American Astronautical Society
Rocky Mountain Section

AAS 14-092

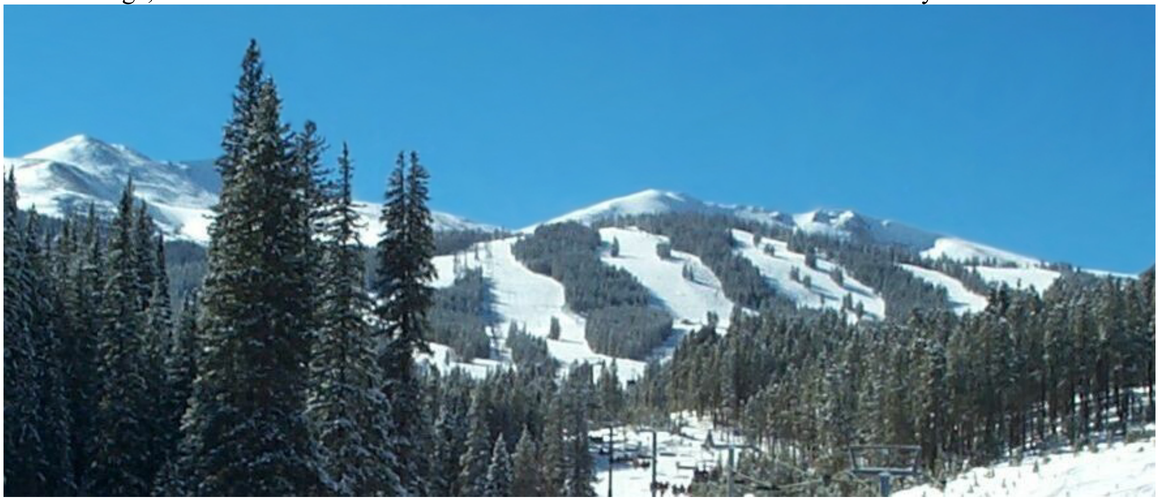
Orion Exploration Flight Test-1 (EFT-1) Absolute Navigation Design

Jastesh Sud, LMSSC – Denver, CO
Robert Gay, NASA – Houston, TX
Greg Holt, NASA – Houston, TX
Renato Zanetti, NASA – Houston, TX

37th ANNUAL AAS GUIDANCE AND CONTROL CONFERENCE

January 31 - February 5, 2014
Breckenridge, Colorado

Sponsored by
Rocky Mountain Section



AAS Publications Office, P.O. Box 28130 - San Diego, California 92198

Orion Exploration Flight Test-1 (EFT-1) Absolute Navigation Design

Jastesh Sud[1], Robert Gay[2], Greg Holt[3], and Renato Zanetti[4]

Scheduled to launch in September 2014 atop a Delta IV Heavy from the Kennedy Space Center, the Orion Multi-Purpose-Crew-Vehicle (MPCV's) maiden flight dubbed "Exploration Flight Test -1" (EFT-1) intends to stress the system by placing the uncrewed vehicle on a high-energy parabolic trajectory replicating conditions similar to those that would be experienced when returning from an asteroid or a lunar mission. Unique challenges associated with designing the navigation system for EFT-1 are presented in the narrative with an emphasis on how redundancy and robustness influenced the architecture. Two Inertial Measurement Units (IMUs), one GPS receiver and three barometric altimeters (BALTs) comprise the navigation sensor suite. The sensor data is multiplexed using conventional integration techniques and the state estimate is refined by the GPS pseudorange and deltarange measurements in an Extended Kalman Filter (EKF) that employs the UDU^T decomposition approach. The design is substantiated by simulation results to show the expected performance.

INTRODUCTION

In 2006, commissioned under the administration of President George W. Bush, Lockheed Martin was awarded the prime contract to develop the next generation man-rate spacecraft called the Orion Crew Exploration Vehicle (CEV) to ferry astronauts to the Low Earth Orbit (LEO), service the International Space Station (ISS) and bolster deep space exploration by returning humans back to the moon and eventually taking them to Mars and beyond. The venture initially fell under the auspices of an umbrella program known as the Constellation, which was later scrubbed by the Obama administration. After surviving through a period of uncertainty that is not uncommon amidst an administration change, the scope of the Orion project was redefined to primarily focus on deep space exploration. Shifting work scope eclipsed by political turmoil precipitated a name change from Orion CEV to Orion Multi-Purpose Crew Vehicle (MPCV).

[1] Jastesh Sud, Systems Engineer Sr, Lockheed Martin Space Systems Company, M/S B3003, P.O. Box 179, Denver, CO 80201, jastesh.sud@lmco.com, 303-971-5826

[2] Robert Gay, Orion NASA Absolute Navigation Lead, Aerosciences and Flight Mechanics Division, NASA Johnson Space Center, Mail Code EG6, 2101 NASA Parkway, Houston, TX 77058, robert.gay-1@nasa.gov, 281-483-6330

[3] Greg Holt, Orion NASA Deputy Navigation Lead, Flight Dynamics Division, NASA Johnson Space Center, Mail Code DM43, 2101 NASA Parkway, Houston, TX 77058, greg.n.holt@nasa.gov, 281-483-0292

