EarthCARE - Earth Clouds, Aerosols and Radiation Explorer
SPECTRA - Surface Processes and Ecosystem Changes Through Response Analysis
WALES - Water Vapour Lidar Experiment in Space
ACE+ - Atmosphere and Climate Explorer
EGPM - European Contribution to Global Precipitation Measurement
Swarm - The Earth’s Magnetic Field and Environment Explorers
REPORTS FOR MISSION SELECTION
THE SIX CANDIDATE EARTH EXPLORER MISSIONS

Swarm - The Earth's Magnetic Field and Environment Explorers
Technical and Programmatic Annex

European Space Agency
Agence Spatiale Européenne
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1 Introduction

This document provides the technical description of the Swarm mission as derived from the preparatory activities at Phase A level, for implementation as an Earth Explorer in the frame of ESA’s Living Planet Programme. It shows how feasible implementation concepts can respond to the scientific mission requirements defined in the Science Report. To this end, the expected system performance will also be described. A summary assessment of the programmatic framework is also provided.

The system description is based on the results of the work performed during two parallel Phase A system studies by two industrial consortia. It is not possible to describe in this report all technical concepts, but, where necessary, two concepts are described in order to present significantly different approaches to meeting the mission observation requirements. The description of particular concepts does not indicate special preferences.

After an overview of the mission architecture and the proposed orbit (in Chapters 2 and 3) the space segment will be described in detail (Chapter 4), followed by the ground segment and operations concept (Chapters 5 and 6). Following an overview of the data products (Chapter 7), the overall performance is described (Chapter 8). The report concludes with programmatic considerations (Chapter 9).
2 Mission Architecture Overview

The Swarm mission architecture is driven by the requirement for separation of the various sources contributing to the Earth's magnetic field. The mission architecture is depicted in Figure 2-2 and consists of:

- Space Segment
- Ground Segment
- Launcher

**Space Segment**

In the original proposal of the Swarm mission the space segment concept consisted of a constellation of 4 satellites in two different near polar orbits at 450 and 550 km altitude. Each satellite provided precise and high-resolution measurements of the magnetic field vector. In combination, the satellites provided the observations necessary for the global high-precision survey of the geomagnetic field that is needed to model its various sources. Two satellites were placed in an orbit as low as possible separated along track by a distance of between 500 and 1000 km. The two other satellites were placed in an orbit 100 km higher and separated by 180 degrees in true anomaly. The inclinations of the two orbits were close to 90 degrees and slightly different, so that the different precession rates would lead to the separation of the orbital planes over the mission lifetime.

Sensitivity studies performed during Phase A by means of an end-to-end performance simulator have demonstrated that the Swarm science requirements for the primary science objectives can be met by a three satellite constellation, two satellites in the lower orbit flying side by side and only one satellite in the higher orbit. Adding a fourth satellite would not contribute significantly to the science return associated with the primary research objectives. There are science issues related to the external field investigations that would benefit from the fourth satellite, properly situated, but this would be beyond the main objectives. The baseline space segment consists therefore of 3 satellites.

Figure 2-1 shows the relation between the Phase A system studies and the end-to-end mission performance simulator study. The simulator study is dedicated to the assessment of the impact of mission and satellite constellation scenarios on the estimation of the magnetic field results.

Figure 2-1: Study interfaces
Ground Segment

The ground segment, which is in charge of satellite command and control operations, and of scientific data acquisition, processing, storage and distribution to the user community, is composed of the following elements:

- Command and Data Acquisition Element (CDAE)
- Mission Operations and Satellite Control Element (MSCE)
- Processing and Archiving Element (PAE)

The CDAE consists of the Kiruna S-Band ground station; an additional ground station can be scheduled in emergency cases.

The MSCE located at ESOC in Darmstadt is in charge of the monitoring and planning of satellite operations. The main tasks of the PAE, managed by ESRIN, are the generation of products from the science data stream, the archive and the interface with the users.

Figure 2-2: Swarm mission architecture

Launcher

Depending on the final satellite configuration the following launchers are taken into consideration to transport the space segment into orbit: DNEPR, ROCKOT, VEGA and COSMOS.
3 Orbit
The selection of the orbit, the constellation geometry and its set-up takes into account the following considerations:

- Near polar orbits are required to obtain global coverage
- Multi-point measurements at global and regional scales, as required for spatial and temporal sampling of the fields, imply the deployment of a constellation
- The launch of the satellites into two different orbits may be accomplished either by direct injection or by injection into an intermediate orbit followed by transfer manoeuvre
- Drift between orbital planes is required to separate the different components of the magnetic field
- A pair of satellites has to fly as low as possible for lithospheric studies.

To meet these requirements, as shown in figure 3.1, the initial Swarm space segment consisted of:

- Four spacecraft in two different orbital planes and orbit altitudes, with two satellites per plane
- A lower orbit with an inclination of 86.8°, two satellites flying at 450 km mean altitude separated by 500 km to 1000 km along track
- A higher orbit with 87.4° inclination, with two satellites flying at 550 km altitude separated by 180° in true anomaly

![Figure 3-1: Swarm, initial 4 satellite orbital constellation](image)

The derivation of the spacecraft delta-V budget and propellant requirements has to take into account the orbit acquisition strategy, the needs for maintaining the orbit, the constellation geometry, the satellite attitude and the potential disposal at the end-of-life. To achieve identical satellite design, the propellant demand for the various tasks needs to be balanced for the spacecraft of both orbits. As for the orbit acquisition strategy, the proposed concept is to release the spacecraft at an orbit of intermediate altitude and inclination. The acquisition of the operational orbits will then be completed by the spacecraft with their own propulsion subsystems.

The satellites will be separated in succession taking advantage of the launcher upper stage propulsion to minimise the requirements on the spacecraft. The acceleration of the upper stage produces a difference in the spacecraft velocities and therefore an increase of the distance between consecutively released satellites. The main effect of this scenario on the spacecraft
orbits consists in a greater orbital energy of the last separated satellite (higher semi-major axis) with respect to the other satellites. This will allow saving part of the fuel employed in the transfer to the final orbits from the intermediate one. The time between the first and second spacecraft separations must be short, so that satellites can be transferred to the lower orbit with similar resources. Waiting longer to separate the remaining spacecraft will permit to save fuel in the transfer to the higher orbit, thanks to the increase in the apogee height produced by the upper stage acceleration. An example for two final orbits at 450 km and 550 km and an intermediate orbit at 490 km is shown in table 3.1.

<table>
<thead>
<tr>
<th>Conditions at launcher release</th>
<th>Operational orbit parameters</th>
<th>Delta V (m/s)</th>
<th>Fuel mass (kg)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Altitude (km)</td>
<td>Inclination (°)</td>
<td>Altitude (km)</td>
<td>Inclination (°)</td>
</tr>
<tr>
<td>490</td>
<td>87.125</td>
<td>450</td>
<td>86.8</td>
</tr>
<tr>
<td>550</td>
<td>87.4</td>
<td>35</td>
<td>21</td>
</tr>
</tbody>
</table>

**Table 3-1: Requirements for an example of optimized orbit injection**

The altitude maintenance strategy (Figure 3-2 and Table 3-2) is more demanding for the lower satellites. The approach is to keep the satellites within an altitude dead-band of 450±1 km until the moment when the spacecraft can be left in free decay without going below 300 km before the end of the mission. Attention has to be paid also to maintain the formation of the lower satellite pair.

