Hybrid Navigation System for Spaceplanes, Launch and Re-Entry Vehicles

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In 2010 the German Aerospace Center (DLR) will conduct the second experiment of the DLR hypersonic SHarp Edge Flight EXperiment Program SHEFEX-2. The purpose is to investigate possible new shapes for future launcher or re-entry vehicles with faceted surfaces and sharp edges and to demonstrate key technologies for re-entry like hypersonic flight control using steerable canard fins.

Accurate control of the vehicle using the canards requires a highly accurate knowledge of the angle of attack and the side slip angle. Both angles can only be derived from the flight path and an attitude measurement. The first can be achieved using GPS measurements. The second can be provided only by the most accurate Inertial Navigation Systems (INS) because drifts due to launch vibrations exceed the accuracy requirements. Therefore, a star tracker will be used to update the attitude information shortly before entry.

The SHEFEX-2 mission describes an entry scenario which is applicable to other entry missions. There is a general need to develop a high accuracy integrated navigation system which can be used for multiple missions. This navigation system should combine the measurements from an inertial measurement unit (IMU), GPS receiver and star tracker with the option to include additional sensors.

This paper will describe the concept of the integrated navigation system with a focus on integrating the star tracker into the SHEFEX-2 experiment.

I. INTRODUCTION

One key technology for reusable space transport vehicles and hypersonic aircraft is autonomous navigation and (aerodynamic) control during ascent, entry, final approach and landing.

The SHEFEX-1 mission demonstrated that sounding rocket systems are suitable to perform entry related flight experiments. SHEFEX-1[1] used a 'passive' re-entry configuration and was stabilized with a conic tail and fins. The SHEFEX-2[2] payload stabilization will be provided by autonomous navigation and an active aerodynamic control system.

This paper will discuss the autonomous navigation system on board the SHEFEX-2 flight. It is one step in the development of navigation systems for future space transportation systems.

A. SHEFEX-2 Payload and Launch Vehicle

SHEFEX-2 uses a Brazilian VS-40 launch vehicle reconfigured for the experiment. It is an unguided solid propellant sounding rocket, consisting of an S-40 motor as the first stage and an S-44 motor as the second stage.

The goal of SHEFEX-2 is to investigate technologies for hypersonic and space transportation systems like aerothermodynamic control during hypersonic flight. The shape of the test vehicle was chosen to create a symmetric re-entry body stabilized by tail fins and 4 movable small canards near the front of the cylindrical payload segments (see figure 1). Within the cylindrical segments, all necessary subsystems are integrated,

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including: navigation platform, power cells, RCS- unit, data acquisition, parachute and recovery system, telemetry, etc.

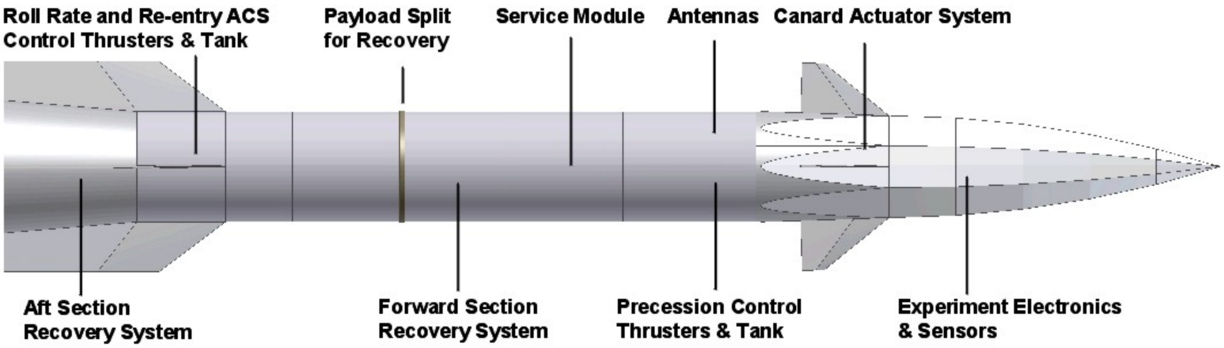


Figure 1. SHEFEX-2 payload in entry configuration after separation from second stage.

The payload is mounted on top of the second stage and will be separated before entry to begin an autonomous flight until the final breaking maneuver and parachute deployment.

During the experiment phase, a maximum entry velocity of Mach 10 to 12 is expected for 60 seconds. Considering these flight conditions, the dynamics and speed are not fully representative for an RLV entry. However, it is close enough to test the autonomous navigation system for ascent and entry.

B. Mission Scenario

The launch site will be in Woomera, Australia. Preliminary trajectory calculations show an expected down range in the order of 830 km. The impact area is expected to be around 830 km northwest of the launch site. The flight will last about 520 seconds with several flight phases (figure 2).