[4] Renato Zanetti, Navigation Engineer, Aerosciences and Flight Mechanics Division, NASA Johnson Space Center, Mail Code EG6, 2101 NASA Parkway, Houston, TX 77058, renato.zanetti@nasa.gov

Political challenges aside, Orion faces a host of technical challenges given the diversity of environments the spacecraft needs to survive. One such challenge is being able to successfully re-enter the Earth's atmosphere and land safely. As a risk mitigation maneuver, in 2010 the project decided to add an uncrewed multi-hour, two orbit test called the Exploration Flight Test-1 (EFT-1) to the flight manifest. The EFT-1 trajectory features a highly elliptical second orbit to replicate high-energy re-entry conditions similar to those that would be experienced when returning from an asteroid. At its farthest point Orion would be approximately 5908 km (3671 miles) above the surface of the Earth. Figure 1 depicts the notional mission profile for EFT-1.

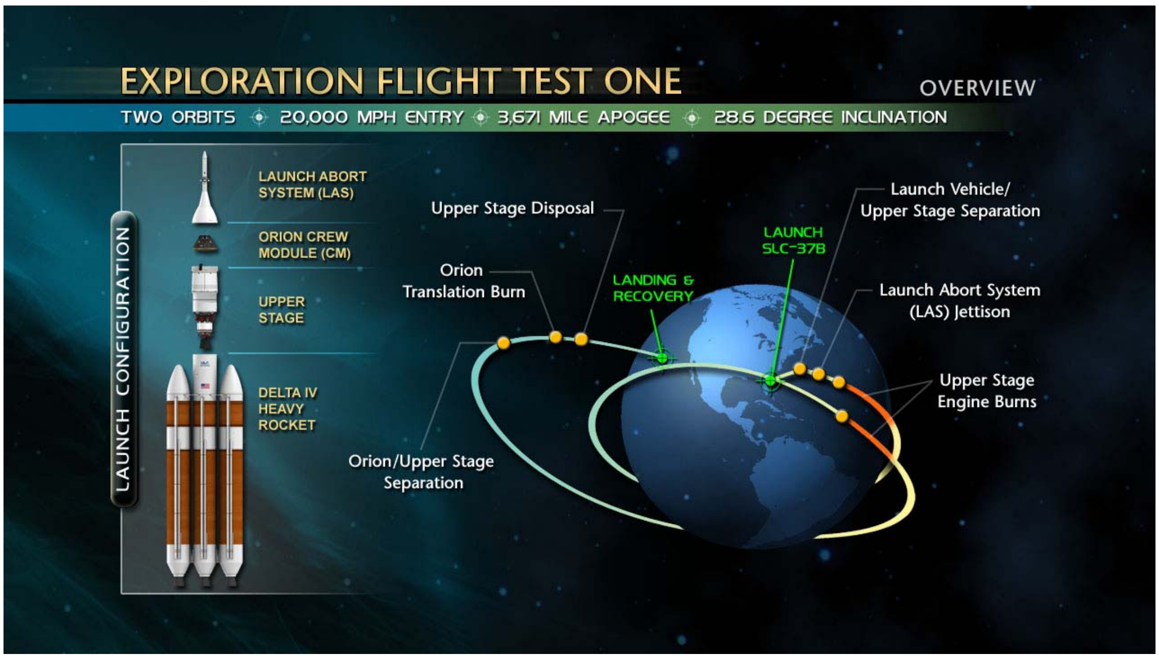


Figure 1: EFT-1 Mission Profile

NAVIGATION ARCHITECTURE

Sensor Manifest for EFT-1

Orion will ultimately be a two-fault tolerant vehicle before it is certified as man-rated. However, the two-fault tolerance requirements were relaxed to just one-fault tolerant for the uncrewed EFT-1 mission. A minimal sensor suite consisting of two Orion Inertial Measurement Units (OIMUs), a GPS receiver (GPSR) and three baro altimeters (BALTs) was selected to satisfy the navigation requirements. The sensors chosen for EFT-1 are intended to serve as the building blocks for future missions, though many of the Fault Detection, Isolation and Response (FDIR) algorithms will be revised as the two-fault tolerance requirements are re-introduced and more sensors are added.

The OIMUs provide high-rate accelerometer and gyro data for translational and rotational state propagation, respectively. The propagated state is updated by GPS pseudorange (PR) and deltarange (DR) measurements. Early in the design cycle the decision was made to process raw PR and DR measurements rather than using the GPSR derived Position, Velocity and Time (PVT) solution. Though there is more complexity associated with ingesting PR and DR measurements, this approach offers more robustness by allowing the EKF to be more selective with each measurement. PR/DR measurement processing over PVT also affords more design flexibility by allowing the EKF to process GPS updates even if there are less than four satellite vehicles (SVs) being tracked by the GPSR.[5] The baseline configuration for EFT-1, however, is to not process GPS measurements unless there are at least five SVs available.

Contrary to most conventional spacecraft, the EFT-1 vehicle lacks an external attitude reference source, i.e. there is no star tracker in the sensor manifest. Instead the attitude is initialized on the pad via a process called ground alignment[6] and propagated using the high-rate gyro data. The EKF employs a unique method to refine the attitude state estimate by coupling the translational and rotational state dynamics. In the presence of non-conservative acceleration the attitude state partials become observable with respect to GPS measurements.