**Figure 3-2: Altitude maintenance strategy for the satellites in the lower orbit**

<table>
<thead>
<tr>
<th>height in km.</th>
<th>Fuel mass for maintenance (kg)</th>
<th>optimum deadband (km)</th>
<th>number of manoeuvres</th>
<th>manoeuvre delta-V (m/s)</th>
</tr>
</thead>
<tbody>
<tr>
<td>450</td>
<td>7.11</td>
<td>449/451</td>
<td>11</td>
<td>1.121</td>
</tr>
</tbody>
</table>

**Table 3-2: Summary of the altitude maintenance strategy**
Following the sensitivity analyses performed with the end-to-end performance simulator introduced in chapter 2, the orbit and constellation geometry was tuned for the three-satellite baseline. The selected space segment orbit and constellation geometry are (Figure 3-3):

- Three spacecraft in two different orbital planes, with two satellites in a plane of 87.4° inclination and with one satellite in a plane of 86.8° inclination
- The two satellites in the 87.4° inclination orbit will fly at 450-km mean altitude, their East-West separation will be 1-1.5°, and the maximal differential delay in orbit will be approximately 10 s.
- The satellite in the lower inclination orbit will fly at 530 km mean altitude

![Figure 3-3: Baseline Swarm with 3 satellite orbital constellation](image)

Since these proposed orbits are in a similar range than the initial constellation, the orbit acquisition strategy described above is applicable and an appropriate intermediate orbit will be selected. As with respect to the initial constellation, the lower orbit altitude is unchanged and the need to maintain the formation is maintained, delta-V budgets and propellant mass estimations for the initial constellation will be similar for the selected three-satellite Swarm constellation. To sum-up, 35 kg of propellant are allocated to orbit acquisition and maintenance as shown in the propellant budget included in chapter 4.

The constellation geometry proposed at the end of phase A is not unique and optimization will continue. Changes in the range of orbits and geometries proposed do not affect at all the satellite concepts nor launch choices and launcher strategies described in this document.

The eclipse period has been analysed over one year time-span for the two Swarm orbits. The simulation shows that for both orbits the average eclipse duration is in the order of 30 minutes per orbit. The minimum value is in the order of one minute and the maximum value in the order 36 minutes. The total eclipse duration for the analysed one-year period is about 30%.
4 Space Segment
As a result of the sensitivity studies performed with the end-to-end mission performance simulator the baseline configuration is composed of three satellites described in section 4.3, each satellite consisting of platform, described in section 4.2, and its payload, described in section 4.1.

4.1 Payload
The payload consists of the following instruments:

- Absolute Scalar Magnetometer (ASM)
- Vector Field Magnetometer (VFM)
- Electrical Field Instrument (EFI)
- Accelerometer (ACC)
- Laser Retro Reflector (LRR)

4.1.1 Absolute Scalar Magnetometer (ASM)
The objective of the Absolute Scalar Magnetometer (ASM) is to calibrate the vector field magnetometer (VFM) to maintain the absolute accuracy in the multi-year geomagnetic field mission. The required main performance characteristics of the ASM are: absolute accuracy of < 0.3 nT (2σ), resolution <0.1 nT within its full-scale range of 15000-65000 nT. For the ASM instrument two feasible concepts are considered.

4.1.1.1 ASM Configuration A
The magnetometer subject of configuration A is based on the Electron Spin Resonance (ESR) principle and makes use of the Zeeman effect which splits the emission and absorption lines of atoms in an ambient magnetic field. The pattern and amount of splitting is a signature of the magnetic field strength. The optically pumped helium magnetometer uses a High Frequency (HF) discharge within a gas cell to excite \(^4\)He atoms from the ground state to the metastable state. This metastable level is split by the Earth magnetic field into 3 Zeeman sublevels. The separation of those sublevels is directly proportional to the ambient field strength and equals half the gyro frequency (eB/2m with m – electron mass).

Transitions between the Zeeman sublevels are excited by an external Radio Frequency (RF) field, which tends to equalize the population of those levels. The resonance signal is usually very weak due to the nearly equal population of the Zeeman levels under thermal equilibrium conditions. In order to enhance the resonance signal, a re-population of the different Zeeman levels is performed by means of a tunable laser (1083 nm), which triggers transitions between the metastable and the short-lived level. According to the non-equal transition probability for the 3 different levels, a non-equilibrium population is achieved which amplifies the resonance signal for several orders of magnitude. A closed-loop operation is obtained according to the decrease in optical cell transmission, as soon as the RF resonance occurs due to the re-population of the Zeeman levels. This signal can be used to lock in phase the RF field frequency, which is a direct measure of the ambient magnetic field (see Figure 4-1).

Optically pumped helium magnetometers have been used on several missions (e.g. Cassini spacecraft and SAC-C). They display a very high bandwidth and have clear advantages in the low-field region compared to Nuclear Magnetic Resonance (NMR) types.
A block diagram of the magnetometer is depicted in Figure 4-2. The instrument consists of a Data Processing Unit (DPU), a sensor head and fibre optical cables. The magnetometer is not yet space qualified but an existing prototype has demonstrated its capabilities. The main items to be worked on for space qualification are the helium cell, the laser, the piezo motor and moving mass as the polarizer and the RF coil to be rotated by the piezo motor. The space qualification of the instrument is expected within a time frame of 2 years.

Table 4-1: Specifications of the helium magnetometer
4.1.1.2 ASM Configuration B

The baseline for the ASM instrument in configuration B is a Proton Free-precession scalar magnetometer. The proton precession magnetometer operates on the principle that the protons are spinning around an axis aligned with the magnetic field. Ordinarily, protons tend to line up with the Earth's magnetic field. When subject to an artificially-induced magnetic field, the protons will align themselves with the new field. When this new field is interrupted, the protons return to their original alignment with the Earth's magnetic field. As they change their alignment, the spinning protons precess, or wobble, much as a spinning top does as it slows down. The frequency at which the protons precess is directly proportional to the strength of the Earth's magnetic field.

The magnetometer consists of a sensor, a harness cable and the electronics unit. The sensor, depicted in Figure 4-3 has an inner ring-shaped container containing a hydrocarbon-rich liquid. The ring is provided with evenly distributed ~1000 turns toroidal coil for polarization and subsequent pick-up of the precession. A DC-current in the coil impresses a large circular polarizing magnetic field for 0.5 s aligning a fraction of the proton magnetic moments in the sample.

After rapid switch-off of the polarization the protons start to precess about the Earth’s magnetic field and the toroidal coil picks up the free-precession signal: a µ-volt level exponentially decaying ~600 Hz – 2.8 kHz sine-wave with about 0.5 s decay time constant. About 0.4 s of the free-precession signal is digitized at ~50 kHz under tight timing control by the local base oscillator. The data sample is then analyzed in the signal processor by an algorithm extracting the center frequency, the time decay constant, the initial signal amplitude and the linear change of frequency during the ~ 0.4 s aperture time. The timing of the single data points, and thus of the sample start, stop and mid-time, is controlled by the base oscillator and is thus traceable to the Universal Time Coordinated (UTC).

![Figure 4-3: The toroidal sensor-coil](image)

The following Table 4-2 summarises the magnetometer technical specification.

<table>
<thead>
<tr>
<th>Specification</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Mass</td>
<td>&lt; 2100 g</td>
</tr>
<tr>
<td>Power Consumption</td>
<td>&lt; 7 W</td>
</tr>
<tr>
<td>Data Rate</td>
<td>100 bps</td>
</tr>
<tr>
<td>Dynamic Range</td>
<td>15000 – 65000 nT</td>
</tr>
<tr>
<td>Absolute Accuracy</td>
<td>0.3 nT (2σ)</td>
</tr>
<tr>
<td>Omni-directional response</td>
<td>&lt; 0.1 nT angular dependence</td>
</tr>
</tbody>
</table>

*Table 4-2: Specifications of the proton precession magnetometer*
4.1.2 Vector Field Magnetometer (VFM)

The Vector Field Magnetometer (VFM) is the prime instrument of the Swarm mission. It will provide ultra linear and low-noise measurements of the Earth's magnetic field vector components. The VFM full-scale range is ±65 μT and it has been allocated a measurement random error of less than 1 nT integrated over frequencies up to 4 Hz.

The proposed VFM (fluxgate type) consists of a Compact Spherical Coil (CSC) sensor, non redundant, mounted on the deployable boom, an internally redundant data processing unit (DPU) and the connecting harness. The CSC sensor is shown in Figure 4-4.

