ROSETTA

System Requirements Specification

RO-EST-RS-2001

Prepared by: Rosetta Project Team

Approved by: J. van Casteren

Approved by: B. Gardini

TABLE OF CONTENTS

Page No.

1. INTRODUCTION

This document together with the

- Management Requirements Specification - RO-EST-RS-1001
- Product Assurance Requirements Specification - RO-EST-RS-1002
- AIV Requirements Specification - RO-EST-RS-2002

Phase B and C/D Statement of Work	RO-EST-SW-0001
Operational Interface Requirements Document (ROIRD)	RO-ESC-RS-5001
Space/Ground Interface Control Document (SGICD)	RO-ESC-IF-5002

form the Rosetta System and Programmatic Requirements and constitute the baseline requirements from ESA for the Rosetta Spacecraft Programme for Phase B and Phase C/D.

In case of contradiction or conflict between the content of these documents and any annex or secondary document called herein, these documents have precedence.

All requirements in this document which require verification are marked with an unique reference.

The International Rosetta Mission is a cometary mission which will be launched in the year 2003 by Ariane 5. After a long cruise phase, the spacecraft will rendezvous with the comet Wirtanen and orbit it, while taking scientific measurements. A Surface Science Package, the Rosetta Lander will be landed on the comet surface to take in-situ measurements. During the cruise phase, the spacecraft will use gravity assist manoeuvres once by Mars and twice by the Earth to gain orbital energy. The spacecraft will also take measurements as it passes close to two asteroids.

The on-board orbiter scientific instruments and the Lander including its instrumentation are conceived and built by Scientific Institutes and funded by National Agencies.

2. APPLICABLE DOCUMENTS

The following documents of the latest issue or as defined herein form part of this specification.

In the event of a conflict between this document and other applicable documents the conflict shall be brought to the attention of the ESA Rosetta Project Manager for resolution.

In the event of a conflict between this specification and reference documents, this specification shall have precedence.

2.1 Applicable Documents

AD-1	ARIANE 5 User's Manual, Issue 2, Rev. 2, February 1998
AD-2	CSG Safety Requirements CSG-RS-10A (VOL 1), CSG-RS-22A (VOL 2) Edition 5, December 1997
AD-3	
AD-4	
AD-5	Telemetry Channel Coding Standard (ESA PSS 04-103) Issue 1, September 1989
AD-6	Ranging Standard (ESA PSS-04-104) Issue 2 March 1991
AD-7	Radio Frequency and Modulation Standard (ESA PSS-04-105) Issue 1, December 1989
AD-8	Packet Telemetry Standard (ESA PSS 04-106) Issue 1, January 1988
AD-9	Packet Telecommand Standard (ESA PSS 04-107) Issue 2, April 1992
AD-10	MIL-HDBK-5: Current issue
AD-11	ESA fracture control requirements (ESA-PSS-01-401) Issue 2, November 1994
AD-12	Space and Planetary Environment, Criteria Guidelines for use in Space Development, (NASA technical memorandum 82478) 1982 Revision (Volume 1)
AD-13	Natural Orbital Environment Guidelines for use in Aerospace Vehicle Development (NASA Technical memorandum 4527)

AD-14	EID-A, RO-EST-RS-3001
AD-15	EID-B - ALICE, RO-EST-RS-3005
AD-16	EID-B - BERENICE, RO-EST-RS-3006
AD-17	EID-B - CONSERT, RO-EST-RS-3007
AD-18	EID-B - COSIMA, RO-EST-RS-3008
AD-19	EID-B - GIADA, RO-EST-RS-3009
AD-20	EID-B - MIDAS, RO-EST-RS-3010
AD-21	EID-B - MIRO, RO-EST-RS-3011
AD-22	EID-B - RPC, RO-EST-RS-3012
AD-23	EID-B - ROSINA, RO-EST-RS-3013
AD-24	EID-B - RSI, RO-EST-RS-3014
AD-25	EID-B - VIRTIS, RO-EST-RS-3015
AD-26	EID-B - OSIRIS, RO-EST-RS-3016
AD-27	LID-B - ROSETTA LANDER, RO-EST-RS-3020
AD-28	Deleted
AD-29	Deleted
AD-30	ISO/DIS/6336 Calculation of load capacity of spur and helical gears
AD-31	Generic Specification for Silicon Solar Cells (ESA-PSS-01-604) Issue date: 1 January 1988
AD-32	Deleted
AD-33	Standard Radiation Environment Monitor (SREM) Rosetta interfaces (SREM/RS/002), Issue 8, 24 Nov 1997
AD-34	ESA Software Engineering Standard ESA PSS-05-0, Issue 2 February 1991
AD-35	Data for Selection Of Space Materials (PSS-01-701) current issue
AD-36	A Thermal Vacuum Test for the Screening of Space Materials

(PSS-01-702) Issue 1, March 1983

AD-37 A Thermal Cycling Test for the Screening of Space Materials and Processes (PSS-01-704) Issue 1, August 1982

AD-38 Material Selection for Controlling Stress Corrosion Cracking (PSS-01-736) Issue 1, May 1981

AD-39 Determination of Susceptibility of Metals to Stress Corrosion Cracking (PSS-01-737) Issue 1, September 1981

AD-40 Power Standard (ESA-PSS-02-10) Issue date 1 January 1992

2.2 Reference Documents

RD-1 ROSETTA Comet Rendezvous Mission (SCI(93)7)

RD-2 ESA Pointing Error Handbook (ESA-NCR-502 Vol 1), issue 1, February 1993

RD-3 Packet Utilisation Standard, ESA PSS-07-101, Issue 1 May 1994

RD-4 Telecommand Decoder Specification (ESA PSS 04-151) Issue 1 September 1993

RD-5 Consolidated Report on Mission Analysis RO-ESC-RP-5500, issue 3, February 1999

RD-6 Rosetta Common Check-out and Control System, Implementation Concept RO-ESC-TN-5001

RD-7 S and S/X bands coherent transponder specification (ESA PSS-48) Issue 1, March 1979

RD-8 Mechanical Engineering of Space Programmes; Mechanisms Engineering Standard (ECSS-E-30-00 Part 2-3) Issue 2 Revision 1 Draft Dec. 96

RD-9 An Engineering Model of the Dust and Gas Environment of the Inner Coma of Comet P/Wirtanen, RO-ESC-TA-5501, issue 1, Part 1 October 1998, Part 2 May 1999

RD-10 Comet Wirtanen Surface Engineering Model RO-ESC-RP-5006, issue 1, May 1999

RD-11 Solid particle environment for Rosetta
ESTEC memo WMA/97-170/GD/ROS, 97.10.31
(RO-EST-FAX-PR/1.1.7.1/GC/253)

3. ROSETTA MISSION

Rosetta has been approved by the ESA Science Programme Committee in November 1993 as Planetary Cornerstone mission of ESA's long-term programme Horizon 2000. It is defined as a mission to rendez-vous with the nucleus of comet Wirtanen. On its way to the comet the spacecraft will have two close fly-bys of asteroids. As in-situ investigation of the cometary nucleus has been regarded of upmost interest by the scientific community, the mission will carry a Surface Science Package, Rosetta Lander to the nucleus and deploy it onto its surface. This Lander will be provided by a consortium of European Institutes, with the main contributions coming from Germany, France and Italy.

3.1 Scientific Mission

Our knowledge of small solar-system bodies, the comets and asteroids, has dramatically improved over the last 20 years. The major milestones were undoubtedly the first fly-bys of Comet Halley by the Giotto, Vega, Sagigake and Susei probes in 1986, the Giotto Extended Mission to comet Grigg-Skjellerup in July 1992, and the first near encounters with main-belt asteroids, Gaspra and Ida, by the Galileo spacecraft on its way to Jupiter. During the same period, telescopic observations performed on the ground or in Earth orbit have greatly expanded and diversified. They constitute the basis for understanding small bodies as a population, since we can now compare observations of a large variety of objects, and can undertake investigations of the variability of cometary activity.

Systematic observations in the visible spectrum are being performed for short-period comets as well as main-belt and near-Earth asteroids, and it is now possible to observe cometary nuclei even at large heliocentric distances from the ground and with the Hubble Space Telescope. Furthermore, small bodies have now been studied systematically at all wavelengths from the ultraviolet, via the infrared to microwave and radio wavelengths.

From this wealth of new information, it is becoming apparent that small Solar System bodies, asteroids and comets, constitute an almost continuous suite of progressively less evolved objects, reflecting the radial gradient in the swarm of planetesimals during the formation of the Solar System. Indeed, the outermost asteroids present spectral similarities with the bare cometary nuclei observed far from the Sun. A better understanding of the relationship between asteroids, comets and planetesimals throughout the solar nebula is an essential step in unravelling the first stages of the formation of our Solar System.

Cometary material has been submitted to the lowest level of processing since its condensation from the proto solar nebula. It is considered likely that pre-solar grains may have been preserved in comets. As such, cometary material should constitute a unique repository of information on the sources that contributed to the proto solar nebula, as well as on the condensation processes that resulted in the formation first of planetesimals, then of larger planetary bodies. While tantalising results have been obtained *in-situ* from cometary grains, and from interplanetary dust particles collected on Earth, the latter cannot be considered as fully representative, in particular in terms of their organic and volatile complement.

Direct evidence on cometary volatiles is particularly difficult to obtain, as species observable from Earth, and even during the Halley fly-bys, result from physico-chemical processes such as sublimation, interaction with solar radiation and the solar wind. The currently available information on cometary material gained from *in-situ* studies and ground-based observations demonstrates the low level of evolution of cometary material. The latter's tremendous potential for providing information on the constituents and early evolution of the solar nebula has yet to be exploited.

3.1.1 Scientific Objectives

Studying cometary material represents a major challenge, owing to the very characteristics that makes it a unique repository of information about the formation of the Solar System, namely its high content of volatiles and organic material. Two solutions to the problem of obtaining unaltered material can be considered: returning to Earth a sample of a cometary nucleus (the original Rosetta concept) or staying close to the comet and performing comprehensive *in-situ* analyses of material from the surface and the coma.

The first approach had the undisputed advantage of bringing the full range of analytical techniques that are, or will be, available in the laboratory to bear on investigations of the Probe material. Limiting thermal and mechanical stresses to acceptable levels during cruise and recovery, and even defining these acceptable levels, represented a significant scientific and technological challenge. The new approach, which results in a less complex and cheaper mission, guarantees by design minimal perturbations of the cometary material, as analyses are performed *in situ* , at low temperatures and in a micro gravity environment. It also provides the opportunity of observing at close range the onset and development of cometary activity, which results in the spectacular displays that have captured the imagination of mankind over the centuries.

A fundamental question that has to be addressed by the mission is to what extent the material accessible to analyses can be considered as representative of the bulk material constituting the comet, and of the early nebular condensates that constituted the cometesimal 4.57×10^{9} years ago?

This representativity issue has to be addressed by first determining the global characteristics of the nucleus (mass, density, state of rotation), which can provide clues concerning vertical gradients, and hence the relationship between the outer layers and underlying material.

The dust and gas activity observed around comets, as well as its rapid response to insolation, guarantees the presence of volatiles at or very close to the surface in active areas. Analysing material from these areas will therefore provide information on both the volatile and refractory constituents of a cometary nucleus. The selection of a proper site for surface-science investigations should be relatively straightforward given the extensive remote-sensing observation phase and the advanced instrumentation, that covers a broad range of wavelengths, being provided on the Rosetta orbiter.

The surface science site can be monitored during surface activities, as well as during a large fraction of the activity cycle, which should bring to light clues concerning the compositional heterogeneity of active regions.

The dust-emission processes are induced by very low density gas outflows and should preserve the fragile texture of cometary grains. These grains can be collected at low velocities (a few tens of metres per second) by the spacecraft after short travel times (of the order of minutes), which will minimise alterations induced by the interaction with solar radiation. Similarly, gas analysed in jets or very close to the surface should yield information on the volatile content of cometary material in each source region.

The Rosetta mission will study the nucleus of comet Wirtanen and its environment in great detail during nearly two years starting the near-nucleus phase at around 3.25 AU, from the onset of activity and following it through perihelion, close to 1 AU. On its long journey to the comet, the spacecraft will pass close to two asteroids, Otawara and Siwa.

The prime scientific objective of the mission as defined by the Rosetta Science Team is to

- study the origin of comets, the relationship between cometary and interstellar material and the implications for the origin of the Solar System.

The measurement objectives to support these goals are:

- the global characterisation of the nucleus, determination of dynamic properties, surface morphology and composition

- the chemical, mineralogical and isotopic compositions of volatiles and refractories in a cometary nucleus

- the physical properties and interrelation of volatiles and refractories in a cometary nucleus

- the study of the development of cometary activity and the processes in the surface layer of the nucleus and the inner coma (dust-gas interaction)

- the global characterisation of asteroids, determination of dynamic properties, surface morphology and composition.

3.1.2 Scientific Instrumentation

Orbiter Payload

The individual instruments of the Rosetta orbiter are listed in Table 1.

Short Name	Objective	Principal Investigator
Remote Sensing		
OSIRIS	Multi-colour Imaging Narrow angle camera 2.35 °x2.35° Wide angle camera 12 °x12 ° A = 250 nm - 1000 nm	H. U. Keller MPI fuer Aeronomie, Katlenburg-Lindau, Germany
ALICE	UV-Spectroscopy (700-2050 Å)	A. Stern, Southwest Research Institute, Boulder, CO, USA
VIRTIS	VIS and IR Mapping Spectroscopy (.25 - 5 µm)	A. Coradini, IAS-CNR, Rome, Italy
MIRO	Microwave Spectroscopy (1.3 mm and .5 mm)	S. Gulkis, NASA-JPL, Pasadena, CA,USA
Composition Analysis		
ROSINA	Neutral Gas and Ion Mass Spectro-scopy; Double-focussing, 12-200 AMU, M/ LIM- 3000 Time-of-flight, 12-350 AMU, M/ LIM- 500 incl. Neutrals Dynamics Monitor	H. Balsiger, Univ. Bern, Switzerland
BERENICE	Isotopic Ratios of Light Elements by Gas Chromatography (D/H; $^{13}C/^{12}C$; $^{18}O/^{16}O$; $^{15}N/^{14}N$)	C. Pillinger, Open University, Milton Keynes, UK
COSIMA	Dust Mass Spectrometer (SIMS, m/ LIm - 2000)	J. Kissel, MPI for Extraterrestrial Physics, Garching, Germany

MIDAS	Grain Morphology (Atomic Force Microscope, nm Resolution)	W. Riedler, Inst. f ur Weltraumforschung Austria
Nucleus Large Scale Structure		
CONSERT	Radio Sounding, Nucleus Tomography	W. Kofman, CEPHAG, Grenoble, France
Dust Flux, Dust Mass Distribution		
GIADA	Dust Velocity and Impact Momentum Measurement, Contamination Monitor	E.Bussoletti Istituto Univ. Navale Naples, Italy
Comet Plasma Environment, Solar Wind Interaction		
RPC	Langmuir Probe, Ion and Electron Sensor, Flux Gate Magnetometer, Ion Composition Analyser, Mutual Impedance Probe	R. Boström, Swedish Inst. of Space Physics, Uppsala, Sweden J. Burch, South West Research Institute, San Antonio, TX, USA K.-H. Glassmeier, TU Braunschweig, Germany R. Lundin, Swedish Inst. For Space Physics, Kiruna, Sweden J.G. Trotignon, LPCE/CNRS, Orleans, France
RSI	Radio Science Experiment	M.Pätzold University Cologne, Germany

Table 1 - Rosetta Orbiter Payload

Lander Payload

The individual instruments of the Lander are listed in Table 2.

APXS	a-p-X-ray Spectrometer	R. Rieder MPI Chemistry, Mainz, Germany
SD2	Sample and Distribution Device	A. Ercoli Finzi Politecnico-Milano Italy
PTOLEMY	Evolved Gas Analyser	C. Pillinger Open University UK
ÇIVA ROLIS	Imaging System (Panoramic cameras, Descent imager, Microscope)	J.P.Bibring IAS, Orsay, France S.Mottola, DLR, Berlin, Germany
SESAME	Surface Electrical and Acoustic Monitoring Experiment, Dust Impact Monitor	D. Möhlmann DLR Cologne, Germany H. Laakso FMI, Finland I. Apathy KFKI, Hungary
MUPUS	Multi-Purpose Sensor for Surface and Sub-Surface Science	T. Spohn Univ. Münster, Germany
ROMAP	RoLand Magnetometer and Plasma Monitor	U. Auster DLR Berlin, Germany I. Apathy KFKI, Hungary
CONSERT	Comet Nucleus Sounding	W. Kofman CEPHAG, Grenoble, France
COSAC	Cometary composition measurement (Evolved Gas Analyser)	H. Rosenbauer MPAe Lindau, Germany

Table 2 Rosetta Lander Payload

3.1.3 Scientific Requirements

This section summarizes some general requirements for the successful completion of the scientific mission in line with the scientific objectives defined in section 3.1.1 Specific requirements from individual instruments would be beyond the scope of this section and are addressed in the specific sections of the individual EID-B's.

SCIR-001 *During the prime mission from 3.25 AU through perihelion passage high resolution mapping of the nucleus (> 80% of the surface) should be possible; i.e., each point of the surface (between local noon, over the terminator, to local midnight) shall be observable at highest possible resolution at any time by the remote sensing suite of instruments - ALICE, OSIRIS, VIRTIS and MIRO - and the Neutral Gas and Ion Mass Spectrometer, ROSINA. For the Narrow Angle Camera of the Imaging System, e.g. this means that most of the nucleus surface will be mapped with a resolution better than 10 cm. The instrument line of sight shall be independently selectable to any point of the comet.*

SCIR-002 *The requirement to collect sufficient cometary material for the dust analysers - MODULUS-Berenice, COSIMA, MIDAS - will lead to 10% of the S/C wall which carries the collectors to be eventually covered by dust. To monitor the dust and gas emission by GIADA and ROSINA, respectively, extended passages through dust and gas jets will be required.*

SCIR-003 *Finally, synergistic investigations from Rosetta and from the Lander require that the orbiter in principle shall be able to reach any position around the nucleus at any given time.*

3.1.4 Cometary Environment

The Comet Nucleus model corresponding to the science operations phase during the mission proper in 2011-2013 has the following characteristics:-

The range of values of the nucleus radius of the comet for design cases will be

r "" 0.5 to 1.5 km, with a ‘most probable’ value of 0.7 km.

The range of values of bulk density of the nucleus for design cases shall be p = .3 to 2.0 gcm^{-3}, with a “most probable” value in the range p = .6 - .8 gcm^{-3}.

The comet dust and gas environment is described in “An Engineering Model of the Dust and Gas Environment of the Inner Coma of Comet P/Wirtanen” (RD-9).

The reference comet nucleus model is discribed in RD-10

3.2 Mission Design

The mission design is performed by ESA/ESOC and/or by a contractor under ESOC supervision. This work is reported in RD-5.

3.2.1 Mission Overview

The Rosetta spacecraft is launched by Ariane 5 from Kourou. Calculations of the Ariane 5 launch vehicle performances show that a delayed ignition of the L9.7 upper stage is required to have sufficient spacecraft wet mass in an escape trajectory for a fly-by with Mars. Utilizing a Mars gravity assist and two Earth swing-bys, a unique mission opportunity has been found to P/Wirtanen with a launch in January 2003.

The orbit remains close to the ecliptic, with maximum Sun distance 5.25 AU, minimum Sun distance 0.9 AU, and maximum distance to Earth 6.2 AU.

The mission has several solar conjunctions (about 1 per year) and several zero declination passages.

3.2.2 Mission Phases

Most operational requirements can be directly identified in a sequence of events. Different modes of operations will be used during various mission phases.

3.2.2.1 Operations Modes

The following major modes of nominal spacecraft operations and the corresponding ground activities can be identified:

The spacecraft is in the **Launch Mode** from removal of umbilical until solar arrays deployment. Batteries are used to supply power to the spacecraft throughout the launch phase. All activities are autonomous.

