GPS-Based Real-Time Navigation for the PRISMA Formation Flying Mission

S. D'Amico[(1)], E. Gill[(1)], M. Garcia[(1)], O. Montenbruck[(1)]

[(1)]*Deutsches Zentrum für Luft- und Raumfahrt (DLR), German Space Operations Center (GSOC)*
82234 Wessling, Germany
Email: simone.damico@dlr.de

ABSTRACT

The paper addresses the design, implementation and validation of the on-board navigation system for the PRISMA technology demonstration mission. The objective of the navigation system is to provide in real-time absolute and relative orbit information for the PRISMA space segment which consists of two satellites flying in formation in Low Earth Orbit. The key drivers for the design of the navigation system are the accuracy requirements on the absolute and relative orbit determination which amount to 2 m and 0.1 m respectively (3D, r.m.s.). Furthermore, a high level of robustness and flexibility imposed by the numerous formation flying scenarios is required during the mission lifetime of about eight months. The paper focuses on the description of the navigation software architecture and algorithms. In contrast to earlier approaches that typically separate the GPS-based navigation task into the independent reconstruction of absolute and relative states, here a single reduced-dynamic Kalman filter has been developed which processes pseudorange and carrier-phase data from both spacecraft in order to exploit the full GPS measurements information at all times. Emphasis is given to the validation of the software via real-world simulations. The flight application is executed on a LEON2 processor in order to evaluate the orbit determination performance in real-time using a representative PRISMA flight hardware.

INTRODUCTION

PRISMA comprises a fully maneuverable micro-satellite (MAIN) as well as a smaller sub-satellite (TARGET) that will be released from MAIN after initial commissioning. The mission schedule foresees a launch in 2009 of the two spacecraft into a low Earth orbit (LEO) with a targeted lifetime of at least eight months [1].

The PRISMA mission objective is to demonstrate in-flight technology experiments related to autonomous formation flying, homing and rendezvous scenarios, precision close range 3D proximity operations, soft and smooth final approach and recede maneuvers, as well as to test instruments and unit developments related to formation flying.

In four major areas DLR/GSOC provides contributions to the PRISMA mission. These comprise the GPS system for both spacecraft, a GPS-based navigation system to support formation flying during all phases, dedicated experiments for relative and absolute orbit control as well as an on-ground automated Precise Orbit Determination (POD) for off-line verification purposes.

One of the main challenges of the PRISMA formation flying is the realization of an on-board navigation system for all mission phases which is robust and accurate even for various spacecraft orientations and frequent thruster firing for orbit control. The requirements for the GPS-based PRISMA real-time navigation software are outlined in [2] and represent the key drivers for the design of the system addressed in this paper. Goal of the absolute and relative orbit determination is to achieve an accuracy of 2 m and 0.1 m, respectively (3D, rms) and provide continuous position and velocity data of the participating spacecraft at a 1 Hz rate for guidance and control purposes as well as for the PRISMA payload. As detailed below, this is achieved by two software cores residing in the MAIN on-board computer. The two cores are executed at 30 s and 1 s sample times to separate the computational intensive orbit determination task from orbit prediction functions with low computational burden. An extended Kalman filter has been developed which processes pseudorange and carrier-phase measurement data issued by the local Phoenix GPS receiver on MAIN and sent via an Inter Satellite Link (ISL) from the remote Phoenix GPS receiver on TARGET.

The filter concept applies an ionosphere-free linear combination of pseudorange and carrier-phase data known as GRAPHIC (Group and Phase Ionospheric Correction) [3] to estimate the absolute orbits and use Single Difference (SD) carrier-phase measurements to implicitly determine the relative orbit with utmost precision. The originality of the design stems from the fact that only the absolute spacecraft states are explicitly estimated by the reduced-dynamic 49-dimension Kalman filter. The accurate raw carrier-phase measurements are differenced among common visible GPS satellites and used, when available, to enhance the information about the relative states of MAIN and TARGET and fulfill the relative navigation requirements. The inherent robustness of the symmetric filter design originates from the fact that common GPS satellites visibility is not a prerequisite to reconstruct the relative state. Even in the case of spacecraft with completely different attitude, the relative state can be determined by simply differencing absolute estimates exclusively based on GRAPHIC data types. As shown in the sequel, the unified filter design simplifies the initialization and the maneuver handling procedures, and, consequently, improves the flexibility of the navigation system and its reliability during the formation flying experiments.

This paper demonstrates the feasibility of the aforementioned concept and shows performance results from software simulations performed on a real-time embedded processor representative of the PRISMA onboard computer. Following a brief introduction of the GPS hardware architecture, the paper focuses on the description of the navigation software architecture and algorithms and emphasizes the simulation scenario and the associated numerical results.