Although the BALT data is not directly processed by the EKF, BALTs offer redundancy during the Entry, Descent and Landing (EDL) sequence. In the event the GPSR fails or the EKF is unable to process GPS measurements during re-entry, the FDIR logic autonomously selects the BALT output as the primary source of altitude. The logic accomplishes this by comparing the EKF covariance in the radial direction with a parameterized threshold value. Obviously this feature is only available when the BALT outputs are valid which only occurs at low-altitudes—approximately below 22.9 km (or about 75,000 ft).

Figure 2 exhibits the data flow from the sensors to the User Parameter Processing (UPP) block. The UPP serves as the "one-stop-shop" for all navigation outputs that are provided to downstream consumers.

[5] In order for a GPSR to calculate a valid PVT solution it must have line-of-sight to at least four SVs.
[6] Ground alignment is analogous to gyro-compassing

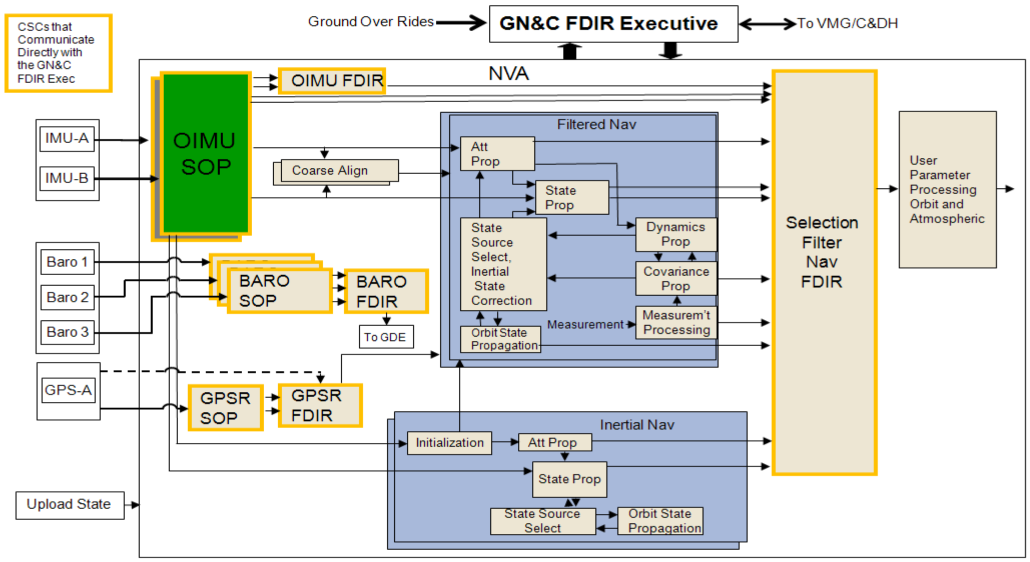


Figure 2: Sensor Data Flow

Processing Rate of Navigation Algorithms

The EFT-1 navigation algorithms are divided into two rate-groups: 1) 40Hz and 2) 1Hz. The 40Hz processing is intended to support the needs of downstream consumers by propagating the state at high-rate. In contrast the 1Hz processing is primarily reserved for the processing of the GPS data.

Navigation Channels

The concept of a navigation channel was developed to provide the capability to "hot-swap" the state estimate in the event a failure occurred. A channel is directly tied to an OIMU meaning Channel1 and Channel2 correspond with OIMU1 and OIMU2, respectively. Within a channel exists an IMU Sensor Operating Procedure (IMUSOP) Computer Software Unit (CSU) that is responsible for parsing the OIMU data, a Coarse Align (CAlign) CSU that is responsible for providing a crude estimate of the initial attitude on the pad, a Filtered Navigator (FILT) CSU that is responsible for multiplexing the OIMU data with the GPS updates and an Inertial Navigator (INRTL) CSU that is responsible for maintaining an un-aided (OIMU-only) state. Figure 3 provides a schematic representation of the 40Hz processing. INRTL and FILT CSUs have counterparts on the 1Hz side. The Inertial Navigator Gravity (InrtlNavGrav) and EKF CSUs on the 1Hz side provide a higher-order gravity estimate to INRTL and state updates to FILT, respectively.

