# Gaia
# Mission Requirement Document

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approved by G. Sarri (Gaia Project Manager)
reference GAIA-EST-RD-00553
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# DISTRIBUTION

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# CHANGE LOG

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| 21-5-2013 | 3 | 1 | 46 | Clarification of SYS-415 (Delta-V requirements) as agreed during GSRR |

# Table Of Contents

# 1 INTRODUCTION

The present Mission Requirements Document (MRD) specifies the general set of requirements applicable to the Gaia mission.

The Gaia mission shall be divided into the following functional elements: the Space Segment and its Ground Support Equipment, the Launcher and the Ground Segment.

The Launcher and the Gaia Ground Segment are provided by ESA.

## 1.1 Background

In October 2000, the Gaia astrometry mission was approved as Cornerstone N°6 in the current Horizons 2000 ESA Scientific Programme, with a launch date target of 2010/2012.

The GAIA mission will rely on the proven principles of ESA's Hipparcos mission to solve one of the most difficult yet deeply fundamental challenges in modern astronomy: to create an extraordinarily precise three-dimensional map of about one billion stars throughout our Galaxy and beyond. In the process, it will map their motions, which encode the origin and subsequent evolution of the Galaxy. Through comprehensive photometric classification, it will provide the detailed physical properties of each star observed: characterizing their luminosity, temperature, gravity, and elemental composition. This massive stellar census will provide the basic observational data to tackle an enormous range of important problems related to the origin, structure, and evolutionary history of our Galaxy.

## 1.2 Objectives

The objectives of this document are:

- to define the main system requirements, and, therefore, represent inputs from the European Space Agency (ESA) for generating a controlled set of requirements from which further lower-level documents shall be derived and documented in the relevant project plans and specifications;
- to specify the interfaces with the external elements including the Ground Segment, the Launcher and the Ground Support Equipment.

## 1.3 Conventions

All requirements in this specification that require verification are marked with a unique reference and are indicated in italic.

Requirements are identified in the following way XXXX – nnn. Where, XXXX is the requirement *theme* and is made of 3 to 4 letters, and nnn is the absolute requirement identifier (sequential nummber of 3 digits).

# 2 DOCUMENTS

In case of conflict between statements/requirements in this document and other applicable documents, the conflict shall be brought to the attention of the ESA Gaia Project Manager.

In the case of conflict between this document and reference documents, the requirements of this document shall take precedence.

The following documents are applicable at the issue and revision on the date of ITT release or at the latest issue and revision agreed with the Contractor.

## 2.1 Applicable Documents

Applicable Documents are normative documents, whose requirements are mandatory unless otherwise specified in the higher-level requirements documents contained in the Gaia data package.

### 2.1.1 General Design Applicable Documents

| AD#-# | TITLE | REFERENCE |
|---|---|---|
| AD1-1 | Gaia - AIV Requirement Document | GAIA-EST-RD-00459 |
| AD1-2 | Gaia - Product Assurance and Safety Requirements | GAIA-EST-RD-00496 |
| AD1-3 | Gaia - Operation Interface Requirement Document | GAIA-ESC-RD-00514 |
| AD1-4 | Gaia - Space/Ground Interface Control Document Volume 1 | GAIA-ESC-ICD-00515 |
| AD1-5 | Gaia - Space/Ground Interface Control Document Volume 2 (Generic Packet Structure) | GAIA-ESC-ICD-00516 |
| AD1-6 | Gaia - Environmental Specification | TEC-EES-05-011/JS |
| AD1-7 | Withdrawn | |
| AD1-8 | Gaia – Statement of Work | GAIA-EST-SOW-00444 |
| AD1-9 | Gaia – Photon Flux Distribution for Reference Stars | GAIA-EST-TN-589 |
| AD1-10 | Gaia – Management Requirements | GAIA-EST-RD-00546 |

### 2.1.2 Launcher Applicable Documents

| AD#-# | TITLE | REFERENCE |
|---|---|---|
| AD2-1 | Soyuz Launcher Manual at CSG* | SOY-100 MUSG |
| AD2-2 | Withdrawn | |
| AD2-3 | Withdrawn | |

| | | |
|---|---|---|
| AD2-4 | Withdrawn | |
| AD2-5 | Withdrawn | |
| AD2-6 | Soyuz User's Manual | ST-GTD-SUM-01 |
| AD2-7 | Launch System/Spacecraft Interface Control File | SOY-DCI-1/50640-B-AE |

* See paragraph 4.3 of AD 1-8.

### 2.1.3 ESA ECSS Applicable Documents

| AD#-# | TITLE | REFERENCE |
|---|---|---|
| AD3-1 | Space Engineering - System Engineering – Functional & Technical Specifications | ECSS-E-10 Part 6A |
| AD3-2 | Space Engineering - System Engineering – Requirements & Process | ECSS-E-10 Part 1B |
| AD3-3 | Space Engineering - Verification | ECSS-E-10-02A |
| AD3-4 | Space Engineering - Testing | ECSS-E-10-03A |
| AD3-5 | Space Engineering - Space Environment | ECSS-E-10-04A |
| AD3-6 | Space Engineering - Electrical and Electronic | ECSS-E-20A |
| AD3-7 | Space Engineering – Communications – Part 1: Principles and Requirements | ECSS-E-50 Part 1A |
| AD3-8 | Space Engineering - Mechanical – Thermal Control | ECSS-E-30 part 1A |
| AD3-9 | Space Engineering - Mechanical – Structural | ECSS-E-30 part 2A |
| AD3-10 | Space Engineering - Mechanical – Mechanisms | ECSS-E-30 part 3A |
| AD3-11 | Space Engineering - Mechanical – Propulsion | ECSS-E-30 part 5.1A |
| AD3-12 | Space Engineering - Mechanical – Pyrotechnics | ECSS-E-30 part 6A |
| AD3-13 | Space Engineering - Mechanical – Mechanical Parts | ECSS-E-30 part 7A |
| AD3-14 | Space Engineering - Mechanical – Materials | ECSS-E-30 part 8A |
| AD3-15 | Space Engineering - Fracture Control | ECSS-E-30-01A |
| AD3-16 | Space Engineering - Software – Part1: Principles and Requirements | ECSS-E-40 part 1B |
| AD3-17 | Lossless Compression | CCSDS 121.0-B.1 |
| AD3-18 | Space Engineering - Radio Frequency and Modulation | ECSS-E-50-05A |
| AD3-19 | Space Engineering - Spacewire – Links, nodes, routers and networks | ECSS-E-50-12A |
| AD3-20 | Ranging | PSS-04-104 |
| AD3-21 | TM Synchronisation and Channel Coding, Blue Book | CCSDS.131.0-B-1 |
| AD3-22 | Space Engineering - Control Engineering | ECSS-E-60A |
| AD3-23 | On-Board Interfaces | MIL1553-B-Notice 4 |
| AD3-24 | Space Engineering - Telemetry and Telecommand Packet Utilisation | ECSS-E-70-41A |
| AD3-25 | Space Engineering - Photovoltaic Assemblies and | ECSS-E-20-08A |

| | | |
|---|---|---|
| | Components | |
| AD3-26 | Software Product Assurance | ECSS-Q-80B |
| AD3-27 | TM space data link protocol, Issue 1 September 2003 | CCSDS.132.0-B-1 |
| AD3-28 | TC space data link protocol, Issue 1 September 2003 | CCSDS.232.0-B-1 |
| AD3-29 | Space Packet Protocol, Issue 1 September 2003 | CCSDS.133.0-B-1 |
| AD3-30 | Space Engineering - Software – Part 2: Document Requirements Definition | ECSS-E40 Part 2B |

## 2.2 Reference Documents

| RD#-# | TITLE | REFERENCE |
|---|---|---|
| RD1-1 | CREMA – Consolidated Mission Analysis | GAIA-ESC-RP-001 |
| RD1-2 | Gaia Object Count Model Access Software | GAIA-UL-010, Rev 1.1 |
| RD1-3 | Gaia Payload Performance Calculations Guidelines | GAIA-EST-TN-00539 |
| RD1-4 | Summary of Sampling Schemes for SM, AF, BP, RP, and RVS | GAIA-CA-TN-NBI-EH-176-3 |
| RD1-5 | BBP Photometric System Evaluation | UB-PWG-028 |
| RD1-6 | MBP Photometric System Evaluation | UB-PWG-029 |
| RD1-7 | Pyxis S/W – Version 2.3 | 2005tdug.conf..335A |
| RD1-8 | Multi-pass scanning across Baade’s Window | Gaia–LL–058 |
| RD1-9 | Gaia Chromaticity Calibration and the BBP filter shape | Gaia–LL–049 |
| RD1-10 | Manufacturing of Filters | GAIA-CUO-146 |
| RD1-11 | GAIA – Composition, Formation and Evolution of the Galaxy – Report on the Concept and Technology Study | ESA-SCI (2000) 4 – July 2000 |
| RD1-12 | GAIA S/C Launcher IF Mission Description | AE/DC/ST/05-92 |
| RD1-13 | GAIA-SOYUZ interface | AE/DC/ST/05-91 |
| RD1-14 | Conventions Document | GAIA-ARI-BAS-003 |

# 3 MISSION REQUIREMENTS

## 3.1 Science Objectives

The scientific measurements of the Gaia mission cover three principal areas: astrometry (the measurement of stellar position, parallax, and proper motion); photometry (the measurement of photometric magnitudes in a number of different spectral bands and at each possible measurement epoch); and spectroscopy (for the acquisition of radial velocities and astrophysical parameters). The basic scientific mission objective consists of generating on-ground, from the measurements made on the satellite, a star catalogue comprising about one billion objects, to a magnitude completeness limit ~20.0 mag, and containing these parameters for as many objects as possible (astrometry and photometry to the above completeness limit, and spectroscopy for object as faint as possible.

Specific attention needs to be given to binaries and multiple stars in all instruments and across all domains of angular separation and magnitude differences, including star detection and measurement. Brown dwarfs and planetary systems will be investigated including planet detection and orbit determination, the detection of multi-planet systems, and the detection of extra-solar transits from photometric data. Solar system (i.e., rapidly moving) objects must be detected and observed down to the limit of ~20 mag. Objects beyond our Galaxy (sufficiently bright stars, galaxies with compact nuclei, and quasars) will also be detected and observed in the same way as stars. Relativistic light bending in the Solar System will be measured to the accuracy limits.

## 3.2 General Requirements

*MIS - 010* *The Gaia spacecraft shall support the science objectives and requirements as specified in chapter 4.*

*MIS - 020* *The spacecraft design shall enable the operational control by the ground segment during all mission phases and modes in both nominal and contingency situations as per AD 1-3.*

## 3.3 Gaia Ground Segment Definition

The ground segment consists of Mission Operations Centre, Science Operation Centre and Ground Stations.

Mission Operation Centre:

- Spacecraft operations, consisting of mission planning, spacecraft monitoring and control, and orbit and attitude determination and control;
- Scientific instrument operations, consisting of implementation of the scanning law, command the Gaia instruments, pointing and auxiliary data, collection of science and transmission to the Science Operation Centre, carry out quality control analysis of the science telemetry data, and intervention in case of payload anomalies according to pre-planned procedures.

Science Operation Centre:

- Analysis of the Science data received from MOC;
- Instrument characterisation and calibration;
- Science data reduction and production of the final scientific products
- Preparation of the Science Operation Plan.

Ground Station:

- During the LEOP and during critical mission phases, the 15-m ESA station at Kourou will be used. The 15-m ESA station at Villafranca will be used as backup;
- During transfer and commissioning the ESA station at Perth/New Norcia will be used for contact with the spacecraft. The ESA station in Kourou will be used as backup;
- The 35-m ESA station at Cebreros will be used for contact with the spacecraft during all other mission phases (see Section 3.6 for the identified mission phases). The 35-m ESA station at Perth/New Norcia has been identified as alternative ground station.

All ESA ground stations interface to the mission operations centre at ESOC.

For the Gaia operations, the spacecraft will collect scientific data 24 hours per day, which means that scientific observation phase will continue during the Ground station visibility period (telecommunications period). The telecommunication period will be used for uplink the telecommands for later execution and dumping of the stored data as well as real-time transmission of science and housekeeping data.

## 3.4 Launcher Vehicle and Launch Site

*MIS - 030* *Gaia satellite shall be launched by Soyuz-Fregat from Kourou.*

*MIS - 040* *Injection into the cruise trajectory to L2 shall be performed by the Soyuz-Fregat upper stage after the first parking orbit.*

*MIS - 050* *To ensure compatibility with Launcher facilities and Launch vehicle ADs reported in par 2.1.2 shall be used for the spacecraft (and GSE) design.*

## 3.5 Mission Lifetime

*MIS - 060* *The nominal mission duration shall be five years.*

The nominal mission lifetime shall commence at the start of scientific observations, i.e., after full completion of:

a) the transfer to L2 and the injection of the spacecraft into the operational Lissajous orbit around L2, and of
b) the spacecraft and payload commissioning and calibration phase.

*MIS - 070* *The extended mission shall be one year beyond the nominal mission duration.*

## 3.6 Mission Phases

Four mission phases shall be considered; the Pre-launch phase, the Launch and Early Orbit Phase (LEOP), the Transfer & Commissioning phase, and the Operational phase.

*MIS - 080* *Up-link and down-link capabilities on the operational orbit shall be sized for a typical ground station operation period of 8 hours/day including overheads as per AD 1-4 and AD 1-5.*

*MIS - 090* *Throughout all mission phases and orbits, ground contact shall be possible at all times. Exceptions shall be clearly identified and reported to ESA.*

*MIS - 100* *Housekeeping telemetry shall be continuously generated and recorded in all modes of operations, including in Survival Mode.*

Note: when a Subsystem or Instrument - which nominally generates or relays HK Telemetry - is in a specific contingency mode (as processor halted/reset), this requirement does not apply to the concerned Subsystem or instrument.

### 3.6.1 Pre-launch Phase

This phase encompasses all pre-launch operations (and starts some weeks before launch). Activities include the final launch simulations, data flow tests, battery reconditioning (if necessary), filling of the propellant tanks, etc. The pre-launch phase ends with the lift-off event.

*MIS - 110* *The spacecraft design shall be compatible with the facilities at the launch site. Specific non-standard facility modifications and/or extensions that might have to be implemented by the launcher authority shall be identified.*

*MIS - 120* *Prior to lift-off the spacecraft shall be in an electrically active state. Thus the on-board TT&C and Data Handling subsystems shall be in an operational mode that is able to receive telecommands, handle telemetry packets and perform on-board monitoring functions, even though the transmitter function is OFF.*

### 3.6.2 Launch and Early Orbit Phase (LEOP)

This phase follows the pre-launch phase and commences at launch (vehicle lift-off). It includes the ascent phase, injection into the low-altitude parking orbit, and injection into the L2 transfer (cruise) orbit. At the end of the launch phase, the orbital unit consists of the satellite plus the launcher upper stage.

*MIS - 130* *The spacecraft shall autonomously detect separation from the launch vehicle. In case of Spacecraft failures, the detection of the separation sequence shall autonomously restart at the step where the failure occurred.*

*MIS - 140* *After separation from the Fregat upper stage, the spacecraft shall autonomously activate one of the transmitter channels and thus allow the ESA ground station network to establish the first contact.*

*MIS - 150* *Full communication capabilities shall be available after spacecraft separation from the Fregat upper stage.*

*MIS - 155* *Full AOCS capabilities shall be available after spacecraft separation from the Fregat upper stage.*

MIS - 160 *After separation from the Fregat upper stage, the spacecraft shall autonomously reduce any remaining body rates.*

MIS - 170 *The spacecraft shall autonomously re-orientate itself to achieve the required solar array alignment with respect to the sun vector and thus terminate any battery discharge.*

MIS - 180 *Following separation from the launch vehicle the spacecraft shall provide the necessary attitude information for attitude reconstitution on the ground.*

MIS - 190 *Spacecraft-launcher separation and subsequent deployment events shall be monitored.*

MIS - 200 *During this phase, all critical operations shall be planned and executed also without the need of ground contact.*

### 3.6.3 Transfer & Commissioning Phase

After completion of the LEOP, when the spacecraft is in its transfer (cruise) orbit toward L2, the commissioning phase can commence. The purpose of the in-flight commissioning of the payload is to establish all initial calibrations and verify the nominal performances of all instruments in all modes to allow on-ground scientific data processing to commence with the accuracy- and precision-levels required to fulfill the scientific objectives of the mission. In-flight commissioning and initial calibration phase activities may include, but are not limited to, the full characterization of:

- the satellite orbit around L2;
- the AOCS system, including feedback/control loops, pointing and rate error budgets, initialization of the scanning law, etc.;
- geometrical calibrations of the focal planes, including optical distortions, CCD positions and rotations, time delays, etc.;
- telescope and instrument transmissions and optical properties, including spectral throughput, wave-front errors, straylight, initial PSFs, vignetting, ghost images, etc.;
- on-board hard/software, including FoV discrimination, sky-background determination, object detection, confirmation, selection, handling of prompt particle events, etc.;
- CCD and detection chain properties, including sensitivities, MTF, fringing, bias, gain, dark/white pixels, column defects, traps, TDI gates, charge injection, full noise properties, EMC coupling effects, CTI effects, linearity, magnitude scale, binning, etc.;

- on-board clocks, including science data time tagging, compression, encoding, packetisation, and storage, etc.;
- the basic-angle monitoring device performance.

*MIS - 210* *The commissioning phase shall start after launcher separation and shall last at most 6 months.*

*MIS - 215* *During the transfer phase, the satellite shall keep an attitude that satisfy the operational conditions on sun aspect angle and Earth aspect angle as reported in RD 1-1.*

*MIS - 220* *The spacecraft AOCS and propulsion system shall support all post launcher-separation attitude and orbit manoeuvres required to correct for injection errors.*

*MIS - 230* *During the spacecraft transfer (cruise) orbit toward L2 and during insertion into the Lissajous orbit around L2, the propellant budget shall account for the requirements to perform the following manoeuvres as reported in RD 1-1:*

- *Compensation for perigee velocity variation;*
- *Removal of launcher dispersion;*
- *Orbit correction and maneuvers required during the cruise trajectory;*
- *L2 orbit insertion;*
- *Correction manoeuvres after insertion;*
- *Station keeping manoeuvres.*

*MIS - 240* *During the spacecraft cruise towards L2 the complete check-out of the spacecraft functions and the verification of all sub-system performances shall be performed.*

*MIS - 250* *During the spacecraft cruise towards L2 the checkout of the Gaia instruments shall be performed to the maximum extent possible compatible with the spacecraft cruise attitude.*

*MIS - 260* *The commissioning phase shall include the instrument performance verification covering all the calibrations and the determination of the performances in all modes.*

The end of the commissioning phase, and the start of the operational phase, can be achieved only when the spacecraft has achieved the operational Lissajous orbit around

L2 and when the complete spacecraft and science instrument functions have been determined.