![Figure 4-4: CSC sensor](image)

The operation of the fluxgate sensor is based on the extreme symmetry of the positive and negative magnetic saturation levels of the ferromagnetic sensor core material. Continuous probing of the core saturation levels by a high frequency excitation magnetisation current enables the sensor to detect deviations from the zero field with only tens of pT noise and sub-nT long term stability.

To obtain the magnetic field vector information the CSC sensor attitude has to be known very accurately. Therefore the VFM and the satellite attitude determination system (star tracker) are mounted together on an optical bench. A summary of the VFM mass and dimensions is provided in Table 4-3. The CSC sensor is similar to the ones flying on Ørsted, CHAMP and SAC-C. The DPU will be miniaturized compared to the VFM’s flying on the before mentioned satellites. The VFM proposed for Swarm is expected to fly on the Proba-II.

<table>
<thead>
<tr>
<th>Unit</th>
<th>Mass (incl. margin)</th>
<th>Size</th>
<th>Power (incl. margin)</th>
</tr>
</thead>
<tbody>
<tr>
<td>CSC Sensor</td>
<td>460 g</td>
<td>82 mm diameter Spherical</td>
<td>150 mW (For 65000 nT)</td>
</tr>
<tr>
<td>Electronics</td>
<td>350 g</td>
<td>100×100×23 mm + feet + lid + connectors</td>
<td>260 mW@±3.3 V</td>
</tr>
<tr>
<td>Cable</td>
<td>12 m @ 30g/m</td>
<td>8mm diameter</td>
<td></td>
</tr>
<tr>
<td>Thermal</td>
<td>16 g</td>
<td>MLI + Insulation</td>
<td>0</td>
</tr>
<tr>
<td>Mech. I/F</td>
<td>10 g</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Total</td>
<td>&lt;1200 g (incl. 12m cable)</td>
<td></td>
<td>425mW @±3.3 V</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
<td>615mW unregulated and incl. interface electronics</td>
</tr>
</tbody>
</table>

10% mass and power margin are included.

*Table 4-3: Summary of the VFM specifications*
4.1.3 Electric Field Instrument (EFI)

The key parameters to be determined by this instrument are the ion density, the drift velocity and the electric field by applying the \((v \times B)\) expression. The proposed EFI consists of three sensors, an Ion Drift Meter (IDM) to measure the cross track components of the ion velocity, a Retarding Potential Analyser (RPA) to measure the total amount of the ion velocity and a Planar Langmuir Probe (PLP), mainly to measure the spacecraft potential (see Table 4-4). The electrical field vector components shall be determined with a random error of less than 3 mV/m (2\(\delta\)) integrated over frequencies up to 0.35 Hz and with a stability of 1 mV/m/ month (2\(\delta\)).

<table>
<thead>
<tr>
<th>The EFI Sensor:</th>
<th>Measurement Parameter</th>
</tr>
</thead>
<tbody>
<tr>
<td>Ion Drift Meter (IDM)</td>
<td>Ion Arrival Angle</td>
</tr>
<tr>
<td>Retarding Potential Analyser (RPA)</td>
<td>Ion velocity</td>
</tr>
<tr>
<td>Planar Langmuir Probe (PLP)</td>
<td>Electron Density, Temperature, Spacecraft Potential</td>
</tr>
</tbody>
</table>

Table 4-4: EFI Sensors and related measurement parameters

All sensors are integrated next to each other in one box together with the data processing unit (Figure 4-5). An aluminum ground plate is mounted on the front side of the box. The sensors are installed behind the ground plate, which has apertures for the sensor’s front sides.

Operating Principle of IDM
A segmented collector behind the square aperture is exposed to the ion stream flowing at supersonic speed. A series of entrance grids provides a field free drift space for the ions before striking the collector. The arrival angle is the result of the superposition of spacecraft velocity and the ion velocities parallel and perpendicular to the spacecraft velocity vector. Using dedicated electronics that produce signals proportional to the logarithm of the generated currents the arrival angle of the incoming ions can be derived.

Operating Principle of RPA
A circular entrance aperture is presented to the incoming ion stream flowing at \(~7.6 \text{ km/s}\), determined by the spacecraft velocity, leading to an energy of \(~1/ 3 \text{ eV per atomic mass unit}\). Using retarding potential grids behind the aperture, the energy of the ions, which have access to the collector, is discriminated. The measured collector current versus the retarding potential (I-V) is used to fit the observations with respect to the ion temperature, ion drift and composition.
Operating Principle of PLP
The PLP is a conductor by which the potential with respect to ground (in general the whole satellite) and the collected current can be monitored. In case of spherical symmetry, the electronic plasma parameters (temperature and density) are obtained from Langmuir’s formula. By sweeping through the probe bias and observing the current, the spacecraft potential can also be determined.

The EFI instrument is mounted on the spacecraft front panel (in-flight), in a cutout to optimise accommodation and field-of-view. The instrument electronic is located inside the spacecraft body. The EFI has a proven design with flight heritage of more than 25 years on a large number of spacecraft. The instrument performed well on several satellites such as Atmospheric Explorer and Dynamics Explorer and is used on all Defence Meteorological Satellite Program (DMSP) payloads.

A summary of the EFI specification is provided in Table 4-5.

<table>
<thead>
<tr>
<th>Mass</th>
<th>6.5kg</th>
</tr>
</thead>
<tbody>
<tr>
<td>Power</td>
<td>5 W</td>
</tr>
<tr>
<td>Dimensions</td>
<td>15.4 cm x 32 cm x 25.1 cm</td>
</tr>
<tr>
<td>Dimensions Ground Plane</td>
<td>28 cm x 40 cm</td>
</tr>
<tr>
<td>Data Rate</td>
<td>1.8 kbps</td>
</tr>
</tbody>
</table>

Table 4-5: EFI specifications

4.1.4 Accelerometer (ACC)
The accelerometer (ACC) will measure the non-gravitational accelerations, caused for example by air-drag, winds, Earth albedo and solar radiation pressure acting on the satellites. In-situ air density measurements together with magnetic data can be used to obtain new insights on the geomagnetic forcing of the upper atmosphere.

The focus on air density investigations relaxes the requirements both for the ACC itself and for the spacecraft in comparison to a pure gravity mission where special care has to be taken to minimize all possible sources of disturbance accelerations from the spacecraft. This relaxes the requirements on the accelerometer accommodation and on the control of the location of the spacecraft Centre of Mass (CoM).

Several candidate sensors have been investigated. In addition to the well-known sensor flying on CHAMP, the MAC (Micro-ACcelerometer) instrument (Figure 4-6) is a candidate to measure non-gravitational forces for the study of thermospheric density structures. The MAC is an integrated instrument box, comprising a sensor and associated electronics. The accelerometer is based on the electrostatic suspension of a free-flying proof mass and the measurement of the restoring electrostatic force to maintain this proof mass motionless in the centre of the sensor cage. It is mounted inside the satellite as close as possible to the CoM. The cage has to be hard mounted to the satellite structure and will be subject to all non-gravitational accelerations applied to the satellite body.

Predecessor instruments of this accelerometer have been flown in Space Shuttle missions. Modifications are foreseen to improve the performance (stability, noise) and for adaptation of the power interface to 28 V.
The instrument specifications are summarized in Table 4-6.

