

SPACE LAUNCH SYSTEMS BLOCK 1B PRELIMINARY NAVIGATION SYSTEM DESIGN

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NASA is currently building the Space Launch Systems (SLS) Block 1 launch vehicle for the Exploration Mission 1 (EM-1) test flight. In parallel, NASA is also designing the Block 1B launch vehicle. The Block 1B vehicle is an evolution of the Block 1 vehicle and extends the capability of the NASA launch vehicle. This evolution replaces the Interim Cryogenic Propulsive Stage (ICPS) with the Exploration Upper Stage (EUS). As the vehicle evolves to provide greater lift capability, increased robustness for manned missions, and the capability to execute more demanding missions so must the SLS Integrated Navigation System evolved to support those missions. This paper describes the preliminary navigation systems design for the SLS Block 1B vehicle. The evolution of the navigation hardware and algorithms from an inertial-only navigation system for Block 1 ascent flight to a tightly coupled GPS-aided inertial navigation system for Block 1B is described. The Block 1 GN&C system has been designed to meet a LEO insertion target with a specified accuracy. The Block 1B vehicle navigation system is designed to support the Block 1 LEO target accuracy as well as trans-lunar or trans-planetary injection accuracy. Additionally, the Block 1B vehicle is designed to support human exploration and thus is designed to minimize the probability of Loss of Crew (LOC) through high-quality inertial instruments and robust algorithm design, including Fault Detection, Isolation, and Recovery (FDIR) logic.

INTRODUCTION

In a recent status report, Associate Administrator William Gerstenmaier outlines the plan forward for NASA Human Exploration and Operations (HEO). Phase 1 of that plan includes building and executing the capability of operating manned spacecraft in the Lunar vicinity while incrementally building a capability for deep space exploration. SLS is a key enabler of this vision.¹ Currently, GN&C at Marshall Space Flight Center is designing the Block 1B launch vehicle while finishing the Block 1 vehicle. The Block 1B vehicle evolves with the new integrated Exploration Upper Stage (EUS). The first 4 Block 1B missions include a cargo mission bound for Jupiter with the Europa Clipper. The remaining three are Exploration Missions, crewed missions with an Orion spacecraft, destined for incrementally proving out capability in cis-lunar and lunar space. Additionally, each EM will be dually responsible for delivering a component of the Deep Space Gateway

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onto a Trans-Lunar Injection (TLI) trajectory.² As part of the evolution, a new stage is being developed. The SLS GN&C and avionics systems, including SLS Navigation must evolve to meet the challenges of post-Block 1 human exploration. The SLS design team is currently in the middle of the second design and analysis cycle (DAC). This paper describes the preliminary navigation system design.

DESIGN OBJECTIVES, CONSTRAINTS, AND REQUIREMENTS

SLS Block 1B vehicle state accuracy requirements are currently stated in terms of the impact to the payload in the delta velocity required by the payload to correct for state errors at the insertion target. A second constraint which must be supported by navigation is to dispose the stage after mission completion.³ For Lunar-bound missions, the EUS will dispose into heliocentric orbit via a lunar assist maneuver. The lunar fly-by is primarily performed with the TLI burn followed by a later maneuver performed with RCS thrusters to correct for orbit perturbations from post-TLI maneuvering, propellant blow down, and venting. The accuracy required for stage disposal via lunar assist has proven to be the driving requirement for navigation accuracy. The navigation system must also support the pointing and attitude accuracy required for SLS-payload proximity operations and the Guidance and Controls interface for all necessary state derived data.

In addition to having to meet mission requirements, other objectives include the desire to reuse the Block 1 Inertial Navigation System (INS) hardware and to utilize a modified GPS Range Tracking Unit for GPS aiding. The system is to demonstrate a high level of robustness to sparse GPS measurements, to GPS outage, and asynchronous communications, which was shown during Hardware-in-the-loop (HWIL) testing with INS and the Flight Computers (FC) to result in occasional sporadic inertial data.