### 3.6.4 Operational Phase

This phase starts after the completion of the commissioning phase. It covers the satellite useful lifetime up to the end of scientific observations.

*MIS - 270* *The spacecraft design shall be compatible with the operational Lissajous orbit around the second Lagrangian libration point in the Earth/Moon¬Sun system. This point is denoted as L2.*

The L2/Earth distance is between 1.33 and 1.68 million km as per RD 1-1 (assuming a solar-S/C-Earth of 15º see figure 3.6.4-1). The Gaia's operational Lissajous orbit chosen around L2 will allow continuous scientific observations during the nominal and extended mission lifetime and the phase of the Lissajous orbit around L2 will be selected such that no solar eclipse caused by the Earth will occur during the operational phase (nominal plus extended mission).

*MIS - 300* *The Spacecraft design shall account for Solar eclipses caused by the Moon as reported in the RD 1-1.*

*MIS - 310* *Considering the attitude constraints imposed by Gaia's scanning law (see Chapter 4), the operational orbit shall be such that the RF beam-pointing requirement for science data downlink shall be satisfied during all ground contacts.*

*MIS - 320* *Throughout the operational phase the spacecraft shall be able to maintain its attitude as specified by the requirements on the scanning law in the operational orbit.*

*MIS - 330* *Orbit maintenance manoeuvres during the operational phase shall be performed in accordance to RD 1-1.*

*MIS - 340* *The selection of the size (amplitude) of the operational Lissajous orbit around L2 shall be such that (during the spacecraft spin-axis precession) the spacecraft–Earth vector will move within a cone, with an opening angle (half angle) up to 15 degrees, the axis of which is centred on the Sun direction.*

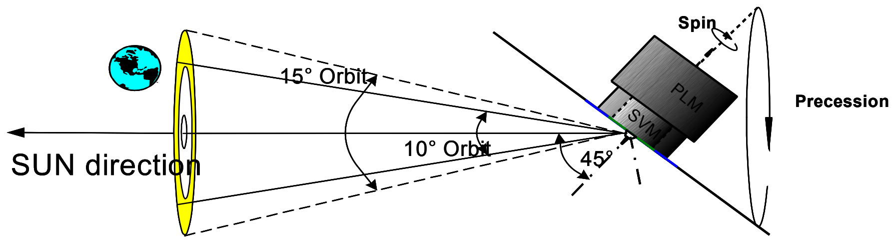



**Figure 3.6.4-1**: Gaia–Earth–Sun angle.

## 3.7 Launch Window Constraints

*MIS - 350* *The spacecraft design shall be compatible with a daily launch slot (lift-off event) of 5 minutes and a time from lift off to spacecraft separation between 15 minutes and 2 hours.*

The launch will be possible on every day during the year except 3 days per month to avoid close approaches to the moon. The liftoff time and the time of the launcher upper stage burn (Fregat) after up to one full revolution in a low circular parking orbit around the Earth, will be different for each launch day and will be subject of the launch window calculation.

*MIS - 360* *In case of a launch abort the spacecraft can be reconnected within 5 hours. The next attempt will then be on the following day. The spacecraft design shall comply with this scenario*

## 3.8 Mission Design Robustness

*MIS - 370* *The spacecraft shall be designed in accordance with the following LEOP & transfer phase parameters contained in the following envelope:*
*- S/C to Earth distance as function of time as shown in* Figure 3.8-1.

- *Sun-S/C to Earth angle as a function of time as shown in* Figure 3.8-2.
- *maximum duration between lift-off and injection into the transfer can be found in AD2-7.*
- *minimum S/C Moon distance of 72,000km*
- *maximum S/C Sun distance of 154,800,000km*

Note (1): The minimum S/C Earth distance at Fregat separation is >750 Km
Note (2): During transfer (after day 2) the Sun-S/C-Earth angle is between 2 to 20 deg. The constrained of maximum Sun-S/C-Earth angle of 15deg is active only after the insertion burn.
Note (3): the max S/C Sun distance is valid also for the operational orbit.

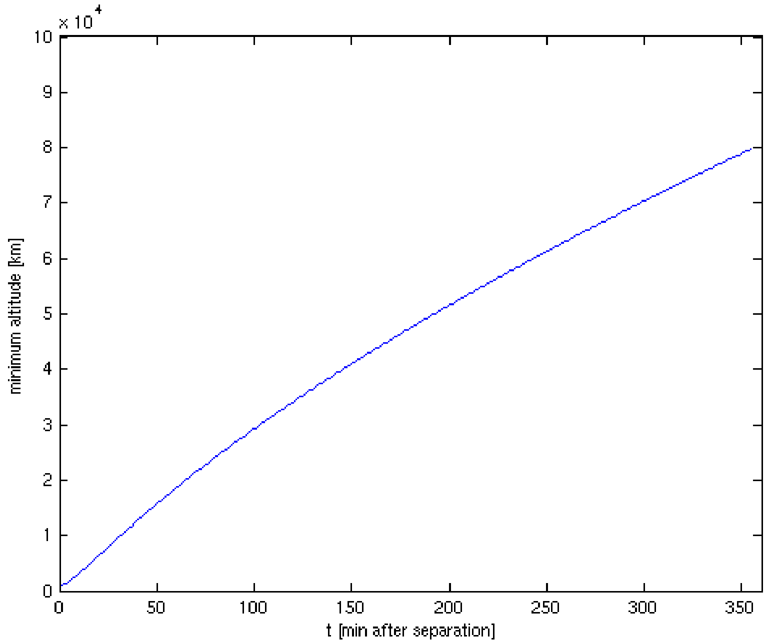



**Figure 3.8-1: Evolution of Earth distance from Fregat separation**

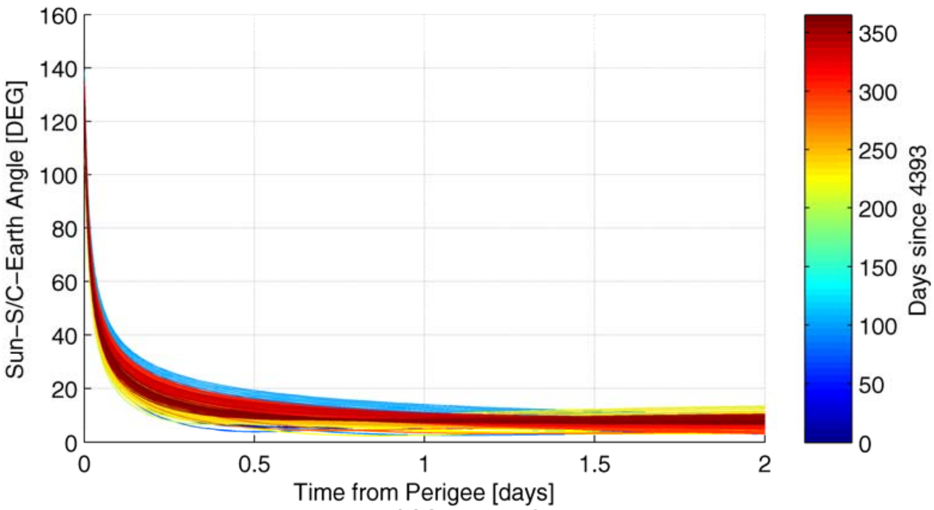



**Figure 3.8-2: Evolution of SSE angle from Fregat separation**

*MIS - 380* *The spacecraft shall be designed in accordance with the following Lissajous family orbits contained in the rectangular box with the following dimensions:*
*±175000km in x*
*±400000km in y*
*±230000km in z*
*symmetric around L2, which has the coordinates (1.50768x10^6km,0,0) in the rotating frame.*

Note (1): The rotating frame is defined with the x-y-plane being the ecliptic and the x-axis pointing in the Sun-to-Earth direction. The origin is the barycentre of the Earth-Moon system.

Note (2): For calculation purposes, the orbit parameters provided in Appendix 1 can be taken as reference.

Note (3): As recalled several times in the MRD, detailed explanations/information about mission parameters for all mission phases are given in RD1.1.

# 4 SCIENTIFIC REQUIREMENTS

## 4.1 General Requirements

Gaia performs micro-arcsecond global astrometry by linking objects with both small and large angular distance in a network in which each object is connected to a large number of other objects in every direction.

Photometry will be performed to:

i) provide the required accuracy for the chromatic correction of the astrometric observations, and
ii) provide astrophysical information for all observed object

Photometric observations will be targeted at the angular resolution of the astrometric observations, for all objects observed astrometrically, over the broadest spectral range, in the largest numbers of spectral bands, and at the highest signal-to-noise ratio.

Spectroscopic observations will be performed to:

i) provide radial velocities by Doppler-shift measurements, and
ii) provide astrophysical information.

Spectroscopic observations will be targeted at the maximum number of objects observed astrometrically, and at the highest signal-to-noise ratio.

*SCI-010* *The satellite shall scan the sky by a continuous, controlled spin motion, in which the satellite Xs axis (see chapter 5 SYS-030) is kept at a nominally fixed angle $\xi$ (solar aspect angle) from the Sun to Spacecraft (S/C) direction of light rays. The angle $\xi$ shall be not less than 45° (defined at L2) and shall be kept constant with a precision in accordance to SCI-735. The nominal scanning law shall be defined independently of the satellite orbit.*

*SCI-020* *The component of the rotation vector ($w_0$) about the satellite Xs axis shall be kept constant, with a nominal value of 60arcsec $s^{-1}$ (the associated nominal spin period equals 6 hours) with a precision in accordance to SCI-730.*

*SCI-030* *This scanning law shall have a negative rotation (clockwise) around the $+X_S$ axis in the satellite coordinate system.*

*SCI-040* *All Gaia instruments' line of sight shall be in the same plane and this plane shall be perpendicular to the Xs spacecraft axis with an accuracy of 3 arcmin with a probability of 99.73%.*

*SCI-050* *The satellite spin axis shall have a forced precession motion (mean value over the year equals 0.173 arcsec $s^{-1}$ considering a solar aspect angle equivalent to 45deg) about the Sun to Earth direction at controlled speed.*

*SCI-060* *For each detected and confirmed object image in all instruments (regardless whether the object image is selected for observation or not) the relevant individual image samples shall be telemetered to the ground. From these individual image samples it shall be possible to determine all relevant astrometric and photometric parameters required on board (including e.g. object position and across-scan and along-scan velocity) and on ground (including e.g. the image position, flux, PSF profile, background).*

Definition of image samples is reported in RD 1-14.

*SCI-070* *Individual image samples containing the detector data for telemetry to the ground shall be defined such as to contain the image flux and sky background in an optimum manner according to object magnitude, whilst satisfying telemetry constraints, imaging requirements, bright star sample modes, double stars, and calibration modes.*

An example of a satisfactory sampling scheme is given in RD 1-4.

*SCI-080* *Measurements performed with Gaia shall allow on-ground calibration of CCD sensitivities, CCD noise, PSFs, CTI effects, magnitude scale and effective wavelengths of spectral throughput during the mission, with accuracies consistent with the prescriptions of RD 1-3.*

Photon flux distributions of the model stars are provided in AD 1-9.

*SCI-090* *For the astrometric, photometric and spectroscopy error assessments, the average sky background shall be assumed to have a uniform brightness of V=22.5 mag/arcsec^2, with the spectrum of an unreddened G2V star as given in AD 1-9.*

Reference object densities per square degree field of view on the sky (where instances of the sky count model are applicable) are defined as follows:

| | |
|---|---|
| Average | 25 000 stars $deg^{-2}$; |
| Typical | 150 000 stars $deg^{-2}$; |
| Design | 600 000 stars $deg^{-2}$; |
| Maximum | 3 000 000 stars $deg^{-2}$. |

Unresolved objects such as unresolved galaxies, quasars, and unresolved binary stars, shall be considered for all astrometric and photometric system requirements as being indistinguishable from stars. Their contribution to the maximum star densities, and the sky count model, is accounted for in the distributions provided in RD 1-2.

*SCI-100* *Solar system objects, including planets, planetary satellites, comets, and asteroids (including main-belt asteroids, Trojan satellites, near-Earth objects, and Kuiper belt objects) shall be detected and confirmed up to V=20, and fully observed over the maximum possible field of view of the astrometric and photometric instruments. Solar system bodies with angular velocities up to 0.04 arcsec/sec in along-scan and/or 0.1 arcsec/sec across-scan directions shall be observed.*

*SCI-105* *The knowledge error, at a given wavelength and for each individual image sample, of the integral non-linearity of the transfer function between the input signal (object incident flux) and the output signal (recorded LSBs at the end of the detection chain) shall be less than 5% over the full dynamic range of the video acquisition chain.*

## 4.2 Astrometric Requirements

SCI-110 *The astrometric measurement principle shall be based upon a continuous scanning of the sky with two fields of view in which object positions are measured.*

SCI-120 *The two astrometric lines of sight (viewing directions) shall be separated by a constant angle, referred to as the basic angle. One of the following values shall be used: 96.8 ± 0.1 deg, 99.4± 0.1 deg, 100.5± 0.1 deg, 105.3± 0.1 deg, 106.5± 0.1 deg, 109.3± 0.1 deg, 109.9± 0.1 deg.*

The ±0.1 deg in this requirement refers to the nominal angle, and shall not be understood as the maximum allowed in-orbit random fluctuations of this angle.

SCI-130 *The number of objects to be repeatedly and consistently observed throughout the mission lifetime shall be at least 1 billion calculated using the galaxy model as specified in RD 1-2.*

SCI-140 *Object detection, confirmation and observation shall be performed autonomously on-board in the same wavelength range as the astrometric measurements.*

An example of a scheme for object detection reflecting real-sky complexities is the Pyxis software developed during the study phase and reported in RD 1-7.

SCI-150 *Nominal observations and all astrometric requirements shall be achieved in the two superimposed fields computed using the design density in one instrument field plus the typical density in the other instrument field.*

SCI-160 *A strategy to observe high-density sky regions (e.g. Baade's windows, Omega Centauri, etc.), with stellar densities up to the indicated maximum, shall be proposed. If higher densities than the stated maximum are encountered, the brightest objects up to the maximum density given shall be observed.*

A potential strategy could be the possibility to have several successive transits of the same field at a reduced precession rate (see RD 1-8).

*SCI-170* *It shall always be possible to distinguish on board, at any time and for all detected and confirmed object images, in which field of view the image is observed.*

*SCI-180* *The flux contained within a rectangle defined by the first diffraction minima of the image sample shall be at least 68% of the total energy of the object reaching the focal plane.*

*SCI-190* *Overlapping of a given field of view between two consecutive great-circle-bands shall be at least 0.07 deg for at least 33% of the great-circle-bands length.*

*SCI-200* *The bright object completeness limit shall be not fainter than:*

- *V=6.0 for unreddened B1V stars,*
- *V=6.2 for unreddened G2V stars, and*
- *V=8.4 for unreddened M6V stars*

*SCI-210* *The faint object completeness limit shall be not brighter than:*

- *V=20.0 for unreddened B1V stars,*
- *V=20.2 for unreddened G2V stars, and*
- *V=22.4 for unreddened M6V stars*.

*SCI-220* *More than 95% per Field of View of the transits of single objects and multiple systems over the magnitude range specified in SCI-200 and SCI-210 shall be observed.*

Missing transits shall be accounted for in the overall error budget.

*SCI-230* *A mapping in unfiltered light shall be made of an area centered on each detected and confirmed object. The extent of the area shall be sufficient to detect object that could affect the astrometric and photometric measurements.*

*SCI-240* *The end-of-mission statistical (parallax) standard error shall be calculated using the guidelines provided in RD 1-3.*

*SCI-250* *The end-of-mission statistical (parallax) standard errors E (averaged over the sky) for unreddened B1V, G2V, and M6V stars shall be as specified in the following table:*

| | *B1V* | *G2V* | *M6V* |
|---|---|---|---|
| *V < 10* | *< 7 μas* | *< 7 μas* | *< 7 μas* |
| *V = 15* | *< 25 μas* | *< 24 μas* | *< 12 μas* |
| *V = 20* | *< 300 μas* | *< 300 μas* | *< 100 μas* |

*SCI-260* *The end-of-mission statistical (parallax) systematic errors for unreddened B1V, G2V, and M6V stars shall be lower than 1μas.*

*SCI-280* *The basic angle fluctuations over the nominal spin period shall be:*
*< 7 μas rms for the random contribution*
*< 4 μas for the systematic contribution.*

The word systematic has to be understood as the amplitude of the fluctuations at the spacecraft spin frequency (inverse spin period) and at the low-order multiples of this frequency.

*SCI-290* *The Basic Angle shall be monitored in flight with accuracy better than 0.5 μas rms for every 5 minutes interval of scientific operation.*

## 4.3 Photometric Requirements

*SCI-300* *The photometric field(s) shall provide multi-colour, multi-epoch photometric measurements for each object image observed in the astrometric field.*

*SCI-310* *The number of objects to be repeatedly and consistently observed throughout the mission lifetime shall be at least 1 billion calculated using the galaxy model as specified in RD 1-2.*

*SCI-320* *Photometric observations shall be provided in the wavelength range 320 nm $\leq \lambda \leq$ 1000 nm.*

*SCI-325* *The wavelength range defined in SCI-320 may be divided into two or more wavelength intervals. For each of these wavelength intervals, the bandwidth is defined as the Full Width at Half Maximum (FWHM) of the overall spectral transmission profile (including the telescope, intermediate optics, and detector). The cut-on and cut-off wavelengths of each bandwidth are defined as the limiting wavelengths of the bandwidth, i.e., the half-maximum -transmission wavelengths. Adjacent wavelength intervals shall share a wavelength overlap region (defined as the distance between the cut-on and cut-off wavelengths of the bandwidths) of at least 20 nm and at most 60 nm.*

*SCI-330* *An adequate number of bands shall provide the photometric measurements at the same epoch and same angular resolution as the astrometric measurements. The number of bands shall cover the whole photometry wavelength range defined in SCI-320.*

A potential proposal is available in RD 1-5.

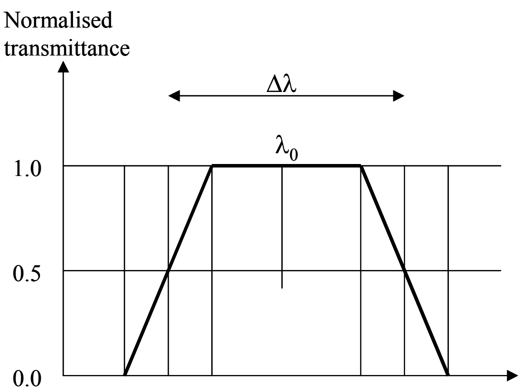
*SCI-340* *The number of photometric bands in SCI-330 shall provide the spectral energy distribution of each object at each epoch of observation with an accuracy sufficient for calibrating chromatic effects with end-of-mission residuals of less than 2.5 µas for objects with V=15 and 30 µas for objects with V=20. The calibration of chromatic effects shall be performed at the same epoch and angular resolution as the astrometric measurements.*

Guidelines for chromaticity evaluation are reported in RD 1-9.