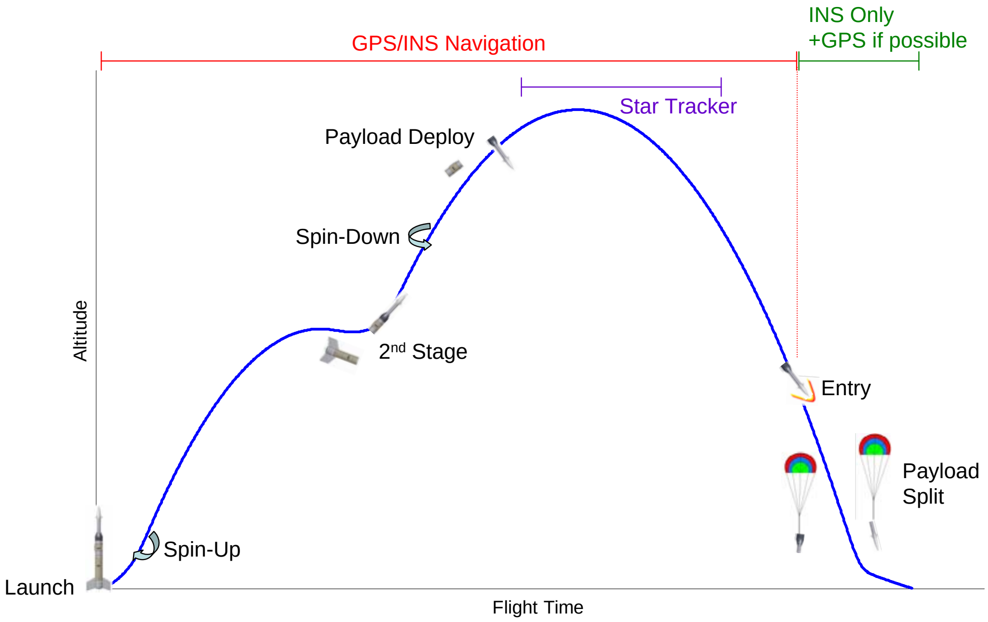


Figure 2. Flight profile of SHEFEX-2 with operational periods of the navigation system sensors.

Soon after launch, the vehicle will spin-up for stabilization using tilted tail fins. After burn-out and separation of the second stage motor, the vehicle is de-spun using a yo-yo system. During the following coast phase, the star tracker lid will be opened and the star tracker will operate. This is followed by the entry phase when GPS signal black-outs are expected. Finally, the payload splits in two and both parts land using parachutes. Both parts (forward and aft section) of the payload will be recovered manually.

II. HYBRID NAVIGATION SYSTEM

A. Motivation

The launcher will be unguided during its propelled flight phases. In between the two stages, an attitude precession maneuver is done using a reaction control system (RCS). The second stage will then be fired in a horizontal direction to create a more horizontal flight path. After separation from the second stage, the payload is spun down using a yo-yo system and the RCS. The RCS is then used for three-axis attitude control. For all flight phases an INS is used as the only attitude measurement.

Launcher is also equipped with a GPS receiver for trajectory monitoring. This GPS is not coupled with the IMU and works independently. It is not used to control the launcher, but is needed for the autonomous flight control experiment. One objective of this experiment is to demonstrate attitude control technology using ceramic fins. Additionally, the flight control system will damp out vehicle oscillations and allowing stable aerodynamic conditions for the passenger experiments.

Some of the passenger experiments require that all attitude oscillations be smaller than 1 degree. To evaluate the experiments' results, the angle of attack (AoA) and angle of side slip (AoS) must be known on the order of 0.25 degrees. Both angles will will be derived from the flight path and attitude. Error propagation results show that an attitude measurement of 0.17 degrees is needed to achieve this accuracy. The flight path angle can easily be determined with the same accuracy from GPS measurements.

Attitude measurements can be provided by the INS, but the accuracy of this system is limited. The gyro compass alignment of the INS before launch can only achieve an accuracy of 0.3 degrees. Additionally, during the propelled flight phases, high vibrations cause the INS to drift an additional 2 to 3 degrees. Therefore, the attitude error after the separation of the payload is one order of magnitude too large.

B. Solution

To achieve the attitude accuracy needed for the flight control experiment a Hybrid Navigation System (HNS) will use measurements from an inertial measurement unit (IMU), a GPS receiver and a star tracker (STR). Apart from providing better attitude accuracy for the control system, the HNS is a technology experiment to test the capabilities of a breadboard integrated navigation system which couples the IMU and GPS. The STR will be added to this coupled system as an additional sensor. It shows the capability to add other sensors to the HNS for other missions. The following sections describe the experimental HNS. The system design, components, current status and tests are described in more detail.

C. Concept

The core of the HNS system couples measurements from the IMU and GPS. Development will be done in three stages. First, a loosely coupled hybridization[5] will be developed where IMU acceleration and rotation measurements and GPS position and velocity measurements are combined together with a Kalman filter. The second step uses a tightly-coupled hybridization using raw GPS measurements (e.g. Pseudo ranges and Doppler measurements). The third step implements GPS receiver aiding to help the GPS re-acquire the tracking loops. An ultra-tight coupling which feeds back accelerations measurements to the GPS tracking loops will not be attempted.