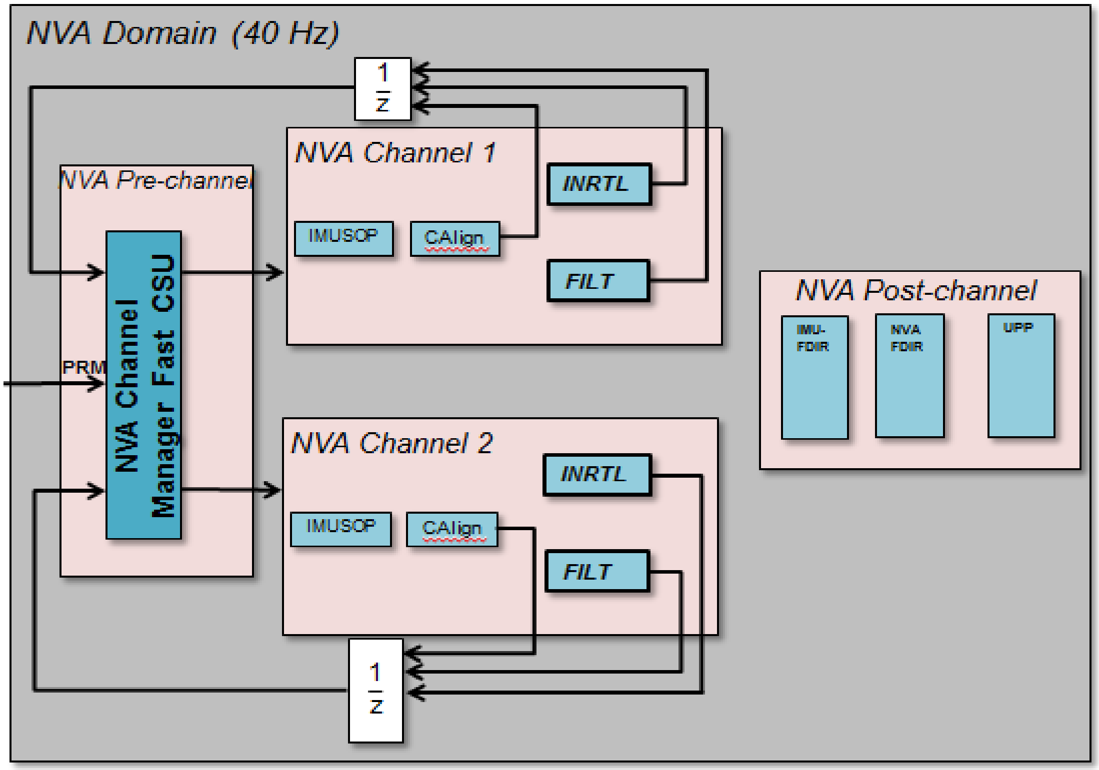


Figure 3: Navigation Channel Concept

The outputs of both Channel1 and Channel2 are funneled into the Post-Channel block where the Navigation FDIR (NAVFDIR) CSU selects the primary state. The NAVFDIR scheme relies on the IMUFDIR outputs and augments the selection criteria with a few checks on the filtered state. One of the checks it relies on is the percentage of PR/DR measurements being accepted by each channel.

EKF Design

The measurements used in the Orion EKF are Integrated Velocity (IV), GPS pseudorange, and GPS deltarange. On the pad, the Integrated Velocity measurement is used for IMU alignment/gyrocompassing, this measurement is not used the rest of the mission and consists in a measured change in ECEF position of zero. The GPS antennas are obscured by the Launch Abort System shroud and therefore the GPS measurements are not available until several minutes into the ascent once the shroud is jettisoned. The GPS pseudorange and deltarange are used for the remainder of the mission.

A high-fidelity mission simulation called Osiris was utilized for tuning the Orion EKF. Osiris uses a TRICK environment running the actual flight software in a variety of dispersed Monte Carlo scenarios.

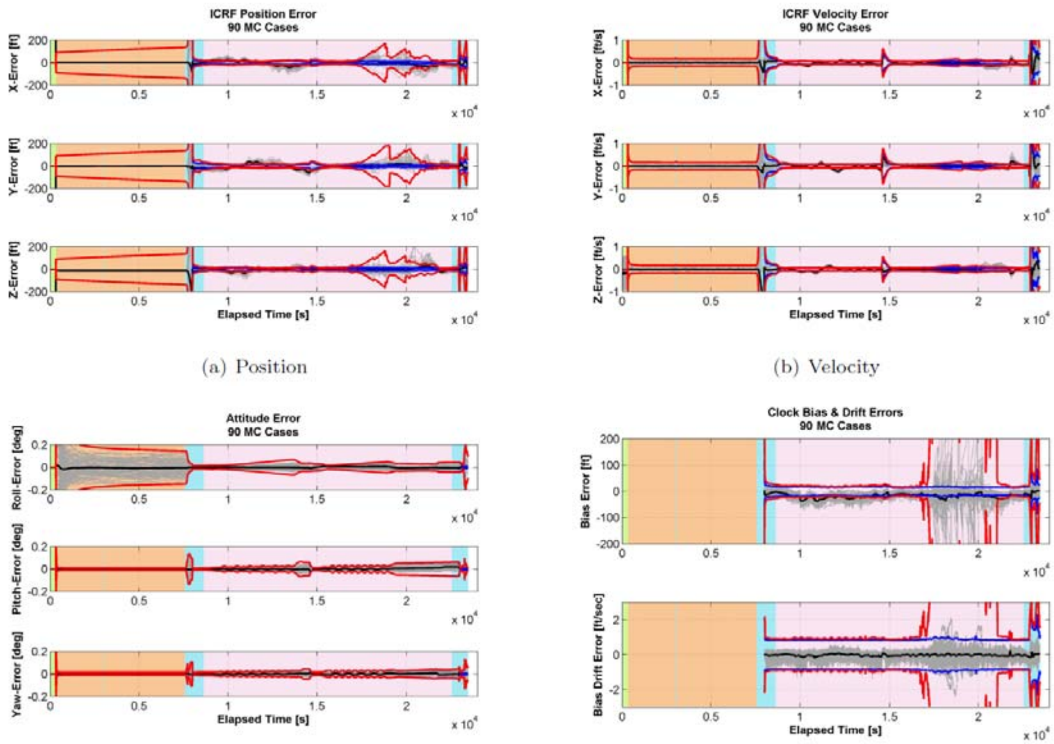
The Orion filter design uses 24 Exponentially Correlated Random Variable (ECRV) parameters to estimate the accelerometer/gyro misalignment and nonorthogonality. By design, the time constant and noise terms of these ECRV parameters were set to manufacturer specifications and not used as tuning parameters. They are included in the filter as a more analytically correct method of modeling uncertainties than ad-hoc tuning of the process noise. Tuning is explored for the powered-flight ascent phase, where measurements are scarce and unmodeled vehicle accelerations dominate. On orbit, there are important trade-off cases between process and measurement noise. On entry, there are considerations about trading performance accuracy for robustness. Process Noise is divided into powered flight and coasting flight and can be adjusted for each phase and mode of the Orion EFT-1 mission. Measurement noise is used for the IV measurements during pad alignment. It is also used for the pseudorange and deltarange measurements during the rest of the flight.