*SCI-350* *Fourteen photometric bands shall be provided in order to derive the astrophysical information for each observed object in the astrometric field. The photometric bands shall be as follows:*

| *Band* | *$\lambda_0$ (nm)* | *$\Delta\lambda$(nm)* |
|---|---|---|
| *C1M344* | *343.5* | *47* |
| *C1M379* | *379* | *24* |
| *C1M395* | *395* | *10* |
| *C1M410* | *410* | *20* |
| *C1M467* | *468* | *20* |
| *C1M506* | *506* | *36* |
| *C1M515* | *515* | *18* |
| *C1M549* | *549* | *22* |
| *C1M656* | *656.3* | *7* |
| *C1M716* | *717* | *26* |
| *C1M747* | *747* | *32* |
| *C1M825* | *825* | *34* |
| *C1M861* | *860.5* | *33* |
| *C1M965* | *965* | *70* |

$\lambda_0$ is the central wavelength and $\Delta\lambda$ is the bandwidth (FWHM) as shown in figure.




*SCI-360* *Withdrawn.*

A proposal for the photometric bands specified in SCI-350, is available in RD 1-6 (Note: the band C1M344 replaces the C1M326). Filters are discussed in RD 1-10.

*SCI-351* *Images shall be spectrally dispersed along the scan direction.*

*SCI-353* *The relative variation of the overall spectral transmission profile of the intermediate components of the photometric instrument (excluding the telescope and detector) over any wavelength range within the bandwidth which extends over up to 90% of this bandwidth shall be smaller than 15% peak to valley (the bandwidth is defined in SCI-325).*

*SCI-355* *The relative variation of the overall spectral transmission profile of the intermediate components of the photometric instrument (excluding the telescope and detector) over any interval of 20 nm within the wavelength range which extends over the 90% of the bandwidth shall be smaller than 5% peak to valley (the bandwidth is defined in SCI-325).*

*SCI-357* *For an unresolved object with a flat spectral energy distribution in wavelength, the total number of photo-electrons recorded by the photometric instrument outside the wavelength range of interest shall be smaller than 5% of the total number of photo-electrons recorded for the same object within the wavelength range of interest.*

Note: The wavelength range of interest is defined as the full width at 0.1% relative overall spectral transmission (including the telescope, intermediate optics, and detector).

*SCI-359* *For an unresolved object with a flat spectral energy distribution in wavelength, the spatial extent corresponding to the spectral domain where the relative overall spectral transmission profile (including the telescope, intermediate optics, and detector) is above 1E-4 shall be smaller than 3.0 arcsec.*

*SCI-361* *For any wavelength within the bandwidth (as defined in SCI-325), the effective point spread function obtained at the end of the acquisition-transmission-reception chain (including optical PSF, optical distortions, detectors and associated electronics, satellite attitude, on-board data treatment, and other effects) shall be sampled in the spectral dispersion direction with more than two pixels per spectral resolution element.*

Note: The spectral resolution element is defined as the central area of the effective PSF containing 76% of the total energy in the dispersion direction.

*SCI-363* *The spatial extent, in the direction of spectral dispersion, of the spectrally dispersed images shall be in the range 1.1 to 1.9 arcsec.*

Note: The spatial extent of a spectrally-dispersed image corresponds to the cut-on and cut-off wavelengths of the bandwidth as defined in SCI-325.

*SCI-364* *A spectrally-dispersed image shall contain (in the case of maximum motion of the image in the direction perpendicular to the spectral dispersion direction), in the direction perpendicular to the spectral dispersion direction, 95% of all photo-electrons generated for that image within less than 1.8 arcsec.*

*SCI-365* *It shall be possible to get enough unsaturated bright objects in the same acquisition conditions as faint ones such as to allow a proper calibration of the photometric instrument.*

*SCI-370* *Withdrawn*

*SCI-380* *One of the fourteen band shall cover the same spectral range as the spectroscopic measurements, reproducing the overall response of the Spectroscopic instrument*

*SCI-390* *Each photometric measurement shall cover an area around each object spectrum at least as large as 1 arcsec$^2$ for derivation of the sky background.*

*SCI-400* *The bright object completeness limit shall be not fainter than:*

- *V=6.0 for unreddened B1V stars,*
- *V=6.2 for unreddened G2V stars, and*
- *V=8.4 for unreddened M6V stars*

*SCI-410* *The faint object completeness limit shall be not brighter than:*

- *V=20.0 for unreddened B1V stars,*
- *V=20.2 for unreddened G2V stars, and*
- *V=22.4 for unreddened M6V stars.*

*SCI-420* *More than 95% of the transits of single objects and multiple systems over the magnitude range specified in SCI-400 and SCI-410 shall be observed.*

*SCI-430* *Observations up to the same design density as the astrometric* *measurements shall be provided.*

This requirement has to be considered applicable only for photometric bands used for the chromaticity correction.

SCI-440 *On-Board autonomous and systematic object detection, confirmation and observation shall be provided over the full magnitude range specified in SCI-400 and SCI-410.*

SCI-450 *The end-of-mission sky-averaged photometric standard errors for all bands shall be calculated using the guidelines provided in RD 1-3.*

SCI-460 *End-of-mission sky-averaged magnitude standard error for unreddened B1V, G2V and M6V stars shall be as follows*:

| *Band* | *V (mag)* | *B1V (mmag)* | *G2V (mmag)* | *M6V (mmag)* |
|---|---|---|---|---|
| *C1M344* | *15* | *< 10* | *< 15* | *< 100* |
| | *20* | *< 150* | *< 1000* | *-* |
| *C1M410* | *15* | *<10* | *< 10* | *< 20* |
| | *20* | *< 60* | *< 200* | *<1100* |
| *C1M549* | *15* | *< 8* | *< 8* | *< 8* |
| | *20* | *< 120* | *< 120* | *< 120* |
| *C1M965* | *15* | *< 20* | *< 10* | *< 10* |
| | *20* | *< 400* | *< 150* | *<10* |

*Where all magnitude standard errors for V = 20 and the magnitude precision for V = 15 for the C1M344 band are considered as design goals.*

## 4.4 Spectroscopic Requirements

SCI-480 *The spectroscopic measurements shall be performed over the full wavelength range 847 nm ≤ λ ≤ 874 nm.*

SCI-490 *The relative variation of the overall response of the spectrometer (including the telescope and any intermediate components) over the wavelength range defined in SCI-480 shall be smaller than 15% peak to valley and local variation shall be minimized.*

SCI-495 *The relative variation of the overall response of the spectrometer (including the telescope and any intermediate components) over any interval of 2 nm within the wavelength range defined in SCI-480, shall be smaller than 5% peak to valley.*

SCI-500 *The total number of photo-electrons recorded by the spectrometer, for an unresolved object with a flat energy density distribution, outside of the wavelength range defined in SCI-480, shall be smaller than 10 % of the total number of photo-electrons recorded for the same object within the wavelength range defined in SCI-480.*

SCI-505 *The irradiance recorded by the spectrometer outside the wavelength range defined in SCI-480 extended on each side by 40%, shall be smaller than 1E-4 of the maximum irradiance recorded within that range. This requirement shall be assumed for a point source object with a flat energy distribution.*

SCI-509 *It shall be possible to operate the spectroscopic (RVS) instrument in two modes: the High Resolution (HR) mode and the Low Resolution (LR) mode. The LR mode shall apply only to objects fainter than a threshold determined from the Gaia RVS (G_RVS) magnitude. The nominal threshold is G_RVS=10 mag.*

Note**:** This threshold shall be selectable over the whole magnitude range specified in SCI-560 and SCI-570.

SCI-510 *The effective point spread function obtained at end of the acquisition-transmission-reception chain (including optical PSF, optical distortions, detectors and associated electronics, satellite attitude, on-board data treatment, and other effects) shall be sampled in the spectral dispersion direction with a number of pixels >= 2 per spectral resolution element for HR spectra and a number of pixels >= 0.80 per spectral resolution element* *for LR spectra.*

Note: the spectral resolution element is defined as the central area of the effective PSF containing 76% of the total energy in the along dispersion direction.

*SCI-520* *In High Resolution mode, the effective spectral resolving power averaged over the wavelength (within the nominal wavelength range as defined in SCI-480) and averaged over the different spectra shall be in the range 10500 < R_effective < 12500. In High Resolution mode, the effective spectral resolving power, for any wavelength (within the nominal wavelength range as defined in SCI-480), for any single epoch spectrum and for any end of mission combined spectrum, shall be above 10000 for ≥90% of individual points of spectra and shall be below 13500 for all individual points of spectra.*

Note: the effective spectral resolving power for a wavelength $\lambda$ is calculated as: $\lambda/\Delta\lambda$, where $\Delta\lambda$ is the spectral width of the spectral resolution element for the wavelength $\lambda$ (the spectral resolution element is defined in the note attached to SCI-510)

*SCI-530* *A spectrum shall contain (in the case of maximum motion of the spectrum in the direction perpendicular to the spectral dispersion direction), in the direction perpendicular to the spectral dispersion direction 90% of all photo-electrons collected for that spectrum within less than 1.8 arcsec.*

*SCI-540* *The spatial extent of the spectrum, in the direction of the spectral dispersion, corresponding to its spectral extent as defined in SCI-480, shall be <= 75 arcsec.*

*SCI-545* *The observing strategy of the spectroscopy shall allow for the acquisition of 40 000 objects/degree² or more in the superimposed fields of view.*

*SCI-550* *The sky-averaged number of spectroscopic transit observations per object over the 5 years of the mission shall be larger than 40 for V<15 mag, well distributed over the mission duration, after accounting for satellite availability and all contributions to dead time.*

*SCI-560* *The magnitudes of the brightest objects to be observed shall be at least:*

| *Stellar type* | *V (mag)* |
|---|---|
| *B1V* | *6* |
| *G2V* | *6* |
| *K1IIIMP* | *6* |

*SCI-570* *The magnitudes of the faintest objects to be observed shall be:*

| *Stellar type* | *V (mag)* |
|---|---|
| *B1V* | *13.0* |
| *G2V* | *17.0* |
| *K1IIIMP* | *18.0* |

*SCI-580* *Adequate sky background measurements necessary for the on-ground reconstruction shall be provided where permitted by the sky object density.*

*SCI-590* *On-board autonomous and systematic object detection, confirmation, observation, pre-processing and transmission to the ground shall be performed over the full magnitude range defined in the requirements SCI-560 and SCI-570.*

*SCI-600* *Object selection shall be made in the spectrometer band at a fixed S/N ratio (the S/N is defined by the faintest of the three reference star).*

*SCI-610* *The end-of-mission sky-averaged radial velocity errors shall be calculated using the guidelines provided in RD 1-3.*

*SCI-620* *The following end of mission radial velocity performance shall be met:*

| *Stellar type* | *V (mag)* | *RV (km/s)* |
|---|---|---|
| *B1V* | *12* | *15* |
| *G2V* | *16.5* | *15* |
| *K1IIIMP* | *17* | *15* |

| *Stellar type* | *V (mag)* | *RV (km/s)* |
|---|---|---|
| *B1V* | *7* | *1* |
| *G2V* | *13* | *1* |
| *K1IIIMP* | *13.5* | *1* |

The following end of mission radial velocity performance are considered design goal:

| Stellar type | V (mag) | RV (km/s) |
|---|---|---|
| B1V | 12.5 | 15 |
| G2V | 17 | 15 |
| K1IIIMP | 17.5 | 15 |

| Stellar type | V (mag) | RV (km/s) |
|---|---|---|
| B1V | 8 | 1 |
| G2V | 14.0 | 1 |
| K1IIIMP | 14.5 | 1 |

*SCI-630* *The maximum instrumental systematic radial velocity error after calibration shall be smaller than 300 m/s.*

*SCI-635* *The undesired spectral orders within the Spectroscopy band shall not contain more than 0.01% of the total power in the desired order if imaged in the focal plane.*

*SCI-637* *It shall be possible to get enough unsaturated bright objects in the same acquisition conditions as faint ones such to allow a proper calibration of the instrument.*

## 4.5 Science Application Requirements

*SCI-640* *The payload shall include the hardware and associated on-board software to acquire, control and process the generated raw science data.*

*SCI-650* *The Science Application shall be able to:*

- *detect all objects (in the spatial domain) of interest;*
- *select objects of interest in order to discard false detection (e.g. cosmic rays);*
- *provide data necessary for the AOCS;*
- *follow all objects of interest as they cross the focal plane (considering the apparent movement of the object on the detectors);*
- *acquire astrometric, photometric and spectrometric data for the selected objects;*
- *provide data necessary to perform instrument calibration during ground testing and in flight;*
- *use uploaded calibration parameters to perform on-board corrections needed to perform on-board operations;*
- *provide instrument(s) monitoring.*

*SCI-660* *The Science Application shall be able to deal with the following type of objects:*

*- single objects including saturated objects;*

*- blended multiple objects;*

*- solar system objects;*

*- extended objects.*

*SCI-665* *For each object, at least the following category of information shall be transmitted to ground:*

*- object samples;*

*- time information;*

*- position information;*

*- object type;*

*- special processing information;*

*- background;*

*- flux intensity;*

*- field information;*

*- extended objects.*

*SCI-670* *If data compression of science data is applied it shall be lossless.*

*SCI-680* *The Science Application shall allow the following operation of the Gaia payload:*

- *nominal science acquisition*
- *calibration during ground and in flight*
- *commissioning (in L2 and during cruise to L2)*
- *reduced capability commanded and after failure.*

## 4.6 Straylight Requirements

Straylight for the Gaia Payload has three different origins:

1) Celestial sources inside telescope FoV
2) Celestial sources outside telescope FoV
3) Artificial sources inside the spacecraft

*SCI-690* *Straylight irradiance (ghost, scattering and diffusion) induced by a point source within the telescope field of view shall be less than 1E-5 of its PSF peak irradiance at an angular distance of 20arcsec or more from the peak of the PSF itself. In addition, within the circle of 20arcsec from the PSF peak location, no ghost of more than 1E-4 of the PSF peak irradiance shall be present.*

*SCI-700* *Straylight irradiance from all astronomical sources outside the telescope field of view shall be lower than 20% of the sky background (as specified in SCI-090). This requirement shall hold for 99.8 percent of the mission time.*

*SCI-710* *Straylight irradiance from artificial light source generated in the S/C shall be lower than 10% of the sky background (as specified in SCI-090). This requirement shall hold for 100 percent of the mission time.*

## 4.7 Additional Requirements

*SCI-720* *Particle events shall be rejected from selection on-board.*

*SCI-730* *The Gaia satellite design shall comply with the pointing and rate error requirements on both Astro line-of-sight as reported in the following table.*

| Parameter | Requirement |
|---|---|
| *Attitude Measurement Error* | *< 20 arcsec (with a probability of 99.73% at all time)* |
| *Rate Measurement Error (along scan)* | *< 0.9 mas.s$^{-1}$ (with a probability of 99.73% at all time)* |
| *Rate Measurement Error (across scan)* | *< 2.7 mas.s$^{-1}$ (with a probability of 99.73% at all time)* |
| *Absolute Pointing Error* | *< 60 arcsec (with a probability of 99.73% at all time)* |
| *Relative Pointing Error (along scan) over τ** | *< 5 mas (with a probability of 99.73% on along-scan axis at all time)* |
| *Relative Pointing Error (across scan) over τ** | *< 10 mas (with a probability of 99.73% on any across-scan axis at all time)* |
| *Mean Rate Error (along scan)* | *< 2 mas.s$^{-1}$ (with a probability of 99.73% on along-scan axis at all time)* |
| *Mean Rate Error (across scan)* | *< 10 mas.s$^{-1}$ (with a probability of 99.73% on any across-scan axis at all time)* |

* τ is defined as the single detector (CCD) integration time
99.73% in the table is intended to be valid during nominal science observations.

Because of the astrometric measurement principle and the way the CCDs are operated, there is often a clear distinction in accuracy requirement between quantities measured parallel to and perpendicular to the plane containing the two Astro lines of sight. The component parallel to this plane is in the following referred to as the along-scan component, and the component perpendicular to the plane as the across-scan component.

*SCI-735* *The Gaia satellite design shall comply with the 3-axis pointing error as reported in the following table.*

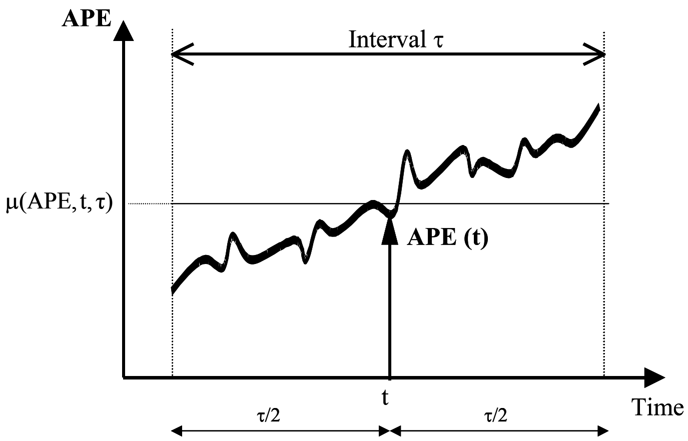
| Parameter | Requirement |
|---|---|
| *Attitude Measurement Error* | *< 20 arcsec (with a probability of 99.73% at all time)* |
| *Absolute Pointing Error* | *< 60 arcsec (with a probability of 99.73% at all time)* |

99.73% in the table is intended to be valid during nominal science observations

**Attitude Measurement Error (AME)** is defined as the instantaneous 3-axis angular separation between the estimated spacecraft attitude (and Astro instruments LoS) and the actual one. This error is normally referred to as "a posteriori knowledge of attitude".

**Rate Measurement Error (RME)** is defined as the mean difference between the estimated spacecraft scan rate and the actual one. The time necessary for the averaging is the time spent by an object from being detected to being confirmed in the Astrometric field.

**Absolute Pointing Error (APE)** is defined as the instantaneous 3-axis angular separation between the actual and the desired spacecraft attitude (and Astro instruments LoS).




**Relative Pointing Error (RPE)** at a given date **t** is defined as the standard deviation of the absolute pointing error over the AF CCD integration time **τ** around t.

$$\mathrm{RPE}(t,\tau)=\sqrt{\frac{1}{\tau}\int_{t-\tau/2}^{t+\tau/2}[\mathrm{APE}(u)-\mu(\mathrm{APE},t,\tau)]^2\,du}$$

with $\mu(\mathrm{APE},t,\tau)=\frac{1}{\tau}\int_{t-\tau/2}^{t+\tau/2}\mathrm{APE}(u)du$

**Mean Rate Error (MRE)** is defined as the mean difference between the actual and the desired satellite scan rate. The time necessary for the averaging is the time spent by an object from being detected to being confirmed in the Astrometric field.