Figure 3 shows the HNS block diagram. IMU measurements are output at 500 Hz to a strapdown navigation and Kalman filter routine running at the same rate. GPS measurements are received at 1 Hz and STR measurements at 2 Hz. The GPS receiver is aided to support signal acquisition with position and velocity information at 1Hz.

An emphasize is laid on synchronizing measurements to minimize errors when combining data with various reference times. All sensor measurements have a corresponding trigger which indicates the reference time of their data. The GPS provides a pulse per second (PPS) signal which is synchronized with GPS time. The STR and IMU are externally triggered by the navigation computer. All triggers are time stamped by the HNS clock to accurately measure each measurement's reference time.

D. Components

To select the HNS hardware some specific conditions of the SHEFEX-2 mission were considered. First, the rocket spin rate is expected to be up to 2.5 Hz. Secondly, the maximum linear acceleration during launch

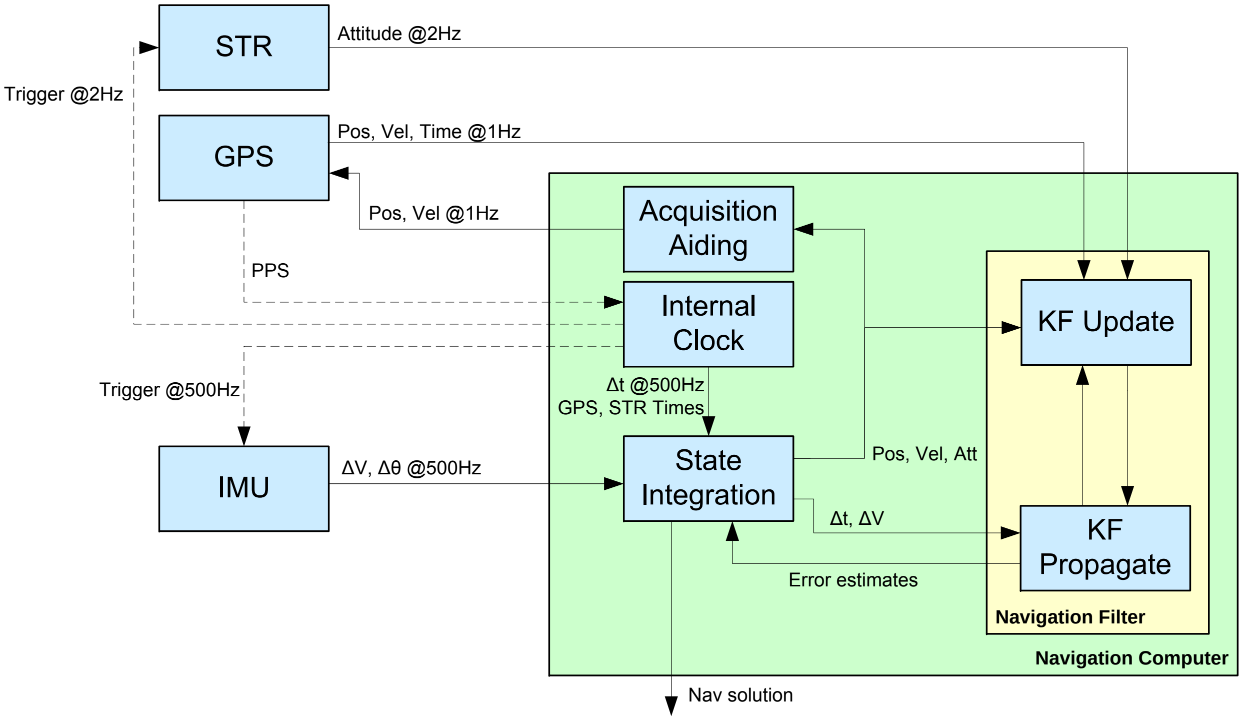


Figure 3. HNS system concept using IMU, GPS and STR sensors.

is about 8.1 g and but is much higher at entry when the payload is split. The atmospheric drag of the separate payload pieces is expected to be in the order of 80 g. Vibration loads during launch also have to be considered.

The thermal environment for the equipment is relaxed. The IMU and electronic components will be located in a water and air-tight section allowing convectional thermal regulation. Furthermore, the total flight time is less than 10 minutes, so the equipment does not need to be space qualified in terms of robustness against radiation events.

Figure 4 shows the configuration of the components on the bulkhead of the service module (as of July 2009). All components seen on top of the circular bulkhead plate will be inside the air-tight section of the service module. The camera, baffle and the shutter release mechanism are outside this section. The camera is mounted on the bulkhead to ensure that the alignment between star tracker camera and IMU can be measured and fixed before integration into the SHEFEX-2 payload.

