AUTONOMOUS SATELLITE FORMATION FLYING FOR THE PRISMA TECHNOLOGY DEMONSTRATION MISSION

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AUTONOMOUS SATELLITE FORMATION FLYING FOR THE PRISMA TECHNOLOGY DEMONSTRATION MISSION

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PRISMA, originating from an initiative of the Swedish National Space Board (SNSB) and the Swedish Space Corporation (SSC), is a precursor mission for critical technologies for formation flying and In-Orbit-Servicing. The PRISMA test bed comprises the fully maneuverable MAIN mini-satellite as well as the smaller TARGET sub-satellite, both built by SSC.

Within PRISMA, a GPS-based onboard navigation system offers precise absolute and relative orbit information in real-time as well as time synchronization. The Phoenix twelve channel single-frequency GPS receiver, selected for PRISMA, provides a code tracking accuracy of better than 0.5 m and a carrier-phase accuracy of better than 1 mm at 45 dB-Hz. The navigation system on MAIN will process in a reduced-dynamic Kalman filter local GPS measurements and raw measurements from TARGET transmitted by an inter-satellite radio link. As a result, real-time position accuracies of 2 m for absolute and better than 0.1 m for relative navigation are expected.

In addition, PRISMA will conduct the Spaceborne Autonomous Formation Flying Experiment (SAFE) with the objective to demonstrate a fully autonomous, robust and precise formation flying of spacecraft. To this end, a fuel-optimized guidance and control algorithm will be employed providing an accuracy of better than 25 m at typical distances of 100 to 2000 m, which is representative of future bi-static radar satellite formation flying missions. The guidance concept applies the eccentricity/inclination vector separation which avoids collision hazard from along-track position uncertainties through the proper separation of the two spacecraft in radial and cross-track direction.

INTRODUCTION

An increasing demand in scientific, technological and commercial applications of space assets will boost the use of multi-satellite systems in the near future. In this framework, PRISMA provides a technology demonstration mission for the in-flight validation of sensor technologies and guidance, navigation and control (GNC) strategies for spacecraft formation flying and rendezvous.

The paper introduces the PRISMA mission, addressing its objectives, orbit characteristics and operations, as well as spacecraft design. Emphasis is on the GPS-based navigation system and an experiment for autonomous formation control. Following a description of the subsystem architecture, the GPS hardware is presented together with the navigation concept as well as the guidance strategy and control approach for autonomous formation flying.
PRISMA MISSION

Mission Objectives

The mission objectives of PRISMA may be divided into the validation of sensor and actuator technologies related to formation flying as well as the demonstration of experiments for formation flying and rendezvous. Key sensor and actuator components comprise (Persson et al. 2005)

- a GPS receiver system of the German Aerospace Center (DE)
- two Vision-Based Sensors of the Technical University of Denmark (DK)
- two Formation Flying Radio Frequency sensors of Alcatel Alenia Space (FR)
- a High-Performance Green Propellant system of SSC, Volvo and FOI (SE).

These will support and enable the demonstration of

1. autonomous spacecraft formation flying,
2. homing and rendezvous scenarios, as well as
3. close-range proximity operations.

Aside from the international cooperation, secondary mission goals originate from the Swedish national space program:

- Flight qualification of new avionics and power systems inherited from SMART
- Software development with Matlab/Simulink and automatic code generation
- Testing of new Electrical Ground Support Equipment
- Testing of silicon-based cold gas micro-thrusters.

Mission Description

The mission schedule foresees a launch of the two spacecraft in the second half of 2008. Both MAIN and TARGET will be injected by a Dnepr launcher into a sun-synchronous orbit with 700 km altitude and 98.2° inclination. A dusk-dawn orbit with a 6 h or 18 h nominal local time at the ascending node (LTAN) is targeted. Maximum eclipse times of 23 min may occur for injections within ±1 h off the nominal LTAN depending on the Sun’s declination (Pokrupa 2005).