SLS is designed for human space flight and includes a high level of redundancy in the avionics systems. The navigation system must manage this redundancy and provide an adequate fault detection, isolation, and recovery for the navigation sensors.⁴ The navigation system must also support failure mitigation strategies in the Guidance system and Controls systems by providing indications of dubious state, state derivative, and state derived data provided at the respective interface. Additionally, an interface with Mission & Fault Management systems is provided to support notification to the crew of potential problems. The design will be augmented with the capability to accept state data from sources external to the vehicle.⁵

NAVIGATION HARDWARE

The SLS Block 1 vehicle consists of an SLS Core Stage and an independent upper stage, denoted as the Interim Cryogenic Propulsive Stage (ICPS). The SLS Core Stage serves the purpose of the ascent vehicle inserting the ICPS and uncrewed Orion payload into a 40.7 km by 1805.7 km (22 Nmi by 975 Nmi) elliptical Earth orbit. The Core Stage navigation hardware consists of a Redundant Inertial Navigation Unit (RINU). For the Block 1 vehicle, the navigation systems for the Core stage and ICPS stage are independent of one another. Both systems perform alignment operations on the ground and Core stage flight is then flown utilizing a navigation solution derived from RINU data. During the ascent phase of flight, the ICPS navigation system passively tracks the navigation state. The hand-off of vehicle GN&C occurs at Core Stage separation, immediately following insertion into LEO.

For the Block 1 vehicle and EM-1 mission, the ascent phase is flown with inertial navigation only. During pre-launch, the RINU utilizes a gyrocompassing alignment algorithm to determine the initial attitude relative to a local-level frame. Supplemental algorithms residing on the SLS

Flight Computer (FC) provide the information required to determine the RINU attitude relative to the navigation frame, Earth Center Inertial (ECI) True of Date (TOD).

The Block 1B vehicle utilizes the new EUS for post-ascent flight. EUS includes the centralization of SLS GN&C for ascent and in-space operations. With the extension of required capability, inertial navigation does not provide suitable accuracy for the SLS Block 1B vehicle. The Block 1B/EUS navigation hardware design currently reuses the RINU and also includes a new redundant set of GPS receivers for navigation aiding. The Block 1B navigation hardware components are described in more detail in the sections below. Not previously mentioned are the Rate Gyro Assemblies (RGA). The RGAs provide angular rate measurements in the vehicle body pitch/yaw plane. These angular rate measurements are used by the controller to mitigate the effect of flexible body dynamics on vehicle stability. The SLS Block 1 and Block 1B navigation software processes these rates for flight control.

Redundant Inertial Navigation Unit (RINU)

The RINU is a redundant inertial navigation system manufactured by Honeywell International, Inc (HI). The RINU is derived from the Fault Tolerant Inertial Navigation Unit (FTINU) INS previously flown on the Atlas V launch vehicle. The RINU features a redundant set of five inertial instruments channels. Each sensor channel consists of a HI GG1320 ring laser gyro, a HI QA2000 accelerometer and associated sensor electronics. The inertial instruments meet the high accuracy required for launch vehicle navigation and gyrocompassing accuracy.

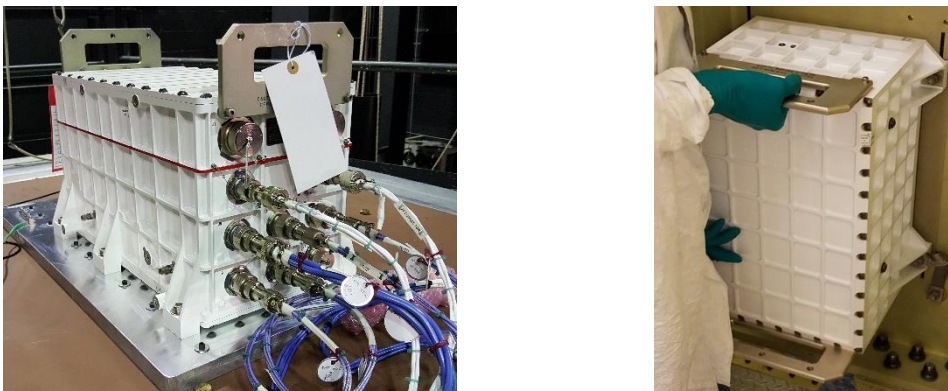


Figure 1. SLS Redundant Inertial Navigation Unit (RINU) on Stewart Platform (Left), SLS Rate Gyro Assembly (RGA) being installed in Block 1 Core Stage Engine Section (Right).