<table>
<thead>
<tr>
<th>Characteristics</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Mass</td>
<td>5.4 kg</td>
</tr>
<tr>
<td>Mass including thermal control</td>
<td>6.2 kg</td>
</tr>
<tr>
<td>Power Consumption</td>
<td>4.5 W</td>
</tr>
<tr>
<td>Dimensions</td>
<td>310 x 155 x 164 mm</td>
</tr>
<tr>
<td>Cubic proof-mass</td>
<td>(~30 x 30 x 30)mm, made from fused quartz</td>
</tr>
<tr>
<td>Measurement Range</td>
<td>$2 \times 10^{-4}$ ms$^{-2}$</td>
</tr>
<tr>
<td>Resolution</td>
<td>$2 \times 10^{-10}$ ms$^{-2}$</td>
</tr>
<tr>
<td>Frequency Bandwidth</td>
<td>$10^{-4} \ldots 10^{-4}$ Hz</td>
</tr>
<tr>
<td>Bias stability</td>
<td>$3 \times 10^{-3}$ ms$^{-2}$/K (thermal)</td>
</tr>
<tr>
<td></td>
<td>$1 \times 10^{-8}$ ms$^{-2}$ (long-term)</td>
</tr>
<tr>
<td>Data Rate</td>
<td>0.278 kbps</td>
</tr>
</tbody>
</table>

**Table 4-6: ACC specifications**

### 4.1.5 Laser Range Reflector (LRR)

The principle of Satellite Laser Ranging (SLR) is the measurement of the two-way round trip time of a laser signal, which is emitted and received by a ground station after reflection by an onboard LRR array. The time-of-flight data yield (via an appropriate atmospheric correction using meteorological data on ground) the precise distance of the optical reference point of the laser retro reflector. Additionally to the time of flight, the exact epoch of the laser shot has to be recorded. By repeated laser firing to the LRR-equipped satellite it becomes possible to measure directly a part of the orbit and to compare it e.g. to the measurements of the onboard tracking system. SLR data can therefore be regarded as an external calibration source for the onboard orbit determination hardware. They can also serve as a backup orbit determination system. The essential drawback of the SLR method is the weather dependence (clear sky is required to transmit and receive the laser signal) and the sparse coverage due to the low number of SLR ground stations.

A proposed design consists of 4 reflecting quartz prisms, which are mounted into a pyramid-shaped aluminium structure by means of fixation rings. Each single prism is glued into the fixation ring by means of two-component silicon rubber. Figure 4-7 shows a view of the complete LRR unit and Table 4-7 provides a summary of the technical data.
Figure 4-7: LRR unit

<table>
<thead>
<tr>
<th>Specification</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Mass</td>
<td>400 g</td>
</tr>
<tr>
<td>Dimensions</td>
<td>100 x 100 x 48 mm$^3$</td>
</tr>
<tr>
<td>Number of Prisms</td>
<td>4</td>
</tr>
<tr>
<td>Spot Far field</td>
<td>2</td>
</tr>
<tr>
<td>Nadir Pointing</td>
<td>$\pm 5^\circ$</td>
</tr>
</tbody>
</table>

Table 4-7: LRR specifications
4.2 Platform
The platform comprises the following subsystems: structure, mechanisms, power, telemetry/telecommand, attitude and orbit control (AOC), thermal control, and data handling.

**Structure/Mechanisms**
The structure consists of the Bus Assembly, Solar Generator Assembly, Boom Assembly, and Launch Adapter. Figures 4-8 and 4-9 show the different elements of the structure for the two proposed satellite configurations.

![Figure 4-8: Structural elements (Configuration A)](image1)

![Figure 4-9: Structural elements (Configuration B)](image2)

The requirements for the magnetic data products of the Swarm satellite are very stringent and impose the need to optimize and control the magnetic properties of the spacecraft. An important design measure is the accommodation of the magnetometer package at a sufficient distance of the platform to minimize the magnetic disturbance. A boom ensures a magnetic clean environment and a very stable accommodation for the magnetometer package. Due to envelope constraints of the launcher fairing the boom needs to be deployable. The above Figures show the boom in the stowed launch configuration.

The optical bench accommodated on the boom will ensure a very stable mechanical interface between the VFM sensor head and the star tracker used to determine the precise attitude of the VFM sensor and as main attitude sensor of the AOCS.

**Power**
The power subsystem is in charge of the generation (solar array), storage (battery), and distribution (PCDU - Power Control & Distribution Unit) of energy for the platform and payload during nominal mission phases. During safe mode only essential subsystems will be powered. The bus topology is characterised as a direct energy transfer system resulting in an
unregulated power bus. The voltage of the bus is determined by the state-of-charge of the battery and the level and direction of the charge current. In order to minimise the charging of the spacecraft, the positive pole of the battery has been connected to structure ground, an approach successfully implemented in the CHAMP and GRACE missions. The solar generator consists of two main panels (partly deployable depending on the satellite configuration) with triple-junction GaAs cells. As the most efficient battery technology a Li-ion battery is selected. A block diagram of the power subsystem is shown in Figure 4-10. The design of the power subsystem takes into account a duty cycle of 100% for the payload.

Telemetry & Telecommand Subsystem (TT&C)
The TT&C consists of an S-band transceiver and is in charge of receiving:

- RF signals from the ground station and deliver the demodulated digital command data stream to the On-Board Computer (OBC),
- the digital data stream from the OBC, modulate it onto the carrier and transmit the resulting RF signal to the ground station.

The S-band communication system consists of two transmitters, two receivers, an RF distribution network and two antennas. The nominal RF transfer from and to ground will be realised by a combined receive / transmit helix antenna pointing towards nadir during nominal flight orientation. On the zenith side patch antennas are accommodated to establish ground contact for off-nominal attitude conditions. While both receivers are running in permanent hot redundancy, only one of the two transmitters will be switched on by the OBC in advance of an expected ground station fly-over.

Science data transmission is also performed via S-band at 1 Mbps. On ground the science data will be separated from the housekeeping data, processed and distributed to the science community. The use of different virtual channels allows an easy separation of science and housekeeping data.

Attitude and Orbit Control Subsystem (AOCS)
The AOCS maintains an Earth oriented satellite attitude within a control band of <5 deg. around roll, pitch and yaw. The subsystem is based on a cold gas propulsion system and magnetic torquers as actuators and star tracker, magnetometer, Coarse Sun / Earth sensors for attitude determination. A GPS receiver provides timing and position determination.
The AOCS operates in the three basic modes, Normal, Orbit Control (OC) and Acquisition and Safe Mode as illustrated in the Figure 4-11. The transitions between modes are either automatic (A) or commanded from ground (TC).

High priority has been put on the design and selection of the magnetic torquers, torquers with air coils have been selected due to their advantage of higher linearity of the magnetic moment and better magnetic cleanliness, i.e. no remanent DC field and no induced field effects. The B-field changes due to magnetic torquer currents will be corrected in the magnetometer measurement data. The residual uncertainty of the magnetic field effect knowledge has been considered in the performance error budget.

**Propulsion**

The cold gas propulsion subsystem is considered the best solution to fulfill not only the delta V requirements for orbit control manoeuvres (and the orbit transfer manoeuvre in the case of spacecraft injection into an intermediate orbit), but also to complement the magnetic torquers as attitude actuators. Each satellite has two propellant tanks carrying each nearly 30 kg of N₂. Thrusters of 20 mN and 40 mN thrust level are arranged to provide sufficient thrust for orbit manoeuvres and control authority for attitude control. A propellant budget for a single Dnepr launch is provided in table 4-8.

<table>
<thead>
<tr>
<th>Propellant demand</th>
<th>Propellant per satellite [kg]</th>
</tr>
</thead>
<tbody>
<tr>
<td>Orbit Transfer and Air Drag compensation</td>
<td>35</td>
</tr>
<tr>
<td>Attitude Control</td>
<td>13</td>
</tr>
<tr>
<td>Margin (10%)</td>
<td>4.8</td>
</tr>
<tr>
<td>Leakage over lifetime and non-useable residuals</td>
<td>1.5</td>
</tr>
<tr>
<td>Total</td>
<td>54.3</td>
</tr>
</tbody>
</table>

*Table 4-8: Propellant budget for a single DNEPR launch*

**On-Board Data Handling (OBDH)**
The OBDH mainly consists of the OBC, the mass memory and the software. It is in charge of:

- reception and decoding of telecommands from the on-board receiver,
- distribution to the on-board users,
- the acquisition and intermediate storage of housekeeping and science data,
- its subsequent transmission to the TT&C during RF ground contact periods.