The **Activation Mode** follows the launch mode and is used until completion of spacecraft check out and payload commissioning (3 months). The spacecraft is three axes stabilised. This mode is characterised by intensive ground activities during time intervals where communication with available ground stations is possible. During initial communications (few hours) the low gain antennas are used for S-band up and down link.

The **Active Cruise Mode** is used during the interplanetary cruise phases in all time intervals around major mission events, such as planetary swing-bys, major orbit manoeuvres and the comet rendezvous manoeuvre. The active cruise mode will also be used during the comet approach phase to a distance of 1000 km from the comet. It is characterised by daily tracking, monitoring and commanding. Whenever analysis, intervention, spacecraft reconfiguration or back up is required in case of on board anomalies, the ground segment will switch back to the active cruise mode. In the active cruise mode, the science payload will be off; the navigation camera will be used for comet detection and approach navigation.

The **Hibernation Mode** is an operational mode without ground support. It is used during all the interplanetary phases between major mission events. The spacecraft shall be as passive as possible, possibly with spin stabilisation to maintain the attitude and to minimise thruster operations. The power consumption in this mode shall be minimised and thermal control shall keep the collectively controlled units above their minimum allowable temperatures.

The **Asteroid Fly-by Mode** is used before, during and after asteroid fly-bys. The scientific payload will be operated during the pre-fly-by navigation and targeting and during the fly-by itself. Communications will use the high gain antenna except for a short period of time corresponding to the close fly-by during which loss of earth pointing is allowed. The spacecraft shall rotate around its axis of lowest inertia to enable a fly-by at the shortest distance with the payload line of sight remaining asteroid pointing. Science data is recorded during this phase and transmitted after reacquisition of the Earth after the fly-by.

The **Near Comet Mode** is dominated by scientific activities of comet characterisation and observation. The instrument line of sight will remain pointing to the nucleus, while the high gain antenna is pointed to the Earth to maximise the down link capability and the solar arrays pointed to the Sun. This mode also covers the delivery phase of the surface science package and extends through perihelion passage of the comet.

3.2.2.2 Sequence of Events

The major events of the ROSETTA mission are:

MAJOR EVENT	NOMINAL DATE
Launch	21 January 2003
Mars gravity assist	26 August 2005
Earth gravity assist #1	28 November 2005
Otawara fly-by	10 July 2006
Earth gravity assist #2	28 November 2007
Siwa fly-by	24 July 2008
Deep space manoeuvre	29 June 2009
Rendezvous manoeuvre (4.5 AU)	29 November 2011
Start near nucleus operations @ 3.25 AU (from Sun)	22 August 2012
Perihelion passage (end of mission)	10 July 2013

3.2.2.3 Mission Phases

The following table lists the mission phases; for details refer to RD-5.

Mission phase
Launch Phase
Commissioning Phase
Earth to Mars Cruise
Mars Gravity Assist
Mars to Earth Cruise
First Earth Gravity Assist
Earth to Asteroid Cruise

Mission phase
Asteroid Otawara Fly-by
Otawara to Earth Cruise
Second Earth Gravity Assist
Earth to Second Asteroid Cruise
Asteroid Siwa Fly-by
Asteroid to Deep Space Manoeuvre Cruise
Deep Space Manoeuvre
Aphelion Cruise
Comet Orbit Matching Manoeuvre
Near Comet Drift Phase
Far Approach Trajectory Phase (up to 90 days)
Close Approach Trajectory Phase
Transition to Global Mapping Phase
Global Mapping Phase
Close Observation Phase
Surface Science Package Delivery Phase
Relay Orbit Phase
Extended Monitoring Phase
Run Down

After comet detection, the rendezvous manoeuvre strategy can be selected subject to availability of battery power, solar array degradation and available propellant.

Nominally, unless a mission extension is agreed and if the spacecraft survives in the cometary environment, the mission ends at the perihelion pass around day 3800.

3.2.3 Orbital Parameters

This paragraph provides major parameters of the current mission design. Details can be found in the Consolidated Report on Mission Analysis (RD-5).

The following figures are applicable to the spacecraft design:

Fig. 3.2.3-1 deleted

Fig. 3.2.3-2 shows the distance of the spacecraft to the Sun from the day of launch.

Fig. 3.2.3-3 shows the distance of the spacecraft to the Earth from the day of launch.

Fig. 3.2.3-4 shows the angle between the Sun and Earth as seen from the spacecraft.

Fig. 3.2.3-5 shows the angle between the spacecraft and Sun as seen from the Earth.

File Contains Data for PostScript Printers Only

Fig. 3.2.3-2
The distance of the spacecraft to the Sun from the day of launch.

File Contains Data for PostScript Printers Only

Fig. 3.2.3-3
The distance of the spacecraft to the Earth from the day of launch.

File Contains Data for PostScript Printers Only

Fig. 3.2.3-4
The angle between the Sun and Earth as seen from the spacecraft.

File Contains Data for PostScript Printers Only

Fig. 3.2.3-5
The angle between the spacecraft and Sun as seen from the Earth.

3.2.4 Mission Design Requirements

3.2.4.1 Interplanetary Mission Phase

The interplanetary mission starts after injection into an escape orbit by the launcher and lasts until the beginning of the rendezvous manoeuvre sequence. In this part of the mission there is no solar eclipse.

MISS-001 *The spacecraft shall support a delayed ignition of the launcher upper stage; in particular it shall run on internal power and withstand the thermal environment.*

MISS-002 *The spacecraft shall support a mission with Earth and Mars gravity assists, with a minimum distance to the Sun of 0.9 AU, a maximum of 5.25 AU and a maximum Earth distance of 6.2 AU.*

MISS-003 *After injection, the spacecraft shall autonomously detect separation, command the deployment of the solar arrays and reorient the spacecraft and solar arrays to achieve sun pointing.*

MISS-004 *The spacecraft shall after initial sun acquisition receive telecommands and provide telemetry to perform spacecraft and instrument commissioning.*

MISS-005 The spacecraft propulsion system shall support a post launch orbit manoeuvre to correct for injection errors and to ensure a launch window of at least 10 days and 20 minutes per day.

MISS-006 *The spacecraft shall function fully autonomously throughout the solar conjunction on cruise to Mars.*

MISS-007 *The spacecraft shall support large orbital manoeuvres associated with the gravity assist swing-bys and comet RDV manoeuvres.*

MISS-008 *The spacecraft design shall ensure communications with the Earth throughout the interplanetary mission for the full range of Sun-S/C-Earth angles as given in fig. 3.2.3-4, except during Sun/Earth/spacecraft conjunction.*

MISS-009 *The spacecraft shall support the Otawara (10.63 km/s fly-by at a distance of 750 km and fly-by distance to asteroid Siwa of 1600 km.*

MISS-010 *The spacecraft shall be able to operate autonomously throughout the aphelion cruise up to the rendezvous manoeuvre for a period of up to 40 month.*

MISS-011 *The spacecraft shall support a rendezvous manoeuvre at 4.5 AU and shall ensure communications with the Earth throughout the manoeuvre. (optionally is a manoeuvre without communications link).*

MISS-012 *The spacecraft shall autonomously and without the need for batteries reacquire sun pointing of the solar arrays upon detection of potential loss of solar power. Off-pointing of the solar arrays from the sun, in case no battery power is available, up to 30 degrees shall be allowed for SSP delivery; (< 3.25 AU) any hardwired safe limit shall be set in excess of 30 deg*

MISS-013 *The spacecraft shall at all times be able to receive telecommands; if telecommunications via the HGA is lost, the spacecraft shall autonomously switch back to antennae with spherical coverage.*

MISS-014 *The spacecraft shall at preset times based upon on-board stored ephemeris autonomously point the HGA to the Earth when telecommunications are unintentionally interrupted.*

MISS-015 *The spacecraft shall carry a Medium Gain Antenna (MGA) mounted on the body as a backup for the High Gain Antenna.*

MISS-016 *Station coverage for the Mars and asteroid fly-bys shall be synchronised with critical operations by adjustment of the arrival times; the time-line for operations shall be based on a minimisation of the delta-V needed for the phasing manoeuvres.*

3.2.4.2 Near Comet Mission Phase

The near comet phase starts with the rendezvous manoeuvre which matches the spacecraft trajectory to the comet orbit.

MISS-020 *The spacecraft shall support full science operation when the comet is not more than 3.25 AU from the Sun.*

MISS-021 *Thrusters shall be capable of providing pure torques (bi-directional) about three orthogonal axes to control the attitude without orbit perturbations; residual perturbations are to be minimised and identified for approval by ESA.*

MISS-022 *Thrusters shall be capable of providing pure forces (bi-directional) in three orthogonal directions without attitude disturbances; residual perturbations are to be minimised and identified for approval by ESA.*

MISS-023 *The propulsion system shall be capable of performing orbit corrections with a resolution of 1 mm/s (design goal 0.1 mm/s).*

MISS-024 *The spacecraft shall carry a navigation camera system for detection of the comet nucleus and for relative navigation near the nucleus.*

MISS-025 *The spacecraft shall provide the capability to scan the celestial sphere with the navigation camera system for detection of the comet.*

MISS-026 *The high gain antenna shall continuously point to the Earth for maximum down-link communication during the near comet phase, possibly except in very specific cases such as SSP delivery. These cases shall be identified and must be agreed with the Agency.*

MISS-027 *The spacecraft shall orbit the nucleus in " circular" or " elliptic" orbits with an orbital plane at a selectable angle w.r.t. the sun vector and w.r.t. the rotation axis of the nucleus at a minimum distance of 1 equivalent nucleus radius and a maximum distance of 25 nucleus radii.*

MISS-038 *The spacecraft shall autonomously eject the SSP in accordance with a pre-defined schedule from either an " elliptic" or " hyperbolic" spacecraft trajectory into a landing trajectory.*

3.2.4.3 Navigation

Navigation is the function to determine and control the spacecraft trajectory on the basis of measurements and estimation algorithms.
This does not include spacecraft attitude estimation and control.

The navigation function for Rosetta will be performed by ESOC on the basis of radiometric measurements via the TT&C subsystem and astrometric measurements of the comet position.

The relative position and velocity of the Rosetta spacecraft with respect to the asteroids or comet nucleus will be refined with the help of processing of images obtained with the optical cameras on-board the spacecraft.

ESOC will be responsible to define the orbit manoeuvre strategy for the entire mission, including planets and asteroid fly-bys, comet detection and approach, nucleus mapping, characterisation and escort, SSP lander delivery and the relay phase.

MISS-040 *The orbit of the spacecraft and the related model parameters shall be determined, and the orbit shall be corrected, by manoeuvres through all mission phases outside hibernation such that the predicted target point at Earth or Mars, at the asteroids or at the comet has the accuracy required by the subsequent mission phase within the allocated propellant budget and guaranteeing safety.*

MISS-041 *During the interplanetary cruise, conventional radio frequency tracking techniques (ranging and Doppler) from ground using S/X band up link and down link shall be used.*

MISS-042 *Before the Mars fly-by radio frequency tracking shall be augmented with optical observables (eg Phobos versus star background) using the navigation camera system.*

MISS-043 *The navigation authorities (i.e., ESOC FD) shall define the orbit strategy for comet detection, comet approach, nucleus mapping and characterization, escort and SSP delivery. The spacecraft shall support these activities.*

MISS-044 *Ground navigation shall be complemented with optical navigation for near comet operations and also for asteroid fly-bys.*

MISS-045 *The navigation authorities (i.e., ESOC FD) shall provide a model of the comet nucleus, equipped with a latitude/longitude grid to allow position identification for SSP delivery and scientific measurements. The spacecraft shall support these activities.*

MISS-046 *The navigation authorities (i.e., ESOC FD) shall determine the comet model parameters:*
1) gravitational constant,
2) gravity field harmonics,
3) kinematic motion of the (relatively) inactive nucleus,
4) outgassing environment of the nucleus and any associated torques.
The navigation shall account for solar radiation and comet gas and particle flux acting on the spacecraft. The spacecraft shall support these activities.

MISS-047 *Near comet orbits shall be planned to avoid collision with the comet in case of failures on the spacecraft or in ground operations. The spacecraft shall support these activities.*

Ground astrometry for initial targeting and comet or asteroid acquisition by the navigation camera shall be organized so that comet and asteroid ephemeris are estimated to an accuracy compatible with acquisition by the navigation camera.

MISS-049 *Image processing on the ground shall be able to provide the following information to the overall navigation system:*
1) resolution of the target as a point direction versus a background given by star references,
2) computation of target line of sight by means of processing images together with information from attitude sensors,
3) processed comet image as an extended object with determination of geometric centre of brightness,
4) processed successive images so as to derive angular rates and range rates of the comet nucleus,
5) identification of landmarks, terminators or limb.

Flight dynamics (ESOC) is responsible for the lander navigation and trajectory; they will define the required separation velocity-increment to be provided by the SSP within the range 0.05 to 0.5 m/s and better than 1% 1 sigma accuracy.

MISS-050 *No spacecraft thruster operation shall be required within 3 hours before and 10 minutes after ejection of the SSP.*

MISS-051 *Deleted.*

The landing site will be specified by the SSP team. It also has the responsibility to ensure that the SSP has the proper attitude at landing.

MISS-052 *Guidance and navigation shall ensure that the post delivery orbit allows communications with the lander for: at least 3 hrs during the first 24 hrs on the surface and at least 15 min. every 16 hrs thereafter at less than 150 km range for the remaining 96 hrs.*

MISS-053 Guidance and navigation shall ensure that the post delivery orbit allows communications continuously from 15 minutes before touch-down to 15 minutes after touch-down.

All products of the navigation process obtained by the ESOC flight dynamics team will be made available to the scientists upon request.

3.3 General Requirements

3.3.1 Spacecraft Configuration

GERE-001 The spacecraft configuration shall allow for a spin stabilised hibernation mode; the stability margin shall be proven to be adequate by analysis to take into account fuel reallocation in the tank-system.

During active phases the spacecraft will be three-axes stabilised.

GERE-002 The spacecraft shall accommodate the orbiter payload instruments as listed in paragraph 3.1.2 in accordance with the relevant EID-Bs (AD-15 to AD-26).

GERE-003 The spacecraft shall accommodate the lander as defined in paragraph 3.1.2 in accordance with the LID-B (AD-27).

GERE-004 The spacecraft shall accommodate the standard radiation environment monitor (SREM) in accordance with AD-33.

GERE-005 The spacecraft shall accommodate all instrument sensors not including the plasma package or SSP on a single side of the main body with their bore-sight perpendicular to the spacecraft side; this general direction will be called “instrument line of sight” or “payload line of sight”.

GERE-006 The spacecraft shall provide two deployable booms of approximately 1.5 m each to accommodate the plasma sensors in accordance with their directional requirements.

The Solar Array Drive Mechanism (SADM) has to be adaptable to a range of orbital periods and to significant changes in the power available from the array as the solar distance varies between 1 and 5 AU.

The final range of comet orbital period will be determined by the orbits selected for the comet observation phase and will not be known until arrival at the comet. It is anticipated that the orbit will have a period of the order of a few days. This factor, together with the possible need to trim the array position during hibernation, requires the SADM to have an accurate position control capability.

The science data down-link is via X band direct to Earth. The necessary antenna size, consequent beam width and requirements for near constant link available during comet operations determines that a mechanism is needed to continuously orientate the antenna toward the Earth.

GERE-010 The spacecraft shall provide a continuous pointing of the HGA to the Earth during the comet operations phase. This is required to maximise the transmission of science data and to support prolonged tracking. Small outages are allowed.

GERE-015 The spacecraft propulsion system shall be a bipropellant system with thrusters capable of being operated in blow-down mode.

GERE-016 The main actuators for attitude control in orbit around the comet nucleus shall be reaction wheels; usage of thrusters shall be minimised to occasional preset periods, for orbit manoeuvres and to off-load the reaction wheels.

GERE-017 The spacecraft thermal design shall be optimised to allow for minimum equipment temperatures at aphelion to minimise the power demand in hibernation.

3.3.2 Spacecraft Resources

ESA is responsible for the procurement of the launch vehicle, of which the performance is not definitively known; therefore ESA shall retain control over the spacecraft wet mass at launch.

GERE-020 The wet mass of the spacecraft (excluding adapter and clamp band) shall not exceed 3004 kg.

GERE-021 Structural dimensioning shall allow for a growth of up to 3190 kg in wet mass. Actual real flight margins shall be determined up to the goal of 10% wet mass increase.

GERE-022 Tank dimensioning and pressurisation strategy shall be compatible with a possible increase in propellant mass of up to 20%.

ESA is ultimately responsible for the mission design and therefore retains control over the delta-V requirements.

GERE-023 The on-board propulsion system shall have a minimum capability of 2130 m/s. This does not include an allocation for attitude control.

Included in this number are a post-launch correction manoeuvre to ensure a 10 day launch window and correction of injection errors and allocations for deep space navigation and near comet manoeuvring.

GERE-024 The contractor shall maintain an ESA reserve of 100 kg in the dry-mass budget.

4. SPACECRAFT SYSTEM

4.1 General Requirements

4.1.1 Lifetime requirements

Nominal lifetime:

LIFE-001 *The spacecraft shall have a nominal lifetime of 11 years in heliocentric trajectory.*

Ground storage / lifetime

LIFE-002 *The satellite shall permit storage in a controlled facility before launch, after integration of experiments, prior to start of transport to the launch site of 12 months.*

Design lifetime

LIFE-003 *Design lifetime shall be as specified elsewhere in this document for electrical, mechanical design with respect to environmental influences and use conditions. Where the design margin on nominal lifetime is not identified, or where the design margin is required for demonstration of resistance to failures modes, a factor of two times the nominal lifetime shall be included as a minimum.*

Consumable lifetime

LIFE-004 *The lifetime of items which degrade with usage shall be designed as two times the nominal mission lifetime usage, except for mechanisms where MRVT-016 shall apply.*

LIFE-005 *None of the mission essential operations after solar array deployment shall depend upon the availability of battery power. Batteries shall be available to enhance the mission. If pyros (and similar devices) are to be fired a long time after launch, alternative means of (temporary) energy storage for fast release shall be made available.*

LIFE-006 *If gyros are used for attitude control alternative methods for control shall exist which do not depend upon the functioning of gyros.*

Micro-meteorite environment

LIFE-007 *The spacecraft shall be designed to withstand the predicted micro-meteorite environment throughout the mission with a probability larger than 0.998. The environment shall be as defined in RD-11.*

4.1.2 Autonomy

Definition of Key Terms

The following terms are used to describe characteristics that are peculiar to the Rosetta mission. These definitions apply to this section only.

autonomy

Autonomous operations are those operations that the spacecraft will itself initiate following an on-board event, these may be implemented by hardware or software or a mixture of both. As such elements of autonomy requirements will be contained in other sections of this ITT.

on-board event

An on-board event is any event not foreseen in the mission time line, typical event could include:-

1. Unit, sub-system or experiment not performing as specified
2. LCL tripping
3. Loss of sun or earth pointing

on-board processor
This is a single processing unit containing it's own RAM, PROM, power supply, bus interfaces and other dedicated hardware.

mission critical
Any function or group of functions is deemed mission critical, if its failure will result in the spacecraft being in an unrecoverable status by ground. Spacecraft mission critical manoeuvres may comprise at least the following:

a) emergency sun pointing;
b) emergency Earth re-acquisition.

General Requirements

AUTO-005 *Spacecraft autonomy shall be limited to the minimum level needed to ensure achievement of all mission goals.*

AUTO-010 *The number and use of safe spacecraft modes after an on-board event shall be minimised. Autonomous recovery from an anomalous event to an operational mode will be preferred, if this is possible without jeopardizing the mission.*

This is in order to continue production of mission products where possible after an event, however this should not be a design driver, rather the spacecraft safety should always be ensured.

AUTO-015 *Autonomy shall be implemented using both hardware and software elements. A Failure Modes, Effects and Criticality Analysis (FMECA) of the system level design shall be carried out to determine failure modes and recovery options, and only then broken down into software and hardware implementations.*

AUTO-020 *The autonomy implemented in software shall be designed using deterministic algorithmic techniques.*

Software should not use advanced software techniques such as model-based, fuzzy-logic, or rules based systems.