As described previously, the IMU error states are modeled as ECRV states in the filter, so they are tuned using a process noise and time constant. The noise values were taken from the component specifications, and the time constants were chosen as two hours (for the IMU biases) and four hours (rest of the IMU error states) to be roughly constant over the duration of the EFT-1 mission. The other filter values to tune were the measurement underweighting, velocity process noise, angular process noise, and measurement weights. In general, the process noise was tuned by propagating the filter without measurements and ensuring the resulting error growth was bounded by the predicted uncertainty. The measurement noise was considerably more difficult to tune, because the measurements themselves were heavily time correlated especially the pseudoranges. This is due to a poor ionosphere correction model and lack of low-elevation satellite masking.

The filter tuning performance was analyzed for a closed loop End-to-End mission scenario. The position components are most impacted by GPS atmospheric errors, while the velocity components are most affected by unmodeled accelerations and IMU model errors. There is still ongoing investigation into the velocity error signature as it is suspected this may be an artifact of unrealistic accelerations supplied by the simulation from the trajectory dynamics file.

SIMULATION RESULTS

This section contains the results of 90 Monte Carlo runs of a complete mission, from pre-launch to landing. These runs confirm the validity of the chosen parameters. Figure 4 shows the performance of the usual eleven states (position, velocity, attitude, GPS receiver clock bias and drift) as well as the performance of the IMU error states. In these plots, the green section is the coarse align, the orange is the fine align, the first blue segment is ascent, the pink section is orbital flight, and the final blue section is entry and landing.



(a) Position (b) Velocity

(c) Attitude (d) Clock

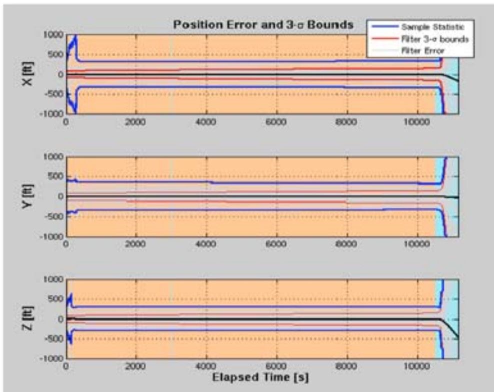
Figure 4: End-to-End Performance

<u>Prelaunch</u>

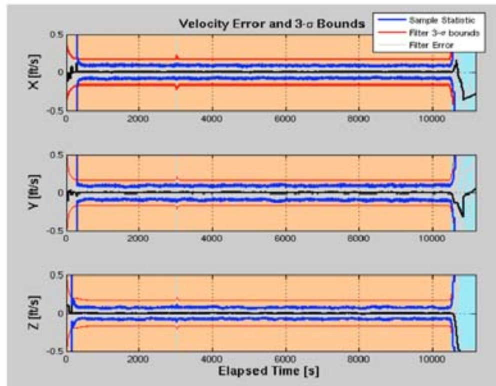
This is the phase prior to launch when the vehicle is on the pad. The only measurement used during fine- align mode is integrated velocity (IV). The pseudorange (PR) and delta range (DR) measurement types are not used during this phase. The main purpose of fine align is to better estimate the attitude and the IMU states, therefore none of these states are considered and they are all estimated.

During fine align the OIMUs sense the opposite of gravity which is a large acceleration. The accelerometer threshold is the same for all flight phases and its value is determined during tuning of the orbit phase. The velocity random walk of the accelerometer is very small, therefore the translational process noise is small and dominated by the unmodeled gravity. The angular process noise is chosen from the gyro's angular random walk spec value. The GPS clock bias and drift noises are chosen from the expected clock physical properties. The coast values for the process noise are not used in this phase; they are only used during the orbit phase. The values are kept the same across flight phases and tuned for orbit.

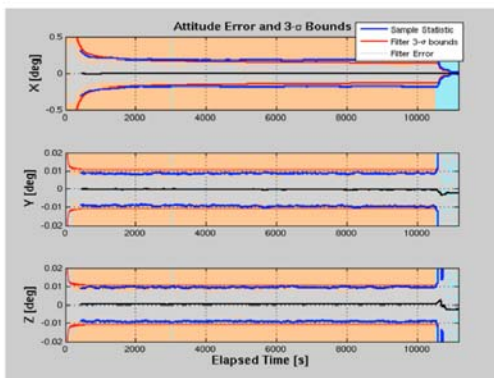
Figure 5 shows the performance of 450 Monte Carlo runs in a Launch Hold scenario which transitions back to fine align for a longer hold prior to launch. It demonstrates the choice of parameters provides reasonable performance even in the worst-case alignment conditions. In this scenario the initial position error on the pad is dispersed. The attitude, position, and velocity are all shown to be well bounded for the entire duration of the alignment and launch hold.