Furthermore the OBDH subsystem provides all necessary processing and memory capabilities to support the operation of application software (e.g. AOC and thermal control), resource management, Failure Detection Isolation and Recovery (FDIR) routines, as well as the control of bus and instrument unit modes. The OBC is internally redundant with a full cross coupling provided on module level.
Thermal
The thermal control design of the Swarm satellites relies on passive means. The orbit local time is drifting. All sides of the spacecraft, except the nadir face will be exposed to sun during the mission lifetime for significant time. For a period of several days, even a nearly eclipse free orbit will be experienced. All dissipating elements, with the exception of a few externally located sensors and appendages, are mounted inside the spacecraft main body. Most of the internally mounted equipment can be maintained within a stable temperature range without additional heater power. Temperature stability is good, orbital variations are in the range of less than 5 degrees, due to a good thermal decoupling from external solar heat loads. The thermal control of the external antennas is passive due to the wide acceptable temperature range. The thermal control of the boom is also passive. The magnetometer sensors are provided with a very good thermal isolation.

4.3 Satellite
The design drivers for the satellite configuration are the following:

- Accommodation of all spacecraft within the launcher fairing
- Accommodation of the instruments according to their specific requirements:
  - The accommodation of the magnetometers is driven by the need to provide a magnetically clean environment at the location of the sensor heads. The spacecraft are provided with a boom to allow for maximum separation of the magnetic field sensor heads from the spacecraft units that produce stray fields, notably the magnetic torquers.
  - The EFI must be accommodated on the ram side of the spacecraft
  - The accelerometer must be accommodated close to the CoM
- Small cross-section for reduced air drag (impact on fuel amount required for orbit maintenance)
- Provision of sufficient solar array area for all solar illumination conditions
- Accommodation of all subsystems, in particular:
  - The propellant must be distributed in two tanks to maintain the CoM as required by the accelerometer
  - The power dissipating equipments are accommodated on the nadir face to facilitate heat rejection

The boom is accommodated in anti-velocity direction. This flight configuration reduces the effect of air drag on fuel consumption and exploits the torque generated by air drag to stabilize the satellite, thus reducing the actuation of the magnetic torquers and the magnetic contamination of the observations. It also facilitates the accommodation of the EFI on the ram side. Two overall satellite configuration concepts have been proposed as depicted in Figure 4-12.
The main spacecraft characteristics are shown in Table 4-9. For some values a range is provided in order to reflect the different configurations A and B.

<table>
<thead>
<tr>
<th></th>
<th>Configuration A</th>
<th>Configuration B</th>
<th>Remarks</th>
</tr>
</thead>
<tbody>
<tr>
<td><strong>Spacecraft Mass</strong></td>
<td>400 kg</td>
<td>240 kg</td>
<td>Incl. margin</td>
</tr>
<tr>
<td><strong>Spacecraft Power</strong></td>
<td>167 W</td>
<td>125 W</td>
<td>Incl. margin</td>
</tr>
<tr>
<td><strong>Solar Arrays Dimensions</strong></td>
<td>4.7 sqm</td>
<td>4.4 sqm</td>
<td></td>
</tr>
<tr>
<td><strong>Power Storage</strong></td>
<td></td>
<td>Lithium-Ion Battery</td>
<td></td>
</tr>
<tr>
<td><strong>Spacecraft Dimension</strong></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td><strong>Length</strong></td>
<td>6.31 m</td>
<td>7.85 m</td>
<td>Boom deployed</td>
</tr>
<tr>
<td><strong>Width</strong></td>
<td>2.02 m</td>
<td>1.50 m</td>
<td></td>
</tr>
<tr>
<td><strong>Height</strong></td>
<td>0.84 m</td>
<td>1.03 m</td>
<td></td>
</tr>
<tr>
<td><strong>Boom length</strong></td>
<td>4 m</td>
<td>5.4 m</td>
<td>Deployed</td>
</tr>
<tr>
<td><strong>Boom Direction</strong></td>
<td>Anti - flight</td>
<td></td>
<td></td>
</tr>
<tr>
<td><strong>Attitude &amp; Orbit Control</strong></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td></td>
<td>Three-axis stabilised (magnetic torquers, cold gas)</td>
<td></td>
<td></td>
</tr>
<tr>
<td><strong>Telemetry/ Telecommand</strong></td>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td></td>
<td>S-Band, 1 Mbps downlink and 4 kbps uplink</td>
<td></td>
<td></td>
</tr>
<tr>
<td><strong>Data Generation per Day &amp; Satellite</strong></td>
<td></td>
<td>125 Mbytes</td>
<td></td>
</tr>
<tr>
<td><strong>Data Storage Capability</strong></td>
<td>512 Mbytes</td>
<td>475 Mbytes</td>
<td></td>
</tr>
</tbody>
</table>

*Table 4-9: Summary of spacecraft characteristics*
4.4 Launchers
The following technical aspects of the Swarm mission affect the launcher selection:

- Number of spacecraft to be launched
- Orbit constellation, 2 orbits with altitudes of 450 km and 550 km
- Accommodation of the spacecraft under the fairing including launcher adapter

After the first launcher evaluation the following potential candidates for Swarm are identified: ROCKOT, DNEPR, COSMOS and VEGA.

A comparison of the different useable fairing volumes and the accommodation of the Swarm satellites within the DNEPR fairing is shown in Figure 4-13.

Figure 4-13: Comparison between the useable fairing volumes of the selected launcher candidates (left) and accommodation of the Swarm satellites within the DNEPR fairing (right)
5 Ground Segment

The ground segment will be based on the infrastructure being developed to support the Earth Explorer and other missions. The breakdown of the ground segment into its constituting elements and functions is outlined in Figure 5-1. It consists of three main elements: the CDAE, the MSCE and the PAE.

The three elements implement the Flight Operations Segment (FOS) and the Payload Data Segment (PDS) functions. The FOS will be developed and operated according to the concept of the “family of missions” by which several missions share resources and staff for reduction of costs and reutilization of expertise. Concerning the PDS, the principles and infrastructure developed for the open – operational “oxygen” initiative will be reused as well as the infrastructure procured for previous Earth Explorer missions.

![Figure 5-1: Ground segment architecture](image)

The CDAE implements the following main FOS functions:

- Telemetry acquisition, telecommand uplink and satellite tracking (TT&C) functions
- Transmission of telemetry to the MSCE

For the PDS, the CDAE implements the following main functions:

- Payload data acquisition
- Demodulation and formatting
- Processing to appropriate level,
- Short-term archiving
- Transmission to the PAE

The CDAE will be located in Kiruna. The capability of the ground station to cope with the Swarm constellation has been analysed. Figure 5-2 shows the 8 possible contact orbits of the
lower Swarm satellites with the Kiruna station and Figure 5-3 shows the possible 9 contact orbits for the upper satellite. The average contact time per day for the lower orbit is approx. 3100 s, with a mean duration of the pass of more than 6 minutes. The average daily contact time for the higher orbit is approx. 4300 s, with a mean duration of the pass of more than 7 minutes.

The required downlink time for all satellites is much less than the daily contact time with the Kiruna station and therefore comfortably compatible with the contact time available. In order to cope with the higher amounts of data stored in the satellite mass memory after a period of satellite autonomy, e.g. 3 days without contact, an additional ground station can be used.