AUTO-025 *As a design goal software elements of autonomy shall be implemented using on-board procedures written in the Spacecraft Control Language, only where it is demonstrated that this is not desirable should separate elements of code be used.*

AUTO-030 *The contractor shall study the use of majority voting for the implementation of mission critical Autonomous actions.*

AUTO-035 *The contractor shall study the use of Boolean AND or OR voting from dual units for the implementation or control of critical Autonomous* actions.

AUTO-040 *The contactor shall study the use of hot stand-by of redundant units during critical mission phases, to avoid waiting for power on of the redundant unit.*

AUTO-047 *The DMS processors shall be in overall control of all autonomous actions. However the lower level functionality may be decomposed and implemented within the other units.*

AUTO-055 *Autonomous actions shall not be started as the result of just one sample of a parameter being in error.*

Techniques such as:-

using more than one sample taken over time of the same parameter, time skewing data gathering by different processors, using data from main and redundant units, validating parameters against defined limit sets should be used

AUTO-060 *It shall be possible for the ground to test non-operational units using built in test modes without interrupting the operation of the spacecraft.*

AUTO-065 *In the event of a fault which is not immediately and autonomously recoverable, Spacecraft Failure Management functionality shall autonomously ensure that the spacecraft is placed in a safe sun pointing configuration and eventually in a commandable state.*

Where Spacecraft is not equipped to autonomously recover from a detected failure mode, it will transition to a pre-defined safe state, and await ground intervention. Ground intervention will not necessarily be possible for some months, depending on the point in the mission.

AUTO-070 *Spacecraft shall provide contingency measures to recover sun pointing of the arrays in case of failure.*

This is a mission critical manoeuvre. During mission critical manoeuvres, Spacecraft shall autonomously manage any necessary switching between redundant equipment in real time.
There could be insufficient time to replace a faulty unit from cold, or recovery via a 'safe mode'.

AUTO-075 *The spacecraft shall support telecommanded override of all autonomous actions, either to inhibit an action, or to cancel an action in progress, should it be determined that the action is undesirable.*

However the contractor shall identify (with justification), where this is not practicable or where additional protection is required.

AUTO-077 *The spacecraft shall provide an independent mechanism for entering a safe mode (sun-pointing and ground-commandable) which will be initiated at pre-configured points in the mission (by time and ephemeris). Such entry points shall be individually cancellable by Telecommand.*

Failure Management

AUTO-080 *The Spacecraft DMS function shall encompass a Failure Management System.*

AUTO-085 *The implementation of the Spacecraft failure management system shall identify critical functions (such as failure detection, identification and recovery software), and distinguish them from routine functions (such as monitor and control, execution of time line etc.). The implementation of the Failure Management System* *shall ensure the independence of critical and routine functions.*

This will permit the independent development of each key area, and will restrict the effect of any change made to one area. Separation of these functions will also allow the best design method to be selected for each area, and will dramatically simplify the testing of this complex function.

AUTO-090 *It shall be possible for the routine functions (control and monitoring etc.), to run in the absence of the failure detection, identification and recovery software.*

AUTO-095 *It shall be possible for the critical functions (failure detection, identification and recovery) to run in the absence of the routine functions.*

Spacecraft Control & Autonomous Control

AUTO-115 *In general autonomous thruster firing is not allowed during nominal operations, all thruster firing shall take place using ground up linked parameters this does not preclude that the actual firing takes place outside ground contact. Autonomous thruster firing is only foreseen for safe mode entry, momentum management, delta-V manoeuvres and possibly during hibernation.*

AUTO-120 *Spacecraft shall autonomously manoeuvre into a pre-defined safe mode on determination of an unsafe situation on-board. This autonomous functionality may be disabled by ground command during certain critical manoeuvres (e.g. during SSP ejection).*

This will be on the basis of selecting a pre-defined set of commands in this case.

AUTO-125 *The spacecraft shall autonomously control its re-orientation to achieve sun pointing and deployment of the solar arrays on separation from the launcher.*

Spacecraft will be powered by battery for the duration of the launch; since the batteries will be sized only to reach this point in the mission, it is necessary for the solar arrays to be deployed for the spacecraft to remain functional. This must be achieved autonomously since at this point, there will be no continuous data link to ground.

AUTO-130 *Spacecraft shall automatically enter hibernation in accordance with a pre-defined on-board schedule.*

The start time of each hibernation phase will be planned on ground.

AUTO-135 *As a baseline the spacecraft will be in hibernation during long cruise phases. Should any low-level on-board monitoring and autonomous actions be required during hibernation, this shall be performed at the lowest level possible without need for a full spacecraft wake-up.*

AUTO-140 *Spacecraft shall autonomously wake up from hibernation in accordance with a pre-defined on-board schedule.*

The end time of each cruise phase will be planned on ground.

AUTO-145 *On wake up from hibernation Spacecraft shall:-*
1. autonomously reassert fine sun pointing of the solar arrays.
Wake up functionality will include re-acquisition of the sun to ensure spacecraft power levels.
2. autonomously establish 3-axis stabilisation.
This will be required to allow establishment of a communications link with Earth.
3. autonomously re-acquire the HGA communications link with the Earth using preset conditions.

Wake up functionality will include re-pointing of the HGA assembly to the Earth, in order to restore the maximum rate communications channel. This will allow the resumption of telemetry.

AUTO-165 Following manoeuvre from SSP ejection orbit to safe orbit, spacecraft shall autonomously re-acquire sun pointing of the solar arrays.

AUTO-170 *Following manoeuvre from SSP ejection orbit to safe orbit, Spacecraft shall autonomously re-acquire Earth-pointing of the HGA assembly using preset conditions.*

This will be necessary to re-acquire the HGA communications link, needed to relay the acquired SSP science data.

Autonomous Control During Hibernation
Comment: If supported by mission analysis, there may be no processor or software activity during hibernation. i.e.,, spacecraft control during hibernation will be entirely passive. This is considered to be the optimum hibernation solution.

AUTO-175 *During hibernation, Spacecraft shall autonomously maintain sun pointing of the solar arrays comensurate with power and thermal constraints.*

This will be achieved either through passive spacecraft attitude control, or through active closed loop control.

AUTO-180 *Any active processor shall be monitored by a dedicated hardware watchdog.*

On tripping, the active processor watchdog shall shut down the tripping processor and wake up an alternative processor to continue the monitoring. This will occur if either the active processor enters an endless loop, or self-detects a fault and ` commits suicide'. Waking up an alternative processor to continue the monitoring process will minimise the power requirements of the monitoring process by deferring any Failure Management action until after the completion of hibernation. This action may require the watchdog to additionally wake up an alternative watchdog to monitor the replacement processor.

AUTO-185 *Any function which persistently initiates false wake-ups shall be autonomously inhibited.*

Redundancy

AUTO-190 *It shall be possible at boot up time to load and operate AOCS/DMS software on any of the processors*

This is to provide more than one level of redundancy given by the traditional main and redundant subsystem with imbedded software.

AUTO-195 *All CDMU processors shall have the capability to be connected to both main and redundant spacecraft OBDH buses.*

This will be necessary to maximise the resilience of the system to multiple faults.

AUTO-200 *All units shall be independent of their redundant alternatives.*

This will be necessary to ensure that any incorporated redundancy removes the single point failure targeted.

On-Board Storage of Events

AUTO-205 *When the SSMM is powered, the spacecraft shall record events, (reported as events packets), in packet stores in the Spacecraft SSMM.*

Events packets shall include:

a) H/W and S/W autonomous actions
b) H/W failures
c) S/W failures
d) daily time stamp of major mission events
e) deviations from pre-loaded mission time line
f) actions started by the on-board monitor

AUTO-210 *Spacecraft critical events shall also be stored in a critical events log.*

A typical critical event is a one-off event such as failures and re-configurations

AUTO-215 *The Spacecraft critical event log shall be stored in non-volatile memory.*

This is to enable ground to interrogate the log even after a switch over or loss of power.

AUTO-220 Deleted

AUTO-225 *All entries in the Spacecraft critical event log shall include the full event packet..*

AUTO-230 Deleted

AUTO-235 *Spacecraft shall support telecommanded down linking of the critical events log, or any selected element of the packet stores in SSMM.*

AUTO-240 *The critical event log or packet stores in SSMM shall only be cleared by ground Telecommand.*

Spacecraft Time

AUTO-245 *An independent wake up timer shall be used to identify the scheduled end of hibernation.*

The wake up timer will be independent of any active processor, since the processor which may be active is variable.

AUTO-250 *A secure wake-up strategy shall be analysed in conjunction with the FMECA (see AUTO-015) during early phase B. This strategy shall consider the use of majority voting on diverse or multiple sampling for critical autonomous actions in the event of an anomaly being encountered during the wake-up.*

AUTO-255 Spacecraft shall autonomously maintain a spacecraft time reference over the whole mission.

The value of spacecraft time will be used to drive all scheduled activity.

Spacecraft Control Language

AUTO-260 Spacecraft shall provide adequate memory for command sequences or Flight Control Procedures written in a Spacecraft Control Language .

AUTO-265 Spacecraft shall support on-board procedures written in a Spacecraft Control Language. This language shall be pre-compiled on ground and interpreted on-board.

The objective is to allow most of the normal on-board control functions and autonomy to be implemented using on-board procedures, this will allow greater flexibility for controlled change during the mission and simplify testing of the whole system.

AUTO-270 The SCL shall allow for the prioritisation of procedures to ensure that critical actions may suspend and resume non-critical actions. Non-critical actions may include the mission time line, therefore the mission time line shall also support suspension and clean re-start at identified points.

AUTO-275 Procedures shall be goal oriented commands.

Goal oriented commands will allow the procedures to take account of the current state of the spacecraft, which will make the intention of the Telecommand more likely to be met.

AUTO-280 Procedures shall be able to be compiled and up linked at any time throughout the mission

AUTO-285 Detailed mission execution shall be via high level procedures stored on-board

AUTO-290 These procedures shall execute steps against an elapsed time line starting from the start of the procedure

AUTO-295 It shall be possible to execute procedures in parallel. The maximum number is to be agreed.

AUTO-300 Normal practice in writing procedures will be for each procedure to only address one unit or experiment. However the onboard system shall not restrict a procedure from addressing any number of units or experiments

AUTO-305 Procedures shall be executed on board cyclically (one instruction per procedure per cycle) so execution can be completely repeatable and predictable by ground

AUTO-310 Procedures shall be able to issue ANY Telecommand

AUTO-315 Procedures can be up-linked to an SSMM file and/or non-volatile RAM and/or working RAM regardless of whether they are currently being executed or not .

This does not imply that the procedure is actually updated while it is executing, but rather that a copy of an executing procedure can be updated, without interrupting on-going actions.

AUTO-320 Procedures shall be modifiable on-board by use of patching

A facility to edit procedures on-board should be provided, not just delete and up-link new ones. Edited procedures should be stored back to where the original was stored.

AUTO-325 Critical & Default procedures shall be defined in phase B and early C/D.

AUTO-330 Critical & Default procedures together with others identified and developed to an agreed status shall be contained as part of the S/W code in PROM

AUTO-335 Functions in a procedure shall be able to be inhibited/enabled at run time, so that if a fault has been detected previously, commands addressing a unit can be inhibited

AUTO-340 Procedures shall be able to but not limited to:

1. *Issue Telecommands*
2. *Verify execution of a Telecommand*
3. *Verify correct state of house keeping parameters*
4. *Time out on expected values of house keeping parameters*
5. *Call another procedure for the same unit (nesting), provided that no parallel execution of OBCP's is demanded. If one OBCP is foreseen to be called by 2 parent OBCP's, the user will define a semaphore in this* OBCP so as to prevent this parallel execution.
6. *Execute another procedure for the same unit, this would force the termination of the first procedure*
7. *Mark other procedures of the same unit as run or don't run*

8. *Enter message in the error log (SGM) by generation of an event packet*
9. *Enable or disable monitoring of a parameter by the on board monitor*
10. *Modify part or all of the monitor tables being used by the on board monitor by down loading from another area of RAM or an Ssmm file*
11. *Branching.*
12. *Conditional execution.*

Mission Time Line

AUTO-345 Spacecraft shall support on-board storage of the mission time line and on-board control procedures for the full mission.

AUTO-350 Mission time line shall execute the mission by executing procedures at specified times or by performing lower level actions.

AUTO-355 Spacecraft shall support without affecting the performance of the time line telecommanded:
** re-scheduling of previously loaded events*
** deletion and insertion of events at any point*

AUTO-360 Time resolution of the time line shall be 1 second

AUTO-365 It shall be possible to store the mission time line in an SSMM file and/or non-volatile RAM and/or normal RAM or a combination of the above

AUTO-370 The mission time line shall be loadable from an SSMM file and/or non-volatile RAM and/or ground command

AUTO-375 It shall be possible to store any Telecommand in the mission time line including software commands to the same software executing the mission time line

AUTO-395 It is foreseen that the majority of all mission tasks shall be performed via high level procedures. Only exceptional tasks will be executed via application programs or single commands

On Board Monitor

AUTO-400 *The Spacecraft DMS function shall monitor the health of spacecraft equipment (including payload) connected to the DMS buses.*

Intelligent devices within the spacecraft will maintain their own housekeeping telemetry, and will report their health on request to the CDMU. Dumb devices will be subject to lower level probing by the CDMU device.

AUTO-405 *The on board monitor shall be able to check House Keeping parameters by accessing the latest telemetry*

AUTO-410 *It shall be possible to copy any cell of memory into the data pool, thus allowing the board monitor to treat the contents of this cell as a parameter (max. number to be handled at any one time shall be fixed to 10).*

This will allow the on-board monitor to trigger on internal software variables if required

AUTO-415 *The monitor shall be table driven*

AUTO-420 *Only one table shall be active at a time but additional tables should be available as an SSMM file or held in RAM*

AUTO-425 *It shall be possible to up load monitor tables whole or part to an SSMM file and/or RAM*

AUTO-430 *The monitor shall be able to apply high low limits or a single value as a pass fail criteria*

AUTO-435 *The monitor shall be able to wait a number of consecutive out of limits before starting any recovery action*

AUTO-440 *The monitor shall report all failures to an error log*

AUTO-445 *The monitor shall be able to stop an executing procedure*

AUTO-450 *The monitor shall be able to start execution of a procedure*

AUTO-455 *The monitor shall be able to stop execution of the time line*

System Level AIV

AUTO-460 *The spacecraft shall support introduction of faults in-order to test the on-board software and autonomy at system level.*

This will be achieved by use of the I-BOB (See EGSE requirements)

AUTO-465 *The spacecraft EQM shall form a test bed for detailed testing of the on-board software and autonomy during the PFM integration and test phase.*

This means that a second EGSE is required.

AUTO-470 *Elements of the Spacecraft EQM (excluding payload) shall form part of the simulator for delivery to ESOC .*

AUTO-480 *All system level functional tests, (ie. not including unit integrations), shall be performed using on-board procedures written in SCL and executed on-board, any deviation from this shall be agreed with ESA in advance.*

4.1.3 Pointing

The following definitions are based upon the reference document RD-2

Absolute Pointing Error (APE)

The absolute pointing error is the angular separation between the actual instantaneous generalised pointing vectors of the spacecraft and the commanded or desired generalised pointing vectors.

Relative Pointing Error (RPE)

The relative pointing error is the angular separation between the actual instantaneous generalised pointing vectors of the spacecraft and the median generalised pointing vectors defined over a time interval containing the reference time instant.
The relative pointing error is also known as pointing stability.

Absolute Measurement Error (AME)

The absolute measurement error is the angular separation between the actual and measured generalised pointing vectors of the spacecraft.

4.1.3.2 Absolute pointing requirements

Pointing requirements are specified in terms of absolute pointing error and relative pointing error specifications, separately applicable during different mission phases and with internal and external perturbations present during these phases. These include all perturbations which influence the pointing budgets e.g.

- vibrations from reaction wheels, solar array drive mechanisms, HGA drives, payload mechanisms etc,
- solar pressure perturbations,
- attitude sensor, actuator and controller noise contributions,
- perturbations from cometary gas and dust impact,
- etc.

The pointing error specifications are expressed in terms of 95% confidence levels, which implies that the specific constraints should not be exceeded with a probability in excess of 5%.

The attitude pointing requirements will be expressed in terms of absolute and relative pointing error specifications separately in terms of a roll about the major direction of activity in a certain mission phase and about axes perpendicular to this major direction.

Major directions are defined as e.g. lines of sight of cameras or payload instruments during observations, lines of thrust during orbital manoeuvres, etc.

Attitude pointing requirements are defined for the following mission phases:

- orbital manoeuvres
- asteroid and comet detection
- asteroid fly-by
- comet nucleus observation
- SSP delivery and data relay
- HGA pointing
- Solar array pointing

For some of these phases relative pointing accuracy may however not be relevant.

4.1.3.3 Absolute pointing accuracy requirements

Orbital manoeuvres

PINT-001 *The absolute pointing error of the thrust vector during orbital manoeuvres shall not exceed 0.5 deg half cone angle, @ 95% confidence level. The magnitude error of an orbital delta-V manoeuvre greater than 1 cm/sec shall not exceed 1% or, below 1 cm/s, 0.1 mm/s of the commanded magnitude, @ 95% confidence level.*

No distinction is being made between orbital manoeuvres in the interplanetary phase and orbit manoeuvres during the near comet phase with respect to absolute pointing requirements.

Asteroid and comet detection phase

PINT-002 *The absolute pointing error of the payload line of sight during the asteroid and comet detection phase shall not exceed 0.3 deg half cone angle, @ 95% confidence level.*

PINT-003 *Ground communication via the HGA and solar power generation shall be maintained during asteroid and comet detection.*

Asteroid fly-by phase

PINT-004 *The absolute pointing error of the payload line of sight during the asteroid fly-by w.r.t. the barycentre of the illuminated part shall not exceed 0.3 deg half cone angle, @ 95% confidence level.*

Comet nucleus observation phase

PINT-005 *The spacecraft shall point the payload line of sight in any inertially fixed direction while maintaining a communications link via the HGA commensurate to TTCS-207 and power from the solar arrays.*

PINT-006 *In addition the spacecraft shall scan the payload line of sight between two inertially fixed directions at rates up to 1 arcmin per sec while maintaining a communications link via the HGA and power from the solar arrays.*

PINT-007 *The absolute pointing error of the payload line of sight during comet observation in any fixed or time variable inertial direction shall not exceed 0.1 deg half cone angle @ 95% confidence level.*

SSP delivery and data relay phase

PINT-008 *The spacecraft shall allow ejection of the SSP in any inertially fixed three axis attitude commensurate with available spacecraft power resources and any pointing constraints of the other payload instruments.*

PINT-009 *The contribution of the spacecraft to the absolute pointing error of the ejection velocity vector shall not exceed 0.1 deg half cone angle, @ 95% confidence level. The contribution to absolute roll pointing error about the ejection velocity vector shall not exceed 0.5 deg, @ 95% confidence level. The contribution to error in the separation velocity magnitude shall not exceed 0.5% of the required value @ 95% confidence level.*

PINT-010 *During the data relay phase, the spacecraft shall point the SSP receive antenna bore-sight to the SSP when the view to the SSP is unobstructed, alternatively, the CONSERT antenna bore-sight shall be pointed to the SSP.*

High gain antenna pointing

PINT-011 The absolute pointing error of the HGA bore-sight, when used in X-band down-link communication shall not exceed 0.15 deg half cone angle, @ 95% confidence level.

Solar array pointing

PINT-012 The spacecraft shall keep the solar arrays sun pointing after solar array deployment and initial sun acquisition apart from preselected periods for asteroid tracking, special comet observation campaigns or SSP delivery.

PINT-013 The absolute pointing error of the solar array rotation axis shall not exceed 5 deg @ 95% confidence level w.r.t. a plane perpendicular to the spacecraft sun line. This applies to periods where experiments are operating except during the SSP delivery phase.

PINT-014 The control accuracy of each of the solar array orientations shall be better than 0.1 deg

4.1.3.4 Relative Pointing accuracy requirements

Asteroid and comet detection phase

PINT-020 The relative pointing error of the payload line of sight shall not exceed 3E-4 deg half cone angle over any 2 sec interval @ 95% confidence level during the asteroid and comet detection phase.