Besides high quality inertial instruments, the second key feature of the RINU is the fault tolerance. Like the FTINU, two different redundancy schemes are employed on the RINU. On the processing side, the RINU utilizes a hot-spare principle for fault tolerance. There are two independent processing channels each with two processing components. The first processing component, denoted by Inertial Measurement System (IMS), interfaces with the redundant sensor channels and handles the high rate gyro and accelerometer sampling. The second processing component, the Navigation System (NS), manages the output interface to the three SLS Flight Computers via the redundant SLS MIL-STD-1553 UF Busses. The remaining functions of measurement compensation, filtering (required for a navigation grade Inertial Measurement Unit (IMU)), sensor FDIR, gyrocompassing and inertial navigation software are shared between the two processors. Each processing component consists of a self-checking pair of robust MIL-STD-1750A processors.^{6,7}

The IMS processors interface with the five electrically isolated sensor channels. The inertial sensors are mounted on a mechanically isolated Inertial Sensor Assembly (ISA) block. The sensitive axes of the sensor pentad are symmetrically aligned about the nominal vehicle roll axis and skewed relative to the pitch/yaw plane. Analysis performed for SLS has shown this as best to enable Fault Detection, Isolation, and Recovery (FDIR) at the sensor level. Inertial measurements along the 3 mount frame axes are derived from the redundant measurements. When redundant measurements are mathematically combined, they provide an increase in effective inertial measurement accuracy statistics over a true triad of similar instruments and a means to detect if a sensor is performing outside of expectation. If a sensor is performing outside of expectation, it can be removed from the inertial navigation solution.⁸ The SLS Block 1 and Block 1B navigation systems leverage heavily on this low level FDIR functionality and mission specific configurability of the sensor FDIR algorithms.⁹

GPS Receiver (GPSR) and Antenna Configuration

As previously stated, the Block 1B vehicle missions require better accuracy than can be provided by inertial navigation due to the significantly increased mission timelines and the unbounded error growth associated with inertial navigation. For this scenario, the GPS was chosen as a navigation aid. The specific GPS receiver design is still being traded. An option under consideration is to combine the required functionality of the GPS Range Tracking Unit (GRTU) used as the primary range tracking source during ascent and the GPS receiver required for navigation.

For SLS, the selected GPS receiver is required to provide GPS pseudorange, range rate, and a Position, Velocity, and time (PVT) solution with specified accuracy through ascent, Low Earth Orbit (LEO), and through Medium Earth Orbit (MEO), defined as 3000km altitude to 8000km altitude for use by SLS navigation. The altitude limit for accuracy requirements was chosen to reduce dependency on GPS constellation geometry and specific launch trajectories.¹⁰ Above 8000km, the receiver is only *required* to operate but *expected* to provide sparse pseudorange and range rate measurements. While operating, the receiver may provide GPS time for FC time discipline and as a common time standard for navigation time standardization.

The SLS EUS avionics will include at least two GPS receivers to support SLS navigation. Each will have two antennas with boresight normal to the vehicle surface, facing outwards, and clocked 180 degrees apart to maximize visibility of the GPS constellation and to minimize attitude dependence for satellite visibility.

Redundant Gyro Assembly (RGA)

The RGA is designed and manufactured by Honeywell International, Inc. and derived from Atlas V RRGU heritage. The RGAs each contain 3 redundant pairs of gyros arranged normal to one another. The sensitive axes are aligned nominally within the SLS vehicle pitch/yaw plane. Accompanying each gyro pair is a suite of sensor electronics and a 1553 interface comprising an RGA channel. The sensors are mechanically isolated and each channel is electronically independent.

PRELIMINARY FLIGHT SOFTWARE DESIGN

The SLS Block 1 and Block 1B Flight Software (FSW) is written in C/C++. The Navigation algorithms are delivered as part of the SLS program-controlled GN&C model per the associated model based design practices.¹¹ The SLS Navigation system can be separated into the hardware components previously discussed and Flight Software. The RINU and RGAs communicate with the FCs over a MIL-STD-1553 bus. The RINU is sampled at the GN&C execution frequency. The RGAs are sampled twice per minor frame. The RINU buffers inertial measurement data for higher rate processing within the navigation software. For the Block 1 design, alignment and inertial

navigation is performed in the RINU. In the Block 1B design, the GPS receiver will communicate over RS-422 and be sampled at a significantly lower rate than RINU. The inertial navigation functionality is performed in SLS FSW on the FC with fully compensated inertial measurements from the RINU. Minimizing additional computational and communication throughput is emphasized in the design due to the large increases in the computational burden on the FCs with the additional EUS systems and the necessity of transferring redundant data to each of the three redundant FCs each minor cycle. Similarly, there is a desire to reuse Block 1 software components and/or to minimize changes to reduce overall effort in development and testing.