**Figure 5-2**: Contacts of each lower orbit satellite with the Kiruna ground station

**Figure 5-3**: Contacts of higher orbit satellite with the Kiruna ground station
Figure 5-4 shows a typical daily distribution of the Kiruna ground station passes for the 3 Swarm satellites.

|                | 00 | 01 | 02 | 03 | 04 | 05 | 06 | 07 | 08 | 09 | 10 | 11 | 12 | 13 | 14 | 15 | 16 | 17 | 18 | 19 | 20 | 21 | 22 | 23 |
|----------------|----|----|----|----|----|----|----|----|----|----|----|----|----|----|----|----|----|----|----|----|----|----|----|----|----|
| KIRUNA_SC_1    |    |    |    |    |    |    |    |    |    |    |    |    |    |    |    |    |    |    |    |    |    |    |    |    |    |
| KIRUNA_SC_2    |    |    |    |    |    |    |    |    |    |    |    |    |    |    |    |    |    |    |    |    |    |    |    |    |    |
| KIRUNA_SC_3    |    |    |    |    |    |    |    |    |    |    |    |    |    |    |    |    |    |    |    |    |    |    |    |    |    |

**Figure 5-4: Daily distribution of the Swarm passes over the Kiruna ground station**

The MSCE will be located at ESOC and will provide the following main functions:

- Overall satellite operations planning
- Satellite monitoring and control
- Flight dynamics and manoeuvre planning
- On-board software maintenance
- Mission simulation
- FOS supervisión
- Interface with the launch site during LEOP

The PAE will be managed by ESRIN and will implement the following PDS functions:

- Acquisition of payload data (including platform ancillary data) from the CDAE
- Acquisition of the required auxiliary data
- Generation of products at Level 1 and 2 by means of Instrument Processing (IPF) and High Level Processing Facilities (HPF) respectively
- Long-term archiving (LTA) of mission products, including re-processing of archived data as needed
- Payload operations planning by means of a Reference Planning Facility (RPF)
- Mission and payload monitoring besides PDS performance monitoring by means of a Monitoring Facility (MF)
- Quality Control (QC) for all products and media distributed to users
- Distribution of mission products to the user community
- Provision of User Services (US) based on the existing EO Multi-mission User Services located at ESRIN
6 Operations and Utilisation Concept

All Swarm satellites will be placed into orbit by a single launch vehicle. After separation from the launcher adapter the satellites subsystems will be activated to allow contact of the satellites with the ground stations. In this launch and early orbit phase (LEOP) at least three ground stations will be involved. The LEOP duration is estimated to be one to two weeks. After LEOP the commissioning phase is entered in order to checkout all satellite subsystems and the payload. The commissioning phase is currently expected to last four to six months.

After the commissioning phase the nominal mission phase of 4 year starts. All instruments are assumed to have a 100% duty cycle; they are operational all time under nominal conditions. A short overview of the mission time line is provided in the Table 6-1.

<table>
<thead>
<tr>
<th>Operations Phase</th>
<th>Estimated Time or Duration</th>
<th>Activities/ Status</th>
</tr>
</thead>
<tbody>
<tr>
<td>Pre- Launch</td>
<td></td>
<td>Payload and spacecraft functional test</td>
</tr>
<tr>
<td>Launch (begin of Launch and Early Orbit Phase-LEOP)</td>
<td>2008/2009</td>
<td>Injection in intermediate orbit or direct injection depending on finally selected approach</td>
</tr>
<tr>
<td>LEOP</td>
<td>1 - 2 weeks</td>
<td>Subsystem activation (for example OBDH boot-up) Initial constellation acquisition</td>
</tr>
<tr>
<td>Commissioning</td>
<td>4 - 6 months</td>
<td>Checkout of all subsystem and payload Final constellation acquisition Calibration activities</td>
</tr>
<tr>
<td>Nominal Operations</td>
<td>4 years</td>
<td>Acquisition of instrument data Spacecraft performance and status monitoring</td>
</tr>
<tr>
<td>Disposal</td>
<td></td>
<td></td>
</tr>
</tbody>
</table>

Table 6-1: Preliminary Swarm mission time line

There are two main modes during the operations phase, the nominal mode and the safe mode. In the nominal mode all subsystems and instruments are switched on. The housekeeping and instrument data are acquired and downloaded daily for all satellites to one ground station. The satellites can also acquire and store on-board science and housekeeping data for 72 hours for later downlink to the ground station. There is very limited need for mission planning other than downlink to the ground station, a deterministic easily programmable process, and manoeuvres to maintain the orbit.

The safe mode shall ensure the survival of the satellites in case of failures/anomalies. The safe mode brings and maintains the satellite in a safe condition, keeping active only essential functions like power, data handling, AOCS, TT&C, at minimum power consumption. Instruments, as long not used as AOCS sensors, are switched off; and no science data are acquired.
Concerning satellite attitude, the safe mode is designed to be as close as possible to the nominal mode maintaining the spacecraft Earth pointing with relaxed attitude requirements.

The transition into safe mode can be initiated by ground command or automatically onboard by the Fault Detection Isolation & Recovery (FDIR). The role of the FDIR function is to keep the spacecraft operational with a minimum of outages and at the same time to support high spacecraft autonomy. The FDIR system includes all means and procedures on-ground and on-board. The onboard FDIR is defined as a hierarchical system allocating specific responsibilities to each layer and make the next higher layer take over in case a certain layer fails. The specific function of each hardware item in terms of FDIR is defined in the Failure Modes, Effects & Criticality Assessment (FMECA). The FDIR mechanisms are tested to the largest extent during the system level tests.
7 Data processing

Data processing in the context of the technical annex is the processing of raw data from a single satellite towards time series of relevant quantities as observed along the satellite orbit. A single Swarm satellite provides data to generate products up to Level 1b. The combination of these Level 1b data from all Swarm satellites leads to Level 2 products.

**Level 0:**
- Spacecraft and instrument housekeeping data (e.g. temperatures and attitude sensor data)
- Instrument science raw data from absolute scalar magnetometer (ASM); vector flux-gate magnetometer (VFM), 50 Hz; electric field instrument (EFI), up to 16 Hz; accelerometer (ACC), 1 Hz; Global Navigation Satellite System receiver (GNSS)

**Level 1a:**
- Time series of merged relevant housekeeping, instrument and auxiliary data needed for processing and calibrating the measurements
- Post-processed orbit and attitude data

**Level 1b:** Time series of relevant quantities as observed along the orbit, corrected, calibrated and converted to physical units.
- Magnetic field magnitude (1Hz), field vector in the spacecraft frame and in the geophysical North, East, Center (NEC\(^1\)) frame (50 Hz) and coarser sampling rate (e.g. 1 Hz)\(^2\).
- Ion drift vector and electric field in NEC frame electron density and temperature (all 2 Hz).
- Acceleration vector, linear and rotational, in spacecraft frame at 1 Hz
- Position, velocity and attitude of spacecraft (all 1 Hz) derived from Precise Orbit Determination system (POD)

---

\(^1\) The NEC (North, East, Center) frame is a local frame with the origin in the geometric center of the vector feedback magnetometer (VFM). The radial component points from the centre of the VFM towards the centre of the Earth (defined in ITRF). The North (N) and East (E) components point from the center of the VFM towards North and East, i.e. along the local tangent to the meridian, respectively the parallel, of the sphere (defined in ITRF) with radius 1 from the center of the Earth to the center of the instrument.

\(^2\) Down-sampling may be done using approaches developed for the Ørsted mission by Philip B. Stark from University of California, Berkeley (http://128.32.135.2/~stark/Preprints/Oersted/writeup.htm)
The overall data processing chain from Level 0 to Level 1b for the magnetometer package is shown in Figure 7-1.

**Figure 7-1: Data processing magnetometer package**

The data processing flow for the electrical field product towards Level 1b data is schematically depicted in the Figure 7-2.

**Figure 7-2: Data processing flow for the electrical field product**
8 Performance Aspects

The system performance has been established by analysis, error budgeting, and verified by simulations. For this purpose a detailed spacecraft simulator has been developed which is depicted in Figure 8-1.

8.1 Magnetic Field Products

Two magnetic field Level 1b data products are defined in spacecraft and NEC reference frames namely the Magnetic Field Magnitude and the Magnetic Field Vector Components.