1. iMAR iIMU-FCIA-E-03 IMU

The IMU provides the only attitude information for most of the flight so the gyro accuracy and bias drift requirements are relatively high. First analyses have shown that a bias drift below 1 deg/h is sufficient. Additionally, due to the spin rate of the launcher, the measurement range of the IMU must cover 2.5 revolutions per second plus 10% margin. So a measurement range of at least 1000 deg/s is needed. A high sampling rate is also needed for high accuracy strapdown navigation while spinning because the IMU is offset from the vehicle spin axis. To minimize navigation errors the sample rate should be as high as possible. An ambitious sampling rate of 500 Hz will be used, which gives a sculling angle of 1.8 deg during the maximum spin rate. To summarize, the IMU must have the following characteristics:

	Gyroscopes	Accelerometer
Range	±1000 deg/s	±20 g
Bias	1 deg/h	2 mg
Data rate	500 Hz (triggered)	

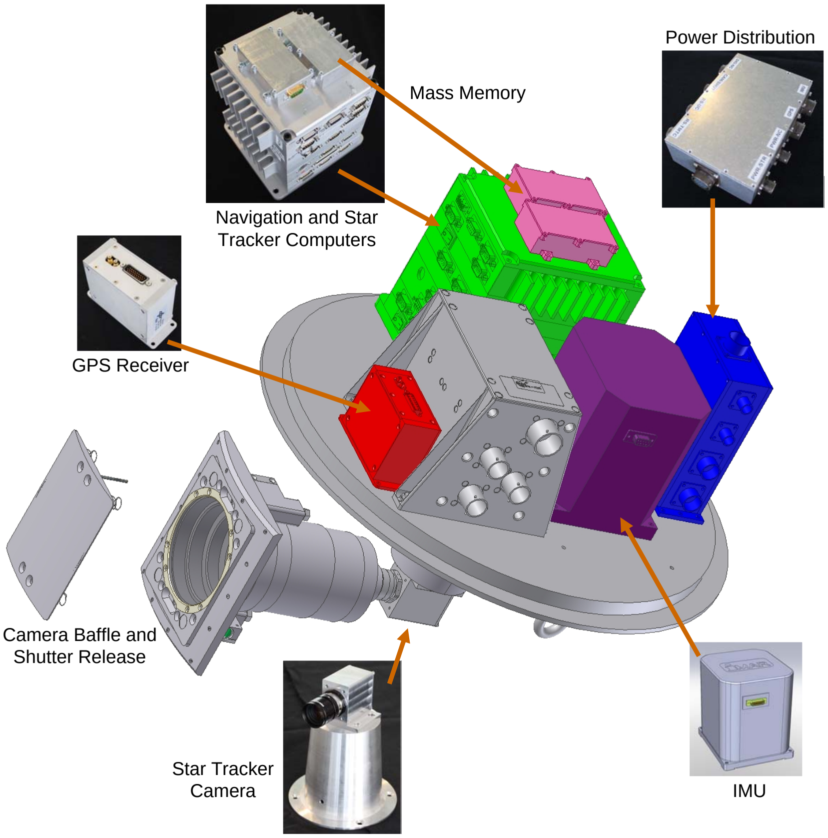


Figure 4. Component Breakdown of the HNS as of July 2009.

Based on these criteria, the iMAR iIMU-FCIA-E-03 was selected. It is a customized small size, low noise IMU. Developed by iMAR, it is composed of an off the shelf IMU with changes to meet the gyro measurement range and sampling rate requirements.

2. Phoenix-HD GPS Receiver

The Phoenix receiver is a spaceborne GPS tracking device specifically designed for navigation of small LEO satellites and a variety of other space vehicles, such as sounding rockets.[6] The receiver represents a low-cost single-board GPS navigation sensor for L1 C/A code and carrier tracking of up to 12 satellites in parallel. The Phoenix receiver combines commercial-off-the-shelf technology with DLR's proprietary firmware, offering precision measurements for advanced navigation applications and robust tracking under extreme dynamics.

A Phoenix receiver based navigation system has already been used on the SHEFEX-1 mission[7] with a dedicated antenna system designed for sounding rockets. This system contained an antenna mounted in the tip of the rocket for signal tracking during the propelled flight phase, which is not feasible for SHEFEX-2 because of the vehicle requirements for the other experiments on board. A traditional wrap-around antenna will be used instead.

E. Star Tracker

Instead of a commercial star tracker, DLR chose to design its own instrument based on off-the-shelf components. This choice allowed DLR to develop the star sensor to suit the mission and gain experience on STR development.[8]

A star tracker is needed in the HNS experiment to compensate attitude errors of the IMU just before entry. A robust and useful utilization of the star tracker in the HNS will be ensured by the following requirements:

- Attitude measurement for the lost in space case (no a priori information)
- Attitude error less than 0.1 deg in three axes (worst case scenario)
- Ability to measure attitude during angular velocities up to 2 deg/s
- Total processing time less than 500 ms (2 Hz) per measurement
- Robustness against unfavorable situations

One principle driving the STR development (especially the related software) is that it is better to generate an error message than to deliver a wrong attitude information. This principle is a major criteria for robust attitude estimation.