Following a separation from the launcher, the two spacecraft will stay in a clamped configuration for initial system checkout and preliminary verification. Once the spacecraft are separated from each other, various experiment sets for formation flying and In-Orbit-Servicing will be conducted within a minimum targeted mission lifetime of eight months.

Spacecraft operations will be performed remotely from Solna near Stockholm making use of the ESRANGE ground station in northern Sweden. The S-Band ground-space link to MAIN supports commanding with a bit rate of 4 kbps and telemetry with up to 1 Mbps. In contrast, communication with the TARGET spacecraft is only provided through MAIN, acting as a relay and making use of a MAIN-TARGET intersatellite link (ISL) in the UHF band with a data rate of 19.2 kbps (Mörtsell 2005).
Experiment Operations

The mission objectives are associated to specific experiment sets which structure the mission timeline and operations. There are four operational and four equipment experiment sets (Berge et al. 2005). The operational experiment sets are summarized in Table 1.

<table>
<thead>
<tr>
<th>Experiment Set</th>
<th>Description</th>
<th>Separation [m]</th>
<th>Key sensors</th>
</tr>
</thead>
<tbody>
<tr>
<td>Autonomous Formation Flying</td>
<td>Closed-loop autonomous formation acquisition and keeping using various guidance and control laws</td>
<td>5000 – 20</td>
<td>GPS</td>
</tr>
<tr>
<td>Homing and Rendezvous</td>
<td>Autonomous approaches emulating rendezvous in GEO, assembly in escape orbits and sample return</td>
<td>100,000 – 3</td>
<td>Vision-based</td>
</tr>
<tr>
<td>Precision 3D Proximity Operations</td>
<td>Technology demonstration using virtual structures for On-Orbit Servicing, Inspection and Assembly</td>
<td>100 – 3</td>
<td>GPS &amp; Vision-based</td>
</tr>
<tr>
<td>Final Approach and Recede Maneuvers</td>
<td>Approaching as close as possible to demonstrate On-Orbit Servicing, Inspection and Assembly</td>
<td>3 – 0</td>
<td>Vision-based</td>
</tr>
</tbody>
</table>

The equipment experiment sets include tests of the High-Performance Green Propellant system, the cold gas micro-thrusters, the vision-based sensors and the formation flying radio-frequency metrology sensor.

Spacecraft Design

The MAIN spacecraft has a wet mass of 150 kg and a size of 80 x 83 x 130 cm in launch configuration (Hellmann 2005). In contrast to the highly maneuverable MAIN spacecraft, TARGET is a passive and much simpler spacecraft with a mass of 40 kg at a size of 80 x 80 x 31 cm (Fig. 1). Electrical power for the operation of the MAIN spacecraft bus and payload is provided by two deployable solar panels (Persson 2005) delivering at maximum 300 W, while TARGET relies on one body-mounted solar panel providing 90 W max.

GNC System

The MAIN spacecraft implements a 3-axis reaction wheel based attitude control and 3-axis delta-V capability (Bodin 2005). To this end, the MAIN GNC sensors comprise two 3-axis magnetometers, one pyramid sun acquisition and five sun presence sensors, five single-axis angular rate sensors, five single-axis accelerometers, two star tracker camera heads for inertial pointing and two GPS receivers. As actuators, three magnetic torque rods, four reaction wheels and six thrusters are employed.

TARGET applies a coarse 3-axis attitude control based on magnetometers, sun sensors and GPS receivers similar to MAIN and three magnetic torque rods as actuators. The nominal attitude profile for TARGET will be sun or zenith pointing, respectively.
Data Handling and Onboard Software

Core of the Data Handling System (DHS) on MAIN is the spacecraft controller based on a LEON3 microprocessor. LEON3 implements a 32-bit processor compliant with the SPARC V8 architecture which is particularly suited for embedded applications (Gaisler 2001). In contrast to its predecessor LEON2, LEON3 recognizes bit flips and is fault tolerant. Its implementation through a Field Programmable Gate Array (FPGA) of Atmel provides a performance of about 20 MIPS (Andersson priv. comm. 2005) and accommodates one Floating Point Unit (FPU). Communication between platform units and spacecraft controller is implemented via a Controller Area Network (CAN) bus.