These products are derived from the VFM and ASM measurements with the star tracker providing the VFM attitude, as needed for conversion of the measurements from VFM axes to the NEC frame. Two different magnetometer types are employed because the VFM is capable of acquiring vector measurements at high bandwidth and low (short term) error but it is susceptible to drifts, while the ASM, with its high absolute accuracy, serves as the calibration reference.

The error contributions for the magnetic field magnitude product include:

- Residual magnetic disturbance at location
- Measurement accuracy of the ASM
- Uncertainty resulting from position determination errors
- Uncertainty resulting from relative timing errors between all electronic elements in star tracker, ASM, VFM and satellite platform.
- Uncertainty wrt to instrument calibration parameters
The error contributions for the vector field components product include:

- Residual magnetic disturbance at location
- Measurement accuracy of VFM (noise, offset, linearity, etc.)
- Measurement accuracy of the ASM wrt. VFM calibration
- Attitude measurement accuracy of the star tracker
- Optical bench errors (inter-alignment VFM/ star tracker/ optical bench, thermo-elastic bending of optical bench, etc.)
- Boom errors (attitude jitter uncertainty resulting from boom dynamics)
- Uncertainty resulting from position determination errors
- Uncertainty resulting from relative timing errors between the elements, star tracker, ASM, VFM and platform

The boom provides sufficient separation of the magnetic field sensor heads from the spacecraft units that produce stray fields, notably the magnetic torquers. The magnetic cleanliness error is driven only by the instability of disturbing fields from magnetic materials. The unknown variations of the stray fields over time also contribute. The temporal variation of the stray fields due to platform units is assumed to be smaller than 10% of the total value.

Table 8-1 provides an overview of the magnetic stray fields at the sensor location for one of the proposed satellite configurations. Main contributors to the bus unit's stray field are the fuel tanks, OBC electronic and the TT &C subsystem.

<table>
<thead>
<tr>
<th>Source</th>
<th>VFM sensor Location</th>
<th>ASM sensor Location</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>Distance [m]</td>
<td>Stray Field [nT]</td>
</tr>
<tr>
<td>Magnet Torquer</td>
<td>6.25</td>
<td>8.15</td>
</tr>
<tr>
<td>VFM</td>
<td>1.5</td>
<td>1.1</td>
</tr>
<tr>
<td>Star Tracker</td>
<td>0.4</td>
<td>0.3</td>
</tr>
<tr>
<td>ASM</td>
<td>1.5</td>
<td>1.3</td>
</tr>
<tr>
<td>Bus units</td>
<td>~5</td>
<td>1.3</td>
</tr>
<tr>
<td>Total Uncertainty</td>
<td>~5</td>
<td>1.3</td>
</tr>
</tbody>
</table>

*Presently insufficient information regarding the ASM instrument bias is available to assess the stability over 3 months. However, considering the functional principal and the low susceptibility to environmental conditions (e.g. temperature sensitivity, 1 pT/°C) and the instrument bias stability is likely well below 0.1 nT. The essential contribution to bias variation over this period will result from changes of the static stray field induced by the platform (<0.06 nT uncertainty) (considering the magneto torquer stray field uncertainty as a random contribution insignificant in this context). The stability requirement will be met.

**Table 8-2: Performance Assessment of Magnetic Field Magnitude Product**
The stability error of the magnetic field vector product is driven by three contributions of about equal magnitude: VFM offset stability, impact of pointing uncertainty and impact of residual uncertainty of stray field at VFM location.

Apart from the contributors listed in Table 8-3 the following error sources have been assessed and found to produce no significant contribution to the stability budgets: impact of pointing rate uncertainty to ASM performance and impact of boom bending to stray field stability.

<table>
<thead>
<tr>
<th>Magnetic Field Vector Product Requirements</th>
<th>Error Budget (2σ Error Contribution)</th>
<th>Numerical Simulation Results</th>
</tr>
</thead>
<tbody>
<tr>
<td>Vector magnetic field random error &lt;1 nT (2σ)</td>
<td>0.98 nT</td>
<td>[0.49, 031, 0.44] nT</td>
</tr>
<tr>
<td>Stability &lt;1nT (2σ) @ 1 year</td>
<td>same as above since noise in measurement bandwidth is negligible</td>
<td></td>
</tr>
<tr>
<td>Vector magnetic field random error with spectral density of 0.1 nT/√Hz at 0.1 Hz linearly decreasing to 10 pT/√Hz at 20 Hz in the AC range</td>
<td>The VFM instrument measurement noise itself meets the spectral requirement. Additional noise sources result from varying magnetic moments of bus units (i.e. resulting from time varying electrical currents in equipment circuits) and from stray field variations due to boom vibrations. Presently there is no indication that any realistic stray field / boom vibration interaction yields a significant impact. Currently no data are available on magnetic moment variation of bus units in the high frequency range. Simulator runs indicate that an unrealistic high fraction of the overall stray field (about 5%) would need to be in the high frequency range to exceed the sensor noise.</td>
<td></td>
</tr>
<tr>
<td>Pointing Knowledge Error using pair of star tracker heads</td>
<td>2.5 arc sec ; Optical Bench: 0.28 arc sec - 0.56 nT</td>
<td></td>
</tr>
<tr>
<td>Timing uncertainty between ASM, VFM, ASC: 5 ms</td>
<td>Error from timing uncertainty versus ASM (rms) 0.5 ms - 0.17 nT</td>
<td></td>
</tr>
<tr>
<td>Magnetic Cleanliness at VFM location: &lt;0.5nT</td>
<td>0.44 nT</td>
<td></td>
</tr>
<tr>
<td>ASM long term stability</td>
<td>0.26 nT</td>
<td></td>
</tr>
<tr>
<td>Noise</td>
<td>0.1 nT</td>
<td></td>
</tr>
<tr>
<td>Linearity</td>
<td>0.12 nT</td>
<td></td>
</tr>
<tr>
<td>Offset</td>
<td>0.58 nT</td>
<td></td>
</tr>
</tbody>
</table>

**Table 8-3: Performance assessment of magnetic field vector component product**

The following diagrams provide an extract of the simulation results obtained by the system simulator. Figure 8-2 shows the errors of the ASM Level 1b data output [nT] and the subsequent Figure 8-3 shows the errors of the VFM Level 1b data

**Figure 8-2:** *ASM, error of post-processed magnetic field magnitude*
8.2 Electrical Field Products

The EFI instrument is used to derive the following products:

- Ion Distribution – Density - Composition
- Ion Drift Vector in NEC frame
- Electric Field in NEC frame
- Electron Density
- Ion Temperature

The ion drift vector product is synthesized from the components of the relative ion velocity distributions (RPA and IDM) corrected for impact of the spacecraft potential measured with the PLP. Electric fields at the orbital altitude of the Swarm satellites can be effectively derived from the vector ion drift velocities using the drift equation:

\[ E = -v \times B \]

\( E \) – electric field vector, \( B \) – magnetic field vector, \( v \) – ion velocity

The expected performance for the EFI products is provided in Table 8-4. The electric field measurement performance is separately budgeted in Table 8-5. The measurement accuracy is compliant with the requirements. The achievable accuracy is dominated by the measurement accuracy of the PLP, which is used to determine the spacecraft potential. The properties of this measurement also determine the off-set stability of the electric field measurement.

<table>
<thead>
<tr>
<th>Electrical Field Components Product and associated Instrument Requirements</th>
<th>Error Budget</th>
<th>Numerical Simulation Results</th>
</tr>
</thead>
<tbody>
<tr>
<td>Random error: &lt;3mV/m (2( \sigma ))</td>
<td>2.7 mV/m</td>
<td>[0.44, 0.69, 0.40] mV/m</td>
</tr>
<tr>
<td>Stability of Bias: &lt;1mV/m/month (2( \sigma ))</td>
<td>Compliant (driven by Langmuir Probe bias)</td>
<td></td>
</tr>
<tr>
<td>Dynamic Range of electrical field measurements: ( \pm 0.2 ) V/m</td>
<td>( \pm 0.18 ) V/m *</td>
<td></td>
</tr>
<tr>
<td>Dynamic Range of ion density measurements: ( 10^7 - 10^{13} ) ions/m^3</td>
<td>( 5 \times 10^7 - 5 \times 10^{12} ) ions/m^3</td>
<td></td>
</tr>
</tbody>
</table>

*Minor deviations in the dynamic range are considered acceptable

Table 8-4: EFI performance assessment
The non-gravitational acceleration acting on the Swarm spacecraft CoM is measured and provided as a data product in the spacecraft coordinate system.