A special condition of the flight experiment is that, unlike stable pointing satellites, the angular rates of the sounding rocket are expected to be on the order of 2 deg/s during the star tracker operating time in the coast phase. This time frame starts shortly after spin down and ends just before entry, when the vehicle acquires the proper attitude for entry.

The availability of the star tracker is also an important issue. The influencing factors are the exclusion angles between the star camera's optical axis and the sun, and between the optical axis and the horizon vector in the star tracker frame. Therefore, the location and orientation of star tracker camera in the rocket must be chosen carefully. The requirements regarding these angles are:

- Sun exclusion angle: 90 deg
- Earth exclusion angle: 50 deg

1. Star Tracker Hardware

Hardware used for the STR include an off-the-shelf camera with a corresponding lens and a PC 104 computer for processing. All equipment has been demonstrated in a high load environment. Integration of the camera together with mounting and baffle hardware is shown in figure 5.

For this experiment, an off-the-shelf camera giving the best images at a reasonable price was needed. It was clear that only sensors with limited sensitivity could be chosen. This leads to a reduced number of stars that can be detected when respecting a required exposure time of 50 ms or less. The need arose to select a camera with a wide field of view, in order to be able to detect enough stars in one image. The wanted limitation of image smear is responsible for the short required exposure times, while still having a maximum signal-to-noise ratio (SNR).

After testing several cameras, the monochromatic CCD camera (model: PROSILICA EC655) was chosen. The lens is 25 mm with manually adjustable focus and brightness. The field of view is 14.87 deg x 11.15 deg, which results in a diagonal FOV of 18.57 deg.

The last component is the baffle, a standard in star tracker hardware. The shape of the baffle results from the exclusion angle requirements. The baffle was designed specifically for SHEFEX-2 to protect the camera from bright sources. An automatic shutter lid was also included, which will open using a cable cutter mechanism before operating the star tracker.

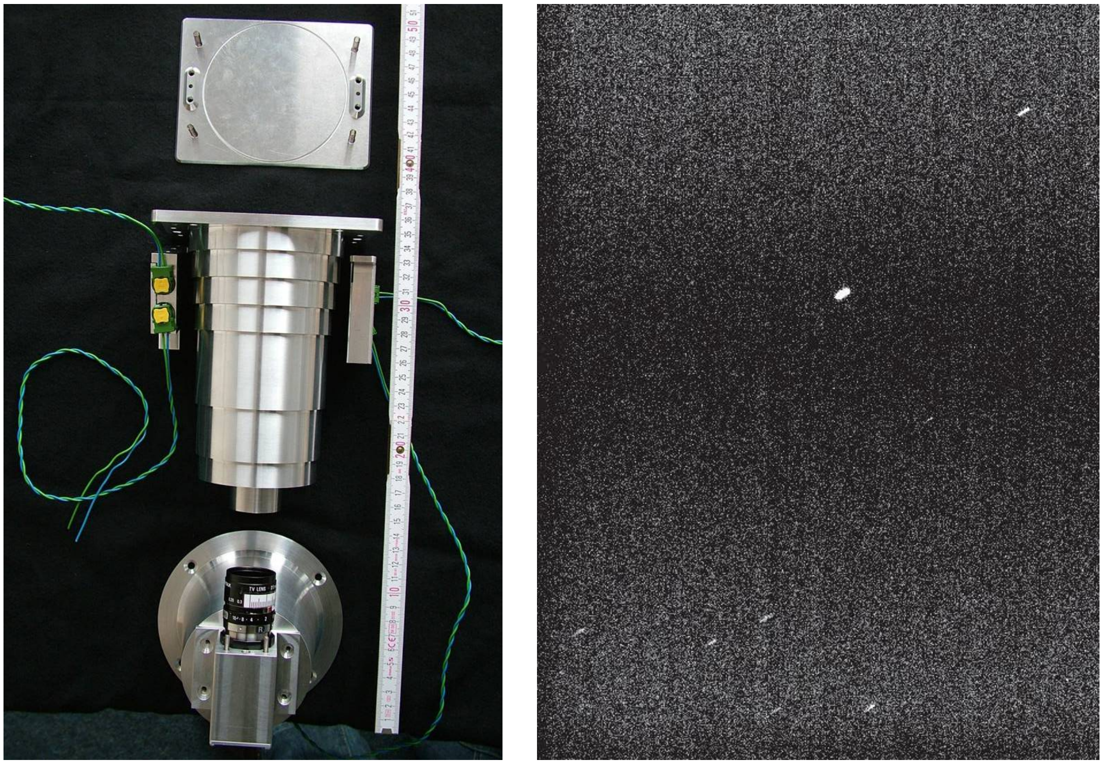
2. Algorithm Development

The attitude estimation processing chain consists of camera control, image processing, star identification and attitude estimation.