(a) Position



(b) Velocity



(c) Attitude

Figure 5: Prelaunch Performance

Ascent

The ascent phase is divided in two parts, the first when GPS measurements are not enabled, and the second when they are. The only difference between Fine Align and Ascent Without GPS is that IV measurement processing flag is set to zero in the latter. Integrated velocity (IV) is only used on the pad, while GPS measurements (PR and DR) are utilized during this phase. To date there is no simulation or any other evidence that attitude or IMU state estimation would corrupt the EKF solution. Therefore none of these states are considered and they are all estimated. The maximum number of processable measurements is set to 12 which is a large enough number to obtain sufficient performance while keeping the flight computer throughput reasonably low. To avoid possible transient issues the first PR measurement is not processed. After a long blackout the covariance becomes very large and the nonlinearity of the DR measurement creates convergence issues. Through numerical simulation it was determined that allowing for multiple PR (30s) to be processed before incorporating a DR mitigates this issue because the PRs shrink the uncertainty before DRs are introduced. If a satellite is not present for a single cycle, the counters are not reset and if the satellite comes back is immediately used as a measurement. If the satellite is absent for more than a cycle the counters are reset.

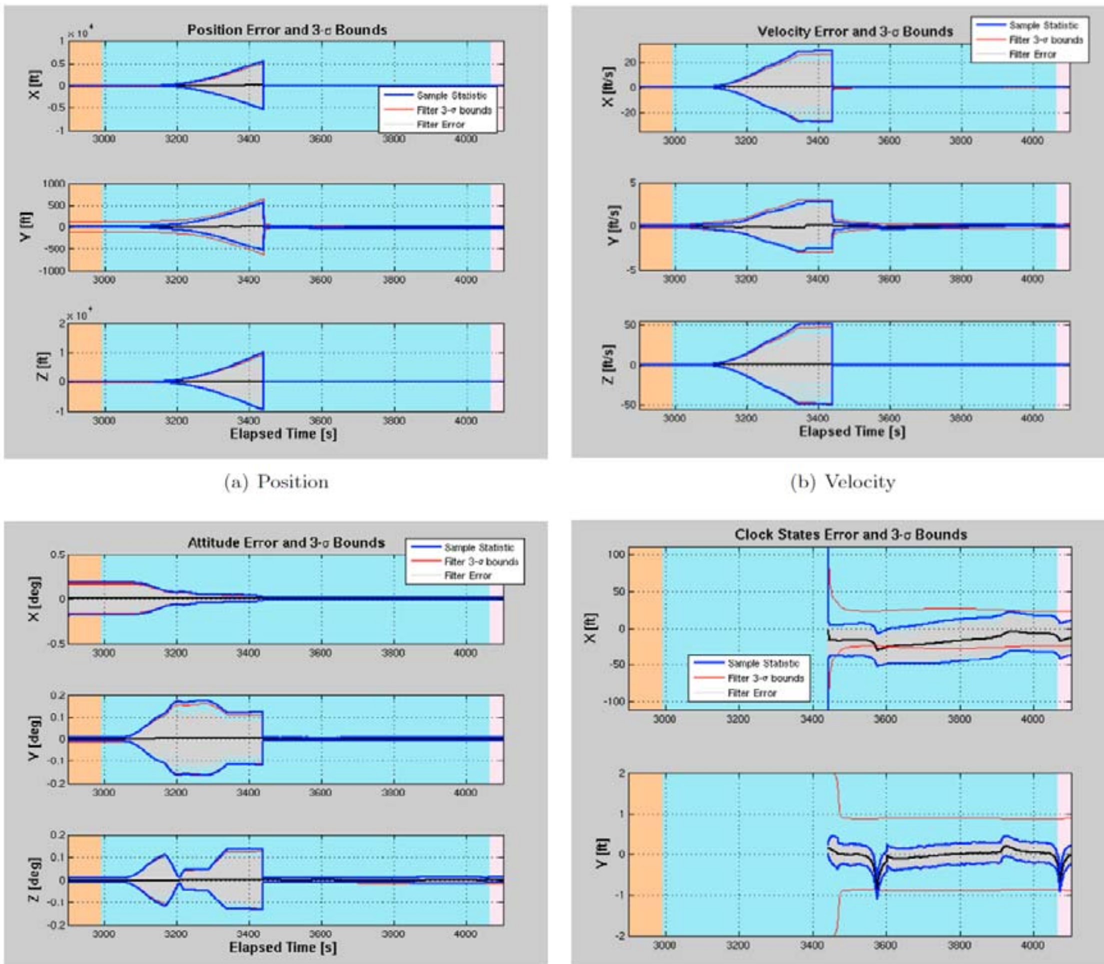
PR and DR noise standard deviations are obtained from GPSR performance. The numbers are kept high to include residual ionosphere error. A larger residual edit threshold is used to compensate for the higher noise values. While we allow for the usage of the PR variance output from the GPS receiver, the EKF has a PR variance floor that will almost always prevent us from using the GPS receiver output value except in the case of very large values. Underweighting is applied when the estimated measurement has an uncertainty greater than 100ft.