*SCI-740* *The Astro telescope line of sight jitter along scan with respect to the inertial frame shall be such that its Power Spectral Density* $\mathrm{PSD}_{\mathrm{APE}}^{\mathrm{AL}}(f)$ *satisfies the following inequality:*

$$\Delta\theta_{\mathrm{AHFD}}(\mathrm{AL})=\left(\int_{1/(10T)}^{\infty}\mathrm{PSD}_{\mathrm{APE}}^{\mathrm{AL}}(f)\,\mathrm{Kal}\,\mathrm{sinc}^2(fT\pi)\;\mathrm{d}f\right)^{1/2}<3.4\,\mu as\ rms$$

Where Kal = $1/(1+1/(2*f*\Delta t)^{12})$ where $\Delta t = 0.5\pi\tau$ s and $T$ is the time for an object to cross the Astro focal plane (i.e. from the 1st to the last AF read-out).

The PSD has to be calculated considering a time interval that is 10 times higher than $T$ and for AHFD calculation an offset equivalent to $T$ between two consecutive time intervals has to be taken into account.

*SCI-750* *The Astro telescope line of sight jitter across scan with respect to the inertial frame shall be such that its Power Spectral Density* $\mathrm{PSD}_{\mathrm{APE}}^{\mathrm{AC}}(f)$ *satisfies the following inequality:*

$$\Delta\theta_{\mathrm{AHFD}}(\mathrm{AC})=\left(\int_{1/(10T)}^{\infty}\mathrm{PSD}_{\mathrm{APE}}^{\mathrm{AC}}(f)\,\mathrm{Kac}\,\mathrm{sinc}^2(fT\pi)\;\mathrm{d}f\right)^{1/2}<100\,\mu as\ rms$$

Where Kac = Kal, where $T$ and the PSD calculation strategy are defined above.

*SCI-760* *All on-board times (including telemetry and telecommand) shall be derived from and synchronised with an on-board master clock.*

*SCI-770* *The detector sample data shall be time-stamped with a precision (standard error) better than 50ns over the nominal satellite spin period of requirement (SCI-020).*

*SCI-780* *The error contribution from the spacecraft to the correlation of spacecraft On Board Time (OBT) with a ground reference time (UTC) shall be lower than 1 μs.*

*SCI-785* *Any clock used for stamping telemetry shall be synchronised and correlated with the SCET.*

Note: the requirement will allow ground to correlate any on-board time stamp with the UTC.

*SCI-790* *In case of need, it shall be possible, at any time, to prioritise and select the science data stream of all payload instrument(s).*

*SCI-800* *In case of need, it shall be possible to independently command each of the Gaia instruments to operate in a reduced science data rate generation mode*

*SCI-810* *The spacecraft shall be able to simultaneously download the science data generated and stored in the SSMM (including housekeeping) and receive telecommands.*

# 5 SPACECRAFT FUNCTIONAL AND DESIGN REQUIREMENTS

## 5.1 General Definitions

### 5.1.1 Metric Standard

*SYS - 010* *Measurement units shall be in the SI system.*

### 5.1.2 Coordinate Systems

*SYS - 020* *All coordinate systems shall be right-handed orthogonal systems.*

#### 5.1.2.1 Spacecraft Coordinate System

*SYS - 030* *There shall be a unique right-handed orthogonal coordinate system for the Gaia spacecraft ($C_S$, $X_S$, $Y_S$, $Z_S$) with the following characteristics (see Figures 5.1.2.1-1/2):*

- *$C_S$, origin: at the centre of the circular satellite interface with the launch vehicle adaptor, in the separation plane;*
- *$X_S$ axis: launcher axis, oriented from the launcher interface to the spacecraft, thus parallel to the conventional launcher coordinate axis $X_{LV}$;*
- *$Z_S$ axis: intersection of the separation plane and the bisector plane of two planes, one of which parallel to $X_S$ and containing the preceding Astro line of sight (LOS1), the other one parallel to $X_S$ and containing the following Astro line of sight (LOS2); $Z_S$ is oriented towards the Astro Focal Plane Assembly (FPA);*
- *$Y_S$ axis: in the separation plane, perpendicular to $Z_S$.*

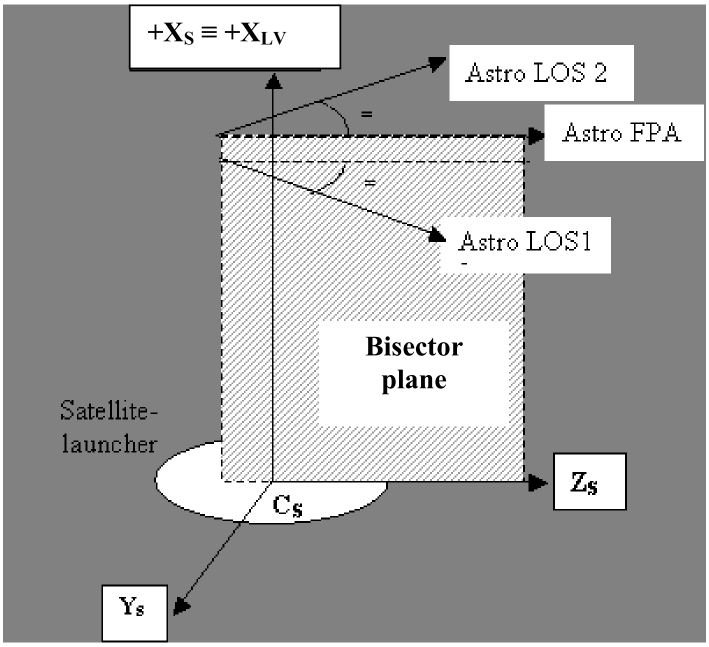



**Figure 5.1-1**: Gaia spacecraft reference system.

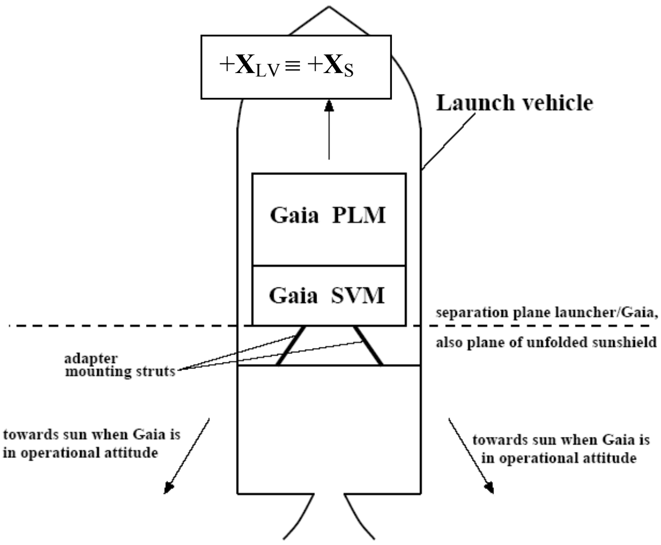



**Figure 5.1-2: Gaia spacecraft reference system.**

#### 5.1.2.2 Unit Coordinate System

*SYS - 040* *The unit coordinate system ($O_U$, $X_U$, $Y_U$, $Z_U$) shall be fixed relative to the unit geometry. It shall be a right-handed orthogonal coordinate system defined as follows:*

- *One of the attachment holes of the unit shall be chosen as mechanical reference hole and shall be identified by an engraved letter "R" on the unit;*
- *The origin ($O_U$) shall be located at the centre of the reference hole at the level of the mounting interface plane;*
- *The $X_U$ axis shall be perpendicular to the mounting interface plane, pointing positively toward the unit;*
- *The $Y_U$ and $Z_U$ axes shall be oriented such that the unit will mostly be included inside the $+Y_U/+Z_U$ quadrant of the mounting interface plane. Moreover, if the unit has a rectangular shape, the $+Y_U$ and $+Z_U$ axes shall be parallel to the unit.*

#### 5.1.2.3 Unit Alignment Frame

*SYS - 050* *The unit alignment frame shall be a right-handed orthogonal coordinate system used for alignment purposes and is defined as follows:*

- *The origin is located at the centre of the alignment cube;*
- *The three axes are identified by the normals to the mirror faces of the optical alignment cube.*

## 5.2 Satellite Lifetime

*SYS - 060* *The satellite shall be designed for a lifetime of 5.5 years from separation with the launcher.*

Reckoning that the commissioning phase may last up to 6 months (Chapter 3), the nominal operational phase is thus 5 years.

*SYS - 070* *The satellite consumables (propellants, solar array power, SSMM(s)) shall be dimensioned for a 6.5 years lifetime from separation with the launcher (extended operational phase, i.e., 1 year beyond the nominal operational phase).*

## 5.3 Autonomy

*SYS - 080* *During LEOP critical phases (e.g. separation, deployments, etc.), the* *satellite shall be able to operate nominally during a ground outage period of at least two hours.*

SYS - 085 *During LEOP, outside critical phases, the spacecraft shall be able to operate nominally during a ground outage of at least 40 hours.*

SYS - 090 *In the operational phase, the satellite shall be able to continue to operate nominally without ground contact for a period of 3 days.*

SYS - 095 *In all mission phases the satellite shall be designed to survive a ground outage period of 7 days.*

SYS - 100 *An unambiguous indication shall be provided in the telemetry every time an autonomous intervention takes place on-board to permit the ground to reconstruct every related action that has been executed on-board, both at sub-system and system level. This information shall be stored on-board for interrogation by the ground, and shall remain in memory (including failures) until reset by the ground.*

SYS - 110 *The satellite shall be able to detect autonomously any failure which affects mission objectives or satellite safety; if such condition is detected the spacecraft shall autonomously react to recover.*

SYS - 115 *In case conditions of SYS – 110 are not met, the spacecraft shall, as a first resort, maintain a quasi-nominal attitude (intermediate safe mode) and keep the spacecraft safe or if this is not possible the spacecraft shall go in sun-pointing attitude ("Safe Mode").*

SYS - 120 *The spacecraft design shall ensure that the "Safe Mode" function is available at all time during the various mission phases also in presence of H/W and/or S/W failures.*

SYS - 130 *The "Safe Mode" shall guarantee TM/TC communications between the ground and the satellite to:*

- *Downlink all the information necessary to ground operators to understand the failure events and to identify the involved hardware/software making efficient usage of the (reduced) downlink bandwidth*
- *Uplink commands to be executed by the satellite for upgrading the configuration and returning to nominal modes*

*and to ensure minimum power bus loading (for spacecraft safety) and* *maximum battery charging capability.*

SYS - 140 *The operational modes of the spacecraft and its payload, subsystems and units shall be clearly identified in terms of both hardware and software.*

SYS - 150 *Initialisation of a mode (at spacecraft, subsystem or unit level) shall include automatic configuration of the necessary hardware (e.g. sensors, actuators), activation of a default periodic telemetry configuration, and all the automatic processes (e.g. automatic control of attitude slews) required to achieve the objective of the mode.*

## 5.4 Availability

SYS - 160 *In the operational orbit, the time available for scientific measurements not disturbed by nominal spacecraft operations (e.g. mode changes, propellant sloshing, AOCS/RCS calibrations, or orbit maintenance manoeuvres) shall be ≥ 97% of the mission time.*

## 5.5 Fault Management

SYS - 160 *Gaia shall automatically detect, isolate, recovery and report any fault, failure, and/or error that cause deviation from its nominal configuration and operational mode or that may adversely affect the mission. This includes hardware and software failures. The spacecraft shall transmit unambiguously in its telemetry to ground the occurrence of failure/fault/error, the associated parameters in each error and their values.*

SYS - 165 *The spacecraft shall transmit unambiguously in its telemetry to ground the occurrence of failure/fault/error, the associated parameters in each error and their values.*

SYS - 170 *The overall FDIR (Failure Detection Isolation and Recovery) concept shall be based on a failure list covering system, sub-system, instrument and unit.*

SYS - 180 *All system used for fault management within FDIR shall be intrinsically fail safe.*

*SYS - 190* *Instrument failures shall not lead to a "Safe Mode" of the satellite.*

*SYS - 200* *The fault management functions at all levels shall be able to access to lower level telemetry data produced by the sub-systems and/or the instrument. This includes in particular non-periodic event packets that can be used to trigger recovery actions at system or sub-system levels as a result of an anomaly occurred (and detected) in another sub-system.*

*SYS - 210* *A predefined set of fault management functions at specific FDIR levels shall carry out consistency verification checks on independent or redundant sensor readings whenever available before starting the recovery actions.*

*SYS - 215* *Failure propagation must be avoided.*

## 5.6 Reliability

*SYS - 220* *The probability of the spacecraft support (S/C excluded the Payload composed by optics, detector and processing chain) of ensuring the nominal (without mission objectives degradation) measurement at the end of the in-orbit operational lifetime shall be 0.75.*

*SYS - 225* *Any single payload (optics, detection and processing chain) failures shall not cause a loss with respect to the nominal payload performances greater than:*

*8% for Astrometry & Photometry,*
*15% for Spectroscopy.*

*SYS - 230* *A failure of one piece of equipment (unit level) shall not cause failure of or damage to another piece of equipment or sub-system, including the scientific instruments.*

*SYS - 240* *Protected switching configurations employing separate "arm" and "active" operations shall be implemented whenever an unintended activation can lead to an operational hazard.*

SYS - 250 *Compliance to the fault tolerance requirements shall be verified by FMECA or by other suitable methods, which shall be subject to agreement with ESA.*

SYS - 260 *Protection systems shall be intrinsically fail-safe and, if appropriate, shall be capable of being disabled either by ground or on-board command. Where not possible, appropriate justification shall be provided.*

SYS - 270 *It shall not be possible to disable/inhibit a protection feature that in case of a single failure at S/C level could lead to the loss of the main primary power bus.*

SYS - 280 *The spacecraft shall have the capability to react autonomously to failures, faults, and errors, including the ability to enter and stay in safe mode.*

SYS - 290 *All DC/DC converters shall be over-voltage protected.*

## 5.7 Redundancy

SYS - 300 *The satellite shall be designed Single Failure Tolerant to failures leading to Critical consequences, as defined in AD 1-2. However, single point failures that cannot be eliminated from the design with reasonable effort (or fault tolerance requirements which cannot be met) shall be summarised in a Single Point Failure/Critical Items List. They shall be subjected to formal approval by the prime contractor/ESA on a case-by-case basis with a detailed retention rationale.*

SYS - 310 *Withdrawn.*

SYS - 320 *The design of the spacecraft shall be such that a single component/part failure shall not cause the failure of spacecraft functions that are essential for mission success (e.g. power supply, data transmission, etc.).*

SYS - 330 *All functions that would lead to critical consequences (see AD 1-2) shall be monitored by at least two independent parameters following independent* *paths.*

SYS - 335 *It shall be possible to read AOCS primary and redundant sensor data separately and simultaneously.*

SYS - 340 *Hot redundancy shall be provided for functions that would lead to Critical and/or Catastrophic consequences in case of failure, as defined in AD 1-2.*

SYS - 350 *Redundant functions shall be physically separated to prevent propagation of failures.*

The physical separation is intended in terms of mechanical, thermal and electrical interfaces.

SYS – 360 *Protection of essential functions shall not be implemented in the same hybrid cavity/package or integrated circuit nor utilize common references or auxiliary supply.*

SYS - 370 *Provisions shall be made to prevent malfunction or elimination of redundant units by a common cause.*

SYS - 380 *It shall be possible by ground command to reverse the switching to a redundant unit provided that such switching capability cannot result in an irreversible undesirable configuration.*

SYS - 390 *Checkout of all redundant or contingency operational modes or units shall be possible during ground testing of the fully integrated spacecraft.*

## 5.8 Resources

### 5.8.1 Mass and Delta-V

SYS - 400 *The satellite total wet mass at launch shall not exceed 2030 kg, including the launcher vehicle adaptor.*

Note: This allocation is calculated considering a launch from Kourou with Soyuz-Fregat

SYS - 405 *If at the time of launch the launcher capacity is exceeding the current* *estimates, it shall be possible to increase the propulsion propellant load (see PROP-190) without re-qualification of the spacecraft.*

*SYS - 410* *The location of the spacecraft centre of gravity and moment of inertia in the launch configuration shall be compliant with the requirement defined in the Launch Vehicle User's Manual.*

*SYS - 415* *The satellite design shall satisfy the Delta-V requirements as defined the table below:*

| *Man* | | *ΔV* | *ΔV to sun angle* |
|---|---|---|---|
| *1* | *Launcher dispersion correction and perigee velocity correction (before day 2)* | *70 m/s* | *≥ 135°* |
| *2* | *Mid-course corrections and launch window reserve (sum)* | *20 m/s* | ***Any direction, possibly using manoeuvre decomposition.*** |
| *3* | *Orbit Insertion (fast transfer)* | *165 m/s* | *From 110° to 130°* |
| *4* | *Avoidance of deep eclipses by Moon* | *10 m/s* | *Non-escape direction* |
| *5* | *Orbit Maintenance for 6 years* | *13 m/s* | *24°-34° or 146°-156°* |
| | *Sum* | *278 m/s* | *Total impulsive ΔV without decomposition & geometric losses* |

Note: Manoeuvre number 5 will occur during the operational phase at L2. Detailed explanations are given in RD 1-1 section 5.1.
For the calculation of the effective ΔV, the satellite contractor shall take into account all losses (geometric, decomposition, ramping, etc).

### 5.8.2 Margin Policy

#### 5.8.2.1 Mass Margins

*SYS - 420* *At equipment level, the following design maturity mass margins shall be applied:*

- *5% for OTS equipment (ECSS category A & B);*
- *10% for OTS items requiring minor modifications (ECSS*

*category C);*

- *20% for new designed/developed items, or items requiring major modifications or re-design (ECSS category D).*

*SYS - 430* *At end of phase B (System PDR), the total mass launch of the spacecraft shall include a system level margin of at least 10% of the nominal mass at launch.*

*SYS - 435* *At System CDR, the total mass launch of the spacecraft shall include a system level margin of at least 5% of the nominal mass at launch.*

*SYS - 440* *An ESA reserve of 100 Kg shall be included in the satellite allocation.*

#### 5.8.2.2 Delta-V Margins

*SYS - 450* *The following Delta-V margin (covering "On the Flight" dispersions and contingencies such as: Launcher dispersions, manoeuvres inaccuracies, navigation errors) shall be applied to the effective Delta-V manoeuvres provided in SYS - 415:*

- *10% for trajectory manoeuvres;*
- *50% for the orbits maintenance manoeuvres over the specified lifetime;*
- *50% for the attitude acquisition and control.*

#### 5.8.2.3 Propellant Margin

*SYS - 460* *A margin of 5% shall be applied on the mass propellant calculation.*

#### 5.8.2.4 Power margins

*SYS - 470* *At equipment level and for conventional units, the following design maturity power margins shall be applied:*

- *5% for OTS equipment (ECSS category A/B);*
- *10% for OTS items requiring minor modifications (ECSS category C;*
- *20% for new designed/developed items, or items requiring major*

*modifications or re-design (ECSS category D).*

*SYS - 480* *The power resources shall be dimensioned with adequate margin providing power up to the end of extended mission as defined in AD 3-6 (ECSS E-20A Electrical & Electronic). However, the margins at EOL for the most critical phase shall be at least 10% including one string failure and one cell failure for the solar array and battery respectively.*

*SYS - 490* *An ESA reserve of 100 W shall be included in the satellite allocation.*

Note: During launch the ESA reserve of 100W shall not be considered.