The 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 control and telecommand and telemetry. A model-based design method is used for the ASW layer which is grouped into functionally encapsulated application components. The onboard software is implemented in Matlab and Simulink as well as C/C++ functions which are then auto-coded with Real Time Workshop and executed under the operating system Real-Time Executive for Multiprocessor Systems (RTEMS) on the LEON3.

Thruster System

The MAIN spacecraft will accommodate a High-Performance Green Propellant (HPGP) system which provides delta-V for relative navigation with respect to TARGET. The HPGP utilizes an ammonium dinitramide-based liquid monopropellant (LMP-103) to provide thrust in all directions using six 1 N thrusters. LMP-103 is preferred over hydrazine because of its 6% higher theoretical specific impulse and its 24% higher
density (Anflo priv. comm. 2005) together with its environmentally benign, non-carcinogenic and low toxicity characteristics.

The HPGP thrusters are capable of providing impulse bits ranging from 0.1 Ns up to continuous burns of 30 seconds with a maximum pulse rate of 1 Hz. Minimum impulse bits translate to single velocity increments of 0.7 mm/s which can be applied for formation control. A total of 11 kg propellant provides a total delta-v of 115 m/s in an accumulated firing time of at least five hours (Anflo 2005).

**GPS-BASED NAVIGATION, GUIDANCE AND CONTROL**

**Objectives**

Within PRISMA, DLR has assumed responsibility for providing the GPS-based navigation functionality which comprises the provision of

1. Phoenix GPS receivers
2. Onboard Navigation System for absolute/relative orbit determination
3. On-ground precise orbit determination (POD).

Among the various experiment sets within PRISMA, DLR will perform the

4. Spaceborne Autonomous Formation Flying Experiment (SAFE)
5. Onboard Autonomous Orbit Keeping (AOK) of a single spacecraft.

The AOK is intended for execution at the end of the PRISMA mission operations phase. In particular, the primary objectives of DLR’s contributions to PRISMA are to

- provide GPS navigation fixes and raw data of MAIN and TARGET
- provide on MAIN a precise absolute orbit solution for MAIN
- provide on MAIN a precise relative orbit solution of TARGET w.r.t. MAIN
- implement a guidance law for a safe separation strategy
- provide a robust control algorithm for formation keeping
- demonstrate autonomous orbit control of close formations
- implement an automated on-ground process for precise orbit reconstruction.

In addition, the secondary objective is to

- demonstrate an autonomous absolute orbit control of the MAIN spacecraft.

**Architecture**

The architecture of the GPS hardware system is depicted in Fig. 2. For redundancy purposes, each satellite will carry two independent GPS receivers. Increased flexibility for handling non-zenith pointing attitudes and to maximize the field of view (FOV) to the GPS constellation is provided by two independent GPS antennas. Using a radio-frequency relay, the active GPS receiver and the associated low noise amplifier (LNA) can be connected to the optimal antenna through ground command or an automated onboard process.
The overall architecture for DLR’s contributions to PRISMA’s GNC system is shown in Fig. 3. Upon tracking, the GPS receivers provide a one Pulse-per-Second (1PPS) for onboard time synchronization and deliver data messages containing raw GPS code and carrier phases to the GPS navigation function. The navigation function performs a reduced-dynamic filtering of GPS raw data and provides absolute MAIN and relative TARGET with respect MAIN orbit information. Orbit information is, furthermore, provided to the autonomous formation control (AFC) and autonomous orbit keeping (AOK) applications for computing the requested velocity increments.