GPS HARDWARE ARCHITECTURE

The GPS receivers to be flown on PRISMA are 12 channel single-frequency Phoenix receivers based on a commercial-off-the-shelf hardware platform [4]. The receivers have been qualified for use in LEO by a series of thermal-vacuum, vibration, and total ionization dose tests. Phoenix offers Coarse/Acquisition (C/A) code and carrier tracking with a noise level of 0.4 m and 0.5 mm, respectively at a representative carrier-to-noise ratio of 45 dBHz. The receivers support aiding with a priori trajectory information to allow a rapid acquisition of GPS signals under highly dynamic conditions. Upon tracking, Phoenix outputs a One-Pulse-per-Second (1PPS) signal and aligns the message time tags to integer GPS seconds which supports onboard clock synchronization and facilitates differential measurement processing, respectively.

The physical architecture of the Phoenix GPS system is identical on MAIN and TARGET. For redundancy, two Phoenix GPS receivers are available, which are connected to two GPS antennas via a coaxial switch. The dual antenna system provides increased flexibility for handling non-zenith pointing attitudes and antennas may be selected by ground command. Only one receiver will be active at any time. The overall physical architecture of the PRISMA GPS system on MAIN and TARGET is shown in Fig. 1.

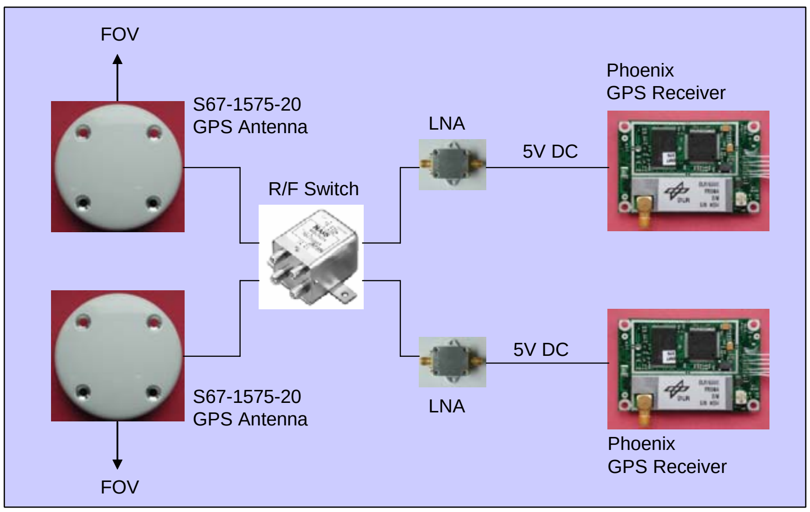

Fig. 1 PRISMA Phoenix GPS system on MAIN and TARGET.

Each GPS receiver is connected to its own low-noise amplifier (LNA) and provides 5 V DC for its operation via the R/F input. Compared to a single LNA placed between the antenna and the coaxial switch, this configuration avoids the need for an external LNA power supply and DC blocks. Furthermore, the adopted design reduces the risk of single-point failures. The use of a passive antenna, finally, allows the insertion of a band-pass or notch-filter prior to the coaxial switch and LNAs, if adverse out-of-band R/F signals should be encountered during interference tests.

PRISMA NAVIGATION SOFTWARE

Software Development Environment

The development of the navigation software for PRISMA is based on C++ and Matlab/Simulink. While Matlab/Simulink offers powerful tools for high level model-based design and real time applications, C++ is a lower level language and is therefore suitable for programming tasks that require high computational load. The processing layer of the software system is implemented in C++, including, for example, numerical orbit integration and data filtering. On the other hand the communication layer is implemented in Matlab/Simulink, including for example input/output interfaces, time synchronization and callback methods. The interface between the two programming layers is given by S-Function pre-build blocks in Matlab/Simulink.

As illustrated in Fig. 2, the flight software is generated using Real Time Workshop (RTW) Embedded Coder for an automatic translation of the Matlab/Simulink blocks into ANSI C code. The automatically generated C-code is then compiled and linked together with the handwritten C++ sources using the Real-Time Executive for Multiprocessor

Systems (RTEMS) cross-compiler system. Finally, the flight application is downloaded from the development host computer into the target, a LEON2 board, for validation and testing.

The LEON2 microprocessor implements a 32-bit processor compliant with the SPARC V8 architecture which is particularly suited for embedded applications [5]. Core of the data handling system on MAIN will be a LEON3 based spacecraft controller instead. In contrast to its predecessor LEON2, LEON3 recognizes bit flips and is fault tolerant. These aspects are, however, not considered as limiting factors for the portability of the navigation software from the test environment at DLR/GSOC to the PRISMA MAIN onboard computer.