The main task of the camera control is to adjust the camera settings during operation. From the many parameters that can be controlled via the camera interface, only the exposure time and the camera gain is needed in a star tracker application.

After using dark image subtraction to avoid bad camera pixels, a centroiding algorithm is used to find the star centroids within the image.[8]

Finally, the star identification algorithm finds the indecies of the cataloged stars which correspond to the imaged stars using a robust and fast technique. Because of the small number of expected imaged stars in the FOV, it was decided to use a geometrical algorithm with a database of inter-star angles. The angles are chosen so that only those smaller than the maximum FOV size are stored. After two sets of imaged and cataloged stars are found, an attitude estimation algorithm based on Wahba's problem is used to find the attitude with respect to inertial.[8]



(a) (b)

Figure 5. (a) The Star Tracker Hardware as of September 2009; (b) Smeared Night Sky Image

3. Star Tracker Night Sky Tests

A night sky test was used to test the star tracker. 24 images were analyzed showing different orientations and different amounts of image smear. They were processed using the uniform Hipparcos catalog. The image processing algorithm was used for all the analyzed images. The results show that the star tracker can reach the required attitude accuracy during a 2 deg/sec rotation and tighter accuracy for slower angular rates.

Additionally, no incorrect star recognition and no "no stars found" cases occured. Many images were identified and processed within the required processing speed. Some analyzed images show significant image smear (for example, the image in figure 5, where a histogram equalization was already used) but these correspond to angular rates of 4 to 5 deg/s, which exceed the expected conditions.[9]

F. Navigation Flight Software

The navigation flight software is composed of a strapdown navigation algorithm and a separate Kalman filter. The strapdown algorithm integrates compensated IMU outputs at a high rate (500Hz) to obtain the vehicle state: position, velocity and attitude. The Kalman filter combines the sensor measurements and vehicle state to estimate the error of the vehicle state and sensor compensation terms.

The strapdown method integrates compensated delta attitude and delta velocity to calculate the vehicle state. The navigation routine is initialized on the launch pad with the following expected accuracy: 10m position, 10deg attitude. IMU data is first compensated by correcting for accelerometer and gyro bias and scale factor terms. The quaternion from the vehicle body to inertial frame is integrated using McKern's 3rd order routine.[10] After rotation corrections, delta velocity and a gravity correction are summed to calculate current inertial velocity. Gravity is calculated at the estimated center of the integration interval using a 9 order 9 degree EGM96 model. Velocity is integrated using trapezoidal integration to calculate inertial position.

An extended Kalman filter combines the measurements from the IMU, GPS and STR. The filter estimates the navigation error states (indirect feedback) and the state transition model is linearized. State estimates are fed back to the strapdown routine at regular intervals. Along with GPS and STR updates, the filter will also use zero Earth-relative motion updates when on the launch pad. Additionally, UDU covariance factorization[11] and covariance scaling (to obtain a good matrix condition number) are used for numerical stability. The following variables are estimated by the filter:

- inertial position error
- inertial velocity error
- attitude error with respect to inertial
- accelerometer bias error and scale factor error
- gyro bias error and scale factor error
- GPS Earth-fixed position offset error

Other errors are present in the system (gravity model, alignment, non-linearity, etc.), but they are not included in the filter because they have a small effect or are not very observable.

To reduce the computational load on the computer, the flight software is broken up into high rate (HR) and low rate (LR) components. The HR process runs the strapdown routine and continually propagates the filter with the IMU data. Kalman filter updates are processed in the LR process using GPS, STR and zero-motion information. The filter states are fed back to the HR process to correct the navigation states.

III. SIMULATION, TEST AND VERIFICATION

To test the hybrid navigation system, the flight software and hardware will be subject to numerous tests in several environments. All flight sensors (GPS, IMU, STR) are included in the tests and are designed to find bugs, measure system performance and stress the system.

A. Software in the Loop

For software in the loop (SWIL) testing, the flight software is embedded into a Matlab/Simulink simulation with high fidelity models for each instrument of the hybrid navigation system. The vehicle model calculates the true dynamics and navigation state of the vehicle while simulating the expected flight profile. The IMU, GPS and STR models take the true data from the vehicle model and corrupt it with instrument errors to get realistic outputs for each instrument. The instrument outputs are sent to the flight software, which is wrapped in a Simulink s-function.

Generic IMU and STR models are used with parameters for the hybrid navigation system's specific IMU and STR in order to closely model the real hardware. The GPS model was developed specifically to emulate the Phoenix GPS receiver and care was taken to simulate the functionality of the receiver as close to reality as possible.

SWIL Simulation Results

The strapdown navigation and filtering algorithms described earlier were tested using the SWIL simulation. A Monte Carlo of 100 runs was completed with random values for IMU, GPS, STR and initialization errors. The simulation contained an additional time of 60sec on the launch pad before the launch. The simulated flight dynamics used for this simulation are the latest values as of July 2009. Later versions of the flight dynamics are available, but navigation performance is expected to be similar. The resulting position, velocity and attitude performance is shown in Figure 6 along with the filter's 1σ and 3σ covariance. Based on these results, we expect that the system performance at the end of the experiment phase will be an accuracy of 1 m position, 0.01 m/s velocity and 0.1 deg attitude.