#### 5.8.2.5 Pin Connector Margins

*SYS - 495* *The pin connectors margin availability shall be as follow:*

- *50% at PDR*
- *30% at CDR*
- *10% at FAR*

#### 5.8.2.6 Data Processing Margins

*SYS - 500* *Requirements in this section shall be computed with worst-case scenarios to be agreed with ESA (at latest for the System PDR).*

*SYS - 510* *Margins to be applied for the size of RAMs, EEPROMs, and PROMs, shall be:*

- *50% at System PDR;*
- *40% at System CDR;*
- *30% at System FAR.*

*SYS - 520* *Margins to be applied for the load of data busses (1553, OBDH or others) loads shall be:*

- *50% at System PDR;*
- *40% at System CDR;*
- *30% at System FAR.*

*SYS - 530* *Margins to be applied for the processing load of all CPUs shall be:*

- *50% at System PDR;*

- *40% at System CDR;*
- *30% at System FAR.*

#### 5.8.2.7 Communication Margins

*SYS - 540* *Links budget and associated margins, for all phases of the mission, shall be computed with at least the following margins:*
- *nominal margin: 3 dB;*
- *RSS worst-case margin: 0 dB;*
- *mean $-3\sigma$ margin: 0 dB.*

## 5.9 Spacecraft Modules

The satellite is composed of a Service Module (SVM), and a Payload Module (PLM), with their software. Sunshield is considered part of the SVM.

*SYS - 550* *The satellite shall be configured in a modular way so that the SVM and PLM can be individually integrated and tested.*

*SYS - 560* *The design shall allow simple mounting and dismounting procedures so that the instrument units can be installed or replaced late in the integration sequence.*

*SYS - 570* *Stimuli and test connectors shall be accessible and positioned together on one specific satellite area.*

*SYS - 580* *The S/C design shall foresee all necessary interfaces and tooling such to enable dedicated health checks and tests on ground at all levels and in open/closed loop configuration after S/C integration. Where the requirement cannot be met, this shall be justified.*

*SYS - 590* *The satellite physical envelope shall be compatible with the payload allocated volume as specified in the Soyuz-Fregat Launcher User Manual AD 2-1.*

## 5.10 Structure Requirements

### 5.10.1 General

General definitions and ESA standards applicable to structure are given in AD 3-9 (ECSS E-30 part 2A).

*STR-010* *The structure shall comply with requirements of AD 3-9.*

### 5.10.2 Functional Requirements

*STR-020* *The spacecraft structure shall provide the physical interfaces to all equipment and payloads in accordance with the spacecraft physical configuration.*

*STR-030* *The spacecraft structure shall provide the physical interface to the launch vehicle.*

*STR-040* *The spacecraft structure shall allow all alignments without disassembling parts after the spacecraft integration.*

*STR-050* *The spacecraft structure shall support all relevant equipment and payloads in the prescribed relative alignment with the required stability during ground and in-orbit operations.*

### 5.10.3 Performance Requirements

#### 5.10.3.1 Stiffness

*STR-060* *The lowest frequencies and effective masses of the spacecraft hard mounted and in launch configuration shall be compliant to requirements reported in the applicable Launch Vehicle User's Manual.*

*STR-070* *The structural design shall provide a minimum margin of 15% over the* *specified frequencies before verification of the spacecraft dynamic properties by test assuming maximum allocated mass.*

*STR-080* *The structural design of secondary structure and equipment shall prevent the dynamic coupling of equipment with the spacecraft structural modes. Whenever this requirement cannot be met, equipment shall be designed to withstand the effects specific to the dynamic amplification caused by the coupling with the spacecraft modes.*

*STR-090* *The stiffness of the spacecraft in-orbit configuration shall preclude the interaction of the spacecraft flexible modes with the attitude control system*

*STR-100* *Adjacent structural elements supporting electronic equipment shall have an electrical continuity between the various parts better than 5 mΩ.*

*STR-110* *The spacecraft configuration including its payload and appendages shall comply with the envelope requirements applicable to ground handling, transportation and launch when subject to the worst case combination of limit loads and root-mean-square manufacturing tolerances.*

#### 5.10.3.2 Strength

*STR-120* *A strength analysis shall demonstrate for every retained worse combination, positive margin of safety after application of the relevant (following) loading:*

- *Yield load, without permanent deformation or any elastic deformation resulting in performance degradation;*
- *Ultimate load, without rupture, or deformation leading to loss of functionality;*
- *Buckling load, without buckling or collapse, taking into account a non-perfect geometry of the failing elements.*

STR-140 The design shall have a positive margin of safety which is defined by the following formula:

$$\frac{\text{Allowable load (or Stress)}}{\text{Design load (or Stress)} \times \text{Safety Factor} \times \text{Additional Safety Factor}} - 1 \geq 0$$

STR-150 *The following minimum design factors of safety shall be applied to both mechanical and additive thermo-elastic loads:*

| ***Structure type / sizing case*** | ***FOSY*** | ***FOSU*** | ***FOSY for verification by analysis only*** | ***FOSU for verification by analysis only*** | ***Additional factors (KD)*** |
|---|---|---|---|---|---|
| *Metallic structures* | *1.1* | *1.25* | *1.25* | *1.5* | |
| *FRP structures (uniform material)[1]* | | *1.25* | | *1.5* | |
| *FRP structures (discontinuities)* | | *1.5[2]* | | *2.0* | |
| *Sandwich structures:*<br>*- Face wrinkling*<br>*- Intracell buckling*<br>*- Honeycomb shear* | | <br>*1.25*<br>*1.25*<br>*1.25* | | <br>*1.5*<br>*1.5*<br>*1.5* | <br>*1.2*<br>*1.2*<br>*1.2* |
| *Glass/Ceramic structures[3]* | | *2.5* | | *5.0* | |
| *Joints and Inserts*<br>*- Failure*<br>*- Gapping*<br>*- Sliding* | | <br>*1.25* | <br><br>*1.25*<br>*1.25* | <br>*2.0* | <br><br><br>*1.2* |
| *Pressurized hardware for the interface loads[4]* | *1.1* | *1.25* | *1.25* | *1.5* | |
| *Buckling* | | *2.0[5]* | | *2.0[5]* | |

*Associated to the above factors to be used, the following assumptions shall be taken into account:*

*[1]) Applicable failure criteria have to be agreed with the customer*

*[2]) This coefficient applies to concentrated stresses*

[3]) *These materials have strength properties that are highly dependant of the manufacturing process and of the surface quality.*

[4]) *This coefficient applies to general stress analysis on internal pressure and external loads. For damage tolerance or safety analysis, refer to ECSS-E30-02*

[5]) *For global buckling, the factor of safety does not include any knock down factor which is included in the results of the buckling analysis.*

*STR-152* *Design loads shall be derived by multiplication of the limit loads by a design factor equal to 1.5, i.e. DL = 1.5 X LL.*

*STR-155* *A reduced Fracture Control Programme as defined in AD 3-15 clause 11 shall be implemented.*

*STR-160* *Under the design loads and where the mass of each unit or component is set at the maximum allocated, the forces fluxes at the launcher interface shall meet the launcher requirements as specified in the Soyuz-Fregat Launcher User Manual AD 2-1.*

*STR-165* *Qualification factors as defined in the Launcher User Manual shall be applied.*

#### 5.10.3.3 Alignment and Stability Requirements

*STR-170* *The stability of the structure shall comply with requirements as defined in Chapter 4 and shall be dimensioned in accordance with the spacecraft pointing requirements.*

*STR-180* *The following causes of misalignment shall be analysed and quantified:*

- *Setting due to mounting procedures;*
- *Setting due to launch distortions;*
- *Misalignment due to gravity release in-orbit;*
- *Thermal deformations under in-orbit temperatures;*
- *Ageing;*
- *Composite structure deformations due to moisture release in-orbit.*

### 5.10.4 Mechanical Analysis

STR-190 *Finite-Element Models (FEM) shall be prepared to support the mechanical and configuration activities at module level and to validate the performance at satellite level. FEM shall be prepared in NASTRAN format.*

STR-195 *The adequacy of the FEM to predict thermo-elastic deformations of the expected order of magnitude shall be demonstrated.*

STR-200 *The FEM of the S/C in its launch configuration shall be detailed enough to ensure an appropriate derivation and verification of the design loads and of the modal response of the various structural elements of the satellite up to 140 Hz.*

STR-210 *The FEM's shall be verified and correlated against the results of the modal survey tests carried out at component, sub-assembly and complete S/C level. The quality criteria for the test-analysis correlation shall ensure that the dynamic response predictions employing the test validated mathematical model are compliant with the accuracy requirements here reported:*

| ***Item*** | ***Quality criterion*** [1] | |
|---|---|---|
| *Fundamental bending modes of a spacecraft* | *MAC:*<br>*and eigenfrequency deviation:* | *> 0.9*<br>*< 3%* |
| *Modes with effective masses > 10 % of the total mass* | *MAC:*<br>*and eigenfrequency deviation:* | *> 0.85*<br>*< 5 %* |
| *For other modes in the relevant frequency range* [2] | *MAC:*<br>*and eigenfrequency deviation:* | *> 0.8*<br>*< 10 %* |
| *Cross-orthogonality check* | *Diagonal terms:*<br>*Off-diagonal terms:* | *> 0.90*<br>*< 0.10* |
| *Damping* | *To take measured values as input for the response analysis.*<br>*To use realistic test inputs for this purpose.* | |
| *Interface force and moment measurements* | *For modes with effective masses > 10 %: deviations of interface forces and moments < 10 %.* | |

STR-220 *Reduced FEM mathematical models to be employed in dynamic response* *predictions shall represent the detailed FEM in compliance with the model quality criteria here reported:*

| ***Item*** | ***Quality criterion*** | |
|---|---|---|
| *Frequencies and modal masses of fundamental lateral, longitudinal and torsional modes* | *Effective mass:*<br>*Eigenfrequency deviation:* | *< 5 %*<br>*< 3 %* |
| *For other modes up to 100 Hz* | *Effective mass:*<br>*Eigenfrequency deviation:* | *< 10 %*<br>*< 5 %* |
| *Modes of reduced FEM (within frequency range of interest)* | *Total effective mass > 90 % of the rigid body mass.* | |

*STR-230* *Adequate models to calculate CoG and MoI shall be prepared to support the mechanical and configuration activities at satellite level.*

## 5.11 Mechanisms

All assemblies featuring parts moving under the action of commendable internal forces shall be considered as mechanisms.

General requirements, definitions and ESA standards applicable to mechanism are given in AD 3-10 (ECSS E-30 part 3A).

*MEC-010* *The mechanisms shall comply with requirements of AD 3-10.*

*MEC-020* *The functional performance of the mechanisms shall be described with:*

- *Kinematics variables of the motion, i.e., as acceleration, velocity, displacement;*
- *Dynamic Variables, i.e., forces and torque applied to the various mobile parts;*
- *Steady State Parameters of the initial and final status of the motion, i.e., relative position or relative velocity w.r.t. a well-identified interface;*
- *Physical Parameters (e.g., mass, inertia, spring forces, friction, hysteresis, adhesion) that entail the kinematic variables.*

*MEC-030* *Release mechanisms (e.g. pyrotechnics) shall be capable of being manually operated for test purpose. In case one-shot initiators are used, simple re-installation of a new device on the spacecraft shall be possible.*

*MEC-040* *The mechanism design shall be compatible with operations in ambient and thermal vacuum conditions and gravity in any orientation.*

*MEC-050* *The mechanism shall feature a feedback technique enabling to determine unambiguously its position.*

*MEC-060* *When latching mechanisms rely on a preload to be applied during launch a method of monitoring (and adjusting if necessary) shall be provided for the spacecraft AIV.*

*MEC-070* *The use of pyrotechnics, as far as practicable shall be avoided.*

If no solution other than pyrotechnics exists, than requirements reported in the AD (ECSS-E-30-3a) will be implemented.

*MEC-080* *In case pyrotechnics and other one-shot device are used, an arm/execute mechanism shall be implemented.*

## 5.12 Thermal Requirements

### 5.12.1 General

General requirements, definitions and ESA standards applicable to platform and payload thermal control are given in AD 3-8 (ECSS E-30 part 1A).

*THE-010* *The spacecraft thermal control shall comply with requirements of AD 3-8.*

*THE-020* *The primary objective of the Gaia thermal control function shall be to provide a temperature environment to the satellite which ensures reliable performance of all units during all phases of the mission, including handling, transportation, testing, storage, and launch.*

*THE-030* *The TCS shall be achieved by passive means (i.e., coating, MLI, conductive paths, insulating washers, etc.) and by heaters.*

*THE-040* *The design shall permit analysis by mathematical models.*

*THE-045* *The TCS must be testable on ground. If special equipments are required to evacuate the heat during the spacecraft functional tests under ambient environment, they shall be compatible with the cleanliness requirements. No TCS item shall prevent the spacecraft from being operated/tested under an attitude required by the thermal environment test.*

*THE-050* *Deviations and temporal degradations from the nominal values of external and internal fluxes, thermo-optical properties, heat capacitances, and conductive and radiative couplings shall be taken into account in the thermal analysis.*

*THE-060* *A thermal reference point (TRP) shall be defined and instrumented for all units. This reference point shall be selected such that its temperature reflects the general thermal status of the equipment. The TRP temperature* *also called "interface temperature" shall be predicted by analysis. The TRP shall be therefore represented in the mathematical models by a thermal node. In addition, the TRP temperature shall be monitored during the thermal tests performed at any level.*

*THE-070* *The TCS shall include and monitor sufficient flight temperature sensors -to evaluate its in-orbit performances.*

*THE-080* *The design limits of units are equal to the qualification limits decreased at both ends by a margin of 10°C and/or equal to the acceptance temperature limits decreased at both ends by a margin of 5°C.*

### 5.12.2 Thermal Analysis

*THE-090* *Mathematical Models in ESARAD, THERMICA and ESATAN shall be prepared to support the thermal analyses as necessary.*

*THE-100* *The GMMs and TMMs of the satellite/modules shall be verified by test and correlated against the measured data as per AD 3-8.*

## 5.13 AOCS Requirements

### 5.13.1 General

The Attitude and Orbit Control and Measurement Sub-system (AOCS) provides the hardware and associated on-board software to acquire, control, and measure the attitude of the satellite during all mission phases and modes according to the specified system requirements.

*AOCS - 010* *The AOCS shall be able to provide all necessary capabilities to satisfy the attitude and orbit control and measurement requirements of all mission phases, operational modes and operational functions compatible with the needs of payload, scientific objectives, and communication and power supply.*

*AOCS - 020* *The detailed requirements of the AOCS shall be derived from the mission performance requirements associated to the functions entrusted to the system, e.g. attitude and orbit control, instrument calibration, etc.*

### 5.13.2 Functional Requirements

*AOCS - 030* *After separation from the launch vehicle, the AOCS shall:*

- *Damp out the residual angular rates*
- *Bring the spacecraft into a power safe pointing attitude within a time compatible with the spacecraft internal electrical energy capability*
- *Maintain the spacecraft into this pointing attitude ready to receive ground commands.*

*AOCS - 050* *The spacecraft shall transmit via telemetry the AOCS unambiguous status information of all command and programme controlled variables and modes and of all parameters required for sub-system monitoring and performance evaluation.*

*AOCS - 060* *The AOCS shall acquire and control the attitude necessary for the correct execution of the various orbit insertion, correction, and maintenance manoeuvres.*

AOCS - 070 *The AOCS shall avoid unsafe attitudes, which might endanger the mission or degrade scientific instrument performance.*

AOCS - 075 *The AOCS shall be able to autonomously detect a violation of the attitude constraints defined in Chapter 4.*

AOCS - 080 *AOCS shall provide the capability for detecting, isolating and recovering any anomaly resulting in the loss of nominal pointing.*

AOCS - 090 *All AOCS functions shall be redundant.*

AOCS - 100 *The AOCS shall be able to monitor a redundant equipment while the nominal is used in the AOCS loop, and provide the telemetry of the redundant equipment to CDMS for house keeping.*

AOCS - 110 *The AOCS shall provide the data necessary for spacecraft attitude reconstruction on ground at all times and any auxiliary data necessary in the orbit determination process.*

AOCS - 120 *After a major on-board failure or a violation of the attitude constraints (safe mode), the AOCS shall maintain:*

1) *a safe attitude within the constraints allowing a continuous supply of power and communication to Earth and shall maintain*
2) *a thermal environment compatible with the spacecraft and essential loads.*

AOCS - 130 *The execution of the AOCS safe mode shall, once started, not be interruptible by an external command: recovery from this mode shall only be possible after a stable Sun-pointing attitude has been acquired or a predefined time-out.*

### 5.13.3 Performance Requirements

AOCS - 140 *The AOCS pointing performances shall comply with the overall spacecraft pointing requirements as specified in Chapter 4 and shall be demonstrated to be met in the presence of perturbing torques, spacecraft flexible modes,* *or liquid slosh if any.*

*AOCS - 150* *The attitude control shall be stable (at least 6 dB gain margin and 30deg phase margin) in all modes in the presence of perturbing torques, spacecraft flexible modes, or liquid slosh during all mission phases and in all operational modes*

*AOCS - 160* *The design of the AOCS shall be such that a single component/part failure cannot cause the failure of functions that are vital for mission success.*

*AOCS - 170* *During the transfer phase, the AOCS shall not make use of the P/L data for spacecraft attitude control purposes.*

## 5.14 Propulsion Requirements

### 5.14.1 General

*PROP - 010* *The propulsion subsystem shall be able to:*

- *Provide attitude control around three orthogonal axes during all mission phases*
- *Correct for launcher injection error*
- *Perform orbit corrections to maintain the orbit parameters within the required range during the mission life*
- *Recover from attitude loss*
- *Support the scan law requirements*
- *Provide thrust for delta-V manoeuvres (as per section 5.8.1) and attitude control torques during all mission phases after separation from the launch vehicle as requested*

*PROP - 020* *The detailed requirements on the Propulsion Subsystem shall be derived from the mission and system performance requirements associated to the functions entrusted to the system, e.g. attitude and orbit control, instrument calibration, etc.*

*PROP - 030* *The characteristics of the thrusters and their accommodation on the satellite shall not cause any deleterious effects on either the spacecraft or the payload during the mission.*

*PROP - 040* *Thruster directions and locations shall be selected to minimize generation of disturbing forces and torques and fuel consumption.*

*PROP - 050* *The layout of the Propulsion Subsystem shall ensure symmetric depletion of fuel in all tanks during thruster firing to minimize the shift of spacecraft centre of gravity during the mission.*

*PROP - 060* *Values of dispersion errors and perturbation with a probability of 99.73% shall be considered in the determination of propellant budgets.*

*PROP - 070* *The location of the fill and drain/vent valve (FDV/FVV) shall be selected to facilitate the propellant loading, draining and the propulsion subsystem evacuation.*

*PROP - 080* *Design of any Propulsion subsystem used on Gaia shall be compliant with the safety and cleanliness/contamination requirements applicable to:*
- *The launcher;*
- *The launch range;*
- *The spacecraft itself.*

### 5.14.2 Functional Requirements

*PROP - 090* *A continuous monitoring capability of main propulsion subsystem items shall be provided for health monitoring, propellant determination and failure detection during both ground operations and in-orbit operations to the ground through telemetry.*

*PROP - 100* *The actuation of Propulsion subsystem shall not generate any electrostatic charge.*

*PROP - 110* *The propulsion subsystem shall provide sufficient data to perform:*
- *Failure detection and isolation;*
- *Switch over to the redundant resources;*
- *Shut down and isolation of malfunctioning parts.*

*PROP - 130* *Any propellant leakage during orbit operations shall not cause unrecoverable spacecraft rates.*

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### 5.14.3 Performance Requirements

*PROP - 140* *Thruster location shall avoid any impingement on sensitive equipment.*

*PROP - 150* *The Propulsion Subsystem shall provide a thrust repeatability, thrust vector stability, thrust controllability, thrust modulation capability and thrust accuracy that are compliant with the mission requirements..*

*PROP - 160* *The Propulsion Subsystem shall provide means for propellant determination within 5% accuracy.*

*PROP - 170* *In order to prevent any detrimental effect on the science performances, means to limit the propellant sloshing shall be adopted so as to avoid the use of on-board estimation.*

*PROP - 180* *The Propulsion Subsystem shall include means to prevent any propellant leakage during ground and launch operations.*

*PROP - 190* *The propellant tank(s) capacity shall cover the full launch mass and allow for an additional potential propellant growth of 15%.*

*PROP - 200* *The propulsion subsystem shall provide safety physical barriers on-ground as requested by Launcher documentation (paragraph 2.1.2 of this document).*

*PROP - 210* *The propulsion subsystem shall allow monitoring of pressure and temperature, during filling operation, after spacecraft complete integration independent of power status of the spacecraft itself.*

## 5.15 Electrical Power Subsystem

### 5.15.1 General

General, definitions and ESA standards applicable to EPS are given in AD 3-6 & AD 3-25 (ECSS E-20A, ECSS E-20-08A).