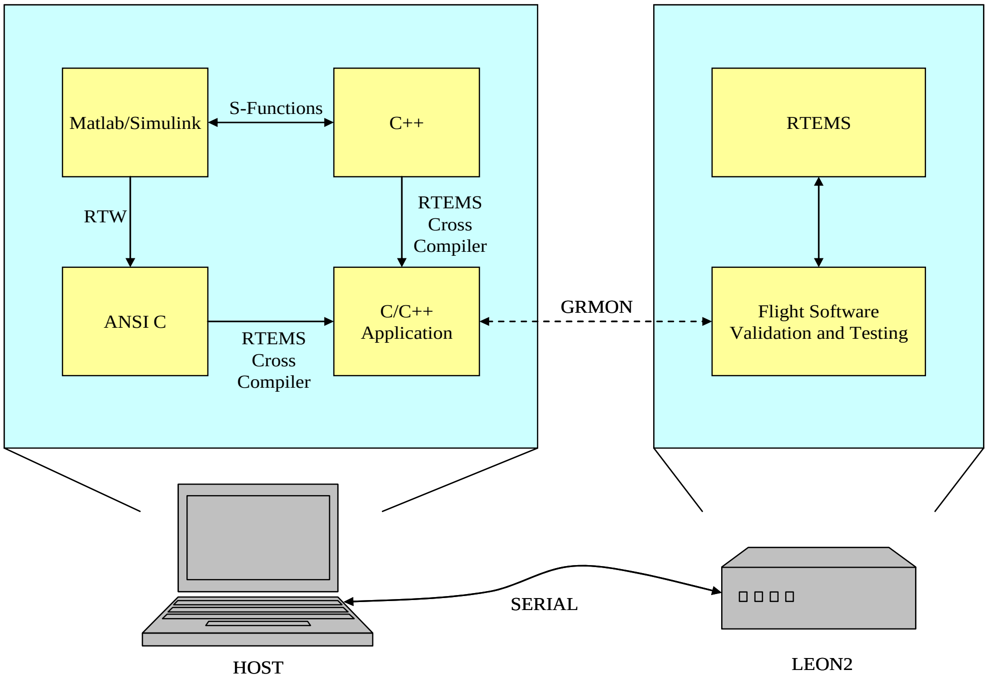

Fig. 2 Schematic of the PRISMA navigation software development environment.

Navigation System Architecture

The PRISMA onboard software (OBS) architecture consists of a layered structure with a Basic Software (BSW) level and an Application Software (ASW) level communicating with each other through dedicated message queues. While the BSW includes basic applications, device drivers and I/O-utilities, the ASW encapsulates all top-level applications like spacecraft navigation, control, telecommand and telemetry. As depicted in Fig. 3, the GPS-based navigation system is split into three modules located in different OBS levels and running at different sample rates.

The GPS interface (GIF) is part of the BSW, runs at 0.25 s sample time and is directly fed with GPS messages issued by the Phoenix GPS receivers on-board MAIN and TARGET. GIF handles GPS raw data formats and ephemerides, and performs data sampling as well as coarse editing prior to the GPS-based orbit determination. The GPS-based Orbit Determination (GOD) and GPS-based Orbit Prediction (GOP) are embedded in the ASW layer as part of the ORB core (30 s sample time) and the GNC core (1 s sample time), respectively.

GOD implements an extended Kalman filter to process GRAPHIC observables as well as single difference carrier phase measurements from MAIN and TARGET. Attitude data from both spacecraft are applied to correct for the GPS receivers antenna offset with respect to the spacecraft center of mass. Furthermore, a history of maneuver data is provided to GOD and taken into account in the orbit determination task. GOD performs a numerical orbit propagation which is invoked after the measurement update and provides orbit coefficients for interpolation to GOP for both spacecraft.

The GOP module interpolates the orbit coefficients provided by GOD and finally supplies the various GNC core functions as well as the PRISMA payload with continuous position and velocity data of MAIN and TARGET. Due to the different data rates of the GPS-based navigation modules, orbit maneuver data have to be taken into account in both GOD and GOP. In particular at each GNC step, the GOP task accounts for maneuvers which have not been considered by GOD in the last orbit determination/prediction process.