PINT-021 The relative roll pointing error about the payload line of sight shall not exceed 6E-3 deg over any 2 sec interval @ 95% confidence level during the asteroid and comet detection phase.

Asteroid fly-by phase

PINT-022 The relative pointing error of the payload line of sight shall not exceed 3E-3 deg half cone angle over any 1 sec interval @ 95% confidence level during the asteroid fly-by at the minimum specified fly-by distance.

PINT-023 The relative roll pointing error about the payload line of sight shall not exceed 2E-2 deg over any 1 sec interval @ 95% confidence level during the asteroid fly-by.

Comet nucleus observation phase

PINT-024 *The relative pointing error of the payload line of sight shall not exceed 3E-4 deg half cone angle over any 1 sec interval @ 95% confidence level during the comet nucleus observation phase.*

PINT-025 *The relative roll pointing error about the payload line of sight shall not exceed 6E-3 deg over any 1 sec interval @ 95% confidence level during the comet nucleus observation phase.*

4.1.3.5 On-ground attitude reconstitution accuracy requirements

Asteroid fly-by phase

PINT-030 *The absolute measurement error in an inertially fixed frame during asteroid tracking at a maximum spacecraft angular rate of 1.5 deg/sec shall not exceed 1 arcmin half cone @ 95% confidence level.*

Comet nucleus observation phase

PINT-031 *The absolute measurement error in an inertially fixed frame during comet nucleus orbiting at a maximum spacecraft angular rate of 10 arcmin/sec shall not exceed 12 arcsec half cone angle, or 60 arcsec roll angle @ 95% confidence level.*

4.1.4 Reliability, Fault Tolerance/Single Point Failures

4.1.4.1 Reliability

A quantitative reliability requirement (numeric figure for probability of success) is not specified for the ROSETTA spacecraft.

RELI-001 *However quantitative reliability assessments shall be prepared and used for comparison of alternate design concepts on system, subsystem, equipment and unit-level.*

4.1.4.2 Fault Tolerance/Single Point Failures

RELI-002 *The design of the spacecraft shall be such that a single component/part failure (e.g. resistor, transistor) cannot, as far as practicable, cause the failure of spacecraft functions which are essential for mission success (e.g. power supply, data transmission).*

RELI-003 *The design of the spacecraft shall be such that two independent failures cannot cause a hazardous situation for hazard catagories “ catastrophic”and “ severe”. (see RO-EST-RS- 1002)*

RELI-004 *Alternate or redundant paths shall be separated or protected such that an event that causes the loss of one functional path will not cause the loss or degradation of alternate, redundant or serial paths.*

RELI-005 *Improper commands or command sequences or software errors (e.g. originating from Single Event Upsets) shall not cause damage of hardware and shall not result in operational conditions which cannot be restored by ground command. This is also applicable for ground testing of flight hardware. Exceptions shall be approved by the agency and shall be reflected in the Rosetta Users Manual.*

RELI-006 *Protected switching configurations employing separate "arm" and " active" operations shall be implemented whenever an unintended activation can lead to an operational hazard (for hazard catagories "catastrophic"and "severe").*

RELI-007 *Protection systems shall, to the greatest extent possible, be intrinsically fail safe and shall be capable of being enabled and disabled by command.*

RELI-008 *Cross strapping shall be incorporated in between chains of redundant units so that the maximum overall reliability is achieved.*

RELI-009 *The activation of a redundant unit shall not require a change of configuration or operational status of another unit.*

RELI-010 *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.*

RELI-011 *Hot redundancy shall be provided for functions which are essential for a continuous, uninterrupted operation needed for mission success.*

RELI-012 *Provisions shall be made as far as practically feasible to prevent malfunction or elimination of redundant units by a common cause.*

RELI-013 *There shall be on-board failure detection capabilities for all failure modes which require switching to:*

- *redundant functional paths,*
- *contingency operating modes,*
- *temporary safe modes, or*
- *survival mode.*

RELI-014 *There shall be a trade-off on a case-by-case basis between centrally organised failure detection/reaction (e.g. monitoring by central processor units) or decentralised detection/reaction (e.g. local current limiters, thermostats).*

RELI-015 *Status indications, including any change-over to a redundant or back-up operating mode and the exact time of occurrence shall be given in the housekeeping telemetry. This information shall also be stored on-board the spacecraft as necessary for later transmission to the ground.*

RELI-016 *Checkout of all redundant or contingency operational modes or units shall be possible during ground testing of the fully integrated spacecraft.*

RELI-017 *Automatic test sequences shall include a special flag and shall require a separate confirmation by the test conductor to initiate the function of any hardware item which is built into the system design as inhibit for (double) failure tolerance for potentially hazardous configurations or operations.*

Further detailed fault tolerance requirements which are resulting from or which may be in addition to the requirements listed above may be defined in other sections of this specification.

Compliance to the fault tolerance requirements shall be verified by FMECA or by other suitable methods which are subject to agreement with ESA.

Single point failures which 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 are subject to formal approval by ESA on a case by case basis with a detailed retention rationale.

4.1.5 Materials and Processes

Materials and processes shall be selected according to their suitability for the intended application and on the basis of previous experience. The following specific requirements apply:

MATE-001 *Materials which are not used inside pressurized volumes on the spacecraft shall have low outgassing properties with in general TML < 1% and CVCM < 0.1% as determined by test method PSS-01-702 or ASTM-E595. Depending on the results of* *contamination analyses which are still to be performed, the maximum tolerable outgassing rate for materials used in specific locations (e.g. in the vicinity of experiment sensors) may have to be lower than the standard rate.*

MATE-002 *Materials shall not be flaking or dusting so that a high degree of cleanliness of the flight hardware can be achieved and be maintained. This is also applicable to materials used on GSE.*

MATE-003 *Materials and combinations of materials bonded to each other shall be resistant to the thermal cycling to which they will be exposed until the end of the Rosetta mission. A standard thermal cycling method is defined in PSS-01-704 but the contractor shall as necessary perform specific thermal cycling tests which are representative of the actual application (for qualification with at least a factor of 1.5 on the number of cycles expected until end of operational life).*

MATE-004 *Materials shall be resistant to the radiation to which they will be exposed during their operational lifetime. The contractor shall determine what the anticipated radiation environment will be for materials used in various locations/applications and as necessary demonstrate by appropriate testing that the properties would not degrade below acceptable limits for significant properties.*

MATE-005 *Materials which are in contact with each other shall be compatible with each other (e.g. fuel-compounds with sealing rings). Compatibility shall be demonstrated by test if insufficient data are available from standard references or previous applications.*

MATE-006 *Materials shall be resistant to corrosion or they shall be suitably protected against corrosion (standard atmosphere/normal air to be assumed for general applications on the spacecraft for a period of 5 years [= estimated time between last surface treatment and launch]).*

MATE-007 *Materials shall have high resistance to stress corrosion for all structural applications but also for applications (like pre-tensioned springs or welded constructions which frequently include residual internal and assembly stresses) in which the materials are exposed for extended periods of time to tensile stresses in the terrestrial atmosphere or other potentially corrosive environments.*

MATE-008 Materials with high/low resistance to stress corrosion cracking and applicable test methods are listed in PSS-01-736 and -737. The contractor shall be responsible to demonstrate adequate resistance to stress corrosion by appropriate tests if insufficient data are available from standard references.

MATE-009 Metals which are in direct contact with each other shall not form a galvanic couple with a difference of more than 0.5 volts EMF (see table 7.2.6 of PSS-01-701); in corrosive environments it shall not be more than 0.25 volts EMF (Electro-Motive Force).

MATE-010 As far as practicable, materials and mechanical parts shall be non-magnetic. In case magnetic material must be used for a particular function (e.g. in a motor or a relay) the magnetic characteristics may have to be determined and depending on the effects on system level, magnetic compensation methods may have to be applied. The overall magnetic moment which is tolerable on system level is defined in section 4.2.5.5 of this document.

MATE-011 Materials which may constitute a safety hazard or can cause contamination shall not be used without specific approval by ESA (RFW). Examples of such materials are:

- *Beryllium-Oxide*
- *Cadmium*
- *Zinc*
- *Mercury*
- *Radioactive Materials*
- *PVC.*

Control of application of such materials in EEE-parts will be dealt with in accordance with RO-EST-RS-1002, section 8.

The Quality Assurance Provisions for Materials and Processes including evaluations, qualifications, approvals and quality control are defined in RO-EST-RS-1002, section 9.

4.1.6 Maintenance, Accessability, Repairability, Testability

MART-001 *The design of the spacecraft shall be constrained by the need for accessibility and the method to be employed for removal of units, placement of test aids, for equipment requiring adjustment or maintenance with the aim of minimising:*

- *the possibility of injury to personnel*
- *the possibility of damage to equipment or facility*
- *the possibility of incorrect assembly*
- *task complexity*
- *use of special tools or equipment*
- *use of non-standard hardware*
- *design complexity*
- *the need for special skills*
- *the activity duration*

MART-002 *It shall be possible to access or remove equipment if they require maintenance control with minimum disturbance to and interference with spacecraft or payloads.*

MART-003 *Items to be removed before flight (red tag items) shall be visible after integration with the spacecraft.*

MART-004 *Items requiring integration for safety, logistical or life reasons, close to the launch, shall be accessible without removing any equipment from the spacecraft.*

MART-005 *Items which require adjustment, servicing or maintenance before launch shall be accessible without removing any equipment from the spacecraft.*

4.1.7 Identification and Marking

IDMA-001 *Each separately identifiable part or sub-assembly shall carry an identification consisting of at least the following information:*

a) Identification number
b) Equipment title
c) Serial number

Items a) and b) may be defined by the contracting authority.

IDMA-006 *All parts or sub-assembly having the same identification number shall be functionally and dimensionally interchangeable.*

4.2 Environmental Requirements

4.2.1 Mechanical Environment

ERME-001 *The Spacecraft shall be designed to withstand all mechanical static and dynamic loads encountered during its entire lifetime, including: manufacturing, handling, transportation, testing, launch and in orbit operations.*
The Contractor shall ensure that manufacturing, handling and transportation loads shall not be design drivers with the exception of the interface points to the MGSE.

ERME-005 *The mechanical environment for the launch phase shall be derived from the Launcher Coupled Loads Analysis.*

ERME-010 *The mechanical dynamic test environment shall meet the requirements of the applicable document reference AD-1, Ariane 5 User's Manual.*

ERME-015 *The mechanical static test environment shall verify the capability of the structure to withstand the Design Loads defined at chapter 4.3, Structural requirements.*

4.2.2 Thermal Environment

ERTE-001 *The Spacecraft shall be designed to withstand all the thermal environments encountered during its entire lifetime, including:*

a) integration transportation and testing,

b) spacecraft preparation at the launch site,

c) pre-launch phase with the spacecraft encapsulated int the launch vehicle,

d) ascent phase,

e) in-orbit operations from launcher separation until the end of the mission.

ERTE-005 *The constrains applicable to b), c), and d) above shall be derived from the applicable document reference AD-1, Ariane 5 User's Manuals.*

ERTE-010 *The constrains applicable to d) above shall be detailed through specific thermal analysis integrated with the launcher performed by the launcher authorities.*

ERTE-015 *The constrains applicable to a) and e) above shall be derived by the contractor.*

ERTE-020 *The in-orbit environment shall be derived from the applicable documents reference AD-12, NASA technical memorandum 8247 and reference AD-13, NASA Technical memorandum 4527.*

ERTE-025 *A model of the comet nucleus and of its environment shall be created by ESA (see chapter 3.1.4) The spacecraft design shall be compliant with the environmental requirements from this model.*

4.2.3 Cleanliness Requirements

CLEN-001 *Materials which are not used inside pressurized volumes on the spacecraft shall have low outgassing properties as defined in paragraph 4.1.5 requirement MATE-001.*

CLEN-002 *The spacecraft shall provide a centralised purging system available for experiments or other sensitive units.*

CLEN-003 *The spacecraft shall be integrated, tested, stored and transported in a clean environment of class 100,000 of US Federal Std No.209 B minimum. (see also RO-EST-RS-2002)*

CLEN-004 *Specific design requirements and cleanliness assurance provisions shall be derived from the identification of sensitive items in EID-Bs and unit specifications (e.g. sun sensors, camera etc.) and they shall be inserted into the appropriate requirements specifications, i.e., platform specification cleanliness specification etc.*

4.2.4 Radiation Environment

RADI-001 *The spacecraft shall be designed to withstand the effects of the varying flux of high energy particles encountered in its mission.*

These effects can be separated at least into three classes:

a) Radiation hazards
During its lifetime the spacecraft and its components will receive an integrated dose that can degrade their performance and possibly cause failures

b) Radiation induced background
Radiation impinging on a detector or its associated electronics will produce an increase of the background noise

c) Single event upset
Cosmic rays and heavy ion impact can provoke single event upsets which may disrupt the operation of sensitive electronics.

The results of a preliminary radiation analysis of the ROSETTA mission carried out by ESA are given in figure 4.2.4-1, 4.2.4-2, 4.2.4-3, 4.2.4-4, 4.2.4-5 and 4.2.4-6.

RADI-002 *The contractor shall be responsible for performing radiation analyses as required using the nominal mission scenario.*

RADI-003 *Electronic components applied in the spacecraft shall either be resistant to the expected radiation levels, to Single Event Upsets and Latch-Up, or suitable provisions shall be made in the design for protection against unacceptable degradation or failure effects.*

RADI-004 *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.*
An exception of this requirement are the solar cells which shall be qualified according to ESA PSS-01-604.

File Contains Data for PostScript Printers Only

Figure 4.2.4-1 The dose in silicon as a function of shielding thickness for the JPL•85 model and averaged particle spectra from the two Earth fly•bys. The "0.54 days" specified in the plot title is incorrect and is an artefact of the software. The figures are for the entire 10 year mission.

File Contains Data for PostScript Printers Only

Figure 4.2.4-2 Equivalent 1 MeV P_{max}•V_{OC} e^{-} fluences for silicon solar cells. Infinite backshielding is assumed as is a proton/electron damage ratio of 3000. The "0.54 days" specified in the plot title is incorrect and is an artefact of the software. The figures are for the entire 10 year mission.

File Contains Data for PostScript Printers Only

Figure 4.2.4-3 Equivalent 1 MeV I_{SC} e^{-} fluences for silicon solar cells. Infinite backshielding is assumed as is a proton/electron damage ratio of 3000. The "0.54 days" specified in the plot title is incorrect and is an artefact of the software. The figures are for the entire 10 year mission.

File Contains Data for PostScript Printers Only

Figure 4.2.4-4 Equivalent 1 MeV V_{OC} e^{-} fluences for Gallium Arsenide solar cells. Infinite backshielding is assumed as is a proton/electron damage ratio of 1400. The "0.54 days" specified in the plot title is incorrect and is an artefact of the software. The figures are for the entire 10 year mission.

File Contains Data for PostScript Printers Only

Figure 4.2.4-5 Equivalent 1 MeV P_{max} e^{-} fluences for Gallium Arsenide solar cells. Infinite backshielding is assumed as is a proton/electron damage ratio of 1000. The "0.54 days" specified in the plot title is incorrect and is an artefact of the software. The figures are for the entire 10 year mission.

File Contains Data for PostScript Printers Only

Figure 4.2.4-6 Equivalent 1 MeV I_{SC} e^- fluences for Gallium Arsenide solar cells. Infinite backshielding is assumed as is a proton/electron damage ratio of 400. The "0.54 days" specified in the plot title is incorrect and is an artefact of the software. The figures are for the entire 10 year mission.

4.2.5 Electromagnetic Compatibility Requirements

4.2.5.1 General and Functional Requirements

EMCR-001 *The spacecraft shall not be susceptible to self generated electromagnetic interference and 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 where applicable) and fields, which fall in either the Payload, Spacecraft or Launcher bandwidth.*

EMCR-002 *The spacecraft shall not be susceptible to electrostatic charge/discharges. (For level definition see EMCR-030)*

EMCR-003 *The spacecraft electromagnetic emissions and susceptibility shall comply with the payload requirements listed in the EIDs (AD-14 through AD-27).*

EMCR-004 *The spacecraft electromagnetic emissions and susceptibility shall comply with the launcher requirements (AD-1)*

The Contractor shall support an EMC Review Board consisting of Experimenters, Industry and ESA in order to review the Spacecraft/Payload EMC performance and to advice and seek solutions to EMC when needed.

4.2.5.2 Design Requirements

EMCR-010 *The spacecraft shall adopt a distributed single point grounding scheme*.

Note: The grounding scheme shall be selected to meet the Payload and Spacecraft Subsystems requirements. This may require a trade-off between the modified single point grounding scheme (ground bar concept) baselined in the EID Part A (AD-14) and the distributed single point grounding scheme (structure ground concept).

EMCR-011 *Galvanic isolation shall be provided between primary and secondary power with the exception of the primary power grounding within the power subsystem.*

EMCR-012 *Signal, power and data lines with their respective returns shall be separated* .

EMCR-013 *Good and reliable bonds shall be provided between the various parts of any electronic box and the spacecraft structure.*

Note: The Contractor shall define the limits acceptable limits for bonding.

EMCR-014 *The power converter frequency bands shall be selected to minimise potential interference within the bandwidth of Payload equipment.*

EMCR-015 *A frequency control plan shall be established and maintained as part of the EMC Programme* .

EMCR-016 *A 6dB interference margin shall exist between susceptibility and specified environment, except for the pyrotechnic subsystem where this margin shall be 20 dB* .

Filters shall be used to reduce conducted emission and susceptibility to a minimum.

4.2.5.3 Electrostatic Cleanliness

EMCR-020 *All spacecraft surfaces exposed to the plasma environment shall be conductive and grounded to the S/C structure.*

EMCR-021 *Exceptions from this shall be minimised and identified by the Contractor for approval by ESA* .

The exceptions shall be justified and characterised by material properties and exposed surface areas.

EMCR-022 *a) The differential charging potential shall not exceed 10 V as a design goal. For the purpose of on-ground verification the Contractor shall define a derived upper grounding impedance for exposed surfaces.*
b) Locally this should be reduced to about 1 V as a design goal in the vicinity of experiments with specific requirements identified in the EID-Bs and approved by the project.

4.2.5.4 Electrostatic Discharge

EMCR-030 *The spacecraft shall not show permanent performance degradation or failure after being subjected to such tests and shall autonomously (i.e., without assistance of ground commands) return to full performance.*

Conducted Electrostatic Discharge (current injected anywhere into the external structure):

Imax: 50 A
Rise Time: < 10 nsec (10-90%)
Duration: 100 nsec (at half amplitude
repetition Rate: 10 Hz

Radiated Electrostatic Discharge (spark gap at 30cm distance):

Energy 15 mJ
Voltage: > 10 KV
Repetition Rate: 10 Hz

4.2.5.5 DC Magnetic Requirements

EMCR-040 The total DC magnetic field generated by the spacecraft shall not exceed 25 nT as a design goal at the tip of the nominal 1.5 m long magnetometer boom.

Potential disturbing items shall be identified and measured, unit level compensation shall be applied if needed.

4.3 Structural Requirements

4.3.1 Definitions

- Primary structures:

  The primary structure is the structure through which the main flight loads are transferred and which defines the major structural frequencies.

- Secondary structures:

  The secondary structures are not responsible for the main load transfer. They are fastened to the primary structure, and transfer unit Loads to the primary structure.

- Unit/Equipment structures:

  Unit/Equipment structures are those belonging to self standing items such as experiment units and sensors, mechanisms, subsystem components and electronic boxes.

SRDF-001 Safety factors:

The Safety Factors shall account for inaccuracies in predicted allowable and applied stresses due to:

- *Analysis uncertainties*
- *Manufacturing tolerances*
- *Scatter in material properties*
- *Setting at interface.*

SRDF-002 The following safety factors shall be used:

See table 1: Safety Factors

See table 2: Additional Safety Factors.