*POW - 010* *The design of the EPS shall be in accordance with AD 3-6 and AD 3-25, shall comply with the system requirements, and shall be compatible with the scientific instruments and the other sub-systems.*

*POW - 020* *The EPS shall be able to generate and distribute sufficient power to the scientific instruments and spacecraft sub-systems to operate these according to the mission requirements, for all operational modes and during all mission phases.*

*POW - 030* *The power shall be provided by means of both solar array and batteries in accordance to the Gaia mission profile.*

According to the characteristics of the Gaia mission, no solar eclipse period caused by the Earth should occur, so ordinary use of batteries should be mainly limited to the initial phases.

*POW - 040* *In case of power contingency mode (e.g., battery undervoltages), all the Non Essential Loads (including instruments) shall be autonomously switched off on board.*

*POW - 050* *Any protection latch not having autonomous reset capability shall be at least resettable from ground command.*

### 5.15.2 EMC and ESD Design Requirements

*POW - 060* *The design principles shall be defined by EMC system analysis. Anyhow, the sub-system shall fulfil the EMC and ESD requirements of Chapter 7. General design principles to be implemented are:*

- *Twisted power lines shall be pre-manufactured by the supplier to achieve the best magnetic compensation;*
- *Other lines shall be twisted as far as practicable with their corresponding return path, or shall run adjacent to the reference grounds to minimise magnetic loops;*
- *Harness shielding, as far as required, shall preferably be made by means of overall shielding, where the signals in the bundles do not interfere with one another;*
- *The shields shall be connected to the equipment housings via connector pins or to the connector case, if no other special arrangement is derived from the EMC analysis of the interface design.*

### 5.15.3 Functional Requirements

*POW - 070* *The EPS shall generate, condition, control, monitor and distribute electrical power to the spacecraft users from a regulated bus, and shall store electrical energy and manage battery charge / discharge to fulfill power demands throughout all mission phases in the presence of all environments actually encountered.*

*POW - 080* *The EPS shall be capable of operating continuously under all operational conditions of the mission including contingency situations. No damage or degradation shall result from intermittent or cycled operation.*

*POW - 090* *The sub-system shall provide adequate status monitoring and telecommand interfaces necessary to operate the sub-system and permit evaluation of its performance (during ground testing and in-flight operations) and failure detection and recovery*

*POW - 100* *The EPS shall restart automatically and autonomously after a complete main bus loss when solar array power reappears.*

*POW - 110* *Essential functions (e.g., synchronisation or auxiliary power supply) shall not rely on centrally generated auxiliary functions. Any EPS equipment shall be able to operate independently of any external synchronisation or auxiliary power supply.*

*POW - 115* *In the case of hot redundant essential functions, latching protection shall not be used, or if applied shall have an autonomous periodic reset. Override of critical on-board autonomous functions shall be implemented* *only if a safety interlock is implemented which prevents the activation of the override feature on both hot and redundant functions.*

*POW - 120* *No single point failure shall endanger full mission performance or cause permanent damage to any essential load. An automatic protection shall be provided to ensure that the power sub-system is able to recover from any malfunction in any load or from any abnormal spacecraft mode of operation.*

*POW - 130* *At power up, restart and upon recovery from any power loss, the SVM shall set the spacecraft electrical configuration (including all sub-systems, units, and instruments) into in a known deterministic and reproducible state. This state shall be safe (full battery charging capability and minimum power bus loading) and shall allow a predefined recovery of the spacecraft and of its sub-systems.*

Note: It is recommended to reset the automatic protections at power up and restart upon recovery from any power loss.

*POW - 140* *All power carrying items within the sub-system shall have double isolation.*

### 5.15.4 Performance Requirements

*POW - 150* *The power conditioning shall be designed such that a regulated DC Main Bus is provided to all satellite users.*

*POW - 160* *Voltage of the main bus shall be carefully selected and justified considering the different drivers (e.g. power losses, solar array technology, COTS hardware, bus impedance and regulation accuracy).*

*POW - 170* *Any power line available from the power subsystem shall be protected against short circuit or overload appearing at user's side. The overload and short circuit protection shall be achieved by current limiters provided with trip off capability. Use of fuses shall be avoided.*

*POW - 180* *The management of the power bus shall be autonomous and completely independent from any control by the on-board computer.*

*POW - 190* *No single point failure in the spacecraft including wiring and connectors shall open or short the main electrical power bus or cause any over voltage.*

*POW - 200* *Main bus regulation accuracy and impedance shall be compliant with requirements of AD 3-6 and mission science needs.*

*POW - 205* *The EPS telemetry monitoring shall allow determination of the battery state of charge from ground to an accuracy of better than 10%.*

### 5.15.5 Design Requirements

*POW -210* *The EPS shall provide all resources needed for the operation of release of deployable items for payload as well as for spacecraft functions.*

*POW - 220* *Whenever two or more blocks are interfaced, the compatibility of the source with the load shall be specified and verified to ensure stability of the cascade. The overall block (cascade of source and load block) shall meet the stability margins specified in ECSS-E-20A section 5.6.3 and the source/load impedances shall respect adequate constraints to be specified by the overall responsible of the cascaded blocks design.*

*POW - 230* *No single failure in the EPS (including connectors) shall result in under/over voltage exceeding limits and recovery time specified in AD 3-6.*

*POW - 235* *The electrical power interface between the solar array(s) and the power control unit(s) and between batter(ies) and power control unit(s) shall be defined and shall result in the specification of the input impedance seen by the power conditioning unit(s).*

*POW - 240* *The solar array electrical performance for each major mission phase shall be predicted. Performance figures shall be based on accepted cell degradation figures and actual cell performance measurement and include all effects that have an impact on the performance (e.g., solar attitude, Sun intensity, radiation, micro-meteorites, etc.).*

*POW - 250* *The solar array design shall be one-string failure tolerant as per requested power margin philosophy.*

*POW - 260* *All solar array cell strings shall have individual blocking diodes and shunt diodes where required.*

*POW - 270* *The solar array shall meet the requirements of AD 3-25.*

*POW - 280* *The electrical network shall be composed of identical electrical sections, as far as possible. It shall minimise the resulting magnetic moment and ensure the insulation of solar network with respect to the solar array structure.*

*POW - 290* *The solar array design (including all units placed on its surface) shall minimise the generation of periodic temperature changes in the SVM and PLM induced by the spin modulation of the solar flux.*

*POW - 300* *The solar array geometrical and electrical configuration shall be such that system EMC requirements for electrostatic cleanliness are met.*

*POW - 310* *The solar cells shall be protected against hot spots.*

*POW-320* *Solar cell degradation shall be derived from electron fluence and coverglass thickness values as given in AD 1-6.*

*POW – 330* *All current carrying tracks and wires after the blocking diodes shall have double isolation.*

*POW - 340* *Battery selection and design, the number of discharge cycles, the operating temperature and temperature gradient and the battery charge/discharge management shall ensure fulfilment of the satellite power requirements to be compliant with the battery depth of discharge requirements.*

POW - 345 *For battery technologies where battery performance and/or lifetime is severely impaired by excessive charge or discharge (e.g. lithium-ion), provisions shall be made to prevent exceeding of these limits. The design shall take into account any residual discharge currents due to leakage through connected circuitry.*

POW - 350 *Protection against excessive overcharge, undervoltage, overheating and freezing shall be provided.*

POW - 360 *Galvanic insulation shall be provided between battery structure, battery cell case, and thermistor leads.*

POW - 370 *The batteries and their regulator units shall be functionally one failure tolerant. Batteries shall be cell failure tolerant. The back-up system shall be completely independent of on-board computer control.*

POW - 380 *To ensure recovery from loss of spacecraft attitude, the minimum charge rate available to the battery shall be either C/10 or such that there is enough margin to ensure effective battery charging with the essential loads connected on the bus and one worse case load connected representing a failure, whichever is the more constraining*.

POW - 390 *The stability of current limiters shall be addressed taking into account the actual loads characteristics.*

POW - 400 *In case the distribution lines are protected by latching, or periodically reset current limiters, it shall be ensured by worst case electrical/thermal analyses and test the inrush energy demanded by the load in normal switch-on and/or in automatic re-triggering does not cause the trip-off of the latching protection with a 20% margin.*

This requirement will be verified by analysis under worst case conditions, and tested under most representative set of cases.

## 5.16 Electrical Distribution Requirements

### 5.16.1 General

The harness provides all electrical connections between all electrical equipment. It includes harnesses for power supplies, signals and pyrotechnic pulses. It includes also harnesses for connections with the umbilicals and test connectors.
The harness includes also all fixation plates, bond straps, clamps support bracketry, metallic brackets used for interface connectors grommets, edge protection, connector savers, and thermal insulation.

*EDR - 010* *The harness shall provide adequate distribution and separation of all power supply lines, analogue and digital data lines, command and actuation pulse and stimuli lines between all units of the SVM sub-systems and those lines to the PLM, the test connectors, the safe/arm brackets and connectors and the umbilical connectors.*

*EDR - 020* *The harness shall be designed in accordance to AD 3-6 and AD 3-7.*

*EDR – 025* *The sub-system shall meet all EMC requirements as reported in Chapter 7.*

### 5.16.2 Functional Requirements

*EDR - 030* *The harness shall transmit all electrical currents in a manner compatible with the requirements of the source and destination unit/interface.*

*EDR - 040* *The signal deterioration due to resistive, inductive and capacitive behaviour of the interconnection lines or coax cables shall be such that all the relevant applicable sub-system specifications are met in the integrated satellite.*

*EDR - 050* *The harness layout shall guarantee the minimum distance of the EMC classes (as defined in AD 3-6) for segregation.*

*EDR - 060* *The power harness DC equivalent resistance from the main regulation point to the input of the load (line and return) shall not exceed the value required by the power standard AD 3-6 par 5.7.1.*

*EDR - 070* *The inductance of the power distribution harness for frequencies up to 100 kHz shall not exceed the value required by the power standard AD 3-7.*

*EDR - 080* *The isolation requirements between leads, which are not connected together and between shield and centre conductor and shield to shield shall be at least 10 MΩ under 500 V DC at both polarities.*

*EDR - 090* *The mechanical construction of the harness shall assure the reliable operation of the spacecraft under all environmental conditions. The stress, which occurs during manufacturing, integration, test, transport, launch preparation, launch and in-orbit operation shall cause no changes in the harness, which might affect the correct functioning of the system. No piece of harness shall be used as a mechanical support.*

### 5.16.3 Design Requirements

*EDR - 100* *Different connector classes shall be implemented in order to separate the different type of links: Power, Signal, and Pyros. Signals falling into different EMC classifications shall be assembled to separate connectors and cable bundles. If not feasible, the separation shall be achieved by a row of unused pins and the cables shall split into their respective categories as soon as they leave the connector or connector backshell.*

*EDR - 110* *All equipment shall use a separate connector dedicated to its functional interface, according to the categories listed above.*

*EDR - 120* *Wiring of redundant systems, sub-systems or units of sub-systems shall be routed through separate connectors and wire bundles.*

*EDR - 140* *Cross strapping of redundant paths and circuits shall not be carried out in the harness.*

| | |
|---|---|
| *EDR - 150* | *The pyrotechnic harness shall satisfy the applicable safety requirements.* |
| *EDR - 160* | *The pyrotechnic harness shall consist of twisted pairs of wires with an overall shield being continuous and connected to the conductive connector shells at all interfaces and grounded to the structure at all intermediate attachment points.* |
| *EDR - 170* | *Connections to the initiators shall be capable of being mechanically broken during ground handling by safe/arm connectors accessible from the outside of the spacecraft.* |
| *EDR - 180* | *The shields of cables shall not be used as return lines.* |
| *EDR - 190* | *All hot/return lines shall be twisted together.* |
| *EDR - 200* | *All harness and all box and bracket mounted connectors supplying power shall have socket contacts.* |
| *EDR - 210* | *Where it is necessary to have a shield connection through a connector, separate pins shall be used.* |
| *EDR - 220* | *The initial design shall ensure that sufficient (including spare) pins are available for all foreseeable sub-system and payload functions.* |
| *EDR - 230* | *All individual wire-to-pin interfaces shall be covered with transparent heat shrink sleeves.* |
| *EDR - 240* | *The possibility of incorrect mating of connectors shall be excluded by design.* |
| *EDR - 250* | *The harness connectors shall be easily accessible, attachable and removable from the corresponding unit connectors; removal of units or disconnection of adjacent connectors shall not be necessary.* |

*EDR - 260* *The harness shall be fixed onto the structure in order to avoid any damage during launch phase. As a general rule it will be fixed:*

- *At equipment level: the harness connectors shall be fitted onto the equipment connectors by appropriate locking systems*
- *At the structure level*
- *At interface level: the connectors shall be fixed on metallic brackets themselves fixed onto the structure.*

*EDR - 280* *The harness restraining systems on the structure shall not bring about any stress at connector level.*

*EDR - 290* *Permanent connections installed for purposes of test at integrated satellite level shall be routed to skin connectors of the modules concerned (module interface connectors are no longer accessible at that level).*

*EDR - 300* *Skin connectors shall also be provided to make-or-break power circuits.*

*EDR - 310* *Caps, bridging connectors, and thermal insulation for flight shall close all these skin connectors. During testing activities these connectors shall be protected by connector savers.*

#### 5.16.4 Interfaces between PLM and SVM

*EDR - 320* *The harness linking equipment mounted in or on the payload module, with the service module shall be such that the heat transferred by conduction is optimized for electrical and thermal performances*

*EDR - 330* *Redundant interface connectors shall be provided for power lines, signal lines, high data lines, coaxial lines if any and pyro lines. The connectors shall be non inter-changeable.*

### 5.16.5 Umbilical and Test Connectors

*EDR - 340* *There shall be umbilical and test connectors to provide electrical interfaces with the launcher and the EGSE. Functions provided shall include all those necessary for supporting AIT and launch site activities.*

*EDR - 350* *The harness interface shall be compatible with the Soyuz-Fregat launcher.*

*EDR - 360* *Safe and arming plugs shall be provided for disabling of hazard functions.*

*EDR - 370* *Test harness shall be provided so that the satellite can be stimulated and monitored during functional testing. Test harness end connectors shall be located at the skin of the spacecraft so that they are accessible also when the spacecraft is fully equipped with MLI. Design of the test harness shall take into account critical lengths. Wherever possible the test harness shall be removed for the flight configuration.*

## 5.17 Telemetry, Tracking & Command Requirements

### 5.17.1 General

*TT&C - 010* *The TT&C Sub-system shall be able to simultaneously receive and demodulate telecommands, modulate and transmit the telemetry, and transpond the ranging signal. Those functions shall be available during all mission phases including pre-launch and launch.*

*TT&C - 020* *The TT&C Sub-system shall interface with the ground segment according to the requirements of the Gaia Space/Ground ICD (AD 1- 4 & AD 1-5).*

*TT&C - 030* *The RF system shall receive and demodulate telecommands, modulate, and transmit the telemetry, and transpond the ranging signals. It shall be compliant with the following ESA standards:*

- *Radio Frequency and Modulation Standard (AD 3-18);*
- *Ranging Standard (AD 3-20);*
- *TM Synchronization and Channel Coding (AD 3-21).*

### 5.17.2 Design Requirements

*TT&C - 040* *The links with the ground stations shall be established using the X-band frequency range (for both uplink and downlink) allocated to Space Research category A missions.*

*TT&C - 050* *The TT&C sub-system shall have the capability of recovering from a failure autonomously. In all cases, it shall be possible to override the autonomous recovery action by use of ground commands.*

*TT&C - 060* *The sub-system design shall ensure that all its relevant operational parameters are acquired via suitable sensors and provided to the CDMS for incorporation into the housekeeping telemetry.*

*TT&C - 070* *The sub-system design shall ensure testability, including hot redundant functions, and failure indication.*

*TT&C - 080* *The sub-system shall meet all EMC requirements as reported in Chapter 7.*

### 5.17.3 Functional Requirements

*TT&C - 090* *The TT&C Sub-system shall support the following modes for the uplink:*
*- Carrier only;*
*- Telecommand;*
*- Ranging;*
*- Simultaneous Telecommand and Ranging.*