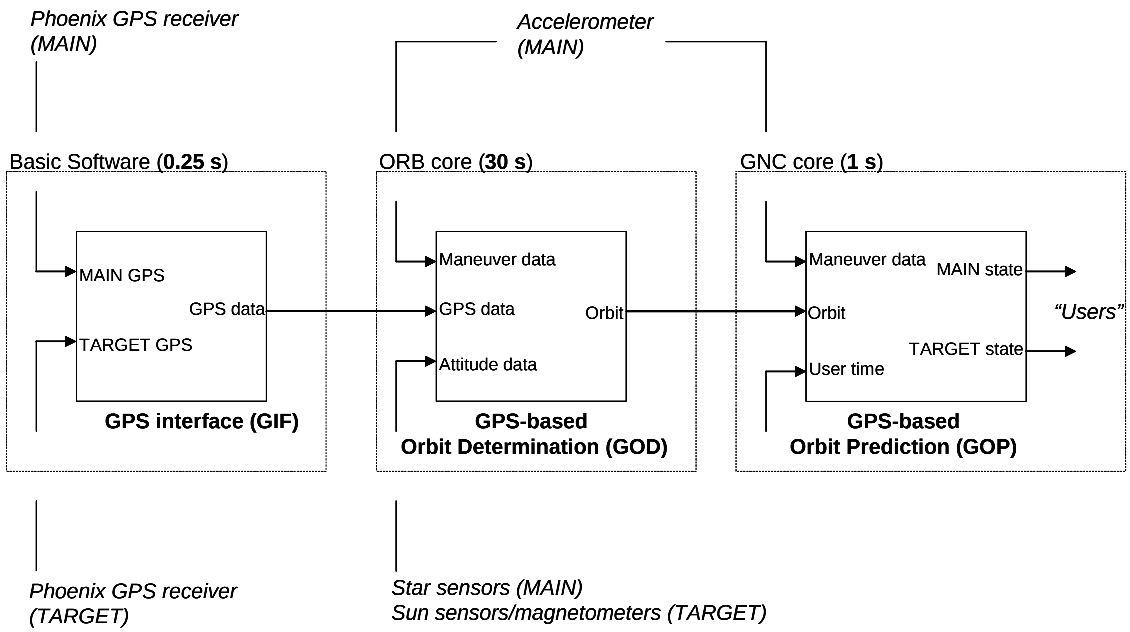

Fig. 3 Schematic architecture of the GPS-based navigation system highlighting software blocks, sensors and actuators.

Maneuver Related Interfaces

The navigation system provides continuous position and velocity data of the PRISMA spacecraft taking into account orbit maneuvers executed by MAIN in the past. To this end, a maneuver data history is available at the GOD and GOP input ports which covers a time interval Δt_M starting from the "current" epoch t_{curr} backwards in time (cf. Fig. 4). In general, t_{curr} refers to the time of the last accelerometers measurements being processed in the GNC core. Due to the time latencies of the data communication chain from the BSW to the ASW layer and to the delay induced by the further processing in the GNC core, t_{curr} will differ from the onboard time t_{GNC}, input to GOP, by a maximum of 1.5 s.

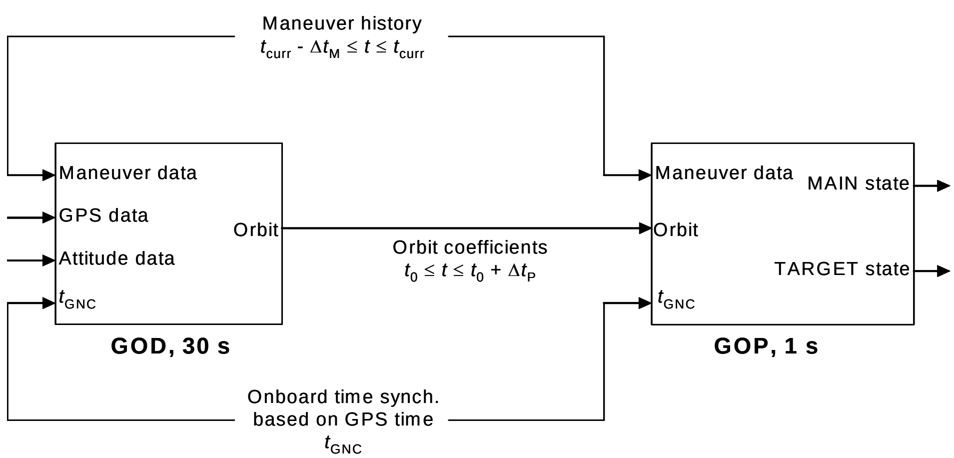

Fig. 4 Schematic view of maneuver-related interfaces in the GPS-based navigation system.

The choice of Δt_M is strictly related to the time properties of the numerical orbit propagation performed in GOD. After each call, GOD generates a set of orbit polynomials for MAIN and TARGET that covers a total orbit prediction interval of length Δt_P. The validity interval of the polynomial coefficients starts at the characteristic time t_0. In case of valid GPS measurements for a measurement update, t_0 equals the time tag of the observations. In case that no GPS measurements are available for a measurement update, t_0 equals the actual onboard time t_{GNC} provided by the GNC core. In the orbit determination process, GOD takes into account all maneuvers which occurred in the time interval between the newly computed t_0 and its previous value. In a conservative scenario, the last valid GPS measurements processed in GOD could be almost 60 s old and an arbitrary number of maneuvers could have been executed up to the time t_{GNC} of the present GOD execution. In contrast to the time tag t_{GPS} of the GPS messages delivered by the Phoenix receivers, t_{GNC} is not aligned to GPS integer seconds. Thus, a safety margin of 2 s is considered in the computation of Δt_M for a total value of 62 s.