Strapdown Inertial Navigation (SDINS) Algorithms

Each minor frame (50 Hz), the FC samples the RINU and collects two buffered 100 Hz incremental angles, $\Delta\theta$, and incremental velocities, Δv . These incremental quantities are valid over a 0.01 second interval. These incremental quantities are converted into angular rate and non-gravitational specific force measurements, ω and a , respectively. Specifically, the angular rates are the rotational rates of the sensor-relative body frame with respect to an inertial frame, expressed in the sensor-relative body coordinates. During HWIL testing of the Block 1 development, it was discovered that, owing to the asynchronous nature of the MIL-STD-1553B communications between the RINU and the FCs, that, in the buffered data collected from the RINU, occasional 100 Hz samples would be missed or repeated.

To use the existing data interface from the RINU, the navigation team was challenged to use a combination of algorithms that detect how many data points of the set of redundant measurements in a minor frame are unique. The team was additionally challenged to use integration algorithms that do not necessarily assume a fixed step-size. For the skipped-sample detection, the corresponding time tags are efficiently analyzed to determine how many new data samples are available. Nominally there are two samples, but as samples are missed or repeated, there will occasionally be only one or even three new data points. The general flow of the angular rate and specific force data through the inertial integration algorithms is shown in Figure 2.

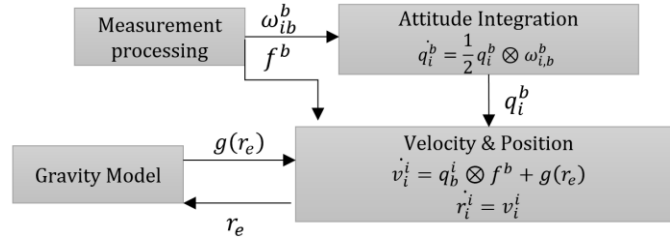


Figure 2 Data flow through the inertial integration algorithms.

For integrating the attitude from the angular rate measurements, we begin with the kinematical equation, Eq. (1).

$$\dot{q}_i^b(t) = \frac{1}{2} q_i^b(t) \otimes \omega_{ib}^b, \quad (1)$$

where $q_i^b(t)$ is the attitude quaternion relating the inertial frame of choice to the body frame—or the sensor frame in our application. Following Sola, at an integration step, t_n , take a Taylor series of $q_i^b(t_n + \Delta t_n)$.¹² Assuming the angular rates are constant over this interval, with length Δt_n , we then construct the quaternion increment, $q\{\omega_n \Delta t_n\}$ resulting from that constant angular rate, ω_n . The updated attitude quaternion is then estimated as, and denoting $q_n = q_i^b(t_n)$,

$$q_{n+1} \approx q_n \otimes q\{\omega_n \Delta t_n\} \quad (2)$$

Now, assume that the angular rates are linear over each integration interval. As demonstrated by Sola, the resulting approximate attitude quaternion is expressed as

$$q_{n+1} \approx q_n \otimes q\{\bar{\omega}_n \Delta t_n\} + \frac{\Delta t_n^3}{24} q_n \otimes \begin{bmatrix} 0 \\ \omega_n \times \omega_{n+1} \end{bmatrix}, \quad (3)$$

where $\bar{\omega} = (\omega_{n+1} + \omega_n)/2$, is the average of the two most recent angular rate samples. Equation (3) is the two-step attitude integration algorithm implemented in the navigation software. Note that the time-step between samples need not be constant.

The algorithm chosen for the position and velocity integrations is based on the well-known Adams-Bashforth method.¹³ Writing the state (position or velocity) to be integrated as $x(t)$ with an associated derivative $\dot{x}(t) = f(x, t)$, the value at time t_{n+1} can be approximated as

$$x_{n+1} \approx x_n + \Delta t \sum_{k=0}^{N-1} b_k f(x_{n-k}, t_{n-k}) \quad (4)$$

As an example, the coefficients for a 4-step Adams Bashforth method are $\{b_0, b_1, b_2, b_3\} = \{\frac{55}{24}, -\frac{59}{24}, \frac{37}{24}, -\frac{9}{24}\}$.¹³

Implicit in the development of Equation (4) is the assumption of a constant time step Δt . Following Rosales and Colomina this assumption can be relaxed at a price.¹⁴ The coefficients b_k in Equation (4) become dependent on the step-sizes. That is, Equation (4) becomes

$$x_{n+1} \approx x_n + \Delta t_n \sum_{k=0}^{N-1} b_{k,n} f(x_{n-k}, t_{n-k}) \quad (4)$$

Rosales and Colomina develop the corresponding coefficients as rational functions of the buffered step-sizes $\Delta t_n = (t_{n+1} - t_n)$ for a four-step Adams Bashforth method. In addition to the Adams-Bashforth predictor algorithm, Rosales and Colomina also develop expressions for a corresponding set of coefficients for an Adams-Moulton corrector algorithm.¹⁴ These weights are also rational functions of the step-sizes.