**Fig. 2 Physical architecture of the GPS system on MAIN and on TARGET.**

**Fig. 3 System architecture of DLR’s contributions to GNC functions within PRISMA.**
GPS Receiver

Phoenix is a miniature GPS receiver for high-dynamics and space applications (Montenbruck & Markgraf 2005). The Phoenix receivers to be flown on PRISMA are twelve channel single-frequency GPS receivers based on a commercial-off-the-shelf hardware platform and qualified by DLR for use in low Earth orbit (LEO). Phoenix offers single-frequency Coarse/Acquisition (C/A) code and carrier tracking and can be aided with a priori trajectory information to safely acquire GPS signals even at high altitudes and velocities. Upon tracking, Phoenix outputs a 1PPS signal and aligns the message time tags to integer GPS seconds which facilitates differential processing.

The receiver is built around the GP4020 baseband processor of Zarlink, which combines the correlator, a microcontroller core with a 32 bit ARM7TDMI microprocessor and several peripheral functions in a single package. Phoenix provides a code tracking accuracy of better than 0.5 m and a carrier-phase accuracy of better than 1 mm at 45 dB-Hz. With a mass of the receiver board of 70 gr and a power consumption of 0.85 W at begin of life, the receiver is particularly suited for small satellite missions.

Navigation

The GPS-based navigation system onboard MAIN will process local raw GPS measurements as well as code and carrier phase measurements transmitted from TARGET by an inter-satellite radio link. Spacecraft attitude information is applied to correct for the GPS antenna offset with respect to the spacecraft center of mass and orbit maneuver data is accounted for in the orbit prediction. The navigation system will provide absolute position and velocity data of MAIN and relative position and velocity data of TARGET with respect to MAIN at a 1 Hz data rate for GNC functions and PRISMA payload.

Measurement Model

Orbit determination of MAIN is based on the linear combination of C/A pseudorange $\rho_{C/A}$ and L1 carrier-phase measurements $\rho_{L1}$. The resulting GRAPHIC (GROup And PHase Ionospheric Calibration) data type $\rho^*$ is modeled as (Yunck 1993)

$$\rho^* = (\rho_{C/A} + \rho_{L1})/2 = \rho + c(\delta t - \delta t_{GPS}) + b$$

(1)

where $c$ denotes the velocity of light, $\delta t$ and $\delta t_{GPS}$ the receiver and GPS satellite clock offsets and $b$ a carrier-phase bias. GRAPHIC data are free from ionospheric errors since the ionospheric group delay in pseudorange measurements is equal in size but opposite in sign to the ionospheric phase change in carrier phase measurements.

For relative navigation, a double-difference processing of carrier phase measurements with phase $\phi$ at wavelength $\lambda$ fully exploits the accuracy potential of this data type. The measurement equation is given as

$$\nabla\Delta(\lambda\phi)^g_{ij} = \nabla\Delta(\rho)^g_{ij} + \nabla\Delta(\lambda N)^g_{ij} - \nabla\Delta(\lambda I)^g_{ij}$$

(2)

where $\nabla\Delta$ denotes the double difference of quantities related to GPS satellites $i$ and $j$ and the receivers on MAIN ($M$) and TARGET($T$). While differencing across receivers
reduces broadcast ephemeris and ionospheric errors, the differencing across GPS satellites eliminates the user clock error and the initial fractional carrier phase. Still, the measurement equation requires the solution of the integer bias $N$ and the treatment of the differential ionospheric error $\nabla \Delta(t)^i_{MT}$.

**Trajectory Model**

While a purely dynamic force model does not provide the necessary accuracy to adequately fit precise GPS measurements over extended data arcs, a kinematic approach would neglect orbit knowledge from the equations of motion to improve the orbit determination accuracy. Thus, a reduced-dynamic approach has been adopted for orbit determination.

To this end, the equations of motion not only account for the complex gravity field of the Earth and atmospheric drag but also introduce empirical accelerations in radial, along-track and cross-track direction to account for residual perturbations which are not explicitly modeled such as luni-solar accelerations and solar radiation pressure.