For the EFT-1 mission we want to test GPS clock stability, clock filter state restarts, and high altitude GPS processing, so we do not want to exercise a GPS clock reset using the clock bias re-initialization timer. Therefore, this value is set to a number larger than the expected duration of the mission.

Initial uncertainty of the clock bias and drift are obtained from GPSR specifications. When accelerating fast under the chutes the attitude dynamics is not accurately represented by the 4Hz IMU buffer. Therefore PR and DR are inhibited above a certain angular velocity, the values are determined from simulation analysis.

A deltarange timing threshold of 1.5 seconds prevents the flight software from trying to incorporate a DR measurement using two non-consecutive PR measurements (GPS measurements are available at 1Hz).

Figure 6 shows the performance of 450 Monte Carlo runs, this is a nominal scenario to demonstrate the choice of parameters provides good performance. These runs are performed without dispersing the GPS constellation in order to obtain the same set of measurement for each Monte Carlo run. As before, the errors seem well bounded and convergence times are reasonable.



(a) Position (b) Velocity

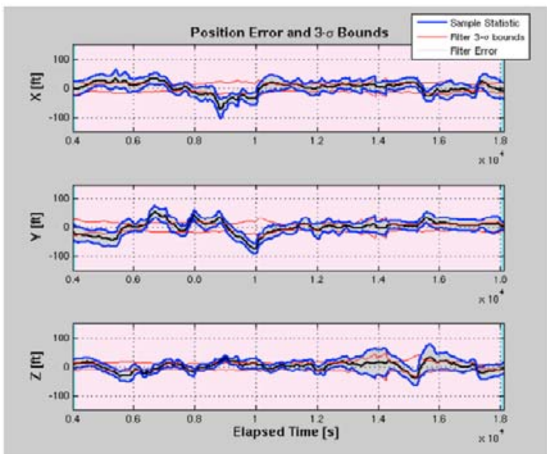
(c) Attitude (d) Clock

Figure 6: Ascent Performance

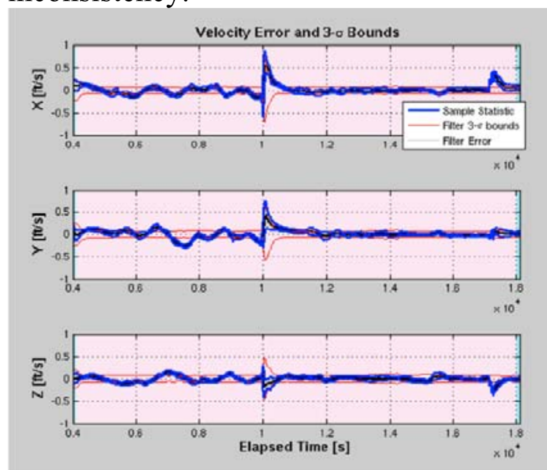
<u>Orbit</u>

The accelerometer bias is constant for a particular trajectory, but random across an ensemble of trajectories. To obtain a worst-case estimate we take three times the standard deviation of the bias, and treat the result as a constant. Each error source applies to each component of the velocity. We assume that the velocity white noise and the velocity random walk are independent. The same GPS parameters are used during orbit as are used during ascent. The only change is that a slightly lower value is used for the underweighting coefficient.

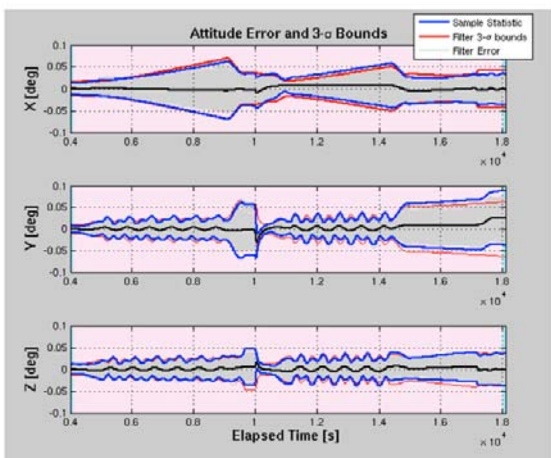
Figure 7 illustrates the performance of 450 Monte Carlo runs. This is a nominal scenario to demonstrate that the choice of parameters provides good performance. These runs are performed without dispersing the GPS constellation in order to obtain the same set of measurement for each Monte Carlo run. Although some of the errors lie outside the three-standard deviation bounds estimated from the filter's covariance matrix, this is a nominal condition. The errors quickly converge to values within the requirement for accuracy of estimation of on-orbit position and velocity. The acceleration models in the trajectory simulation file are suspected to not match the velocity, so the oscillating error behavior is likely at least partially due to this inconsistency.