Similar considerations apply for the choice of Δt_P. The orbit prediction has to cover the time interval between the previously determined t_0 and its new value. A conservative assessment suggests this time span not to be smaller than 60 s. Considering in addition the asynchronous triggering of the GOD and GOP functions with respect to t_{GPS}, a safety value of 62 s has been chosen for Δt_P. With these particular settings, it is assured that t_{GNC} is always within the orbit polynomials validity interval.

The maneuver history data comprise execution time and size of the maneuver executed by MAIN on a 1 Hz rate. In particular, for each second of accelerometer measurements an equivalent impulsive maneuver is computed with its total velocity variation mapped in the radial, along-track, cross-track reference frame.

Navigation Filter Design

The GPS-based navigation filter for PRISMA comprises two steps: the time update and the measurement update. In this section, the dynamic model for the time update and the measurement equations for the measurement update are presented. As introduced earlier, the filter employs a Kalman filter for the absolute states of MAIN and TARGET and avoids the need for an explicit relative state. The state vector is given by

$$\mathbf{x} = (\mathbf{r}_M, \mathbf{v}_M, \mathbf{a}_M, C_{DM}, \delta t_M, \mathbf{B}_M, \mathbf{r}_T, \mathbf{v}_T, \mathbf{a}_T, C_{DT}, \delta t_T, \mathbf{B}_T, \Delta \mathbf{v}_{eq})^T \tag{1}$$

which comprises position $\boldsymbol{r}$ (3), velocity $\boldsymbol{v}$ (3), empirical accelerations $\boldsymbol{a}$ (3), drag coefficient C_D (1), receiver clock offset δt (1) and GRAPHIC biases $\boldsymbol{B}$ (12) for MAIN (subscript M) and TARGET (subscript T), as well as the velocity variation of the equivalent impulsive maneuver $\Delta \boldsymbol{v}_{eq}$ (3) performed by MAIN in radial, along-track and cross-track directions.

The time update, or propagation step, is based on dynamics model for the orbits of the individual spacecraft, for the two independent Phoenix receiver clocks, and for the GRAPHIC ambiguities. The absolute orbit model for each spacecraft includes gravitational and non-gravitational forces. More specifically, the GRACE GGM01S gravity field [6], up to order and degree 15, is adopted to obtain the acceleration due to the Earth's static gravity field and analytical expansions of luni-solar coordinates are used to compute Sun and Moon point mass forces. Accelerations due to surface forces include solar radiation pressure and atmospheric drag. As the atmospheric density is difficult to model, the drag coefficient C_D is estimated as part of the force model parameters. Furthermore, empirical accelerations in the radial, a_R, along-track, a_T, and cross-track, a_N, directions are modeled as a first-order Gauss-Markov process [7] and estimated by the reduced-dynamic Kalman filter to compensate for modeling deficiencies in the employed dynamic models. The Phoenix receiver clock is handled as a random walk process. Empirical accelerations and receiver clocks are characterized by the respective steady sate variances, σ_a and σ_{clk} and their process noise time scales τ_a and τ_{clk}.

The measurement update uses pseudoranges, $\rho^{C/A}$, and carrier-phase, ρ^{L1}, measurements from the L1 GPS signals received by the Phoenix receivers on MAIN and TARGET. The standard models used for the raw GPS measurements are presented for MAIN (subscript M). The same equations apply for TARGET (subscript T).

$$\begin{aligned} \rho_{jM}^{C/A} &= \rho_{jM} + c(\delta t_M - \delta t_j) + I_{jM} \\ \rho_{jM}^{CP} &= \rho_{jM} + c(\delta t_M - \delta t_j) - I_{jM} + N_{jM} \end{aligned} \tag{2}$$

Here, ρ_{jM} is the real range between the MAIN Phoenix receiver and the GPS satellite j, $c\delta t_M$ is the MAIN Phoenix receiver clock offset, $c\delta t_j$ is the GPS satellite clock correction, I_{jM} is the ionospheric path delay, or range-equivalent TEC at the L1 frequency, and N_{jM} is the carrier-phase measurement ambiguity. The observables defined in (2) are linearly combined into the GRAPHIC (ρ^*) and single difference ($\Delta\rho$) data-types as follows

$$\begin{aligned} \rho_{jM}^{*} &= (\rho_{jM}^{C/A} + \rho_{jM}^{L1})/2 &&= \rho_{jM} + c(\delta t_M - \delta t_j) + B_{jM} \\ \Delta\rho_j &= \rho_{jM}^{L1} - \rho_{jT}^{L1} &&\approx (\rho_{jM} - \rho_{jT}) + c(\delta t_M - \delta t_T) + 2\cdot(B_{jM} - B_{jT}) \end{aligned} \tag{3}$$