*TT&C - 100* *The TT&C Sub-system shall support the following modes for the downlink:*
*- Carrier only;*
*- Telemetry;*
*- Ranging;*
*- Simultaneous Telemetry and Ranging.*

*TT&C - 110* *The TT&C sub-system shall accept uplink signals and provide a demodulated digital telecommand signal to the CDMS for further processing. This function shall always be enabled without any possibility of switching it off for both main and redundant.*

*TT&C - 120* *The TT&C sub-system shall accept a digital telemetry signal from the CDMS and modulate it onto a downlink carrier. It shall be possible to enable/disable this function by telecommand.*

*TT&C - 130* *The TT&C sub-system shall provide a range and/or range rate measurement capability in accordance to AD 3-20.*

*TT&C - 140* *It shall be possible to operate the TT&C sub-system in coherent and non-coherent mode and this mode shall be selectable via telecommand.*

*TT&C - 150* *Hot redundancy shall be provided for the receive function and cold redundancy for the transmit function.*

*TT&C - 160* *The receiver outputs shall be cross-coupled with the inputs of the CDMS command decoders.*

*TT&C - 170* *The configuration shall be such that both receivers can receive and both decoders can decode simultaneously.*

*TT&C - 180* *The transmitters shall be able to receive the telemetry stream from both parts of the redundant CDMS.*

*TT&C - 190* *In order to ensure that the antenna switching is never time-critical, the antenna configuration shall ensure sufficient coverage and up- and downlink rate and ranging capabilities for all mission phases and in any spacecraft attitude.*

*TT&C - 200* *The TT&C sub-system shall provide the required telecommand and ranging capabilities at maximum distance from the Earth and in any spacecraft attitude.*

*TT&C - 210* *The receiver shall provide a status signal indicating the presence of an uplink signal.*

*TT&C - 220* *A Medium-Gain Antenna (MGA) shall provide the primary communication for the downlink during the scientific operations phase and during the Commissioning and Performance Verification Phases.*

### 5.17.4 Performance Requirements

*TT&C - 230* *The uplink/downlink signals shall be in the range 7190-7235 MHz for telecommands and 8450-8500 MHz for telemetry. Category A Mission with turnaround ratio of 749/880 for coherent operations.*

*TT&C - 240* *The link budget calculations and associated margins shall be according to ECSS-E-50-05A, ANNEX D3.*

*TT&C - 250* *The probability of frame loss (FLP) on the downlink shall be < 1.0E-5. The telecommand bit error rate (BER) at the input of the telecommand decoder shall be < 1.0E-5.*

*TT&C - 260* *The LGAs shall support an uplink high command rate of at least 4 kbps, using the primary ground station, and a low command data rate of at least 125 bps using the 15-m ESA ground station at Kourou, up to the maximum*

*Earth-spacecraft distance.*

*TT&C - 270* *The LGAs shall support the downlink of a low telemetry data rate of at least 2 kbps, using the primary ground station, and at least 62.5 bps using the 15-m ESA ground station at Kourou, up to the maximum Earth-spacecraft distance.*

*TT&C - 280* *The MGA shall allow a data telemetry downlink rate (total bit rate including all packetisation and error correction overheads before the convolutional encoder input) of 5 Mbps to the primary ground station and to the ground station at New Norcia and > 250 Kbps to the Kourou ground station, up to a distance from Earth of 1.68E6 km, at ≥ 5° elevation.*

*TT&C - 290* *In order to (almost) double the downlink data rate, the TT&C shall be designed to allow to enable/disable the convolutional encoder by telecommand.*

Note: in case convolutional encoder is disabled, violation to requirement TT&C – 250 might be considered acceptable.

*TT&C - 300* *The TT&C subsystem shall be designed to allow selection of possible all different telemetry data rates via telecommands.*

### 5.17.5 Ground Station Interface Characteristics

See AD 1-4 and AD 1-5.

### 5.17.6 Ground Compatibility Test

*TT&C - 300* *Tests shall be carried out to ensure the full compatibility between the spacecraft and the ground segment.*

## 5.18 Control and Data Management System

### 5.18.1 General

*CDMS - 010* *The Control and Data Management System (CDMS) shall collect all the data from the spacecraft. The data shall be conditioned, digitized, encoded, and formatted for transmission to ground.*

*CDMS - 020* *The Control and Data Management System (CDMS) shall process the uplink signal received by the Radio Frequency system, and validate the commands and distribute them to the users/applications for execution.*

*CDMS - 025* *The Control and Data Management System (CDMS) shall store data for later transmission and shall perform data compression.*

*CDMS - 030* *The CDMS shall be compliant with the following ESA Standards:*
- *Telemetry and Telecommand Packet Utilisation (AD 3-24)*
- *TM Synchronization and Channel Coding (AD 3-21)*
- *TM Space Data Link Protocol (AD 3-27)*
- *TC Space Data Link Protocol (AD 3-28)*
- *Space Packet Protocol (AD 3-29).*

*CDMS - 040* *The CDMS shall be compatible with the Gaia Operations Interface Requirements Document (AD 1-3) and the Gaia Space to Ground Interface Document (AD 1-4 and AD 1-5).*

### 5.18.2 Functional Requirements

#### 5.18.2.1 General

*CDMS - 050* *The CDMS shall perform the following general functions:*
- *Telemetry acquisition, encoding, and formatting;*
- *Telecommand acquisition, decoding validation, and distribution;*
- *Data storage;*
- *Data compression, if not already performed by the experiments;*
- *Time distribution and time tagging;*
- *Autonomy supervision and management;*
- *On Board Control Procedure Management (OBCP) functions.*

*CDMS - 060* *The CDMS shall provide an interface to the Electrical Ground Support Equipment (EGSE) as needed for the system level checkout. This shall at least include telemetry, telecommand, timing signals and fast access for memory loading.*

*CDMS – 070* *A safeguard function (e.g., Safeguard Memory) shall be implemented which can store at least the spacecraft configuration and all data (e.g. monitoring table, etc.) that are necessary to safely continue the mission after a reconfiguration.*

*CDMS - 080* *The CDMS shall perform the Telecommand decoding, verification, expansion, storage, execution and post verification.*

*CDMS - 090* *The CDMS shall be able to perform On Board Time management as per AD 1-3, AD 1-4, and AD 1-5.*

*CDMS - 100* *The CDMS shall be able to perform the Telemetry conditioning, acquisition, packetisation and storage as per AD 1-4, and AD 1-5.*

*CDMS - 110* *The CDMS shall be able to perform the Data storage required by all on-board actions.*

*CDMS - 120* *The CDMS shall be able to provide the Failure Detection Isolation and Recovery mechanisms both hardware and software as per SYS-160.*

*CDMS - 130* *The CDMS shall provide synchronisation signals and timing signals as required by the science instruments or spacecraft units.*

*CDMS - 140* *The CDMS shall maintain on-board Central Time Reference (CTR) and shall distribute this time to all on-board users. The CTR shall be settable by telecommand.*

#### 5.18.2.2 Telemetry

*CDMS - 150* *The CDMS shall acquire the scientific and periodic and non-periodic housekeeping data from the scientific instruments and spacecraft sub-*

*systems.*

CDMS - 160 *The CDMS shall condition, process, format, packetise, and code the data for on-board storage and for telemetry transmission to the ground station and for on-board supervisory functions as required.*

CDMS - 170 *Adequate telemetry modes shall be implemented to acquire, store and transmit the housekeeping and scientific data.*

CDMS - 180 *The CDMS shall allow parallel acquisition, storage and transmission of telemetry data.*

CDMS - 190 *The CDMS shall exchange TM/TC packets with all on-board units that can encode/decode TM/TC packets, for another units TM/TC packets shall be encode/decode by CDMS itself.*

#### 5.18.2.3 Telecommand

CDMS - 200 *The CDMS shall acquire, decode, validate, and distribute telecommands generated and stored on-board (e.g., from mission time line or On-board Control Procedures, etc) from the TT&C system to the various users.*

CDMS - 210 *The decoder shall protect the spacecraft against erroneous commands, either due to noise transmission conditions or from command signals sent to other spacecraft.*

#### 5.18.2.4 Data Compression

CDMS - 220 *The CDMS shall provide H/W and S/W resources to perform off-line lossless compression on the science data to be to optimize the usage of the bandwidth available during periods of ground contact.*

CDMS – 225 *The data compression technique shall be robust against channel transmission bit-flip by confining the error at sample level.*

CDMS - 230 *It shall be possible to enable/disable the use of data compression by telecommand function.*

#### 5.18.2.5 Data Storage and Memory

*CDMS - 240* *The CDMS shall provide the capability to store science and housekeeping data, as well as On-board Control Procedures, Time lines, and software programmes or software patches.*

*CDMS - 250* *The data storage shall be organised as a file system as a minimum for Spacecraft services.*

*CDMS - 260* *A direct link between the SSMM(s) and the TT&C subsystem shall be established to enable data to be direct downloaded to ground.*

*CDMS - 270* *In case of reconfiguration (e.g. due to errors) of the SSMM, the content shall not be lost.*

### 5.18.3 Performance Requirements

#### 5.18.3.1 General

*CDMS - 280* *The CDMS shall perform its own initialization and monitoring.*

*CDMS - 290* *The CDMS shall be able to distinguish between permanent faults and transient ones (single event upset, e.g.) and shall be able to reconfigure or adopt a safe mode autonomously as well as by ground command.*

*CDMS - 300* *The CDMS shall be able to route and store the peak TM and TC traffic without degradation, as required by instruments and on-board users.*

#### 5.18.3.2 Telemetry

*CDMS - 310* *Telemetry packets with no segmentation shall be used on the space to ground link.*

#### 5.18.3.3 Telecommand

*CDMS - 320* *During ground contacts with maximum TC uplink rate on-board operation shall continue without degradation.*

*CDMS - 330* *Any invalid telecommand received shall be indicated in the telemetry.*

*CDMS - 340* *Telecommand packets with no segmentation shall be used on the ground to spacecraft link.*

*CDMS - 350* *Capability shall be provided to send high priority commands (pulse and/or register load commands with only hardware support) to configure essential spacecraft items.*

A high priority command is intended as a single digital signal that is given as command to a unit. A high priority command is used to command essential function under ground control or to override on board actions. The decoding and execution of a high priority command shall therefore require the minimum of on-board resources, and in particular being executed only with simple and highly reliable hardware support. The items referred in the requirement as essential shall be identified after the spacecraft system design, and shall include on-board commands of high criticality for the spacecraft and/or that are meant to override on board functions. Examples of such commands are power distribution reconfiguration, RF network, and on-board computer reconfiguration.

#### 5.18.3.4 Data Storage

*CDMS - 360* *The size at End-of-Life of the of the SSMM(s) shall, as minimum, allow acquisition and storage of science data compatible with the scientific accuracy requirements reported in chapter 4 plus 3 days of house-keeping data, as well as On-board Control Procedures, Time lines, and software programmes or software patches.*

*CDMS - 370* *The input data rate of the on-board storage medium shall allow the acquisition of the worst-case volume of scientific and housekeeping data.*

CDMS - 390 *At all times, the CDMS shall record key on-board events in a Spacecraft Key Event Log (SKEL) stored in the safeguard memory. Key events include at least:*

- *Hardware and SW autonomous actions;*
- *Hardware and SW critical failures;*
- *Hardware and SW reconfigurations;*
- *SW start-up and shutdown;*
- *On-board Control Procedure start-up and completion;*
- *Recording of the major Mission Timeline (MTL) events.*

*This shall be performed in addition to the nominal storage in SSMM(s) and to the RT downlink of these events.*

CDMS - 400 *The Spacecraft Key Event Log (SKEL) shall be partitioned such that the various types of records can be easily separated and handled as separate logs.*

CDMS - 410 *All entries in the SKEL shall be time-stamped using the on-board spacecraft time reference.*

CDMS - 420 *The CDMS shall allow downlinking of any selected partition of the SKEL.*

CDMS - 430 *It shall be possible to clear the SKEL (or any SKEL partition) by ground command.*

CDMS - 440 *The size of the SKEL shall be such that recording of key events of up to 72 hours of nominal operation shall be possible with a margin allowing the triggering of one major failure.*

## 5.19 On-Board Software Requirements

### 5.19.1 General

All requirements in this Section apply to all Gaia on-board software.

*SFTW - 010* *All software production and test shall follow the ESA Software Standards AD3-16 and AD3-26 to the extent specified in the PA requirements (AD 1-2).*

*SFTW - 020* *The On board software shall be able in conjunction with hardware to execute all the tasks identified in CDMS and scientific requirements in chapter 4.*

*SFTW - 030* *The on-board CDMS S/Ws shall be developed using the same development environment.*

*SFTW - 040* *Withdrawn.*

Note: The standard development environment comprises standard processor type, standard development language, and standard software development tools.

*SFTW - 050* *The on-board software shall be developed using a high-level language, except where explicitly exempted in agreement with ESA.*

*SFTW - 070* *The on-board software shall be implemented with a layered structure separating application logic, basic services and operating system.*

*SFTW - 080* *The spacecraft shall support post-launch modifications of all on-board software.*

*SFTW - 090* *The on-board SW shall be structured such that modifications to any individual code module have minimum impact on other modules.*

*SFTW - 100* *All mission and safety critical SW, to the extent specified in the PA requirements (AD 1-2), shall be verified by an Independent Software Verification & Validation (ISVV) Team different from the software developer.*

*SFTW - 105* *A Software Validation Facility (or testing time at an existing SVF), together with all necessary software items (e.g. software documentation, source codes, etc.), shall be made available to the ISVV supplier in order to allow him to perform the independent testing activity.*

Note: the ISVV team is considered independent when from managerial, technical and financial point of view there is no link with the software supplier.

*SFTW - 110* *On-board software shall be stored on safeguard memory and run from RAM.*

*SFTW - 120* *At power ON, the boot-up sequence shall autonomously start and bring the spacecraft to a safe configuration (power, thermal, pointing and communication). The boot-up sequence shall be mission dependent (pre-launch, separation, LEOP and nominal operation mode).*

*SFTW - 125* *At boot-up the software shall perform a self-test and generate a dedicated report that shall be stored in the SKEL and downloaded to ground.*

*SFTW - 130* *The total boot sequence up to the first frame sent including all the self-tests shall have a maximum duration compatible with attitude control requirements.*

### 5.19.2 Functional Requirements

*SFTW - 140* *Software engineering parameters shall be available in telemetry housekeeping to enable ground to fully diagnose the status of the software.*

The data handling shall be supported with the capabilities for its fault detection, isolation and switching to its redundant items.

### 5.19.3 Performance Requirements

*SFTW - 150* *Software and the system shall protect against infinite loops, computational errors and possible lock ups resulting from an undetected hardware failure.*

*SFTW - 160* *Starting or stopping processes shall not affect the execution or performance of other running processes.*

*SFTW - 170* *Fixed areas of on-board memory shall be dedicated to:*
- *Code*
- *Fixed constants*
- *Variable parameters.*

*SFTW - 180* *On-board monitoring shall be the activity continuously ensuring that the on-board units under supervision are in an acceptable safe state. This shall be achieved by regularly checking the state of individual parameters in the housekeeping information.*

*SFTW - 190* *The software code used to boot up a processor shall be characterised in terms of time to initialise (cold/warm cases).*

*SFTW - 210* *The Contractor shall demonstrate that the software can be scheduled under all circumstances including mission worst case and nominal scenarios (to be developed by industry) and operational scenario provided by ESA.*

### 5.19.4 Software Maintainability

*SFTW - 220* *The software maintenance environment shall provide the means to generate and prepare software patches or full images and associated checksums for uplink to the spacecraft.*

*SFTW - 230* *Failed on-board software shall write a "Death Report" to safeguard memory before final FDIR actions, e.g. reboot. The "Death Report" shall contain all details necessary to diagnose the reason for failure. This shall include all task statuses, all processor registers and trap/exception type.*

*SFTW - 240* *The on-board software shall be fully patchable and dump-able without any restriction. Multiple selections shall be possible in one telecommand.*

*SFTW - 250* *All on board software shall be re-programmable on ground and in orbit without any restriction with the exception of PROMs.*

*SFTW - 260* *At FAR the software shall be delivered without patches.*

*SFTW – 270* *All CDMS software shall be stored redundantly in separate EEPROM memories.*

*SFTW - 280* *Ground shall be able to select which CDMS software memory, nominal or redundant, is used and subsequently copied into working RAM during initialisation.*

### 5.19.5 Verification Requirements

Software Verification Requirements are reported in the AIV Requirement Specification (AD 1-1).

## 5.20 Standard Radiation Environment Monitor

*SREM - 010* *Withdrawn*

*SREM - 020* *Withdrawn*

*SREM - 030* *Withdrawn*

# 6 External Interfaces

## 6.1 Launcher Interfaces

*EXIF-010* *The spacecraft shall be compatible with a launch on Soyuz-Fregat from Kourou.*

*EXIF-020* *The spacecraft design and operations shall comply with all performances, requirements, interfaces, and operations specified in the Soyuz-Fregat Launcher User Manual and relevant applicable documents reported in par. 2.1.2.*

*EXIF-030* *The spacecraft shall be designed to satisfy the launcher safety requirements as defined in ADs in paragraph 2.1.2.*

*EXIF-040* *The Prime Contractor shall provide the test adapter that needs to be representative of the launch vehicle adapter interfaces.*

*EXIF-050* *The flight adapter together with the separation mechanisms (e.g., including clamp-band) shall be provided either by the Launch Vehicle Authority or by the Gaia Prime Contractor (see AD 1-8).*

*EXIF-060* *The telemetry and telecommand standards applied to the satellite shall be compatible with those used on the launch site in order to perform the launch test and to monitor the satellite during the launch preparation.*

*EXIF-070* *The spacecraft design shall allow handling, transportation and integration with the launcher in both horizontal and vertical configurations.*

## 6.2 Ground Facilities

*EXIF-080* *The spacecraft design shall be compatible with the relevant ground test facilities.*

*EXIF-090* *In case specific non-standard facility modifications/extensions might have to be implemented, then a justification shall be submitted to ESA for approval.*

Note: facility modifications if needed, are considered part of the contractor cost and responsibility.