**Filter Model**

An extended Kalman filter will be employed for absolute and relative orbit determination. The filter adjusts the spacecraft position and velocity components, the empirical accelerations, clock errors and GRAPHIC biases along with integer carrier phase ambiguities as float values and, potentially, differential ionospheric path delays.

A process noise matrix is employed to cope with residual modeling deficiencies and to keep the Kalman filter receptive to new measurements. Scalar measurement updates are applied to avoid time-consuming matrix-vector operations.

**Concept Validation**

The presented concept for GPS-based absolute and relative navigation has been validated within a hardware-in-the-loop (HIL) demonstration. To this end, a signal simulator generated GPS signals for a LEO space scenario which were received by ORION GPS receivers (predecessor of Phoenix with comparable performance). The raw GPS measurements were transferred through radio modems to a central navigation computer for subsequent filtering (Leung & Montenbruck 2005). For the relative motion of two spacecraft, an ellipse with 4 km maximum along-track separation and 2 km radial separation was adopted which is representative for the PRISMA mission.

Results from HIL demonstrations for relative navigation, covering the filter initialization and subsequent convergence, are depicted in Fig. 4. Here, the relative navigation accuracy (3D RSS) after convergence is about 1.5 mm in relative position and 5 $\mu$m/s in relative velocity. At larger separations of 10-50 km, this accuracy degrades to 10 cm max. due to differential broadcast ephemeris and ionospheric errors.

While this accuracy may not be achievable for PRISMA due to multipath effects, coverage problems and heavy maneuver activity, real-time relative navigation accuracies of better than 0.1 m (3D RSS) are expected. For absolute navigation, a position accuracy of better than 2 m will be achieved which is largely governed by GPS broadcast ephemeris errors.
On-ground Verification

The on-ground post-facto trajectory reconstruction will provide precise absolute and relative position and velocity data for verification and calibration purposes. In this context, ionosphere-free code-carrier combinations allow to estimate absolute positions of the two PRISMA spacecraft with a 3D accuracy of better than 0.5 m (Gill & Montenbruck 2004), and, subject to multipath avoidance and good common satellite visibility, relative positions with centimeter accuracy.

The improvements over an on-board processing stem from the use of precise GPS clock and ephemeris data, distributed by the International GPS Service (IGS), as well as the use of more elaborate data editing and ambiguity resolution algorithms.

![Image](image_url)

Fig. 4 Relative position accuracy for two spacecraft flying in a 2 x 4 km radial-tangential ellipse. GPS time is given in seconds. The figure is adopted from Leung & Montenbruck 2005.

Formation Guidance and Control

In addition to the GPS-based navigation system, DLR will contribute to PRISMA mission the Spaceborne Autonomous Formation Flying Experiment (SAFE). SAFE complements other PRISMA experiment sets with the objective to demonstrate a fully autonomous, robust and precise formation flying of spacecraft. To this end, a fuel-optimized formation flying at typical distances of 100 to 2000 m is foreseen, which is representative of future bi-static radar satellite formation flying missions (Gill & Runge 2004).

Guidance

The guidance concept applies the eccentricity/inclination (e/i)-vector separation, which has originally been developed for the safe collocation of geostationary satellites.
Based on the absolute eccentricity $e_i$ and argument of perigee $\omega_i$ for the satellite $i = (1, 2)$, the relative eccentricity vector (D’Amico & Montenbruck 2005) can be formed according to

$$\Delta e = e_2 - e_1 = e_2 \begin{pmatrix} \cos \omega_2 \\ \sin \omega_2 \end{pmatrix} - e_1 \begin{pmatrix} \cos \omega_1 \\ \sin \omega_1 \end{pmatrix} = \delta e \begin{pmatrix} \cos \phi \\ \sin \phi \end{pmatrix}$$

where $\delta e$ denotes the amplitude and $\phi$ the relative phase of the vector. Similarly, the relative inclination vector depends on the absolute inclinations $i_i$ and right ascension of the ascending nodes $\Omega_i$,

$$\Delta i = \delta i \begin{pmatrix} \cos \theta \\ \sin \theta \end{pmatrix} = \begin{pmatrix} \Delta i \\ \Delta \Omega \sin i \end{pmatrix}$$

with amplitude $\delta i$ and phase $\theta$, which can be expressed as differences in inclination and right ascension of ascending node according to $\Delta i = i_2 - i_1$ and $\Delta \Omega = \Omega_2 - \Omega_1$.