where $B_{jM} = N_{jM}/2$ is the GRAPHIC bias and the single difference ionospheric path delay (I_{jM} - I_{jT}) has been neglected in the formulation of single differences because of the small distances between the satellites of the PRISMA formation (<5 km). The resulting observables can be treated as ionosphere-free and are characterized by a small noise if compared with the native measurements in (2). The maximum number of biases to be estimated by the filter equals the amount of tracking channels (i.e. 2*12 for the Phoenix GPS receivers on MAIN and TARGET), while the maximum number of measurements to be processed within the measurement update equals 36 (24 GRAPHIC + 12 single differences) provided that the same GPS satellites are tracked in the Phoenix receiver channels onboard MAIN and TARGET.

A proper reordering of the filter state and covariance matrix is done at each step between the time and measurement updates depending on the current GPS satellites visibility. Whenever GPS satellites are no longer observed, the corresponding GRAPHIC ambiguities are removed from the state and covariance matrix. Viceversa whenever a new GPS satellite is tracked, the corresponding GRAPHIC bias is initialized and introduced in the filter. This reordering is necessary to incorporate the change in observed GPS satellites and is of vital importance for an efficient implementation of the system in real-time. Unnecessary operations are avoided and, in the absence of maneuvers, a total filter size of (22 + n) is guaranteed at all times, where n is the number of currently tracked GPS satellites from MAIN and TARGET.

The filter design has to support the occurrence of several maneuvers executed by MAIN in the time interval between consecutive calls of the navigation filter. Let us define as m the total number of impulsive maneuvers $\Delta\boldsymbol{v}_i$, executed at times t_i, to be considered in the orbit determination process ($1 \le i \le m$). A so-called equivalent impulsive maneuver $\Delta\boldsymbol{v}_{eq}$, executed at time t_{eq}, is then modeled as

$$\Delta\mathbf{v}_{eq} = \sum_{i=1}^{m} \Delta\mathbf{v}_i ; \qquad t_{eq} = \sum_{i=1}^{m} \|\Delta\mathbf{v}_i\| t_i \bigg/ \sum_{i=1}^{m} \|\Delta\mathbf{v}_i\| \tag{4}$$

and incorporated in the estimation process. In particular, the Kalman state and covariance matrix are extended with three additional parameters, namely the radial, along-track and cross-track components of $\Delta\boldsymbol{v}_{eq}$. The estimation of the equivalent maneuver is performed via the introduction of the partial derivatives of the state

$$\frac{\partial \mathbf{r}_M}{\partial \Delta\mathbf{v}_{eq}} = (t_0 - t_{eq})\mathbf{U}; \qquad \frac{\partial \mathbf{v}_M}{\partial \Delta\mathbf{v}_{eq}} = \mathbf{U}; \qquad \frac{\partial \Delta\mathbf{v}_{eq}}{\partial \Delta\mathbf{v}_{eq}} = \mathbf{I} \tag{5}$$

in the state transition matrix. Here, **U** represents the rotation matrix from the radial, along-track, cross-track reference frame (in which the velocity variations are given) to the Earth Centered Inertial (ECI) reference frame (to which position and velocity of MAIN refer to), and **I** is the identity matrix. Estimation parameters linked to the impulsive maneuver via the state transition matrix are the position and velocity of MAIN and the velocity variations itself. The partial derivatives of the measurements with respect to the velocity variations are neglected.

FLIGHT SOFTWARE VALIDATION

Simulation Test Scenario

This section presents numerical results obtained from the test and validation of the PRISMA navigation flight software on the LEON2 microprocessor. The simulation test-bed comprises a reference trajectory generator using a GGM01S 20x20 gravity field model, third body gravitational perturbations from Sun/Moon, atmospheric drag and solar radiation pressure. The true trajectory is used together with a Phoenix receiver software emulator to generate highly realistic native messages emulating the Phoenix GPS receivers onboard MAIN and TARGET. The Phoenix emulator models the main sources of errors for a GPS receiver in LEO, including receiver dependent as well as constellation dependent characteristics. For the adopted scenario the emulator applies a phase-locked loop bandwidth of 9 Hz for carrier tracking and a delay-lock loop bandwidth of 0.08 Hz for code tracking, resulting in carrier-phase and pseudorange measurement noise levels of 0.5 mm and 0.4 m, respectively. Broadcast ephemeris errors of the GPS satellites of 2 m are assumed. The ionospheric path delays correspond to a Vertical Total Electron Content of 10 TECU. All applied errors are modeled as Gaussian random distributions with zero mean.

Table 1 Settings for the PRISMA GPS-based real-time navigation filter used throughout the 48 h test simulation.

