fall in Phase 2b, when the orbit period is longest and the apogee is away from the Sun the correlation is expected. Overall, the agreement between the Reference Orbit and ETE results demonstrates that the MMS formation can be flown without the need to control the entire formation to a fictitious spacecraft that follows a reference path.

Table 2: Comparison of the baseline requirements between the Reference Orbit and ETE Monte Carlo Runs for a Nov. 1st launch date

Metric	Requirement	Reference orbit value	ETE orbit value
Phase 1a GSE Latitude, λa (deg)	$0 < \lvert\lambda a\rvert \leq 20$	[redacted]	[redacted]
Phase 1b GSE Latitude, λb (deg)	$0 < \lvert\lambda b\rvert \leq 25$	[redacted]	[redacted]
Neutral sheet dwell time, TNS (hours)	$TNS \geq 100$	[redacted]	[redacted]
Maximum umbra duration, MU (minutes)	$MU \leq 216$	[redacted]	[redacted]
Maximum umbra + 50% penumbra, MUP (minutes)	$MUP \leq 231$	[redacted]	[redacted]
Max. shadow in the first 2 weeks, MS (minutes)	$MS \leq 60$	[redacted]	[redacted]

As far as the formation requirements are concerned, the ensemble average over all the Monte Carlo trials show comparable behavior to those shown in Fig. 8. Specifically, the ensemble average of the time averages for T_Q were 94.4 and 84.9 for Phases 1 and 2b, respectively. In addition, the average time between maneuvers across the mission is approximately 16 days, exceeding the 14-day requirement. Figure 9 shows the ensemble statistics for mean, maximum, minimum and 3-sigma low for fuel remaining. The average fuel usage per spacecraft is 337 kg out of the 410 kg allocation, leaving 73 kg of fuel remaining. However there is a fairly large statistical variation in fuel remaining across the Monte Carlo ensemble. This variation is attributable to the effects of maneuver execution error and navigation uncertainty and for this reason these models are carefully tracked as MMS evolves.

5. Conclusions and Future Work

Currently, the ETE code serves as the primary MMS flight dynamics requirements verification tool. It is arguably the most complex and sophisticated flight dynamics simulation that has been produced at GSFC and, using it, the Flight Dynamics team has verified that the mission design requirements can be met with the current operations concept and fuel loading. Despite this success, the ETE code has its limitations particularly in the amount of intelligence that is currently built in to handle various maneuver scenarios.

As the MMS mission moves from design into fabrication and testing, the ground system is beginning development of the operational systems. The ETE code will become the primary prototype and regression test system for a new system that is envisioned to have a more powerful decision engine and a fully integrated navigation and maneuver planning simulation capability.

The lead author would like to acknowledge the exceptional work and dedication of the entire MMS team and Ms Cheryl Gramling who were instrumental in bringing the End-to-End code to fruition.

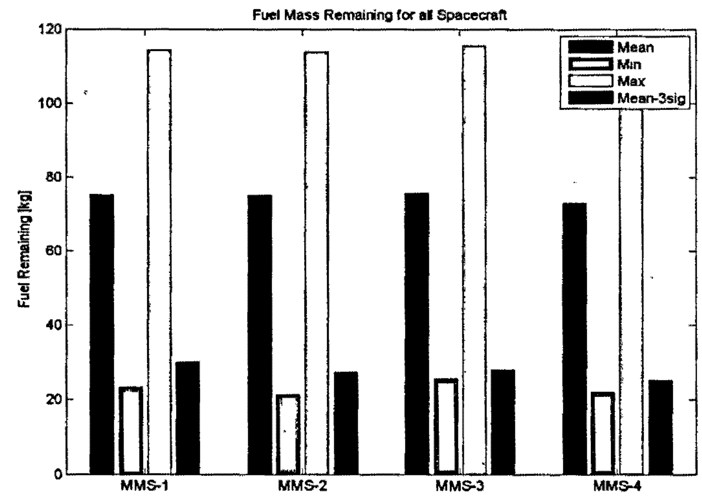


Figure 9: Fuel remaining statistics for the MMS formation

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