Parallel relative eccentricity and inclination vectors constitute a collision-free geometry, where radial and cross-track separations never vanish at the same time. This implies a coordinated selection of the relative orbital elements which results in an elliptic relative motion perpendicular to the flight direction as depicted in Fig. 5.

![Fig. 5 Relative satellite formation geometry for PRISMA based on parallel (e/i)-vector separation in an orbital frame. Radial, tangential and normal directions are depicted by $R$, $T$, $N$, respectively.](image)

For PRISMA, this concept avoids collision hazard from along-track position uncertainties through the proper separation of the two spacecraft in radial and cross-track direction. The ($e/i$)-vector separation ensures a maximum operational safety in contingency cases and proves robust under the presence of maneuver execution errors as well as data and communication losses.

**Control**

A convenient orbit control concept maintains the safe ($e/i$)-vector separation of the PRISMA formation at an affordable expenditure in terms of thruster activations and propellant consumption. To that end, deterministic orbit maintenance maneuvers will be performed to counteract secular perturbations of the relative ($e/i$)-vectors which tend to disturb the initial nominal configuration. A continuous control of the relative orbit is thus avoided which is a prerequisite for most scientific formation flying missions and proves
adequate in terms of accuracy for the majority of formation flying missions. SAFE aims at a position control accuracy of 10/20/10 m (R/T/N 1σ).

Maneuver executions are conveniently based upon requests for velocity increments issued by SAFE. Since the velocity increments are provided in an orbital frame, SAFE requires no attitude knowledge and complex propulsion system operations such as thruster pre-heating and thrust profile modeling are avoided. Including the formation acquisition phases, a velocity increment budget of 5 m/s is allocated for SAFE.

Operations

A total of six weeks will exclusively be allocated for SAFE operations including one week of initial calibration and two weeks of open-loop operations. While initial calibration refers to the verification of GPS-based absolute and relative navigation, open-loop operations denote formation control phases which are not entirely performed autonomously.

Prior to closed-loop operations, a manual formation acquisition will be performed. To this end, a commanded transition from an initial relative geometry to the targeted eccentricity and inclination separation is performed. Within closed-loop operations, SAFE will be given guidance and control authority for MAIN. Autonomous formation flying phases at 100 to 2000 m maximum separations are foreseen to cover representative separations of future bi-static radar satellite formation flying missions and to allow an analysis of separation-dependent control errors.

SUMMARY

PRISMA is a technology demonstration mission for the in-flight validation of sensor and actuator technologies and guidance, navigation and control strategies for in-orbit servicing and spacecraft formation flying. An overview of the mission is provided including mission objectives, mission description and operations as well as spacecraft design.

PRISMA comprises GPS-based navigation, guidance and control capabilities to demonstrate in real-time completely space-based closed-loop autonomous formation flying. To this end, high-grade GPS receivers will provide raw data for absolute and relative position determination onboard the MAIN spacecraft with accuracies of better than 2 m and 0.1 m, respectively.

A guidance concept based on the separation of relative eccentricity and inclination vectors has been introduced which ensures a maximum operational safety in contingency cases and proves robust under the presence of maneuver execution errors as well as data and communication losses. The guidance and control concept will demonstrate autonomous formation control with a typical accuracy of 20 m for spacecraft separations between 100 to 2000 m.

PRISMA will demonstrate for the first time in Europe a GPS-based fully autonomous closed-loop formation flying of spacecraft. This is considered a key milestone for autonomous formation flying and paves the way for advanced formation flying missions in low Earth orbit.
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