Table 1: Safety Factors (SF)

Item	*Yield SF*	*Ultimate SF*	*Buckling SF*
Conventional Materials Metallic	*1.25*	*1.4*	*1.4*
Conventional Material non Metallic	*1.25*	*1.4*	*1.4*
Unconventional Materials	*1.7*	*1.85*	*1.85*
Inserts and joints	*1.7*	*1.85*	*NA*

Table 2: Additional Safety Factors , for material allowables (ASF)

Item	*With specific test, (Qualification or development)*	*Without specific test*
Bonding, Structural inserts (axial loading)	*1.1*	*2.0*
Honeycomb in tension	*1.65*	*forbidden*
All materials if ultimate/yield < 1.2	*1.0*	*1.7*

SRDF-006 If structural integrity, buckling or stability cannot be demonstrated by test for practical reasons, at component, unit or secondary structure level, use of unconventional materials is forbidden and an additional safety factor of 2 shall apply.

- Conventional Materials:

  All materials, also composites, provided sufficient statistical data are available to derive A values as defined in the applicable document reference AD-10, MIL-HDBK-5 section 1.4.11.

- Unconventional Materials (or Advanced Materials):

  All those materials for which sufficient statistical data are not available.

SRDF-011 *The Design Allowable for each material shall correspond to the A values as defined in the applicable document reference AD-10, MIL-HDBK-5. The Contractor shall perform material testing as necessary to establish the Design Allowable. The test programme and interpretation of the results are subject to ESA approval.*

SRDF-016 *Margin of Safety (MOS):*

The Margin of Safety is calculated as follows:

$$MOS = \frac{\text{(relevant Strength Capability)}}{\text{(Design Loads times requested SF)}} - 1$$

The Yield Margin of Safety (YMOS) compares the Yield Strength Capability of the structural elements to the Yield Loads.

The Ultimate Margin of Safety (UMOS) compares the Ultimate Strength Capability of the structural elements to the Ultimate Loads.

The Buckling Margin of Safety (BMOS) compares the Buckling Strength Capability of the structural elements to the Buckling Loads.

The Strength Capability is the Load which induces exactly the Design Allowable stresses for the material.

- Limit Loads:

  Limit Loads are the load combinations which have a 99% probability of not being exceeded with a 95% interval of confidence, during the entire life of the structure, including manufacturing, handling, transportation, ground testing, launch and in-orbit operations.

SRDF-021 *Design Loads:*

Design Loads are simplified load cases which shall envelope the Limit Loads combined with others (e.g. thermal stresses, transportation...) and the Qualification loads of the dynamic environmental tests. The contractor shall establish and maintain during the course of the programm a summary of Design Loads applicable at system, subsystem and unit level.

- Yield Loads:

  Design Loads multiplied by the yield safety factor and relevant ASF's, where applicable.

- Ultimate Loads:

  Design Loads multiplied by the ultimate safety factor and relevant ASF's, where applicable..

- Buckling Loads:

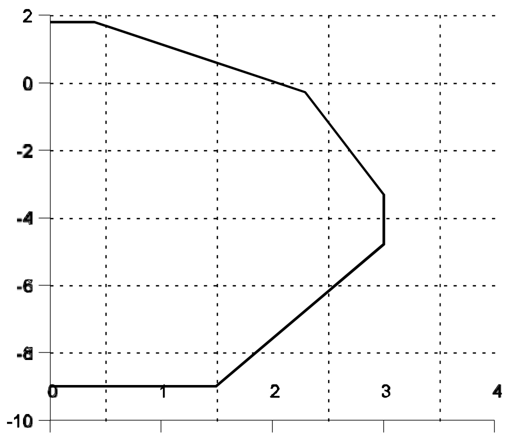
  Design Loads multiplied by the Buckling safety factor and relevant ASF's, where applicable..

SRDF-026 *The Design Loads applicable to the units shall be greater or equal to the Design Loads of the supporting structures.*

SRDF-031 *The interfaces of any unit shall be designed against the Design Loads of the unit applied at the centre of mass of the unit.*

SRDF-036 *Preliminary Design Loads:*

In order to initiate the design phase, the primary structure shall demonstrate the capability to withstand the following domain of Design Loads.
The negative sign represents a compression load, the lateral loads are circular envelopes. Axial and lateral Design Loads shall be combined.



Lateral (g)	*Axial (g)*
0.0	*1.8*
0.4	*1.8*
2.3	*-0.3*
3.0	*-3.3*
3.0	*-4.8*
1.5	*-9.0*
0.0	*-9.0*

The following Preliminary Design Loads shall be applied to all units and secondary structure: *30 g spherical envelope.*

4.3.2 Functional Requirements

SRFR-001 The interface to the Launcher shall be part of the primary structure.

SRFR-006 The primary and secondary structures shall comply with the I/F requirements of the payload defined in EID-A and B and of the SSP defined in LID-B.

4.3.3 Performance Requirements

4.3.3.1 Stiffness

SRPR-001 Structure resonance frequencies shall meet the launcher requirements.
The longitudinal frequency shall meet the B criterion of the section 5.4.3.2 of the applicable document reference AD-1, Ariane 5 User's Manual.

SRPR-003 Secondary structure, subsystems and brackets shall be designed to be decoupled from any major frequency of the S/C, the ratio of the frequency of secondary structure, subsystems and brackets to any major frequency of the spacecraft shall be greater than 1.4.

SRPR-006 The eigen frequencies of compact equipment and boxes in hard mounted condition shall be above 140 Hz.

SRPR-011 A stiffness margin on every part of the spacecraft shall be included to cover design, and material uncertainties; the mass of each unit or component shall be the maximum allocated mass.

4.3.3.2 Strength

SRPR-016 The structure shall withstand at element, subassembly, or complete S/C level the following:

- *The Yield Loads, without permanent deformation or any elastic deformation resulting in performance degradation,* e.g. stiffness.

- *The Ultimate Loads, without rupture or deformation leading to the loss of functionality.*

- *The Buckling Loads, without elastic or plastic buckling or collapse. A local buckling may be tolerated in the case of webbed structures provided it does not result into a loss of functionality.*

where the mass of each unit or component is the maximum allocated mass.

SRPR-021 *Under the Design Loads and where the mass of each unit or component is set at the maximum allocated, the force fluxes at the launcher I/F shall meet the launcher requirements as specified in the applicable document reference AD-1, Ariane 5 User's Manual.*

SRPR-026 *The S/C structure shall withstand the complete set of tests at qualification level without the need of any refurbishment for flight.*

4.3.3.3 Stability

SRPR-031 *The stability of the structure shall be commensurate with the spacecraft pointing requirements defined in the section 4.1.3. The following causes of misalignment shall be taken into account:*

- *Setting due to mounting procedures*
- *Setting due to launch distortions*
- *Misalignment 1 g/0 g environment*
- *Deformation caused by evolution of the orbital temperatures over the overall mission*
- *Ageing*
- *Creeping*
- *Composite structure deformations due to moisture absorption.*

In addition the effects of mechanical perturbations induced by the mechanisms shall be analysed and assessed.

4.3.4 Verification requirements

4.3.4.1 Modelling

SRVM-001 *All structural Finite Element Model (FEM) shall be delivered in NASTRAN format.*

SRVM-006 *The spacecraft FEM shall be detailed enough to ensure an appropriate derivation/verification of the Design Loads and of the modal response of the various structural elements of the satellite up to 140Hz.*

SRVM-011 *The spacecraft FEM shall be supported by additional and more detailed models for the analysis and the design of specific aspects, (strength verification, thermal stress analysis, thermo-elastic analysis, interface stiffness analysis, as required).*

SRVM-016 *The spacecraft FEM shall be reduced at system level into an Interface FEM (IFEM) for the Launch Vehicle Coupled Dynamics Analysis (LVCDA). This IFEM shall fulfil the requirements established by the Launch Vehicle Authorities, as defined in AD-1, Ariane 5 User's Manual.*

SRVM-021 *The FEM's shall be correlated against the results of the modal survey tests carried out at component, subassembly and complete S/C level. As a result of that all the modes with an eigen mass greater or equal to 10% of the rigid mass of the tested item shall satisfy the following:*

- *Frequency deviation less than 5%*
- *Error for the damping factor less than 20%*
- *Modal Assurance Criterion (MAC) greater than 0.9*

$$MAC = \frac{(X_{ai}^{t}\; X_{tj})}{(X_{ai}^{t}\; X_{ai})\;(X_{tj}^{t}\; X_{tj})}$$

Where the Xa and Xt are respectively the analytical and test eigen vectors.

In addition, the above requirements are apllicable at spacecraft level regardless of the effective mass if the mode can generate on any spacecraft item or payload item loads in excess of the design loads.

Finally, the major modes of the spacecraft shall be better correlated for the needs of the LVCDA and the following applies to the first two bending modes and to the first axial mode of the spacecraft.

- *Frequency deviation less than 3%*

- *Modal Assurance Criterion (MAC) greater than 0.95*

4.3.4.2 Analyses

SRVA-001 *All the Design Loads applicable to the various parts, subassemblies or complete structure shall be substantiated by analyses of significant events of the complete life time.*
They shall be reassessed after each test at subassembly or system level.

SRVA-006 *The stiffness analysis shall demonstrate compliance to the stiffness requirements as indicated at 4.3.3.1. The analytically predicted frequencies shall be higher than the minimum requirement specifications by the following percentages:-*

- *Start of phase B : 20%*
- *At System Design Review : 15%*
- *At Critical Design Review : 10%*

- *At Flight Acceptance Review: 0%*

At s/c level, the axial frequencies shall meet the above requirements when the cumulative eigen mass has been increased by the same percentage.

SRVA-011 The stiffness analysis shall also demonstrate the fulfilment of the eigen frequency requirements specified for the various flexible appendages of the S/C.

SRVA-016 The stress analysis shall yield positive MOS and cover loading originating from mechanical, thermal or moisture desorption effects combined adequately altogether.

SRVA-021 Strength values for mechanical parts shall not be assumed to be higher than the values specified for the relevant qualification and acceptance tests.

SRVA-026 Fatigue analysis shall be carried out in detail where relevant and shall demonstrate a positive MOS after application of 4 times the most constraining life cycles.

SRVA-031 Fracture mechanics analysis shall be carried out for the items which are potentially hazardous for the ground operations and meet the requirements specified in the applicable document reference AD-11, ESA-PSS-01-401, ESA fracture control requirements.

SRVA-036 The stability analysis shall be budgetized according to section 4.3.3.3.
Each potential cause of misalignment shall be compliant with its allocation.

4.3.4.3 Testing

SRVT-001 Verification of the performance of the structure shall be possible by test at element, subassembly or system level.

SRVT-006 Any structural element shall demonstrate by test the capability of withstanding the Limit Loads times the Test Factor:

- *Test factor = 1.5 for qualification*
- *Test factor = 1.1 for acceptance*

SRVT-11 The S/C primary structure shall be qualified by test; other secondary structure or units may be qualified under the proto flight approach.

SRVT-016 All the standard potted inserts shall be proof-tested up to 110% of their allowable loads before delivery of the assembly. The special inserts shall be qualified independently up to their relevant design loads.

The test procedures are subject to ESA approval.

4.3.4 Design Requirements

SRDE-001 The mechanical Design of the launcher I/F shall meet the launcher requirements specified in the applicable document reference AD-1, Ariane 5 User's Manual.

SRDE-006 Unless a cavity is hermetically sealed, adequate means of venting shall be provided in the design.
The method of venting shall prevent the contamination of the cavity by the external environment and prevent the release of contaminants from the cavity toward a sensitive equipment or component.

SRDE-011 All threaded parts shall be positively locked.

4.4 Mechanism Requirements

4.4.1 Definitions

- All subassemblies featuring parts moving under the action of commandable internal forces shall be considered as mechanisms. Deployable appendages, reaction wheels, antennas or solar arrays drives at least belong to this category.

- The functional performances of the mechanism shall be described with the following:

  - The motion of a mechanism shall be described by Kinematic Variables i.e., acceleration, velocity, displacement, and Dynamic Variables, i.e., forces and torques applied to the various mobile parts.

  - The initial or final status of the motion shall be described by so called Steady State Parameters, i.e., relative position or relative velocity w.r.t. a well identified I/F.

  - The Kinematic Variables are the results of the interaction of various parameters e.g. mass, inertia, spring force, friction, hysteresis, adhesion, etc. These parameters shall be called the Physical Parameters.

4.4.2 Functional Requirements

MRFR-001 The mechanism shall support an equipment or a P/L unit of the S/C and shall change its relative position with respect to the S/C in a predictable manner, and where applicable in accordance with the Pointing requirements, section 4.1.3.

MRFR-006 *The mechanism shall feature an unambiguous feedback control technique which is compatible with the accuracy requirements applicable to the Steady State Parameters. For deployables, indication of correct end position and locking is sufficient.*

MRFR-011 *The monitoring of the mission critical Steady State Parameters shall be accessible to the S/C telemetry and shall be of the absolute type (i.e., not relative).*

4.4.3 Performance Requirements

4.4.3.1 Structural performances

MRPR-001 *The structural parts of the mechanisms shall meet the requirements specified in section 4.3.*

MRPR-006 *Stiffness requirements shall be established and demonstrated compatible with the structural and functional needs at system level for all the envisaged configurations of the mechanism.*

MRPR-011 *In addition to the limit/design loads specified in section 4.3 the structural parts of the mechanism shall withstand the loads induced by its operations.*

MRPR-016 *The following structural additional safety factors shall apply to the identified components:*

-Cables: 2.0
-Mechanical stops, shaft shoulders and recesses: 1.6

4.4.3.2 Thermal performances

MRPR-021 *The mechanisms shall satisfy the requirements specified under chapter 4.5, Thermal requirements.*

MRPR-026 *Motors shall withstand their stalling torque for a period of time of at least 24 hours under the maximum predicted operating voltage without degradation under the predicted worst case flight thermal environment. Qualification shall be performed with a 10° C margin.*

4.4.3.3 Functional performances

MRPR-031 *Global and local envelope volumes shall be defined for all moving parts within which they are free to move and shall be demonstrated at system level and at unit level without risk of mechanical interference.*

MRPR-036 *The tolerances of the Steady State Parameters shall be specified and demonstrated compatible with the functional needs.*

MRPR-041 *The parameters restricting the design of the Dynamics Variables shall be established and demonstrated to be compatible with the functional needs at system level. For instance: the total duration of an operation or a maximum acceleration.*

MRPR-046 *Redundancy concepts shall be selected to minimize single points of failure and to satisfy the mission reliability requirements.*

MRPR-051 *Where a single point of failure is identified and cannot be avoided, a rational for retention shall be proposed for approval by ESA.*

MRPR-056 *Where redundancy cannot be achieved by the provision of a complete mechanism, elements of mechanisms such as bearings, sensors, motors, actuators, switches and electronics etc. shall be redundant.*

MRPR-057 *Failure of one element shall not prevent any other element from performing within the tolerances of the functional requirements.*

4.4.3.4 Maintainability

MRPR-061 *The mechanisms shall not require any periodic maintenance for their entire ground storage period. Activations and maintenance after a period of storage is acceptable to verify the functional performances.*

MRPR-066 *The mechanisms shall not require any periodic maintenance for any Hibernation Phase of the mission.*

4.4.4 Verification Requirements

4.4.4.1 Modelling

The following Mathematical Models shall be established:

MRVM-001 *Structural models representative of the various configurations of the mechanisms according to the requirements specified in section 4.3.*

MRVM-006 *Thermal Mathematical Models, (TMM) representative of the various configurations of the mechanisms according to the requirements specified in section 4.5.*

MRVM-011 *Dynamics Models, (DM) representative of the various envisaged displacements and capable of describing the Kinematic and Dynamic Variables w.r.t. the time.*

MRVM-016 *The mathematical models shall be correlated through specific tests carried out at unit, subassembly or system level; the relevance of the level shall be substantiated.*

- *The Structural Model shall meet the correlation requirements specified in section 4.3.*
- *The TMM shall meet the correlation requirements specified in section 4.5.*
- *The DM shall be correlated taking into account the alterations of the performances induced by the ground environment.*

4.4.4.2 Analyses

MRVA-001 *The structural analysis shall demonstrate that:*

- *All parts withstand the predicted limit/design loads and the functional loads with a positive margin of safety as required in section 4.3. Attention is drawn to the Hertzian contact pressure of some specific parts, which are very sensitive to deformations originated by thermo-elastic effects, ageing, etc....*
- *The stiffness requirements are met with the margins specified in section 4.3.*

MRVA-006 *The thermal analysis shall demonstrate that all the parts stay within their design ranges for the worst operational cases. The case of the stalling torque/ 24 hours shall be identified separately.*

MRVA-011 *A budget (Static Budget of motorisation) corresponding to a static case (no relative acceleration and no relative velocity) shall compare the sum of the minimum driving forces/torques to the sum of the maximum resistive forces/torques.*
The comparison shall demonstrate that the motion can restart if stopped at any point of the displacement for the worst cases derived from the qualification thermal environment. The following factors specified in the reference document, RD-8; ECSS-E-30-00, issue 2, revision 1, Draft dated 10 December 1996 shall be used to initiate the design.

In order to derive the factored worst case quasi-static resistive torques (or forces), the components of resistance, considering worst case conditions, shall be multiplied by the following minimum uncertainty factors.

Component of resistance		*Initial Factor*	*See nota*
Inertia	*It (or If)*	*1.1*	
Spring	*S*	*1.2*	
Friction	*Fr*	*3.0*	*(1.5) #*
Hysteresis	*Hy*	*3.0*	*(1.5) #*
Others (harness)	*Ha*	*3.0*	*(1.5) #*
Adhesion	*Hd*	*3.0*	

Hence the minimum required actuation torque (or force) against worst case resistive torques (or forces) is defined by the equations:

Minimum required actuation torque

Tmin = 2.0(1.1 It + 1.2 S + 3 Fr +3 Hy + 3 Ha +3 Hd)*

Minimum required actuation force

Fmin = 2.0(1.1 If + 1.2 S + 3 Fr +3 Hy + 3 Ha +3 Hd)*

Where the factor 2.0 is the motorisation factor.

Nota: the specified uncertainty factors marked by # may be reduced to 1.5 providing that the worst case measured torque/force resistive components to which they refer are determined according to a test procedure approved by the customer and demonstrating adequacy of the uncertainty factor with respect to the dispersions of the resistive component functional performances.

MRVA-016 The static budget established for the flight model shall be substantiated by tests run on a sufficient number of different subassemblies. Test procedures are submitted to ESA approval. The uncertainty factors shall be re-assessed on the basis of the test results. The motorisation factor shall be greater than 2.0. It must be greater than 1.0 if one of two redundant driving forces designed to run nominally in parallel has failed.

MRVA-021 The Dynamical analysis shall demonstrate that the performance requirements identified in section 4.4.3.3 are met for the range of scatter of the physical parameters assumed for the static budget. As a minimum two cases shall be envisaged:

- *the maximum driving forces against the minimum components of resistance.*

- *the minimum driving forces against the maximum components of resistance.*

Each case assumes the qualification thermal environment leading to the predicted worst cases.
If required the analysis shall be repeated with the model of redundant items.

MRVA-026 The worst case operational loads shall be derived from this analysis, including the shocks at the end stops if any.

4.4.4.3 Testing

MRVT-01 The mechanisms shall be qualified on dedicated units which shall not be used for flight.

MRVT-06 Verification of the performances of the mechanism shall be possible on ground at unit and assembly level. Verification of the operation shall be possible at system level, use of special jigs or spacecraft attitude is acceptable..

MRVT-16 The lifetime of the mechanism shall be demonstrated/qualified by test in a configuration representative of the predicted worst case conditions of the flight model.

For the test demonstration, the number of nominal cycles predicted for the flight items shall be multiplied by the following factors:

Type/number of predicted cycles.	***Applicable factor for test***
Ground operations before flight	*4 (the minimum number is 10)*
In orbit predicted cycles from 1 up to 10	*10*
In orbit predicted cycles from 11 up to 1 000	*4 (the minimum number is 100)*
In orbit predicted cycles from 1 001 up to 100 000	*2 (the minimum number is 4 000)*
In orbit predicted cycles greater than 100 000	*1.25 (the minimum number is 200 000)*

During actuation, the full output cycle or the full revolution of the mechanism, or another worst case as agreed by the Agency, shall be applied. The number of cycles to be demonstrated by test shall be the accumulation of each case multiplied by its individual factor.

Attention is drawn to short amplitude movements which may degrade the lubricant locally.