For integrating the position and velocities, the SLS Navigation Team implemented the four-step Adams-Bashforth variable step-size model from Rosales and Colomina.¹⁴ Additionally, the team is currently trading the use of the variable step-size Adams-Moulton corrector as well as using a less computationally expensive two-step method.

As the accelerometers in an inertial system only sense specific force, a model of the gravitational acceleration must be included in the navigation software. A detailed implementation of a full spherical harmonic model was implemented in the Block 1B navigation design following the development found in (Reference 15). An advantage of this particular implementation of the gravity model is that the same recursive relationships associated with the terms in the series expansion are re-used to compute the gravity partials used in the extended Kalman filter. A number of computation efficiencies are being explored to reduce the throughput requirements associated with higher-order gravity models. Several alternative implementations for computing the gravitational potential and its gradients under consideration are the Clenshaw (forward-column) method as well as methods using the Pines formulations.¹⁶

To further improve the computational footprint of the navigation software, a trade is currently underway to evaluate the effects of using the full gravity model at a lower rate, e.g., only during frames when the gravity partials are computed for the EKF, than the rest of the navigation system.

In between the gravity updates, the gravitational acceleration is propagated with a first or second order model.¹⁷

Navigation Filter Design

The SLS Block 1B navigation design uses a tightly coupled filter architecture for GPS/INS integration. In this context, tightly coupled is defined as utilizing GPS pseudorange and delta-range measurements versus a loosely coupled design which would utilize estimated whole states from the GPS receiver such as position and velocity. Neither designs necessarily provide feedback to the GPS receiver. Loosely coupled designs are arguably easier to implement but have inherent drawbacks. The first is a dependency on the software within the GPS receiver used to estimate the states. Commonly this is accomplished through the use of a Kalman filter within the receiver. Cascading filters can result in providing the second filter with correlated measurements without the ability to represent the correlation in the GPS/INS filter.¹⁸ A tightly coupled implementation avoids this issue by centralizing the GPS/INS processing through the direct use of pseudorange and/or deltarange measurements. The primary disadvantages of this approach is that additional states representing clock errors must be estimated and the number of measurements being processed dramatically increases. A less noted disadvantage to the tightly coupled approach is with the associated analysis and test burden. A significantly higher fidelity GPS model is required to design, analyze, and test a tightly coupled filter. The key advantage to the tightly coupled approach is that the filter will produce corrections with sparse GPS availability while the loosely coupled design must have 4-6 satellites in view depending on what type of integrity monitoring is used. For missions which must operate at high orbital altitudes, the tightly coupled design is clearly better. The tightly coupled filter replaces a loosely coupled design developed and used for the first DAC for analysis, vehicle trades, and requirements definition. A detailed GPS model was implemented to support development.¹¹

The navigation filter is of feed-back discrete Extended Kalman Filter (EKF) design. The preliminary EKF design includes only 17 states and is consistent with a multiplicative EKF in the attitude state and update.^{19,20} To meet early design objectives, the number of filter states is kept at a minimum in an effort to reduce algorithmic complexity and to minimize FC throughput. The state vector is defined in Eq. (5). The first two states are the position and velocity errors relative to the navigation frame. The third state is the attitude error relative to the navigation frame. The 4th and 5th states are the accelerometer and gyro biases. The 6th and 7th states are the pseudorange and range rate errors due to GPS receiver clock drift bias and drift.

$$x = [\delta \vec{P}_N \quad \delta \vec{V}_N \quad \delta \vec{\psi}_{N+}^T \quad \vec{b}_a \quad \vec{b}_g \quad \delta \rho_{rc} \quad \delta \dot{\rho}_{rc}]^T \quad (5)$$

The general form of the Kalman filter equations consists of the covariance propagation, the Kalman gain equation, the state update equation, and the covariance update equations. These are listed as Eq (6-9).

$$P_k^- = \Phi_{k-1} P_{k-1}^+ \Phi_{k-1}^T + Q_{k-1} \quad (6)$$

$$K_k = P_k^- H_k^T (H_k P_k^- H_k^T + R_k)^{-1} \quad (7)$$

$$x_k^+ = K_k y_k \quad (8)$$

$$P^+ = (I - K_k H_k) P^- (I - K_k H_k)^T + K_k R_k K_k^T \quad (9)$$

Note that, with the exception of the receiver clock terms, the state is not retained by the filter post-update, as seen in Eq (8). The form of the covariance update equation, Eq. (9), is the Joseph form.¹⁹ Although more computationally expensive, early revisions of the filter had issues with maintaining

a symmetric and positive semidefinite covariance matrix. An additional mechanism is also used to enforce symmetry in P as well.