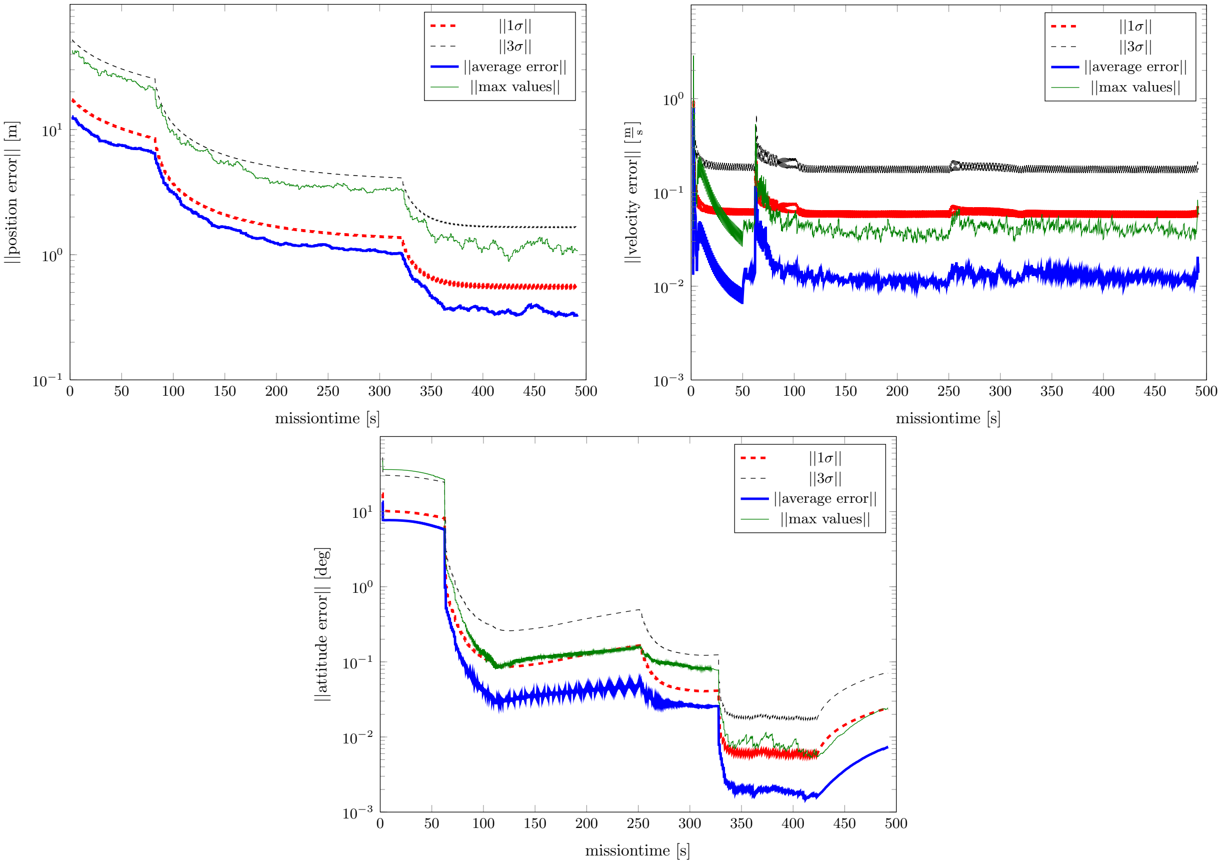


Figure 6. Position, velocity and attitude accuracy during SWIL simulation. The plot shows the average and maximum errors over a Monte Carlo of 100 runs, as well as the combined 1σ and 3σ filter covariance values.

B. Hardware in the Loop

Hardware in the loop (HWIL) tests the flight hardware under simulated flight and laboratory conditions. Testing is done in three stages, with a dSpace real time simulator controlling the simulation using the SWIL models. In the first HWIL setup, all of the navigation computer's instrument interfaces are simulated (Figure 7). This setup will verify the navigation software, its implementation on the computer and its ground station and CCC interfaces. The second HWIL setup replaces the simulated GPS instrument with a Spirent GPS signal generator and a GPS receiver, which will verify the GPS instrument and interface. The third HWIL setup additionally replaces the simulated IMU instrument with a 3 axis rotation table and IMU (Figure 8) and uses a simplified test scenario. This will verify the IMU instrument and interface with the expected flight rotations, but with laboratory accelerations.

IV. CONCLUSION AND OUTLOOK

This paper presented the development status of the Hybrid Navigation System experiment for the second SHarp Edge Flight EXperiment (SHEFEX-2). The concept, hardware selection, algorithms and software- and hardware-in-the-loop-tests of this system were discussed.