Any element in a chain of actuation (motor, bearing, gear etc.) shall be compliant with the maximum number of cycles applicable to any of the remaining elements in the chain.

Lifetime of the critical mechanisms components is declared successful if the following is met:

- *No metal to metal contact identified in the I/F of solid lubricated surfaces.*
- *Functional performances met within tolerances specified in section 4.4.3.3 for the entire life test.*
- *No rupture or loss of functionality of any part.*
- *Stiffness requirements met for the entire life test.*
- *Amount and size of wear product compliant with the contamination requirements.*

4.4.5 Design Requirements

MRDE-001 Existing parts and components used in mechanisms shall be fully qualified for the intended application. The qualification procedures are subject to ESA approval.

MRDE-006 Sliding surfaces shall be avoided. If they cannot be avoided, lubricated hard surface coatings shall be applied to prevent fretting.

MRDE-011 Dry lubrication is preferred; use of liquid lubricant shall be fully justified.

MRDE-016 Liquid lubricated mechanisms shall be appropriately sealed to confine outgassing, creeping and possible sources of contamination.

MRDE-021 When the migration of liquid lubricants to sensitive equipments shall be avoided or when it may cause a change of the lubricant amount, anti-creep barriers shall be used.

MRDE-026 In order to verify the correct application of anti-creep barriers, UV-detectable anti-creep barrier coatings shall be used.

MRDE-031 The mechanisms shall operate in standard clean room environment without special precaution, with the exception of any 1g compensation if needed.

However, where it is demonstrated to be required, flushing with inert gas may be designed to protect parts of the mechanism which are sensitive to operation in air due to the presence of moisture, particles or other deleterious contamination.

MRDE-036 Latches, and other release devices must be resettable or replaceable for ground operation. They shall be capable of being operated manually for ground operation or for protection during handling and transportation.

MRDE-041 Emergency mechanical end stops shall be designed to prevent a moving part from protruding from the volume envelopes defined in section 4.4.3.3

MRDE-046 Relative sliding of contact surfaces shall be avoided to prevent degradation of the surface properties which may affect the mechanical accuracy or increase the friction.
This is not applicable to one shot devices..

MRDE-051 All ball bearings shall be pre-loaded.
Pre-loading shall prevent sliding at the bearing mounting interface.

MRDE-056 All gears shall be designed and sized according to the applicable document reference AD-30, ISO/DIS/6336.

MRDE-061 The mechanisms shall comply with the requirements specified at chapter 4.9 Power Supply and Conditioning Requirements and at chapter 4.13 Harness Requirements.

MRDE-066 Windings shall be designed to withstand a high voltage of 250 VAC (operating voltage up to 50 V) or 500 VAC (operating voltage up 100 V), applied between each other or between windings and the structure for a period of time of at least 2 minutes without causing disruptive charges.

MRDE-071 Torsional loads shall not be applied to electrical cables which change their configuration, e.g. wires around an hinge.

MRDE-076 All release devices (e.g. Pyrotechnics, Thermal Knives, Memory Metal and paraffin actuators) shall be redundant. Redundancy shall be provided by duplication up to and including the level of initiators, heating elements or equivalent for non pyrotechnic devices and its pertinent power supply.

MRDE-081 As a minimum, the pyrotechnics shall be qualified for properties and performances such as strength, shock impulse, redundancy, performance output, sealing and operation at temperature extremes under vacuum.

MRDE-086 The operation of release devices shall be compatible with the cleanliness requirements and shall not endanger the operation of the mechanism due to debris generation.
Suitable means of debris containment shall be included if necessary.

MRDE-091 The materials which are in contact with each other (e.g. during launch) and are to be separated later (e.g. for deployment) and parts which are sliding against each other shall be sufficiently different in composition, even if a lubrication is applied on their surfaces, to avoid the risk of cold-welding or fretting.

MRDE-093 All deployable appendages shall be positively locked in their end-position and confirmation shall be provided via telemetry that the required end-position and locking have been achieved at the end of deployment. Monitoring of intermediate positions during the deployment cycle is required if needed for corrective actions/commanding to overcome non-nominal situations.

4.5 Thermal Requirements

4.5.1. Definitions

TRDF-001 The units and equipments mounted onto the S/C shall be subdivided into two groups:

- *The Collectively Controlled, (C/C) units: their TRP temperature is monitored and controlled by the S/C.*
- *The Individually Controlled, (I/C) units: their TRP temperature is monitored and controlled by the S/C if they are non-operational. If operational, the units control by themselves their temperature, while the S/C monitors and controls their thermal environment. The thermal design of the I/C unit is the unit supplier responsibility and will be based on environmental and spacecraft interface data provided by the spacecraft Contractor.*

TRDF-006 Temperature ranges definition

TCS Design Temperature Range

The TCS design temperature range is specified at TRP and STP for the operating and non-operating mode of a unit. This temperature range represents the requirements for the TCS design activities. It shall be guarantied by the TCS.

TCS Predicted Temperature Range

The TCS predicted temperature range (operating, non-operating, switch-on) is a temperature range obtained from the relevant TCS calculated temperature range after addition of all possible uncertainties. The TCS predicted temperature range is based on nominal worst case considerations excluding failure cases.

The TCS predicted temperature range shall be enveloped by the relevant TCS design temperature range with a TCS design margin which shall be justified by the spacecraft Contractor.

Unit Acceptance Temperature Range

The unit acceptance test temperature range is identical to the TCS design temperature range. The acceptance temperature range is the extreme temperature range at which a unit shall be tested for a limited period of time. Fulfillment of all the functional performances (operational and non-operational) of the unit shall be demonstrated by test for the complete acceptance temperature range. This test is mandatory for all FM units prior to delivery to the spacecraft.

Unit Qualification Test Temperature range

The unit qualification test temperature range is the extension of the unit acceptance test temperature range by 10°C both ends. The qualification test temperature range is the extreme temperature range at which the unit shall be tested for a limited period of time to qualify its design. Fulfillment of all the functional performances (operational and non-operational) of the unit is mandatory for the complete acceptance range. Partial deviation from the performance requirements may be accepted within the item qualification test margins, provided they do not affect the interfaces to the S/C and they are reversible.

Unit Design Temperature Range

The unit design temperature range represents the requirements for the unit internal thermal design activities. The internal thermal design of a unit shall envelope the unit qualification test temperature range at TRP and STP for the operating and non-operating modes of this unit.

Margin Philosophy

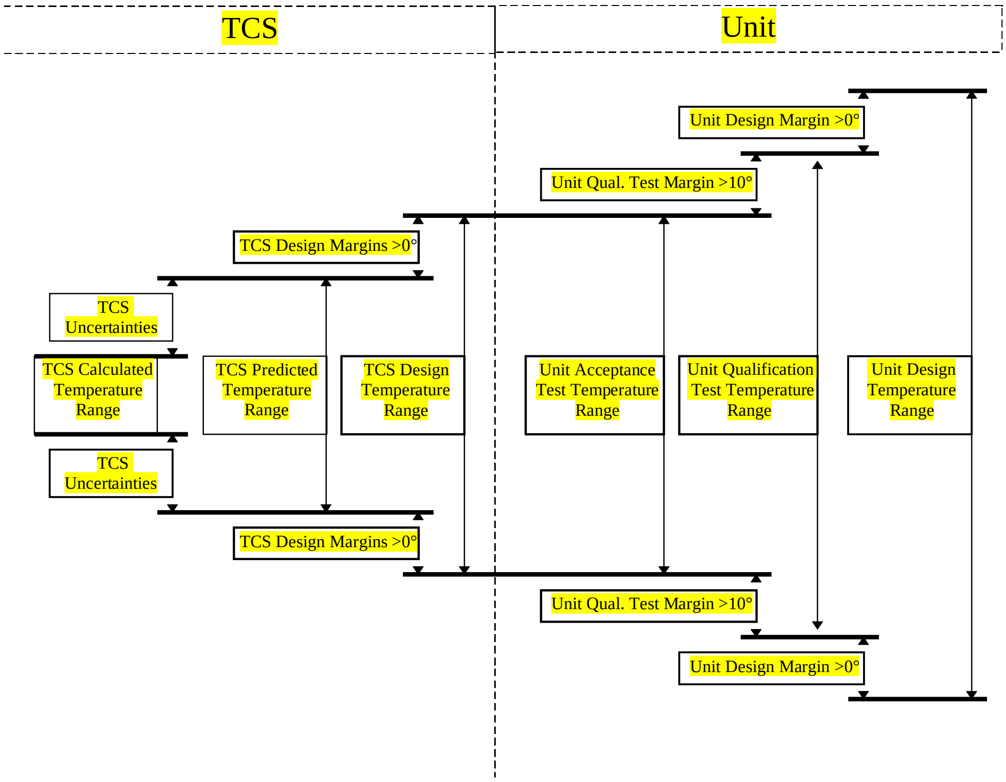
The various definitions defined above and the related margin philosophy are illustrated by the chart below.

Temperature reference point TRP

The TRP is a point located in/on the unit. The TRP shall be selected by the unit thermal designer in order to be the most representative of the unit temperature. The TRP shall be instrumented during the Qualification/Acceptance testing of the unit and during the TB/TV test at system level and for the flight.

System Interface Temperature Point STP

The STP is a point located at the mechanical interface between an I/C unit and the spacecraft on the spacecraft side. The STP shall be selected by the prime contractor in order to be the most representative of the temperature of the structural interface to the unit. It shall be instrumented during all the thermal test at system level and for the flight.



Radiative Sink Temperature R S T

The R S T is a virtual black body radiation temperature used to define the equivalent radiative load on a unit. This includes both the natural environment load (solar, planetary albedo and infrared fluxes) and the radiative exchanges with other items on the spacecraft.

4.5.2. Functional Requirements

TRFR-001 The TCS (Thermal Control Subsystem) shall monitor and control the TRP and STP temperatures for all phases of the mission with the exception of the TRP's of operational I/C units.

4.5.3. Performance Requirements

TRPR-001 The TCS shall ensure that the TRP and STP temperatures are kept within their TCS design ranges for all phases of the mission with the exception of the TRP's of operational I/C units.

TRPR-006 The TCS shall be protected against any single point of failure of any item of the chain of the temperature measurements by appropriate redundancy.

TRPR-011 The TCS shall be protected against the failure of any active device (e.g. heaters, louvers, blinds....) by appropriate redundancy of the complete chain.

TRPR-016 The TCS shall be tolerant to any single failure at system or unit level.

TRPR-021 Any structural device shall meet the requirements specified in section 4.3, Structural Requirements.

TRPR-026 Any active device featuring mechanisms shall meet the requirements specified in section 4.4, Mechanism Requirements.

4.5.4. Verification Requirements

4.5.4.1 Modelling

TRVM-001 Detailed Thermal Mathematical Models (DTMM) and Detailed Geometrical Mathematical Models (DGMM) shall be created for analytical predictions representative of all the phases of the mission including the ground tests.

TRVM-006 The models shall unambiguously identify the flight and the test monitoring points, (TRP and STP).

TRVM-011 Any thermal mathematical model shall be delivered as ESATAN file. Sub-modelling techniques shall be used down to a level agreed with the Agency.

TRVM-016 Any geometrical mathematical model (GMM) shall be created and delivered as ESARAD files. The ESARAD files shall consist as a minimum of the geometrical files and the ESATAN formatting file where the merging of the surfaces is described.

*TRVM-021 The DGMM's and the DTMM's shall be correlated against the environmental test results.
After correlation, the temperature differences between predictions and measurements shall be analysed for all the measurements points.*

TRVM-026 The thermal Mathematical Model of the test facility shall be compatible with the data measured during the environmental test.

*TRVM-031 A Reduced Thermal Mathematical Model (RTMM) of the entire S/C in launch configuration shall be derived from the DTMM for the integrated analyses with the launcher.
The format of this model shall meet the launcher requirements.*

4.5.4.2 Analysis

TRVA-001 The compliance to the thermal performance requirements shall be demonstrated by analysis for the nominal operational and non operational cases.

TRVA-006 A set of agreed failure cases shall be simulated for which positive TCS design margins shall be demonstrated for any STP or TRP. After correlation of the DTMM with test results, the TCS Design margins may be reduced down to zero for the agreed failure cases.

TRVA-011 TCS uncertainties shall be added to the TCS calculated temperature ranges when comparing the TCS predicted temperatures ranges with the TCS design ranges applicable to TRP's and STP's, see chart at 4.5.1, Definitions. They shall be substantiated by appropriate sensitivity analysis and agreed by ESA.

TRVA-016 The differences between the test measurements and the test simulations calculated by the correlated DTMM shall be taken into account for the final flight predictions.

TRVA-021 The differences between the results yielded by the RTMM and the DTMM running similar cases (transient and steady state) shall be taken into account for the final flight predictions.

4.5.4.3 Testing

TRVT-001 The thermal design shall be validated by a thermal balance test with solar simulation.

TRVT-006 The test cases shall cover the extreme environmental conditions envisaged for the complete mission and the most critical predicted thermal situations.

TRVT-011 *The S/C thermistors shall be continuously monitored during the test and may be used for the assessment of the stabilisation. If their accuracy is compatible with the need, they shall be used for the correlation.*
Additional measurement points shall be provided by test Thermo-Couples, mainly for complementing the flight measurement plan and monitoring local or general environmental data.

TRVT-016 *The environment induced by the test facility shall be continuously monitored during the test with the level of detail required by the complexity of the external surfaces of the S/C.*

TRVT-021 *The Thermal Vacuum test at system level shall be designed to bring all the S/C and P/L units to their worst predicted flight environment without exceeding their qualification range.*

TRVT-026 *For a C/C unit TV test the temperature of any point on the structural housing of the unit shall envelope the temperature ranges specified for the test.*

4.5.5. Design

TRDS-001 *The materials shall meet the requirements specified in section*

4.1.5 Material and processes.

TRDS-006 *Use of material with stable thermo-optical properties is required. Any deviation shall be adequately justified and the ageing law assumptions shall be substantiated.*

TRDS-011 *The TCS design shall allow easy access to the units and equipments.*
All parts belonging to the TCS shall stay within their allocated volume envelopes.
The MLI design shall ensure a minimum clearance of 20 mm between the MLI and any moving part. If the MLI is attached to a moving part, the clearance shall be of 35 mm at least.

TRDS-016 *The TCS shall 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.*

TRDS-026 *The design shall allow trimming to correct the temperature biases predicted after correlation of the thermal models.*

TRDS-031 The design shall comply with the EMC requirements specified in section 4.2.5.

TRDS-036 Venting of all the cavities including the interspacing of the MLI shall be designed to avoid damaging overpressure during the ascent. The method of venting shall prevent the release of contaminants toward the sensitive components of the P/L or of the S/C.

4.6 Attitude, Orbit Control and Measurement Requirements

4.6.1 General Requirements

AOCS-001 The AOCMS shall provide hardware and associated on-board software to acquire, control and measure the required spacecraft attitude during all phases of the mission, and to produce, control and monitor all the necessary reaction wheel momentum management manoeuvres and delta-V burns for the complete mission.

The attitude and orbit control thrusters are part of the Propulsion System, for which requirements are specified in section 4.8, but their operation is controlled by the AOCMS.

AOCS-005 The AOCMS shall provide during all mission phases:
- an autonomous capability of detecting (the need for autonomous isolation and re-configuration being mission phase dependent) any anomaly resulting in the loss of the nominal pointing attitude;
- an autonomous sun-pointing safe and hold mode initiated by the DMS, to ensure the spacecraft systems survival with respect to their thermal environment, and to guarantee adequate power generation.

AOCS-010 The AOCMS shall perform autonomous recovery of the high gain antenna earth pointing at preset times (e.g. at hibernation exits, or after a given duration following a scientific or operational event through a time-tagged entry in the mission time line) and upon ground command.

AOCS-015 The AOCMS shall perform, under the supervision of the on-board data management subsystem, autonomous attitude and orbit adjustment manoeuvres required by the mission operations during periods when ground contact is not available or ground response times inadequate.

AOCS-020 The AOCMS shall accept ground or on-board telecommands to perform attitude and orbit manoeuvres. It shall also provide attitude measurement data via telemetry to the ground for attitude reconstitution. In addition it shall provide information via *telemetry to ground to allow diagnosis of on-board failures.*

AOCS-025 *The AOCMS shall provide offset pointing capability in closed loop attitude control, e.g. in order to allow to compensate for global experiment misalignment.*

AOCS-030 *The AOCMS shall make use of a dedicated star tracker which has the capability to provide the inertial orientation of its reference frame by tracking multiple stars and using an on-board star catalogue.*

AOCS-035 *The AOCMS shall make use of a three axis set of accelerometers to guarantee that delta-V manoeuvres with specified magnitude accuracy are generated.*

To achieve the required magnitude accuracy it may be necessary to re-calibrate the accelerometers in flight.

4.6.2 Functional Requirements for different Mission Phases

In addition to the general requirements, specific system and pointing requirements are defined for each individual mission phase, which drive the overall Rosetta AOCMS functional requirements, taking into account the applicable mission constraints.

4.6.2.1 Commissioning Phase

AOCS-040 *Upon separation from the Ariane 5 launcher interface in a coarse 3-axis pointing mode, the spacecraft AOCMS shall damp out the residual angular rates and acquire the sun along a specific spacecraft axis within 15 minutes after exit from eclipse and within a cone compatible with the power and thermal requirements for the subsequent deployment of solar arrays and antenna booms.*

The Ariane 5 User's Manual, specifies the maximum residual rates imparted by the Ariane 5 separation system (@ 99% confidence level):

- less than 1° /s about the transverse axes
- less then 0.6° /s about the longitudinal axis

The Ariane 5 upper stage guidance system will orient the specific spacecraft axis within 1 deg (half cone) from the spacecraft-sun line.

AOCS-045 *After sun acquisition, the AOCMS shall provide stable 3-axes attitude control before and after the deployment of the solar arrays and during the deployment of the high gain antenna.*

It is anticipated that during solar array deployment possibly dangerous interactions between thruster operations and the relative movement of the solar array panels shall be avoided.

AOCS-050 *The AOCMS shall then maintain a sun-pointing attitude and upon initialisation from the ground perform star acquisition and acquire full three axis inertially stabilised attitude control with a predetermined roll angle about the spacecraft-sun line.*

AOCS-055 *The AOCMS shall allow a first orbit correction manoeuvre to be performed within 3 days after separation in order to compensate for the launcher upper stage injection errors, followed by orbit trim manoeuvres if necessary to reach the nominal cruise phase orbit.*

AOCS-060 *The commissioning operations will typically last for three months, during which the AOCMS shall be capable of acquiring and maintaining any attitude as required by the sequence of mission operations, such as commissioning of the spacecraft and payload systems, early validation of specific mission modes (e.g. hibernation), tracking for orbit determination, early trajectory maintenance and correction manoeuvres, health checks, etc..*

In this phase it is anticipated that both the solar arrays and the HGA shall be pointing in agreement with the pointing requirements in section 4.1.3.

The pointing accuracy in this phase is expected to be considerably less stringent than during scientific observation phases and compatible with the pointing requirements for solar array and HGA pointing and with the requirements of specific commissioning activities.

AOCS-063 *Thrusters shall not be used for attitude control during ground tracking intervals in order to avoid degrading the orbit determination accuracy within the reaction wheel capability defined by AOCS-212.*

It is anticipated that for most of this phase attitude control will be performed using reaction wheels in order to provide a quiescent environment for the commissioning activities.

AOCS-065 The AOCMS shall operate redundant reaction wheel configurations for attitude control in the commissioning phase, and in the active mode of the cruise phase. Reaction wheels shall also provide the angular momentum capability and torque level for payload boresight attitude control during asteroid flybys, for spacecraft manoeuvring in preparation of delta-V manoeuvres, and spacecraft attitude control during comet and asteroid detection and comet nucleus observation.

AOCS-070 The AOCMS shall be able to reacquire sun pointing attitude and realign solar arrays to the sun from any initial orientation after initial sun acquisition, solar array deployment and other commissioning phase activities.