The state dynamics matrix, F, is defined in Eq. (10). The state transition matrix, Φ defined in Eq. (11), is derived from F using a third order Taylor expansion. The state transition matrix is defined by Eq. (12). Some of the higher order terms have been neglected, e.g. some terms with Δt^3 in numerator.

$$F = \frac{\partial f(x)}{\partial X} = \begin{bmatrix} 0 & I & 0 & 0 & 0 & 0 & 0 \\ \partial G/\partial P & 0 & -[C_B^N \vec{f}_B] \times & C_N^B & 0 & 0 & 0 \\ 0 & 0 & 0 & 0 & C_N^B & 0 & 0 \\ 0 & 0 & 0 & -1/\tau_{ba} & 0 & 0 & 0 \\ 0 & 0 & 0 & 0 & -1/\tau_{bg} & 0 & 0 \\ 0 & 0 & 0 & 0 & 0 & 0 & 1 \\ 0 & 0 & 0 & 0 & 0 & 0 & 0 \end{bmatrix} \quad (10)$$

$$\Phi = I + F\Delta t + \frac{1}{2}F^2\Delta t^2 + \frac{1}{6}F^3\Delta t^3 \quad (11)$$

$$\Phi = \begin{bmatrix} I + \frac{1}{2}\frac{\partial G}{\partial P}\Delta t & \Delta t + \frac{1}{6}\frac{\partial G}{\partial P}\Delta t^3 & -\frac{1}{2}[\Sigma C_B^N \Sigma \Delta \vec{V}_B] \times & 0 & 0 & 0 & 0 \\ \frac{\partial G}{\partial P}\Delta t & I + \frac{1}{2}\frac{\partial G}{\partial P}\Delta t & -[\Sigma C_B^N \Sigma \Delta \vec{V}_B] \times & \Delta t \Sigma C_B^N & 0 & 0 & 0 \\ 0 & 0 & 0 & 0 & \Delta t \Sigma C_B^N & 0 & 0 \\ 0 & 0 & 0 & I - \frac{\Delta t}{\tau_{ba}} & 0 & 0 & 0 \\ 0 & 0 & 0 & 0 & I - \frac{\Delta t}{\tau_{bg}} & 0 & 0 \\ 0 & 0 & 0 & 0 & 0 & 1 & \Delta t \\ 0 & 0 & 0 & 0 & 0 & 0 & 1 \end{bmatrix} \quad (12)$$

In the current implementation, the process noise covariance, Q, is static and diagonal. The process noise represents the expected noise and unmodeled error in the reference model. The reference model is the SDINS computed state and thus is primarily a function of inertial instrument errors. The position component is non-zero, but notionally small. The process noise covariance associated with the velocity error state is defined based upon expected RINU accelerometer output noise and velocity random walk. The components associated with attitude error are a function of angular random walk and quantization noise. The gyroscope and accelerometer bias process noise covariance is based mostly on the expected instability in those instrument error terms. Without a mature GPS receiver design, the process noise covariance associated with the receiver clock states are based on assumptions made in GPS receiver clock modeling.

The initial state covariance is diagonal and defined based upon the well characterized uncertainty in the initial state at launch. Just prior to filter execution, the initial SDINS state is reset to the expected RINU position. The initial position variance is defined based upon the expected uncertainty in the position of the launcher plus the uncertainty in the mounting of the RINU. The initial velocity state is computed from the initial position; the variance is mostly defined from the uncertainty in the velocity state due to initial position uncertainty. The attitude variance is a function of expected gyrocompassing performance. The instrument terms are derived from RINU instrument capability estimates. Currently, the receiver clock variance terms are defined to be approximately consistent with GPS modeling but will be refined as the receiver design information becomes available and the model matures.

As previously stated, the navigation filter uses GPS pseudorange and deltarange measurements. The filter measurement, or filter innovation, is an error between the state observation and the state estimate. The general measurement model is defined in Eq. (13).

$$\hat{Z} = h(\hat{X}) + n(t) \quad (13)$$

The continuous function h relates the state to the observations and n is the noise on the estimated observation.