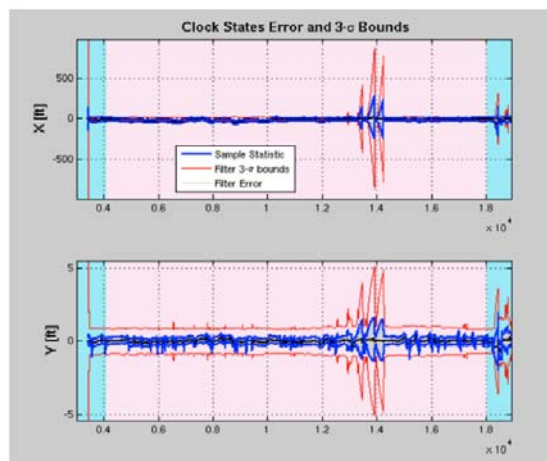
(a) Position



(b) Velocity



(c) Attitude

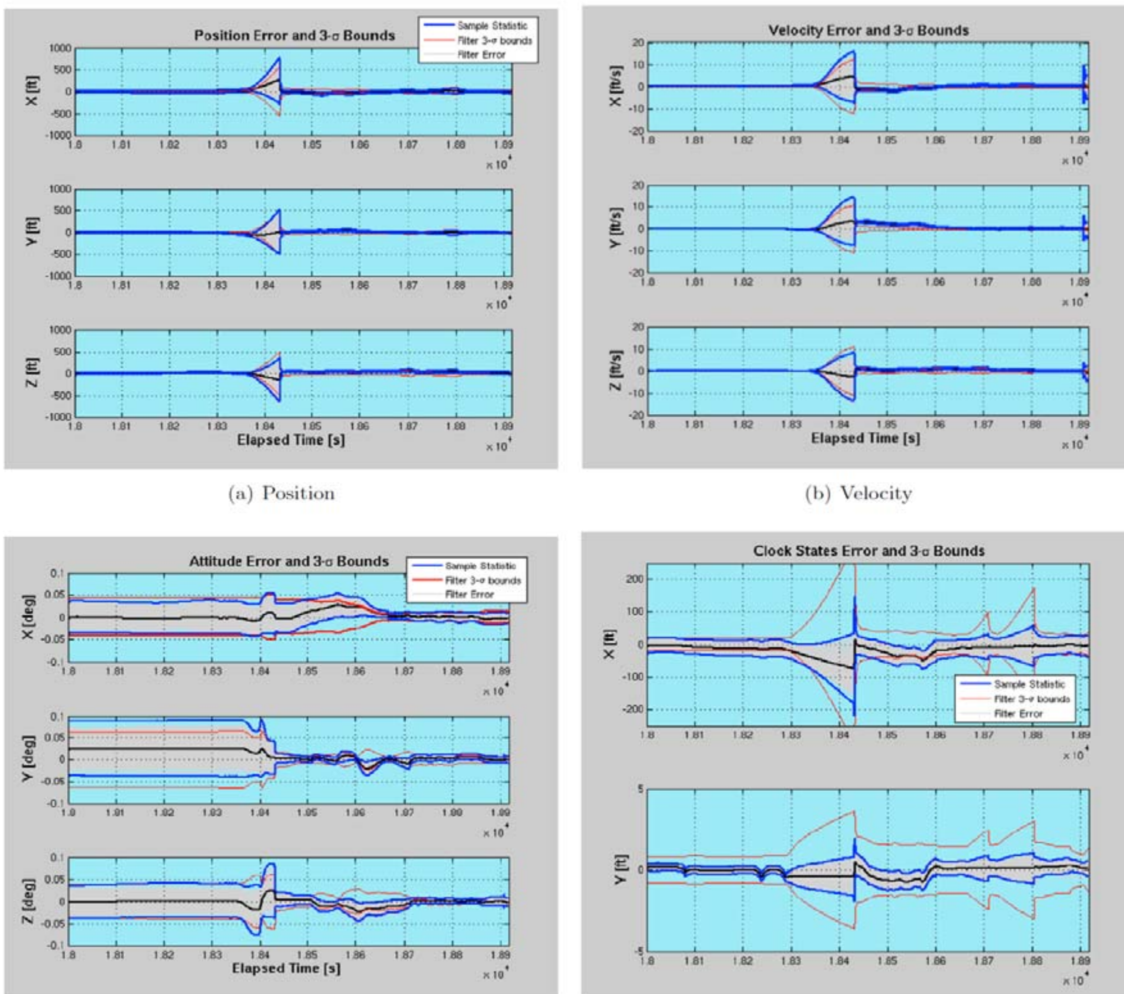


(d) Clock

Figure 7: Orbit Performance

<u>Entry</u>

Figure 8 shows the performance of 450 Monte Carlo runs, this is a nominal scenario to demonstrate the choice of parameters provides good performance. These runs are performed without dispersing the GPS constellation in order to obtain the same set of measurement for each Monte Carlo run.



(a) Position (b) Velocity

(c) Attitude (d) Clock

Figure 8: Entry Performance

CONCLUSIONS

This paper presents the design of the absolute navigation architecture and filter for the Orion Exploration Flight Test I mission. There are still some aspects of the filter tuning and simulation environment that are under investigation. The oscillating velocity error signature during Low Earth Orbit is suspected to be at least partially due to numerical issues with the input trajectory file. Future implementations of the filter design will include masking of GPS signals with high ionospheric interference, which should mitigate the position excursions outside the 3σ filter uncertainty. In spite of this future work, to date the filter's performance has been proven robust and sufficient to meet mission requirements in both high fidelity simulation and hardware-in-the-loop testing.