4.6.2.2 Cruise Phase - Active mode

The Rosetta mission is expected to contain during the cruise phase several periods when the spacecraft will be active with the preparation for typically three months of major mission phases and with their finalisation for about one month. During these active periods, the main spacecraft operations will include:

- ground tracking for orbit determination;
- in flight calibration of scientific or spacecraft units;
- small orbit correction manoeuvres;
- scientific data relay to download on-board memory content;

It is anticipated that in this phase both solar arrays and HGA shall be pointing in agreement with the pointing requirements of section 4.1.3. The spacecraft pointing accuracy and stability shall be compatible with the requirements for solar array and HGA pointing and the instrument calibrations envisaged.

As in the commissioning phase no thrusters shall be operational for attitude control during ground tracking.

AOCMS requirements for this phase are covered by those of the commissioning phase.

4.6.2.3 Cruise Phase - Hibernation mode

The Rosetta mission is expected during cruise phase to contain several periods with the spacecraft in passive hibernation mode in order to reduce the workload of the ground segment and to save on spacecraft components which degrade with usage.

AOCS-075 *The spacecraft AOCMS concept for hibernation shall be designed to be compliant with the following constraints:*

- *no ground contact during long time intervals varying from several months to several years;*
- *spacecraft power consumption should be minimised;*
- *coarse sun-pointing within power and thermal constraints;*
- *minimum number of active AOCMS equipments; especially those which degrade with usage (e.g. gyros, reaction wheels), or those which are believed to present a non-negligible risk* with respect to very long ground contact outages (e.g. propulsion system prone to leakages due to open thruster failures) and could endanger the spacecraft survival or would lead to prohibitively complex and costly autonomy requirements to overcome this risk;
- *the AOCMS shall automatically enter the hibernation mode upon command;*
- *The AOCMS shall perform automatic recovery of the high gain antenna earth pointing at wake-up times;*

4.6.2.4 Planet Gravity Assist Phase

AOCS-080 *The AOCMS shall provide the spacecraft with the capability to produce delta-V manoeuvres between a few m/s and a few hundred m/s during the Mars and Earth gravity assist phases.*

These delta-V manoeuvres shall satisfy the pointing and magnitude requirements specified in section 4.1.3.

AOCS-085 *The AOCMS shall allow these manoeuvres to be performed automatically, time-tag commanded but otherwise without ground support.*

4.6.2.5 Comet and Asteroid Detection Phases

AOCS-095 *The AOCMS shall enable the spacecraft to perform comet and asteroid detection in a three axis inertially stabilised attitude satisfying the absolute and relative pointing requirements specified in section 4.1.3.*

These specifications guarantee that sequences of pictures can be taken with the navigation camera which overlap sufficiently and such that each picture is sufficiently focussed.

AOCS-097 *Ground communication via the HGA and continuous solar power generation shall be maintained during the comet and asteroid detection phases.*

4.6.2.6 Asteroid Fly-by Phases

Absolute and relative pointing requirements applicable of this phase are specified in section 4.1.3.

Continuous tracking of the asteroid by the scientific payload will start several hours before the closest encounter, with absolute and relative pointing requirements as specified in section 4.1.3.

Fine pointing of the HGA to the Earth simultaneously with asteroid tracking by the science instruments is not mandatory, all the science data being stored on-board and down linked when ground contact is restored.

AOCS-100 *In case of AOCMS anomaly during fly-by, the AOCMS shall switch to a safe and hold mode to guarantee adequate thermal control and power generation and wait for intervention.*

AOCS-105 *The AOCMS shall, after asteroid flyby, enable the spacecraft to reacquire the communication link with the ground and adequate sun pointing for power generation.*

4.6.2.7 Comet Rendezvous Phase

This phase includes the execution of a sequence of delta-V manoeuvres, in total more than 1000m/s, to insert the spacecraft into a cometary orbit. This implies that the overall manoeuvre duration will add up to several hours. The manoeuvre strategy is not yet defined in detail and shall be derived from mission analysis studies to assess its impact on the LIV budget. Autonomous re-configuration of the spacecraft AOCMS in case of anomaly may not be mandatory since exact timing of the delta-V manoeuvres is not as critical as for gravity assist manoeuvres and reoptimisation of the subsequent manoeuvres after an anomaly may be beneficial.

This phase starts after the detection of the comet nucleus with the Navigation Camera, for which pointing specifications have been given in section 4.7.
Then the spacecraft navigation camera or imaging system will be used to improve the characterisation of the comet nucleus (kinematics, shape, mass, optical properties), and also to obtain improved comet orbit determination accuracy. Delta-V manoeuvres shall be performed to inject the spacecraft into an orbit around the comet nucleus where all payload instruments will be used for remote sensing. The pointing requirements for these delta-V manoeuvres and for the comet detection and observation activities have been specified in section 4.1.3.

AOCS-110 The AOCMS shall provide capability to produce a sequence of sizable delta-V manoeuvres totalling about 1200m/s to annihilate the relative velocity with respect to the comet nucleus and to inject the spacecraft into an orbit around the comet.

AOCS-115 The AOCMS shall minimise generation of perturbation forces in the weak gravity field of the comet by using reaction wheels for attitude control and pure torques for momentum desaturation, except during orbit correction and maintenance manoeuvres.

AOCS-120 The spacecraft AOCMS shall be compatible with the dust environment around the comet nucleus (especially at low altitudes).

4.6.2.8 Comet Observation Phase

During this phase it is the intention to map at least 80% of the comet nucleus surface within 40 days. Moreover experiments should be able to observe dark parts of the nucleus and slew between interesting points of observation.

AOCS-125 The AOCMS shall orient the payload line of sight in any inertially fixed direction, compatible with maintaining a communication link with the ground via the HGA and providing power from the solar arrays.

AOCS-130 The AOCMS shall be capable of slewing the experiments line of sight, based on ground update of an on-board path control algorithm.

AOCS-135 The AOCMS shall provide the capability to adjust the spacecraft orbit around the comet compatible with requirements from the scientific payload and compatible with spacecraft survival.

4.6.2.9 Probe Ejection Phase

This phase requires the AOCMS to acquire a specific orientation and velocity for the SSP ejection with adequate accuracy and stability, which are specified in section 4.1.3.

AOCS-140 *The AOCMS shall satisfy the following constraints during the probe ejection phase:*

- any attitude shall be allowed for the SSP ejection while ensuring the conditions specified in PINT-008.

- the AOCMS shall allow recovery of the communication link with the SSP and the ground within 30 minutes.

AOCS-145 *After SSP ejection the AOCMS shall manoeuvre the spacecraft into a safe data relay orbit and switch to an attitude or attitude profile compatible with reception of the data link from the descending and landing SSP for 3 hours during the first 24 hours from Lander touchdown. Thereafter visibility shall be ensured for 30 min. every 16 hours for the remaining 96 hours.*

The data relay shall make use of the SSP receive antenna when the view of the SSP is unobstructed or of the CONCERT payload antenna in the alternate situation.

4.6.3 Performance Requirements

AOCS-150 *The AOCMS shall be compliant with the absolute and relative pointing and attitude measurement requirements specified in section 4.1.3.*

4.6.4 Design Requirements

AOCS-155 *The AOCMS shall be fully functional at start up after separation.*

AOCS-160 The attitude control shall be a closed loop design for all three axes in all mission phases, possibly with the exception of the hibernation mode.

It is advised for the design of the AOCMS controllers to consider using Multi-Input, Multi-Output (MIMO) digital design techniques to account for the multi body interactions and to minimise the sampling frequency.

AOCS-165 *Orbit control shall be open loop under ground command by up linking appropriate command sequences.*

AOCS-170 *All attitude and orbit control and measurement functions shall be redundant.*

AOCS-175 *The AOCMS software shall be reprogrammable in flight.*

AOCS-180 *Major design parameters used in the AOCMS on-board software shall be adjustable by ground command. Telecommanding of control parameters shall be performed such that a specific parameter or a set of associated parameters is updated as a block.*

AOCS-185 *The AOCMS shall transmit via telemetry unambiguous status information of all command and programme controlled variables, modes and of all parameters required for subsystem monitoring and evaluation and for reconstitution on ground of the attitude and attitude and orbit control manoeuvres.*

AOCS-190 *The AOCMS shall contain a redundant safe and hold mode to recover coarse sun pointing from any initial attitude at any time in the mission, to guarantee uninterrupted adequate solar power supply and thermal control.*

AOCS-195 *The operation of the safe and hold mode shall, once started, not be interrupted by an external command; recovery from this mode shall only be initiated after the safe and hold mode has reacquired stable sun pointing.*

AOCS-200 *The AOCMS time shall be the spacecraft on-board time.*

AOCS-205 *The implementation of mechanical gyroscopes shall be avoided. Non-mechanical gyroscopes shall only be implemented if their use is deemed essential to achieve performance or to avoid complexity during some mission phases or modes.*

AOCS-210 *When gyroscopes are implemented, gyroless operations shall be implemented as backup, possibly with acceptable performance degradation or increased complexity.*

AOCS-212 *4 Reaction wheels of ± 40Nms capacity each used for attitude control shall be configured to guarantee:*
- attitude control of the payload line of sight to follow the asteroid during the baseline fly-by (with-out redundancy),
- attitude control during comet observation phases with any 3 wheels operating.

4.6.5 AOCMS Autonomy Requirements

AOCS-215 *All software based AOCMS functionalities shall be executable in a single dedicated microprocessor.*

AOCS-220 *The AOCMS software shall provide automatic closed loop 3-axis spacecraft attitude stabilisation, including automatic acquisition of sensor data, automatic attitude determination and automatic commanding of the attitude control actuators.*

AOCS-225 *The AOCMS shall provide a hibernation mode, possibly spin stabilised, and time tagged or ground initiated transition into and out of this mode.*

AOCS-230 *The AOCMS shall provide hot redundancy of essential functions during mission critical manoeuvres.*

AOCS-235 *The AOCMS shall allow for DMS initiated SSP ejection, subsequent spacecraft attitude and orbit manoeuvring and recovery of the communication links with the SSP and ground.*

4.7 Optical Navigation Camera Requirements

The baseline is to carry a science camera system produced by the science community as part of the scientific payload and a separate camera system to perform relative navigation.

The science cameras include a NAC and a WAC with filter wheels, reflective optics, a mechanical shutter system, CCD's with 2048x2048 pixels, a 14 bit dynamic range etc.

The navigation camera shall be built with maximum reliability in view i.e., no mechanical shutter, no filter wheels etc. Analysis shall be made to show whether the navigation requirements can be met with a single camera, possibly an average FOV, visual wave length only, a CCD with 1024x1024 pixels etc. but with sufficient dynamic range to cope with detection of a faint object and imaging of the nucleus near the comet nucleus in perihelion.

Navigation requirements

An optical camera system is required for comet nucleus detection and approach navigation, for comet nucleus characterisation and for near nucleus navigation. It may assist in asteroid detection and approach to enable control over the fly-by distance and illumination angle during the close fly-by and in planetary swing-bys.

NAVI-001 *The camera for optical navigation shall provide positional data of the payload bore-sight axis and orientation about it on the nucleus during data take periods of payload instruments.*

NAVI-002 *A model of the nucleus shape, with a latitude and longitude grid, shall be produced as positioning interface between the navigation team and the scientists for definition of areas for observation or landing sites for the SSP. The spacecraft shall support this activity which is performed on ground.*

NAVI-003

NAVI-004 *The navigation camera shall have a detection capability equivalent to a visual magnitude of 11. The required integration time of the navigation camera at the detection distance shall be commensurate with the spacecraft pointing stability.*

NAVI-005 *The camera shall be designed to allow to detect target movements of less than one pixel against the back-ground stars.*

NAVI-006 *Once the comet nucleus is detected images taken at less than 12 hr intervals shall be telemetered to the navigation team for far approach navigation. The accuracy of the images shall be better than 0.03 deg (1 sigma) in an inertial reference frame. The dynamic range of the camera shall allow simultaneous observation of reference stars.*

NAVI-007 *At a distance of about 300 nucleus radii close approach navigation starts with the estimation of the nucleus kinematics and the definition of a coarse shape model.*

NAVI-008 *The resolution of the navigation camera shall be sufficient to observe at least 5 landmarks. The number of images required during the close approach phase is at least 1 every 4 hrs. Refer to chapter 3.1.4 for the comet nucleus model.*

NAVI-009 *During the global mapping phase the shape model is refined and gravitational parameters defined. The navigation camera shall be capable to distinguish between at least 5 landmarks. During the global mapping phase the number of navigation images to be down linked shall be maximised.*

NAVI-010 *During the close observation phase, the navigation camera shall be capable of detecting objects on the surface of the cometary nucleus which are commensurable with the Rosetta Lander dimensions.*

NAVI-011 *The cometocentric latitude/longitude grid of the shape model shall have an accuracy of 0.05 deg (3 sigma) on the surface to allow the exchange of positioning data with the scientists.*

NAVI-012 *The number of navigation camera images during the nucleus escort phase is typically 4 per day, subject to pericenter height and comet nucleus gravitational field asymmetry.*

NAVI-013 *The navigation camera system as defined above shall also be used for asteroid detection, approach navigation and close fly-by tracking of the asteroid.*

NAVI-020 *The navigation camera system shall be made available to the scientists for photometric calibration for a duration of 2 months.*

NAVI-021 *All parameters which are required for emulation of the camera shall be provided.*

4.8 Propulsion Requirements

4.8.1 Propulsion System Requirements

PROP-001 *Deleted.*

PROP-010 *The propulsion system shall contain a set of Attitude and Orbit Control thrusters.*

The spacecraft shall carry a set of low level thrusters (10-20 N) to provide for orbit control and fly-by manoeuvres, for attitude and spin rate control wheel off loading and to support the rendezvous manoeuvre sequence.

PROP-015 *The propulsion system shall provide the capability to perform major orbit manoeuvres with a minimum thrust level of nominal 40 N (i.e., 4 thrusters of 10 N); and it shall guarantee full redundancy for these manoeuvres.*

Major orbit manoeuvres are those to complement gravity assist manoeuvres and to perform comet rendezvous.

Thrusters arrangement on the spacecraft.

PROP-020 *Pure torque thrusters for attitude control and wheel off-loading shall have minimum residual forces.*

PROP-025 *Pure force thrusters for orbit manoeuvres shall have minimum residual torques.*

PROP-030 *The characteristics of the thrusters and their accommodation on the spacecraft shall not cause any deleterious effects on either the spacecraft or experiment payload during operation. Thruster nozzles in line of sight to experiments and radiators shall be shielded.*

PROP-035 *Thruster operations before solar array deployment shall be possible to detumble the spacecraft to allow proper S/A deployment and sun pointing.*

PROP-040 *The performance of the propulsion system in terms of total impulse and margin shall satisfy the requirements imposed by the ROSETTA mission, the trajectory analysis and the overall system requirements.*

PROP-045 *The propellant feeding to supply the AOCMS thrusters with propellants shall be performed in pressure blow down mode for the entire mission. Major orbit manoeuvres may be performed in constant pressure mode.*

Growth Capability

PROP-050 *The tank mass margin allocated during the subsystem design phase shall be used in the design of the tank to optimize the tank volume.*

The intention of the requirement is to maximise the ullage volume and thereby to allow a larger fuel load, if later in the program margins proof to be insufficient.

Single Point Failure

PROP-055 *The design of the Propulsion System shall be such that a single component/part failure cannot, as far as practicable, cause the failure of functions which are vital for mission success.*

Single point failures shall be identified via Failure Mode Effects and Criticality Analyses and, for those which cannot be avoided with reasonable effort, a Rationale for Retention shall be prepared which shall be subject to approval by ESA.

4.8.2 Design Requirement

Plume Impingement and Contamination

PROP-060 *The thruster directions and locations shall be selected to minimize plume impingement on the spacecraft, contamination of sensitive equipment and the generation of disturbance torques and forces.*

Thruster Redundancy

PROP-065 *The propulsion system shall contain redundant Attitude and Orbit Control thrusters. Each thruster shall be operated with dual valves at which the upstream valves takes over the conventional latch valve function.*

Pipework and Tank Layout

PROP-070 *The layout of the Propulsion System pipework and the design of the propellant tanks shall take into consideration the spacecraft three axis and the spin stabilised modes to operate properly the thruster during these conditions. The simple draining of the pipework and the tanks during spacecraft system tests shall be a design goal.*

Fuel Depletion

PROP-075 *Fuel depletion management shall be such that the requirements on maximum wobble angle are met during all mission phases. The layout of the propulsion system and the arrangement of the tanks shall ensure a symmetrical depletion of fuel and oxidiser during all thruster firings so that a shift of the COG position is minimized.*

Fuel Quantity Determination

PROP-080 *Adequate means for determination of remaining propellant quantities shall be provided. It shall be possible to achieve an accuracy of better than ± 8 kg at any time.*

Component Design

PROP-090 *Components (e.g. valves, regulators) using sliding surfaces for their actuation and operation as well as components with bellows inside the fluid containing part shall not be allowed. For those which cannot be avoided with reasonable effort, a rationale for retention shall be prepared which shall be subject* *to approval by ESA.*

PROP-091 *All materials used in contact with propellant/simulant propellants shall be mutually compatible. All materials selected shall, in particular w.r.t. the mission duration, neither degrade or reduce functionality of components nor degrade propellants.*

PROP-093 *The thruster control valve design shall be such that the valve closes in case of failure.*

Pyrovalve Design

PROP-095 *Pyrovalves shall not allow the actuator gas from flowing into the propellant or gas feed line passages.*

Magnetic Compensation

PROP-100 *Magnetic compensation of thruster and feed line valves shall be foreseen.*

Propulsion System Thermal Control

PROP-105 *The Propulsion System shall have thermal control capability to prevent freezing of propellant.*

PROP-110 *The Propulsion System shall have thermal control capability to prevent condensation of propellants in the Pressurant gas control assembly / circuitry upstream of the non return valves.*

PROP-115 *The Propulsion System shall have thermal control capability to prevent condensation of propellants outside the reach of the propellant management device (PMD) within the tanks.*

Propulsion System Structural Design

PROP-120 *The Propulsion System structural design shall take into account the requirements as specified in Section 4.3 (Structures).*

Safety Requirements for Pressurised Fluid Systems

PROP-130 *For on-board pressurised fluid systems the requirements as specified in the CSG Safety Regulations apply.*

4.8.3 Propulsion System AIV Requirements

Propulsion System Test Provisions

PROP-135 *Fill and drain valve (and test port) locations shall be compatible with all requirements for testing and for loading and unloading of propellant and pressurant gas. Interfaces shall be provided through the spacecraft skin connector for providing critical value states (pressure, temperatures etc.) and shall allow valve operation independent of spacecraft power.*

Pyro Valve Accessibility

PROP-140 *The design of the spacecraft and the location of pyrovalves shall provide an easy access to the pyrovalves for initiator and booster installation during the launch preparation.*

Simulant Propellants

PROP-145 *Suitable simulant propellant shall be selected which shall be as far as practicable density representative.*

NOTE: The use of Trichlorotrifluoroethane (e.g. FREON) is not allowed.

4.9 Power Supply, Storage and Conditioning Requirements

4.9.1 General Requirements

POWR-001 *The power subsystem shall provide all power required by the ROSETTA spacecraft for all pre-launch, launch and mission modes through the entire duration of the mission.*

The power shall be provided by means of both solar cell array and batteries.

POWR-002 *The solar cell array shall be sized with 15% margin for the worst case power situation and provide power up to the end of mission.*

POWR-003 *The power subsystem shall be designed in accordance to the ESA POWER STANDARD PSS-02-10, if not specifically stated here otherwise.*

POWR-004 *The subsystem shall have the capability of a predefined automatic start up after a complete loss of all main bus power.*

POWR-005 *The subsystem shall provide all resources needed for the operation of pyrotechnical devices for experiments as well as for spacecraft functions .*

4.9.2 Functional Requirements

POWR-006 *The subsystem shall condition, control, store and distribute electrical power on the ROSETTA spacecraft.*

POWR-007 *The subsystem shall provide adequate status monitoring and telecommand interfaces necessary to operate the subsystem and permit evaluation of its performance.*

POWR-008 *Where not specified in detail the subsystem shall provide adequate failure tolerance and protection circuitry to avoid failure propagation and to ensure recovery from any malfunction within the subsystem and/or load failure.*

POWR-009 *There shall be sufficient telemetry parameters assigned such that the power available and demanded can be established.*

POWR-010 *Redundancy and protection circuitry shall be provided to avoid failure propagation and enable recovery to specification requirements from any malfunction within the subsystem and/or loads.*

POWR-011 The subsystem shall provide <u>safe</u> 'keep-alive' lines to users in subsystems and experiments.