The measurement innovation, Eq. (14), includes the observation from the GPS receiver and estimate of that observation with corrections. The filter innovation can be written as a function of the trajectory reference model Eq. (15).²¹ Note that the hat notation denotes an estimate and the tilde notation denotes that the quantity is observed. Subscript N denotes the navigation frame, a denotes the assumed antenna location, and SV denotes satellite in view.

$$y = \tilde{Z} - h(X) = \tilde{m}_{GPS} - (\hat{m}_{SDINS} + \delta \hat{m}_{SDINS}) = \begin{bmatrix} \tilde{\rho}_{CH} \\ \tilde{\rho}_{CH} \end{bmatrix}_{GPS}^{1..N_{CH}} - \begin{bmatrix} \hat{\rho}_{CH} \\ \hat{\rho}_{CH} \end{bmatrix}_{GPS}^{1..N_{CH}} \quad (14)$$

$$\begin{bmatrix} \hat{\rho}_{CH} \\ \hat{\rho}_{CH} \end{bmatrix}_{GPS}^{1..N_{CH}} = \begin{bmatrix} \sqrt{[\vec{r}_{N,SVCH}^N(\tilde{t}) - \hat{r}_a^{N-}]^T [\vec{r}_{N,SVCH}^N(\tilde{t}) - \hat{r}_a^{N-}] + \delta \rho_{RC}^-} \\ [\vec{u}_{SVCH}^N]^T [\vec{v}_{N,SVCH}^N - \hat{v}_a^{N-}] + \delta \hat{\rho}_{RC}^- \end{bmatrix}^{1..N_{CH}} \quad (15)$$

The measurement matrix linearly relates the observation to the state. We assume a function which linearly relates the states to observations at time t_k . $h(x(t_k)) = H_k x_x$, leads to $H_k \triangleq \frac{\partial h}{\partial X}$.

For this implementation, a number of simplifications are made in H . The partial derivatives of pseudorange with respect to attitude error is neglected. The partial derivatives of range rate with respect to position, attitude, and gyro bias are also neglected. This is due to a weak direct relationship between the measurements and the other states, e.g. it is desired to estimate a correction to attitude, but it is preferred that the correction be primarily derived from the stronger relationship to the change in the velocity state through Φ .²¹ The partial derivatives with respect to clock error states are simplified. The resultant H matrix is shown in Eq. (16) and is only a function of the unit vector from the estimated GPS antenna position to the SV position relative to the navigation coordinate frame. A future version will likely include the radial component of the lever arm between the RINU mount reference frame and the plane containing the antenna boresights.

$$H_k \triangleq \frac{\partial h}{\partial X} \cong \begin{bmatrix} [\vec{u}_{a,SV_i}^N]^T & [0,0,0] & [0,0,0] & [0,0,0] & [0,0,0] & 1 & 0 \\ [0,0,0] & [\vec{u}_{a,SV_i}^N]^T & [0,0,0] & [0,0,0] & [0,0,0] & 0 & 1 \end{bmatrix}_{SV_i} \quad (16)$$

The measurement noise covariance matrix, R , is diagonal and contains pseudorange and pseudorange rate variance. Currently, the form and statistics are defined to be approximately consistent with GPS modeling. Eventually the composition of the R matrix will be redefined based upon vendor provided information and test validated.

Navigation Execution

As stated in the avionics section above, the RINU and the GPS receivers operate asynchronously in time with respect to one another. This results in different epochs for discrete samples between the devices. Further, the GPS measurements are expected to be considerably more latent than the inertial measurements from the RINU due to a larger amount of inherent latency in the receiver processing and a significantly lower sampling rate on the SLS FCs. This poses an implementation issue for the EKF given that the filter assumes consistency in time between the non-linear reference

model, the measurement innovation and components, and the state. This leads to the subdivision of the filter code to an executive and a wrapper within the navigation filter.

The EKF executive contains the core filter implementation in code. The EKF executive is modularized and operates on a filter-specific data structure that contains the working memory required for the filter including the input, the retained P, Q, and R matrices, and a timestamp for when they were last valid. The receiver clock states are also maintained. The input from the GPS measurement processing, through the EKF wrapper, currently consists of the GPS measurement epoch, a set of compensated pseudorange and range rate measurements, the associated ECEF relative satellite position vectors computed from GPS ephemeris data, and a new/valid measurement flag. This data set may be expanded as the GPS receiver design and filter measurement noise model matures. The inertial input to the filter consists of a position, velocity, and attitude state and the components of the state transition matrix which must be computed at a higher rate, e.g. total delta-velocity. The inertial data is assumed by the executive to be valid at the GPS measurement epoch. The primary output of the function are the estimated states. The EKF executive may also be called without a new/valid measurement to propagate the filter covariance.