The error contributions for the electric field vector product include:

- ACC thermal bias variation
- ACC long term bias drift
- Knowledge of distance between ACC and CoM
- Spacecraft attitude knowledge

The performance of the ACC Level 1b products has been analysed considering that the accelerometer proof mass is not precisely located at the CoM. In the present design the CoM will move by a few centimeters due to the fuel consumption. During spacecraft integration the accelerometer position can be adjusted according to the measured CoM. Therefore the maximum distance of the ACC proof mass from the CoM over the mission time is limited to a few centimeters. Nevertheless a correction of the measured acceleration using knowledge of CoM position, pointing rate and pointing acceleration is needed.

Accelerometers are very sensitive to vibrations in the frequency range between 0.1 Hz and 1000 Hz (especially 1 Hz and 100 Hz). The AOCS of Swarm is designed such that the resulting accelerations remain within the dynamic range of the accelerometer.

The performance assessment overview is provided in Table 8-6.

### Table 8-5: Electric field measurement performance

<table>
<thead>
<tr>
<th>Contribution</th>
<th>Assumption (pessimistic)</th>
<th>Conversion to velocity error</th>
<th>Result</th>
</tr>
</thead>
<tbody>
<tr>
<td>Instrument Accuracy IDM</td>
<td>cross: 2 m/s, sum: 10 m/s</td>
<td>vector sum</td>
<td>10 m/s</td>
</tr>
<tr>
<td>Altitude Uncertainty from Startracker to ERI</td>
<td>$\alpha = 0.1\degree$</td>
<td>$v_0 = \gamma_{\text{Acc}} \tan \alpha$</td>
<td>13 m/s</td>
</tr>
<tr>
<td>CHSS uncertainty</td>
<td>1 m/s</td>
<td>1:1</td>
<td>1 m/s</td>
</tr>
<tr>
<td>BCG potential uncertainty (effect in terms of velocity error for CIR ions)</td>
<td>PLL resolution: 0.05 s</td>
<td>$\Delta v = \sqrt{v_0^2 + \frac{2 \cdot \Delta y_{\text{BCG}} \cdot \Delta y_{\text{BCG}}}{m_{\text{BCG}}}}$</td>
<td>39 m/s</td>
</tr>
<tr>
<td>Ambient magnetic field uncertainty</td>
<td>0.256 A m$^{-2}$ dipole at 0.5 m from sensor</td>
<td>see above Eq. 3.4-2</td>
<td>&lt;1 m/s</td>
</tr>
<tr>
<td>Total velocity Error</td>
<td>-</td>
<td>-</td>
<td>43 m/s</td>
</tr>
<tr>
<td>Equivalent electrical field error</td>
<td>at specified maximum magnetic field of 65 T</td>
<td>-</td>
<td>2.8 mV/m</td>
</tr>
</tbody>
</table>

**Table 8-6: Accelerometer performance assessment**
The following diagrams provide an extract of the simulation results obtained by the system simulator. Figure 8-4 shows the errors of the ACC level 1b data for the simulation scenario 1 (Date: 20.06.98) and Figure 8-5 shows the errors of the ACC level 1b data output for the simulation scenario 5 (Date: 20.06.01). The three-year time difference between the two simulations shows that the propellant depletion has not affected the accelerometer performance.

![Figure 8-4: Errors of the ACC level 1b data](image1)

![Figure 8-5: Errors of the ACC level 1b data output](image2)

These results include also the periods where the spacecraft boom and body are excited due to thruster firing (observe the spikes in the Figures above). The following colour assignment holds for the Figures containing vector-type data: Red: 1st axis (x) Green: 2nd axis (y) Blue: 3rd axis (z). No "red" date is visible as the accelerations along the x-axis (orbit radius) are negligible.

**Performance Summary**

The predicted performance based on error budgets and the numerical simulation results show that the required performance can be met with the proposed satellite bus and the payload complement.

The proposed instruments either have already demonstrated adequate performance in flight (VFM, EFI, ASM-configuration B) or the design is based on heritage of instruments that already demonstrated adequate performance (ACC) or have demonstrated adequate performance as breadboard (ASM configuration A).

The measurement environment as provided by the spacecraft with respect to magnetic cleanliness, low acceleration environment and spacecraft electrical potential is adequate to allow the full exploitation of the instrument capabilities.
9 Programmatic Aspects

This chapter briefly outlines the programmatic aspects of the Swarm implementation as analysed by the Phase A contractors. The technical maturity of the Swarm mission is consistent with a launch in 2008/2009. The platform and the payload have a strong heritage from ongoing missions.

No critical item has been identified neither for the instruments nor for the platform.

The phase B of the Swarm implementation phase is planned with duration of 8 months. The objectives of this phase are the selection of technical solutions for the system concept identified in phase A and their further definition. A System Requirement Review (SRR) is planned early in the phase B to review the System Requirement Document (SRD) and the system specification delivered by the prime contractor. After the SRR the main part of the phase B will be dedicated to refine the equipment specifications in order to have agreed versions with the subcontractors / suppliers at the time of the Preliminary Design Review (PDR), which is planned at the end of the phase B.

The phase C/D covers all activities required to have the space segment designed, built, verified and tested. The critical design review (CDR) at the end of phase C, is the stage where all development testing is completed and the detailed design is established. The phase C/D is scheduled with a duration of 38 months and ends with the Flight Acceptance Review (FAR) where records of the spacecraft integration, ground segment operation, critical item status and launch campaign plan and procedures will be reviewed. The high maturity of Swarm allows the adoption of a proto-flight model (PFM) approach for the first satellite. For the other two satellites only the flight models (FM) will be built. Concepts have been developed to ensure an efficient manufacturing, assembly, integration and verification flows.

Phase E starts with the transportation of the satellites to the launch site after the successful Flight Acceptance Review (FAR). This phase comprises the launch campaign, launch and in-flight acceptance of space elements and it corresponds to operation and maintenance of the space segment. Phase E is divided into two sub-phases E1 and E2. The sub-phase E1 is an overall test and commissioning phase with a scheduled duration of 4 to 6 months. The sub-phase E2 is the utilization phase itself with a scheduled duration of 4 years.

To account for the possibility of damage of flight hardware during handling and testing on ground, provisions will be made to ensure replacement of components and repair / refurbishment within a duration of some weeks.
An overall schedule of the Swarm mission implementation and operation is depicted in Figure 9-1. The schedule is assuming a phase B kick-off beginning 2005 and a launch of the spacecraft end of 2008/beginning 2009.

<table>
<thead>
<tr>
<th>Project Task / Phase</th>
<th>Duration</th>
<th>2003</th>
<th>2004</th>
<th>2005</th>
<th>2006</th>
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<th>2008</th>
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<th>2010</th>
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<tr>
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</table>

*Figure 9-1: Swarm implementation schedule*

### 9.1 Critical Areas

All instruments intended to be used for the Swarm mission have a strong heritage from previous space missions. For the ASM, space pre-qualification is ongoing for a higher performance sensor already demonstrated on ground versions. The ASM pre-qualification will be completed in phase B. No other pre-development activities are necessary.

There are no risks associated to the platform. The ground segment is based on ESA infrastructure. The PAE can use the heritage of Ørsted and CHAMP concerning instrument processors and data processing chains.

The Swarm mission is very timely from the technical point of view as it augments and builds upon the considerable heritage accumulated by Ørsted, SAC-C and CHAMP.