SHEFEX-2 is planned to launch in autumn 2010. Until then, a lot of development steps have to be completed. The flight hardware will be complete at the end of October. This will be followed by interface tests within the experiment and to other vehicle subsystems. Flight hardware delivery is expected at the end of 2009/beginning of 2010. Software tests will continue using an engineering model, which is a close copy of the flight system.

The flight experiment is expected to confirm the estimated performance of the navigation system and

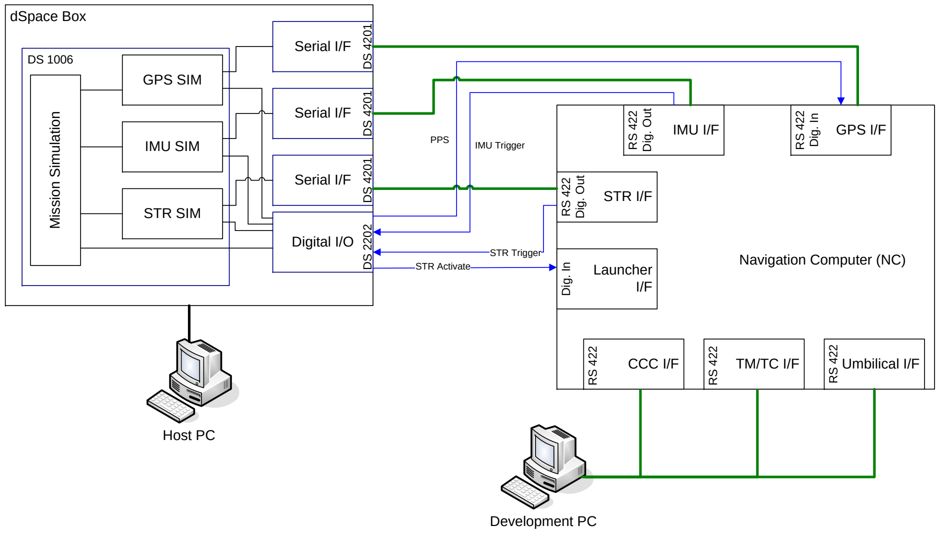


Figure 7. HWIL setup for the navigation computer alone.

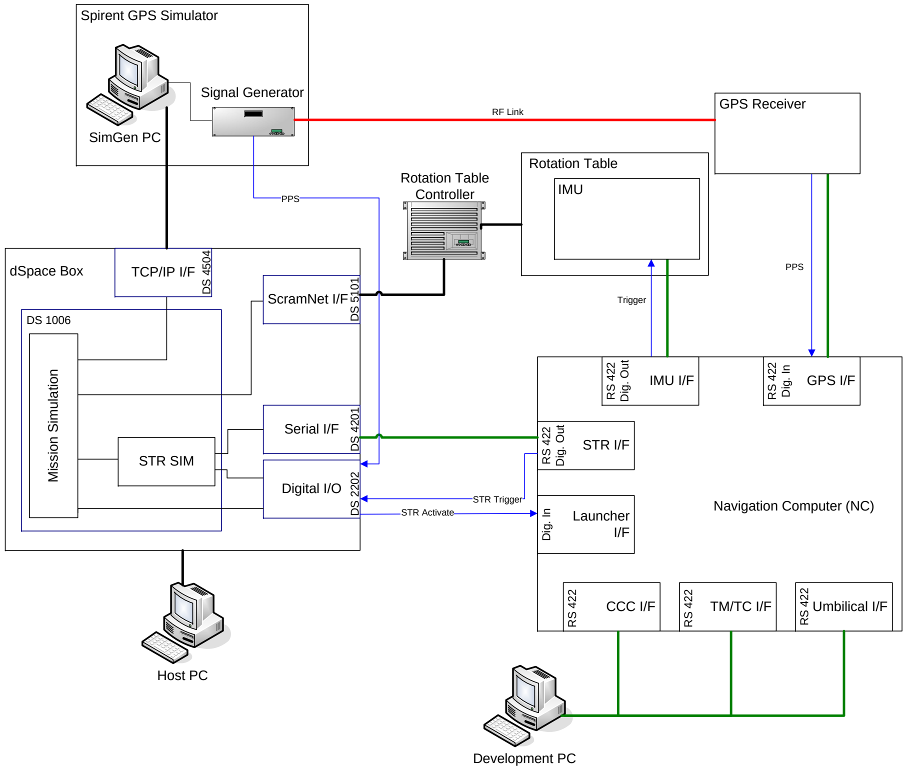


Figure 8. HWIL setup for the main hybrid navigation system components.

deliver flight data to be used for further algorithm and software development. One goal is to develop an autonomous navigation system for hypersonic flight systems, sounding rockets, space transportation and re-entry systems. The next step is to develop dedicated space qualified hardware which will be tested on suborbital flights or entry missions.

V. ACKNOWLEDGEMENTS

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