Given the feedback design, the EKF wrapper is executed at 50Hz after the SDINS algorithms have propagated through all of the available corrected inertial measurement data. Given that the RINU and FCs are asynchronous, this may be 10-40 milliseconds forward. The EKF wrapper has the responsibility of populating the EKF executive input and managing the output to affect the SDINS state. Upon execution, the EKF wrapper will store the inertial measurements and SDINS state data into a buffer and ensure that the filter covariance is current within the buffer. The buffer is sized to be approximately twice the expected GPS receiver measurement latency. If a new measurement is available, the EKF wrapper will check whether the measurement epoch is within the time frame of the buffered inertial and use the closest discrete timepoints to interpolate the state to the GPS measurement epoch. Linear and Spherical Linear (SLERP) interpolation is used to approximate to the exact measurement epoch. The EKF Executive is then called with the appropriate data for state estimates at the GPS measurement epoch. If the EKF Executive returns with an estimated correction, the estimated states are propagated to the nearest discrete timepoint within the buffered inertial navigation data, underweighted if necessary. A new instance of the SDINS function is then iteratively called to propagate the latent corrected SDINS state forward with corrected inertial measurement, effectively re-navigating through the buffered data to the current SDINS epoch. During this process the previously buffered state data is overwritten with quantities propagated from the corrected data. The primary SDINS structure is then overwritten with the re-navigated data for the next cycle. Rotating buffers and indexing are used for efficiency.

ALGORITHM ROBUSTNESS AND FAULT MANAGEMENT

The SLS Block 1B Navigation system is designed for a high level of reliability and robustness that extends beyond the redundancy in the hardware to the algorithms. The approach chosen for adding robustness to the algorithms is to protect critical quantities and interfaces within the GN&C system. Table 1 defines the protected quantities and names the algorithms used for fault mitigation and robustness. For estimates of angular rate, acceleration, and navigation quality ΔV and $\Delta \theta$ derived from RINU measurements, the RINU FDIR algorithm is leveraged - offering a high degree of low level protection. The RINU FDIR configuration is mission specific and developed to maximize mission success and minimize loss of crew probabilities.⁹ The navigation Sensor Data Quality (SDQ) and downselection algorithms are re-used from Block 1 and described by Park et. al.²² The state derivatives, not used by inertial navigation, are additionally and appropriately anti-aliased with linear filters for their intended use. The Navigation FDIR algorithm consists of rotating par-

allel filters similar to both the algorithms described by Da and that used for Space Shuttle rendezvous. The Navigation FDIR algorithm offers the ability to check the filter state and re-initialize if necessary.²³ Innovation editing is a χ^2 test on the matrix ratio of the filter innovation and the innovation covariance and used to gate filter measurements.²⁴ Underweighting will be employed but remains under development at the current time.

Table 1. Navigation algorithms for Fault detection, mitigation, and robustness.

Protected Quantity		Algorithm
Output for Controls Rate Blending	Angular Rate (RINU)	RINU FDIR + Mission specific configuration
	Angular Rate (RGA)	Navigation rate SDQ and downselection
Specific Force for G & C	Filtered Delta-Velocity (RINU)	RINU FDIR + Mission specific configuration
Inertial Measurements to SDINS	Compensated DV for SDINS	RINU FDIR + Mission specific configuration
	Compensated DTheta for SDINS	RINU FDIR + Mission specific configuration
EKF Input	GPS $[\rho, \dot{\rho}]$ for EKF innovation	GPS Sensor data quality
	SDINS data for EKF innovation	Navigation State FDIR
	EKF Innovation	Measurement Editing
EKF Output	State Correction/estimate	Measurement Underweighting
Aided Inertial Navigation State	Navigation State	State Navigation FDIR
	Navigation EKF Covariance	State Navigation FDIR

CONCLUSIONS AND FORWARD WORK

This paper discusses the preliminary Navigation system architecture and design for the SLS Block 1B vehicle. The change in architectures from Block 1 to Block 1B represents a significant effort. The navigation hardware is in various states of maturity, most of which would be considered flight ready due to carry-over from Block 1. The GPS receiver is expected to pass a Preliminary Design Review this year. At the time of publication, SLS GN&C and Vehicle Management is in the midst of the second DAC. Multiple trades are scheduled for the current DAC which will affect implementation and filter states. The source code for the software is scheduled for initial delivery to FSW in June of this year. Changes in implementation may result from their review, testing, and profiling of the code. The flight software is in the early stages of development but rests on a very mature architecture. Captive carry and other HWIL testing is in the process of being scoped out but will likely not occur in 2018. Once fully realized, the SLS Block 1B Navigation system is expected to robustly meet mission requirements with a high level of reliability.

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