Parameter	Value	Parameter	Value
Order and degree of gravity field	15	*Process noise*	
A-priori standard deviation		σ_{aR} [nm/s^2]	4.0
$\sigma_{\mathbf{r}}$ [m]	1000.0	σ_{aT} [nm/s^2]	10.0
$\sigma_{\mathbf{v}}$ [m/s]	1.0	σ_{aN} [nm/s^2]	10.0
σ_{aR} [nm/s^2]	100.0	σ_{clk} [m]	500.0
σ_{aT} [nm/s^2]	60.0	*Auto-correlation time scale*	
σ_{aN} [nm/s^2]	60.0	$\tau_{a(R,T,N)}$ [s]	900.0
σ_{CD} [-]	1.0	τ_{clk} [s]	100.0
σ_{clk} [m]	500.0	*Measurement standard deviation*	
$\sigma_{\boldsymbol{B}}$ [m]	0.05	σ_{ρ^*} [m]	0.05
$\sigma_{\Delta\boldsymbol{v}}$ [%]	10.0	$\sigma_{\Delta\rho}$ [m]	0.001

A representative PRISMA formation is simulated over 48 hours at an altitude of 700 km with typical spacecraft separations below 1 km. The main force model parameters are the cross-section area for drag computation, A_M=0.67 m^2, A_T=0.23 m^2, the satellite mass, M_M=150 kg, M_T=50 kg, the drag coefficient, C_{DM}=2.3, C_{DT}=2.1, and the solar radiation pressure coefficient, C_{RM}=1.3, C_{RT}=1.4, respectively for MAIN and TARGET. The a-priori state variances and filter settings applied for the 48 hours simulation arc can be found in Table 1. The same values apply for the MAIN and the TARGET spacecraft.

Numerical Results

Figures 5 and 6 depict the relative and absolute orbit determination errors obtained by comparing the output of GOP with the reference trajectory, both sampled at 0.1 Hz and mapped into the radial, along-track, cross-track directions. The relative navigation solution shows a performance of 0.04 m 3D r.m.s., while the accuracy of the absolute navigation solution is around 2 m 3D r.m.s. During the 48 hours simulation, four orbit control maneuvers are executed by MAIN in velocity and anti-velocity direction. Despite the spikes in the relative navigation output, especially in radial direction (cf. Fig. 5), the maneuvers are smoothly absorbed by the filter as shown by the empirical accelerations in Fig. 7.

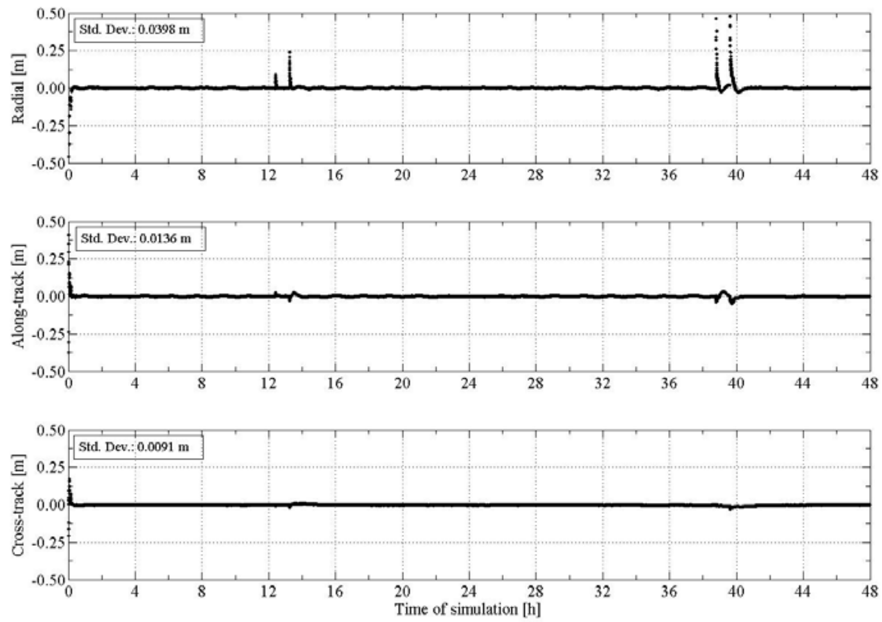

Fig. 5 Error of the relative position solution of the navigation filter mapped into the orbital frame (MAIN-TARGET).