*EXIF-100* *Electrical/mechanical access for adequate system testing shall be ensured.*

*EXIF-110* *The spacecraft design shall be such to allow the transportation of the complete spacecraft as well as the separate payload and service modules by standard commercial means.*

*EXIF-120* *The design of the satellite and all operations and handling on the site shall be compliant with the safety regulations of the launch site (ADs in paragraph 2.1.2).*

## 6.3 Ground Segment Interface

*EXIF-130* *To ensure compatibility with existing ground systems, the spacecraft shall be compatible with the ground segment interfaces as defined in the AD 1-4 and AD 1-5.*

# 7 SPACECRAFT ENVIRONMENT

## 7.1 Mechanical Environment

*SENV-010* *The Gaia satellite shall be designed to withstand all the mechanical static and dynamic loads encountered during its entire life, including manufacturing, assembly, handling, transportation, testing, launch (as defined in the Soyuz-Fregat User Manual AD 2-1) and in-orbit operations.*

*SENV-020* *Manufacturing, handling and transportation loads (except for the MGSE interface points themselves) as well as test loads shall not be design drivers.*

*SENV-030* *The mechanical dynamic test environment shall meet the requirements of the Soyuz-Fregat User Manual (AD 2-1).*

## 7.2 Thermal Environment

*SENV-040* *The spacecraft shall be designed to withstand all thermal environments encountered during its entire life including:*

*a) Integration, Transportation, and Testing*
*b) Spacecraft preparation at launch site*
*c) Pre-launch phase with the spacecraft under the fairing*
*d) Ascent phase*
*e) In-obit operations from launcher separation until the end of the extended mission (including partial eclipses of the Sun by the Moon).*

*SENV-050* *The constraints applicable to b), c), and d) shall be derived from AD 2-1 and through a specific thermal analysis of the spacecraft integrated with the launcher.*

*SENV-060* *The constraints applicable to a) and e) above shall be derived by the Contractor in accordance to the RD 1-1.*

## 7.3 Contamination Environment and Requirements

### 7.3.1 General

General requirements about Contamination Environment are specified in the Product Assurance Requirement Specification (AD 1-2)

*SENV-070* *Specific design requirements and cleanliness assurance provisions shall be derived from the identification of sensitive items of the payload and unit specification (all optical equipment and surfaces, e.g. mirror, sensors, CCDs, etc.) and they shall be inserted into the appropriate requirements specifications.*

*SENV-073* *The S/C scientific performances to be achieved at the end of the mission as reported in Chapter 4 (and notably SCI-250, SCI-460, and SCI-620) shall include all effects of molecular and particulate contamination such as, for example, transmission losses, straylight, etc.*

Note: Cleanliness shall be considered with the highest attention during all programme phases (mainly AIT phases).

### 7.3.2 Chemical Cleanliness Requirements

*SENV-075* *The exposure of all optical surfaces to the ambient environment shall be minimised.*

*SENV-080* *Withdrawn.*

## 7.4 Particulate Cleanliness Requirements

*SENV-090* *Withdrawn.*

*SENV-100* *Withdrawn*

## 7.5 Specific Design Requirements

*SENV-110* *The spacecraft shall provide means to minimize contamination of the telescope and instruments in-orbit, during the initial (outgassing) phase and during the thruster firings. If needed, deployable protective covers shall be implemented as necessary on the telescope apertures for minimising optics contamination during AIV operations, launch preparation, and in-orbit initial manoeuvres.*

*SENV-120* *The contamination of the telescope mirrors shall be controlled during all the ground activities including (pre-launch).*

*SENV-130* *The spacecraft shall be designed to allow the decontamination of the optical surfaces in the early orbital phase.*

*SENV-140* *Protections against chemical contamination during ground operations shall be implemented.*

Sensitive areas like Astro focal plane detector that may be used on ground at low temperature (170 K) behave like contamination traps, since the molecular mobility is much lower at these temperatures than ambient temperature. In these cases special provisions such heaters/covers or special design of spacecraft elements near the detector shall be provided.

*SENV-150* *Means to decontaminate critical surfaces nominally operating at low temperatures (e.g. FPA's, CCD's, etc.) shall be foreseen.*

## 7.6 Radiation Environment

### 7.6.1 General

*SENV-160* *The Contractor shall be responsible for performing radiation analyses* *using the nominal plus extended mission scenario and taking into account data from Gaia Environmental Specification (AD 1-6).*

*SENV-180* *A list of all external surfaces (e.g., thermal blankets, thermal paints, windows, etc.) shall be maintained and it shall be demonstrated that any degradation of properties due to radiation dose over the 6.5 years (extended mission) is acceptable from a performance point of view.*

*SENV-190* *The following applicable document AD 3-5 shall be used as a guideline.*

## 7.7 Radiation Dose

*SENV-200* *The satellite and its components shall be qualified (either based on existing or new test data) to withstand twice the expected levels of radiation. Radiation testing shall be included in the lot acceptance testing, if the margin is small and if the variation of radiation resistance between lots is large or insufficiently known.*

### 7.7.1 Radiation Induced Background

Radiation impinging onto a detector or its associated electronics can produce an increase in noise, which in turn can produce a significant decrease of performance. Such changes can last until well after the radiation dose has stopped (remittances)

*SENV-210* *The spacecraft design and component selection shall be such as to minimize temporary reduction of performance due to radiation background effects, including any necessary means to ensure the most rapid restoration of nominal performance.*

### 7.7.2 Single Event Effects

*SENV-220* *The spacecraft shall be designed to withstand cosmic ray and heavy ion impacts, which can provoke Single Event Effects (SEE) (i.e., Single Event Upset (SEU), Single Event Latch-up (SEL), Single Event Transient (SET), Single Effect Burn Out (SEBO) or Single Event Gate Rupture (SEGR)) in all spacecraft devices (e.g. bi-stable elements such as memories, comparators, or gate latches, etc).*

*SENV-230* *Any device used (registers, ASICs, FPGAs, etc.) for which the correct functioning is critical to mission objectives, mission life or spacecraft safety, shall have protection against SEEs.*

*SENV-240* *Single event effects (SEE) occurrence during in-orbit operational life shall be dimensioned through the Cosmic Rays integral flux versus Linear Energy Transfer values and Solar proton energy spectra, as given in AD 1-6. SEE rates for a given equipment shall be derived from the relevant tested LET threshold associated with the equipment as well as proton testing.*

*SENV-250* *Electronic components applied in the spacecraft shall either be resistant to the expected radiation level and to SEE (SEU, SEL, SET, SEBO and SEGR), or suitable provisions shall be made in the design for protection against unacceptable degradation or failure effects.*

*SENV-260* *The rate of un-correctable errors in spacecraft memories (PROMs, EEPROMs, and RAMs) shall be better than $1*10^{-11}$ (error/bit/day) at the end of mission.*

*SENV - 265* *The rate of un-correctable errors in spacecraft SSMM(s) shall be better than $1*10^{-14}$ (error/bit/day) at the end of mission.*

*SENV-270* *The spacecraft design shall withstand the effects of SEU without interrupting the on-going operations and it shall include proper error detection, transient filters and correction schemes.*

*SENV-280* *Analog/digital status lines shall be filtered to prevent disturbing propagation of the SET.*

*SENV-290* *Any SEE effect detected by the on board computer shall be reported to ground with its location and time occurrence.*

*SENV - 300* *The Random Access Memory (RAM) and EEPROM shall be protected by an Error Detection And Correction (EDAC) function. This shall allow correcting single bit errors and detecting double bit errors. An analysis shall demonstrate that the occurrence of a bit-flip inducing a mission outage shall be less than one per 6.5 years.*

*SENV - 310* *Each processing unit shall count and store location of each single bit corrections by the EDAC and the uncorrectable double errors. These values shall be part of the telemetry housekeeping information.*

*SENV-320* *The processor design shall ensure that the processor internal registers are refreshed at a rate sufficient to avoid cumulation of deposited charges leading to errors.*

## 7.8 Meteoroids

*SENV-330* *The spacecraft shall be designed to withstand and perform nominally after the predicted meteoroids environment throughout the mission with a probability larger than 0.998.*

*SENV-340* *The sporadic meteoroid impact occurrence shall be derived from the AD 1-6.*

## 7.9 Electromagnetic Cleanliness Requirements

### 7.9.1 General Requirements

*SENV-350* *The spacecraft shall not be susceptible to self-generated electromagnetic interference and shall ensure satisfactory payload performance during the mission, as well as non-hazardous operation of the spacecraft in ground test and launch environment. The term EMC shall cover all frequencies (including DC) that fall in either the payload, spacecraft, or Launcher bandwidth.*

*SENV-360* *The spacecraft electromagnetic emissions and susceptibility shall comply with the launcher requirements (applicable documents in paragraph 2.1.2).*

*SENV-370* *The spacecraft shall not be susceptible to electrostatic charge/discharges expected in the operational space environment.*

*SENV-375* *The GSE used for ground testing shall be designed to be compatible with spacecraft EMC environment/requirements.*

### 7.9.2 Design Requirements

*SENV-380* *A distributed single point grounding scheme shall be adopted. This implies that galvanic isolation shall be provided between primary and secondary power.*

*SENV-390* *Good and reliable bonds shall be provided between the various parts of any electronic box, connectors, harness shields, structure including thermal blankets (foil and MLI), and the spacecraft structure.*

The contractor shall define the limits acceptable for bonding.

*SENV-400* *A frequency control plan shall be established and maintained as part of the EMC programme.*

*SENV-410* *A 6 dB margin shall exist between susceptibility and specified* *environment, except for any pyrotechnic devices where this margin shall be 20 dB.*

SENV-420 *Signal, power, and data lines and their respective returns shall be separated.*

### 7.9.3 Electrostatic Cleanliness

SENV-430 *Any space-exposed surface or surface coating shall be electrically conductive with a surface resistivity less than 100 kΩ/square, and bonded to structure with a resistance less than 100 kΩ.*

SENV-440 *All metallic items that do not perform any electrical function shall be bonded with a resistance of less than 100 Ω to the interface grounding point.*

SENV-450 *There shall be no harness dielectric exposed to plasma environment.*

SENV-460 *Spacecraft design and materials shall be selected such to ensure that no parts of the spacecraft are charged to high potentials.*

# 8 ASSEMBLY, INTEGRATION AND VERIFICATION REQUIREMENTS

The Assembly, Integration and Verification (AIV) and GSE Requirements are specified in the System AIV Requirement Document (AD 1-1).

# 9 PRODUCT ASSURANCE

The Product Assurance requirements are specified in the Product Assurance and Safety Requirements (AD 1-2) document.

# APPENDIX 1: LIST OF MISSION DATA TO BE USED FOR CALCULATION PURPOSES

The state vectors/orbits presented in this appendix are sample orbits only. It should be noted that a reference orbit in the classical sense does not exist in the operational phase as each station-keeping manoeuvre is designed to get back to the nearest local non-escape orbit, rather than a given reference state. Also, for the LEOP there will be a set of reference state vectors at perigee plus the deterministic manoeuvre that is used to bias the launcher dispersion correction towards the solar hemisphere. The apogee altitude and argument of perigee of those states are binned in the launcher flight programs.

To be noted, this orbit shall not be used to drive spacecraft design, where for this a specific orbit envelope requirement have been defined (chapter 3) but only for the calculation/verification of the following aspects

- Science Performance Modelling
- cold gas budget

Any additional uses shall be agreed on a case-by-case basis with ESA.

The reference transfer orbit for spacecraft analyses is characterised by the following parameters:

o Orbit at s/c separation: 2012/07/04 – 22:45:07 (4568.94800)

Orbital parameters

A 500217.546 km

E 0,986525

I 15.0 deg

BOM 32.63 deg

SOM 90.0 deg

F 36.12 deg

RP 6740.166 km

RA 993694.927 km

o The reference operational Lissajous orbit (at L2) for spacecraft analyses is characterised by the following parameters:

osculating Lissajous elements at TU (time of upper crossing)

A1 -3821.63243

A2 -5131.10656

AX 107469.96059

AY 342531.40599

AZ 90035.31648

ΦXY -4.49773

ΦIZ 220.00096

# APPENDIX 2: ACRONYMS & ABBREVIATIONS

| | |
|---|---|
| **A** | |
| AC | Across Scan |
| AD | Applicable Document |
| AIT | Assembly, Integration, and Test |
| AIV | Assembly, Integration, and Verification |
| AF | Astro Field |
| AHFD | Attitude High Frequency Disturbances |
| AL | Along Scan |
| AME | Absolute Measurement Error |
| AOCS | Attitude and Orbit Control Sub-system |
| APE | Absolute Pointing Error |
| ARE | Absolute Rate Error |
| ASIC | Application-Specific Integrated Circuit |
| ASM | Astrometric Sky Mapper |
| Astro | Astrometric Instrument |
| | |
| **B** | |
| BA | Basic Angle |
| BBP | Broad-Band Photometer |
| BER | Bit Error Rate |
| BoL | Beginning of Life |
| | |
| **C** | |
| CCD | Charge-Coupled Device |
| CCSDS | Consultative Committee for Space Data Systems |
| CDMS | Command and Data Management Sub-system |
| CDMU | Command and Data Management Unit |
| CDR | Critical Design Review |
| CE | Conducted Emission |
| CoG | Centre of Gravity |
| CoM | Centre of Mass |
| CPU | Central Processing Unit |
| CR | Cosmic Ray |
| CreMA | Consolidated Report on Mission Analysis |
| CS | Conducted Susceptibility |
| CTE | Charge Transfer Efficiency |

| | |
|---|---|
| CTI | Charge Transfer Inefficiency |
| CTR | Central Reference Time |
| | |
| **D** | |
| DC | Direct Current |
| DHS | Data Handling Sub-system |
| DMS | Double and Multiple Star |
| DPC | Data Processing Center |
| DSA | Deployable Sunshield Assembly |
| DoD | Depth of Discharge |
| | |
| **E** | |
| ECSS | European Cooperation for Space Standardisation |
| EDAC | Error Detection And Correction |
| EEPROM | Electrical Erasable Programmable Read Only Memory |
| EGSE | Electric Ground Support Equipment |
| EIRP | Equivalent Isotropic Radiated Power |
| EMC | Electro-Magnetic Compatibility |
| EMI | Electro-Magnetic Interference |
| EOL | End of Life |
| EOP | Early Orbit Phase |
| EPS | Electrical Power Sub-system |
| ESA | European Space Agency |
| ESARAD | Software tool for analysis of spacecraft radiation environment |
| ESATAN | Software tool for spacecraft thermal analysis |
| ESD | Electro-Static Discharge |
| ESOC | European Space Operations Centre |
| ESTEC | European Space Research and Technology Centre |
| | |
| **F** | |
| FAR | Flight Acceptance Review |
| FCL | Foldback Current Limiter |
| FCV | Fill Control Valve |
| FDIR | Failure Detection, Isolation, and Recovery |
| FDV | Fill and Drain Valve |
| FEM | Finite-Element Model |
| FID | Function Identifier |
| FLP | Frame Loss Probability |
| FMECA | Failure Modes, Effects, and Criticality Analysis |
| FoV | Field of View |

| | |
|---|---|
| FVV | Fill and Vent Valve |
| FPA | Focal Plane Array/Focal Plane Assembly |
| | |
| **G** | |
| GC | Great Circle |
| GEO | Geostationary Earth Orbit |
| GMSK | Gaussian Minimum Shift Keyed |
| GSE | Ground Support Equipment |
| GST | Gaia Science Team |
| | |
| **H** | |
| HFAD | High-Frequency Attitude Disturbances |
| HGA | High-Gain Antenna |
| H/K | Housekeeping |
| HK | Housekeeping |
| H/W | Hardware |
| | |
| **I** | |
| ICD | Interface Control Document |
| ID | Identifier |
| IPLM | Integrated Payload Module |
| HW | Hardware (not used) |
| ISVV | Independent Software Verification & Validation |
| | |
| **J** | |
| | |
| **K** | |
| kbps | kilo-bit per second |
| | |
| **L** | |
| L2 | Second Lagrange point of Earth/Moon-Sun system |
| LCL | Latching Current Limiter |
| LCLA | Launcher Coupled Loads Analysis |
| LEOP | Launch and Early Orbit Phase |
| LET | Linear Energy Transfer |
| LGA | Low-Gain Antenna |
| LoS | Line of Sight |
| LV | Latch Valve |
| LV | Launch Vehicle |
| LVA | Launch Vehicle Adapter |

| | |
|---|---|
| LVA | Launch Vehicle Authority |
| | |
| **M** | |
| mag | Magnitude |
| mas | milli-arcsecond |
| MBP | Medium-Band Photometer |
| Mbps | Mega-bit per second |
| MeV | Mega electron-Volt |
| MF | Master Function |
| MGA | Medium-Gain Antenna |
| MGSE | Mechanical Ground Support Equipment |
| MLI | Multi-Layer Insulation |
| mmag | milli-magnitude |
| MOC | Mission Operation Center |
| MPSK | M-Phase Shift Keyed |
| MRD | Mission Requirements Document |
| MRE | Mean Rate Error |
| MSC/NAS TRAN | Software for finite-element analysis |
| MTF | Modulation Transfer Function |
| MTL | Mission Time Line |
| | |
| **N** | |
| NEO | Near-Earth Object |
| NIEL | Non-Ionising Energy Loss |
| | |
| **O** | |
| OBCP | On-Board Control Procedure |
| OBDH | On-Board Data Handling |
| OTS | Off-The-Shelf |
| | |
| **P** | |
| PDR | Preliminary Design Review |
| PLM | Payload Module |
| ppm | Parts Per Million |
| PROM | Programmable Read Only Memory |
| PSD | Power Spectral Density |
| PSD | Packet Structure Document |
| PSF | Point Spread Function |
| PT | Pressure Transducer |
| P/L | Payload |

| | |
|---|---|
| PLM | Payload Module |
| | |
| **Q** | |
| | |
| **R** | |
| RAM | Random Access Memory |
| RCS | Reaction Control Sub-system |
| RF | Radio Frequency |
| RME | Rate Measurement Error |
| RPE | Relative Pointing Error |
| RTC | Real-Time Clock |
| | |
| **S** | |
| SA | Solar Array |
| SAA | Solar Aspect Angle |
| S/C | Spacecraft |
| SCET | Spacecraft Elapsed Time |
| SEBO | Single Event Burn-Out |
| SEE | Single Event Effect |
| SEGR | Single Event Gate Rupture |
| SEL | Single Event Latch-up |
| SET | Single Event Transient |
| SEU | Single Event Upset |
| SGM | Safeguard Memory |
| SKEL | Spacecraft Key Event Log |
| SoW | Statement of Work |
| SPF | Single Point Failure |
| SREM | Standard Radiation Environment Monitoring |
| S/S | Sub-system (not used) |
| SSMM | Solid State Mass Memory |
| SVF | Software Validation Facility |
| SVM | Service Module |
| S/W | Software |
| | |
| **T** | |
| TC | Telecommand (Packet) |
| TCS | Thermal Control Sub-system |
| TDI | Time Delay Integration |
| TM | Telemetry (Packet) |
| TML | Total Mass Loss |

| | |
|---|---|
| TT&C | Telemetry, Telecommand, and Command |
| | |
| **U** | |
| UTC | Universal Time Coordinated |
| | |
| **V** | |
| V | Johnson visual magnitude |
| | |
| **W** | |
| WCA | Worst-Case Analysis |
| w.r.t. | With reference/respect to |
| | |
| **X** | |
| | |
| **Y** | |
| | |
| **Z** | |
| | |
| **Other** | |
| μas | micro-arcsecond |
| μs | micro-second |