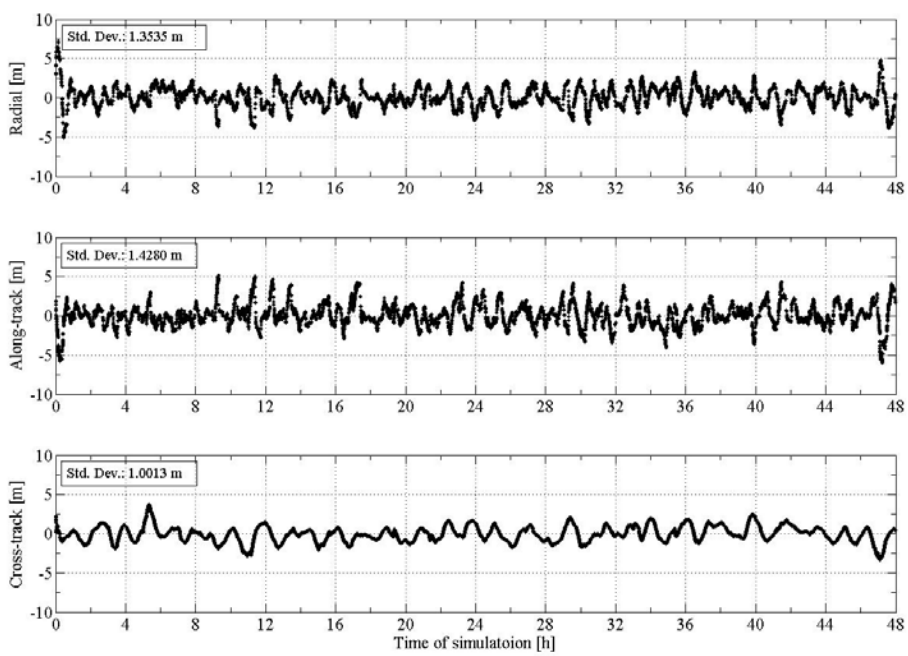

Fig. 6 Error of the absolute position solution of the navigation filter mapped into the orbital frame (MAIN).

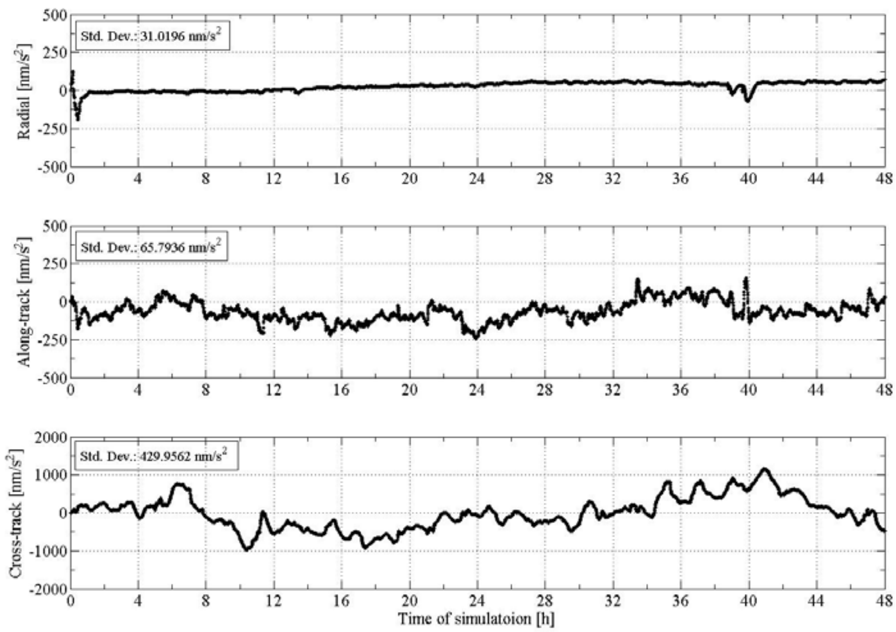

Fig. 7 Estimated empirical accelerations mapped into the orbital frame (MAIN).

SUMMARY

The real-time GPS-based navigation filter developed for the PRISMA formation flying mission has been presented. The filter concept applies single frequency pseudorange and carrier-phase measurements issued by the Phoenix receivers onboard the participating spacecraft. The raw GPS measurements are linearly combined into a powerful set of quasi-ionosphere-free data types, namely GRAPHIC and single-difference carrier-phase measurements from commonly visible satellites. This enables a consistent treatment of all measurements for an explicit estimation of the absolute spacecraft states and at the same time for the implicit determination of the relative states of MAIN and TARGET. Real-time simulations executed on the LEON2 microprocessor, representative of the PRISMA onboard computer LEON3, prove that requirements on absolute and relative navigation accuracy have been met. In particular, accuracies of 2 m 3D r.m.s. and 0.04 m 3D r.m.s. are observed for the absolute and relative orbit determination solutions, respectively. Orbit control maneuvers are incorporated in the filter as part of the state vector and do not degrade the navigation performance. A careful tuning of filter parameters, which was beyond the scope of this investigation, is expected to improve especially the absolute navigation accuracies.

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