POWR-012 The subsystem shall condition, control and distribute the power required for the pyrotechnic devices in subsystems and experiments.

POWR-013 The power subsystem shall contain all electronics necessary:

a) to provide electrical power from the solar cell generator and/or batteries to all users,
b) to charge, discharge and recondition all batteries
c) to give the capability for automatic and commanded control of the operation of the subsystem

It shall also include all power switching and protection electronics for the experiments and other spacecraft subsystem users.

POWR-015 The subsystem shall reliably ensure the pyrotechnic initiated change of state at the mechanism output interface.

<u>4.9.3</u> Performance Requirements

4.9.3.1 Power Conditioning and Distribution

POWR-016 The subsystem shall be capable of operating continuously under all operational conditions of the mission. No damage or degradation shall result from intermittent or cycled operation.

POWR-017 The design of all electronic circuits shall be such as to provide appropriate control loop stability margins.

POWR-018 The power conditioning shall be designed such that a regulated 28 Volt main bus is provided to all users.

POWR-019 The subsystem shall be designed such:

a) that in all operating modes where the power available from the solar cell generator exceeds the main bus and battery charge demand the surplus electrical energy is left in the solar arrays.

b) that large circulating currents between the solar cell generator and the S/C are avoided.

c) that in all operating modes where the power demanded from the main bus exceeds the available power from the solar cell generator, the battery charging will be stopped and a maximum power point tracker (MPPT) will operate the solar cell generator in its maximum power point automatically.

d) that in all operating modes where the power available from the solar cell generator is still insufficient to satisfy the load demand, battery discharge regulators will provide the required electrical power from batteries automatically.

The above functions shall be designed into one control loop covering all three domains.

The design of the maximum power point tracker (MPPT) shall be made using simple, reliable circuitry.

4.9.3.2 Main Bus

POWR-020 *The main bus voltage regulation requirements shall be in accordance with the requirements of PSS-02-10. The voltage regulation accuracy shall be ground testable via a unit connector.*

POWR-021 *The main bus impedance shall be in accordance with the requirements of PSS-02-10. The main bus impedance shall be ground testable via a unit connector.*

POWR-022 *Bus transients due to interdomain operation shall be in accordance with requirements of PSS-02-10. The transients shall be testable on ground via a unit connector.*

POWR-023 *Transient voltage excursions due to step load changes shall not interfere with the operation of other subsystems or experiments.*

POWR-024 *No single component failure shall cause an over-voltage or permit short circuit on the main bus.*

POWR-025 *Full protection against short circuit or overload at all outputs shall be provided by limiting the maximum current in any of those lines. Lines in which overload has occurred shall be switchable. The use of fuses must be avoided.*

POWR-026 *No undesirable operating mode shall occur in the event of a failure in other subsystems (including harness) or in one or more experiments.*

4.9.3.3 Batteries

POWR-027 *The batteries shall be sized to cover all nominal spacecraft demand during the entire duration of the mission in cases of insufficient or no solar generator power. The batteries shall supplement the solar generator power for peak power requirements (e.g. power surges, transients etc.) in nominal orbit, if necessary.*

POWR-028 *The batteries and their Battery Regulator Unit (BRU) shall be functionally one failure tolerant up to the input of the Power Control Unit (PCU) including failures in power transmission elements, i.e., connectors, harness etc .*
A multi modular battery plus BRU package design shall be considered in order to achieve maximum failure tolerance.

POWR-029 *A maximum Depth-of-Discharge (DoD) of 75% in nominal cases and 90% in failure cases shall not be exceeded.*

POWR-030 *Adequate means and telemetry shall be provided for the ground to be able to determine the state of charge of the batteries throughout all mission phases to an accuracy better than 10%.*

POWR-031 *Protection against excessive over-charge, over-discharge, and over-heating shall be provided.*

POWR-032 *Ground commandable individual cell reconditioning via cell dedicated series connected diode, resistor networks shall be provided for all batteries.This requirement shall not be applicable to Li-Ion batteries comprising self compensating cell matrix arrangements. In case Li-Ion batteries are used, discharge must be allowed by telecommand.*

POWR-033 *The thermal environment control of the batteries shall be optimized in order to guarantee maximum performance over the entire mission duration. Battery temperature monitoring shall be provided for all batteries.*

4.9.3.4 Solar cell generator

POWR-034 *The solar cell generator shall be designed such, that its single string performances allow the efficient use of a maximum power point tracker (MPPT) during the power critical phases of the ROSETTA mission.*

POWR-035 The solar cell generator performance for each major mission phase shall be predicted. Performance figures shall include all effects which have an impact on this performance (e.g. solar attitude, sun intensity, radiation, micro meteorites, MPPT tracking error etc) and shall be based on accepted cell degradation figures and actual cell performance measurements.

POWR- 036 The solar cell generator shall be designed such that it is two (2) string failure tolerant.

For power transmission elements such as connectors, harness and (if used) Solar Array Drive Mechanism (SADM) slip ring usage, up to the input of the PCU a two (2) failure tolerance requirements shall be fulfilled as far as practical.

POWR-037 Solar cells used in the array shall meet all requirements of the ESA Specification PSS-01-604 for silicon cells and the agreed changes for GaAs cells.

POWR-038 All solar array cell strings shall have individual blocking diodes and shunt diodes where required.

4.9.3.5 Pyrotechnics

POWR-039 The actuation of pyrotechnical devices shall be possible while in any state within the operational envelope and having been subjected to all defined environments encountered up to the time of activation

POWR-040 The pyrotechnical subsystem shall ensure reliable maintenance of the original states when providing it with any but the correct stimuli up to the defined limits.

POWR-041 The output and properties of the Pyrotechnic Subsystem shall be compatible with the requirements of mission and all other subsystems.

POWR-042 The interface to the experimenter pyrotechnic initiators shall comply with the pyrotechnic interface requirements as stated in the EID part A and B.

POWR-043 Redundancy shall be provided for each function by duplication up to at least the initiators.

POWR-044 High reliability and safety shall be provided by the use of approved practices including the use of initiators meeting the applicable technical and PA requirements, the shielding of all wires and electronics, and the use of the correct test equipment and procedures.

POWR-045 The requirements of the launcher, transportation and all test facilities shall be met by the design of, and procedure with, the subsystem. In particular the requirements of ARIANE and the Centre Spatial Guyannais shall be met.

4.10 Telecommunication Requirements

4.10.1 General Requirements

TTCS-001 The TT&C System shall receive and demodulate telecommands, modulate and transmit the telemetry, and transpond the ranging signal.

TTCS-005 The TT&C System is required to interface with the ESA ground segment and the NASA Deep Space Network (DSN). The applicable requirements for this interface are defined in the Space/Ground Interface Control Document (SGICD) RO-ESC-IF-5002.

TTCS-010 The links with the ground shall be established using: S-band and X-band for the uplink and S-band and X-band for the downlink.

TCCS-015 Commanding *via X-band uplink is the nominal procedure and will never be simultaneous with S-band uplinking.*

TTCS-020 The TT&C system shall be compatible with the following ESA standards:
- Telemetry Channel Coding Standard (ESA PSS-04-103)
- Ranging Standard (ESA PSS-04-104)
- Radio Frequency and Modulation Standard (ESA PSS-04-105)
- Packet Telemetry Standard (ESA PSS-04-106)
- Packet Telecommand Standard (ESA PSS-04-107)

TTCS-035 The spacecraft shall have no requirements for telemetry operation during the launch phase.

4.10.2 Functional Requirements

TTCS-050 The TT&C system shall be capable of simultaneously handling of telemetry, ranging and commands.

TTCS-055 The TT&C System shall accept S-band and X-band up-link signals and provide a demodulated telecommand signal to the Data Management System (DMS) for further processing.

TTCS-057 *Simultaneous transmission of S-band and X-band shall be possible without any interference.*

TTCS-060 *Hot redundancy shall be provided for the receive function and cold redundancy for the transmit function.*

TTCS-065 *The receiver output shall be cross coupled with the command decoders*

TTCS-070 *The configuration shall be such that both receivers can receive and both decoders can decode simultaneously.*

TTCS-072 *The receiver shall provide a signal which indicates the level of the uplink. This signal shall be such that it can be used to control the HGA pointing to earth.*

TTCS-075 *The transmitters shall be able to receive the telemetry stream from both parts of the redundant DMS.*

TTCS-080 *Each transponder shall, upon command, be able to operate either in non-coherent or in coherent mode, the latter depending on the lock status of the receiver.*

TTCS-085 *When switching between antenna's it shall not be necessary to switch off the transmitter.*

TTCS-090 *The transmit frequency may be derived from an optional external Ultra Stable Oscillator (USO) on request by telecommand..*

TTCS-095 *The TT&C System shall support the following modes for the uplink:*
- Carrier only
- Telecommanding
- Ranging
- Simultaneously Telecommanding and Ranging

TTCS-100 *The following down links shall be selectable by telecommand:*
- Carrier only
- Telemetry
- Ranging
- Simultaneously Telemetry and Ranging

TTCS-102 *The telemetry modulation on the S-band and X-band shall be switchable separately.*

TTCS-115 *In the event of a loss of correct attitude, the TT&C subsystem shall provide the telecommand capabilities at maximum distance and in any of the spacecraft attitudes.*

TTCS-120 *Omni-directional coverage shall be provided by LGA's at minimum bit rates for any attitude of the satellite, for both the uplink and the downlink with sufficient overlap such that antenna switching is never a time critical item.*

TTCS-125 *The MGA's shall provide emergency uplink and downlink default communication after Sun pointing mode of the S/C is reached.*

TTCS-130 *The HGA shall provide the primary communication for both uplink and both downlink frequency bands.*

TTCS-140 *Testability and failure indication shall be taken into account in the design (e.g. RX-lock signal on a spacecraft test connector,..).*

TTCS-145 *It shall be assured that hot redundant functions can be tested.*

TTCS-150 *Attention shall be paid to the antenna thermal design with regard to the plume impingement effects of the RCS Thrusters.*

TTCS-155 *A test with a spacecraft mockup-up shall be performed to verify the antenna pattern and performance for LGA and SSP antennae including the measurement of the coupling between these two antennae. The pattern of the CONSERT and MIP antennae shall also be measured.*

4.10.3 Performance Requirements

TTCS-160 *The uplink/downlink signals shall be in the range*

2110.243 to 2117.746 MHz for telecommand at S-band

and 7149.598 to 7188.897 MHz for telecommand at X-

band and 2291.666 to 2299.814 MHz for telemetry at S-

band and 8402.777 to 8440.802 MHz for telemetry at

X-band.

Category B Mission with turnaround ratios of:
221/240 (S-/S- Band)
221/880 (S-/X- Band)
749/880 (X-/X- Band)
749/240 (X-/S- Band)

TTCS-165 The frequency stability shall be 10^{-12} (Allan variance over 3 seconds) using the external USO.

TTCS-170 RHC polarization will be used for uplink and downlink throughout the mission.

TTCS-175 The maximum telecommand bit error rate (BER) shall be better than $1 * 10^{-5}$

TTCS-180 Link budgets shall be computed as defined in ESA PSS-04-105: RF and Modulation Standard, including nominal, adverse, favorable, mean -3 sigma and worst case RSS (root sum square).

TTCS-182 The Contractor shall control the subsystem design by producing the link budgets for the different phases of the mission and updating them during the course of the project.

TTCS-185 The probability of frame loss on the telemetry shall be < $1*10^{-4}$.

TTCS-190 The LGAs omni-directional coverage shall guarantee at least 1000 bps uplink and 256 bps downlink in S-band during near Earth mission phases (0.025 AU for 32 m Perth station).

TTCS-192 The LGAs shall support an uplink command rate at S-band of 7.8 bps up to a distance of 6.5 AU for DSN 70 m, and up to 1.1 AU for the Perth 32m station.

TTCS-195 During any near earth phases the LGA's shall support a minimum down-link bit rate of at least 16 bps (up to 0.04 AU at 10 deg elevation) for the Kourou 15m.

TTCS-200 The MGA shall allow at least a downlink rate of 16 bps at a distance of 4.0 AU with the Perth 32m at 10 deg elevation in X-band.

TTCS-202 The MGA's shall allow an uplink bit rate of at least 16 bps up to 6.25 AU for the Perth 32 m in S-band and X-band.

TTCS-205 The HGA shall allow at least an uplink rate of 2000 bps in S-band and X-band and a downlink rate of 5000 bps at a distance of 4.5 AU with the Perth 32 m station at 10 deg elevation.

TTCS-207 *The HGA shall allow Earth pointing in all nominal mission phases. A margin of 10 degrees shall be accommodated on both pointing axes (needed range ± 10 degrees) It shall not be necessary to 'flip over' (180 degrees azimuth change) the antenna if the pointing is within the ± 10 degrees margin zone.*

4.10.4 Ground Compatibility Test

To ensure the full compatibility between the spacecraft and the ground segment, compatibility tests will be carried out. Such compatibility tests shall be accomplished by means of a spacecraft RF-suitcase.

TTCS-210 *The RF-Suitcase shall contain flight representative hardware sufficient to test all up- and down links for both functional and performance characteristics.*

TTCS-220 *The RF-Suitcase shall enable the verification of all telemetry, telecommand and ranging functions and combination thereof, as well as spectral analysis.*

TTCS-225 *The RF-Suitcase shall be a self contained item easy to transport, and shall include flight representative DMS and RF units as well as the RF switches and cables. The unit shall allow to control and monitor all integrated units.*

TTCS-230 *The RF-Suitcase shall be computer controlled to allow the commanding of any function of the units and to display all telemetry data on request. The housekeeping parameters of the integrated units shall be permanently displayed.*

TTCS-235 *The telemetry/telecommand bit streams and clocks shall be available to measure Bit Error Rate.*

4.10.5 Design Requirements

TTCS-240 *The TT&C system shall contain no single point failures, except for the radiating elements of antennas and their associated cabling, and shall have the capability of recovering from a failure either autonomously, or by assistance from other on-board subsystems.*

TTCS-245 *The interconnection between the RF elements shall be chosen such that the RF-losses are minimised.*

TTCS-246 *The subsystem design shall ensure that all relevant operational parameters are acquired via suitable sensors and provided to the Data Management System for incorporation into the Housekeeping (HK) downlink.*

TTCS-250 *The analogue readings shall be scaled in such a way that the useful reading is uniformly spread over the full range of the DMS input range of 5 V.*

4.11 Data-Handling Requirements

4.11.1 General Requirements

The Data Management System (DMS) shall collect all the telemetry data from the spacecraft. The data shall be conditioned, digitised and encoded for transmission to ground. It shall also process the uplink signal received by the Radio Frequency system, and validate the commands and distribute them to the users for execution. It shall store data for later transmission and shall perform data compression to increase the science data return. The transmission of housekeeping information shall be reduced to a minimum to allow a higher bandwidth for the experiments. More detailed housekeeping information can be transmitted on request.

DMSS-010 *The DMS shall be compatible with the following ESA standards:*
- Packet Telemetry Standard (ESA PSS-04-106)
- Packet Telecommand Standard (ESA PSS-04-107) Suitably adapted for this deep space mission and jointly agreed.
- Telecommand Decoder Specification (ESA PSS-04-151)

DMSS-020 The autonomy on-board shall be supervised by the DMS.

DMSS-025 The DMS Software shall be reprogrammable

The main functions to be performed by the DMS are:

DMSS-030 *Telemetry acquisition and formatting for transmission*
DMSS-035 *Data Compression*
DMSS-040 *Data Storage*
DMSS-045 *Telecommand decoding and distribution*
DMSS-050 *Timing distribution and tagging*

DMSS-065

DMSS-070 *The DMS shall be switchable in a low power mode with only a wake up timer running.*

DMSS-075 *The DMS shall be fully redundant including cross strapping to improve reliability.*

DMSS-080 *The DMS shall be fully functional after start up (i.e., no data to be loaded from ground)*

DMSS-082 *The DMS 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.*

4.11.2 Functional requirements

The data management system shall provide the following functions to fulfill the mission requirements:

Telemetry

DMSS-085 *Collect data from subsystems, experiments and Solid StateMass Memory (SSMM).*

DMSS-090 *Format and encode telemetry data.*

DMSS-095 *Deliver telemetry data to SSMM.*

DMSS-100 *Modulate encoded telemetry and deliver the video signals to the TT&C subsystem*

DMSS-105 *Concatenated coding, i.e., convolutional and Reed-Solomon coding shall be implemented. It shall be possible to inhibit all telemetry encoding (Convolutional, Reed-Solomon and transfer frame formatting) implemented in hardware and software and replace it with software implemented coding during the mission.*

DMSS-110 *Telemetry packets with no segmentation shall be used on the space to ground link.*

DMSS-111 *Hardware buffering on all data transmission paths shall be sufficient to allow a polled structure of the software all the way through the data path. Software will use a simple cyclic executive with the minimum of hardware interrupts.*

Telecommand

DMSS-115 *Acquire, decode, validate and distribute telecommands from the TT&C subsystem.*

DMSS-120 *Distribute commands generated on-board (e.g. from mission time line, On-Board Control Procedures (OBCP's), etc.*

DMSS-125 *Telecommand packets with no segmentation shall be used on the ground to space link.*

DMSS-127 *TT&C decoder shall protect the spacecraft against erroneous commands, either due to noise transmission conditions or from command signals sent to other spacecraft.*

DMSS-128 *Any invalid telecommands received shall be indicated in the telemetry.*

DMSS-129 *The DMS shall support the “Aggregation” of TC packages in the same TC frame.*

Data Compression

DMSS-130 *Data compression shall be performed on the telemetry downlink to optimize the transmission of data to Earth.*

DMSS-131 *The data compression algorithm shall be selectable from ground.*

DMSS-135 *The compression shall be loss-less for housekeeping and science data and lossy/lossless (depending on selected mode) for the image data. It shall be possible to apply the compression algorithm(s) on all data.*

Storage

A Solid State Mass Memory (SSMM) shall be available for the on-board storage as follows:

DMSS-140 *The SSMM shall be able to store science and housekeeping data, as well as Flight Operation Procedures, Time lines and S/W programmes or S/W patches.*

DMSS-141 *- The SSMM should behave as a disc unit to the users and support a filing system*

- The SSMM shall support random access.

- The SSMM should manage free space and automatically mark bad areas as unusable

- Information should be made available on request about free space, files stored and bad areas

- The SSMM should allow multiple read write access simultaneously to all possible users

- It should be possible for a user to read write update or delete a file

- The SSMM shall support a single write/copy command to files which are duplicated in the SSMM

- It should be possible to select files to be compressed

- The SSMM should hold binary images in files of on board software for all units. More than one version for each unit should be supported

- The SSMM shall support simultaneous read and write operations.

- Multiple redundant processors in the DMS together with direct links from high rate instruments shall necessitate multi-porting of the storage device, and simultaneous support of access over any of these links and processors.

- A high bandwidth bi-directional link between the SSMM and the DMS RAM shall exist.

- The SSMM shall support a OBDH bus interface for read or write. Data read or written on this bus will be treated identically to data transferred via high rate links.
This provides a back-up interface to acquire data over the OBDH bus at lower rates should the direct link fail.

DMSS-142 *There shall be a non-volatile memory available for storage of boot-up sequences and essential software functions (e.g. PROM).*

DMSS-145 *Non-volatile reprogrammable memory shall be available to store important parameters and programme code (e.g. EEPROM).*

DMSS-147 *The Random Access Memory (RAM) shall be protected by an Error Detection And Correction (EDAC) function. This shall allow to correct single bit errors and to detect double bit errors.*

DMSS-148 *Each processing unit shall count the single bit correction by the EDAC and the un-correctable double errors. These values shall be part of the TM housekeeping information.*

DMSS-149 *RAM allocated to one processor shall not be accessible by another processor. This will remove a number of untraceable single point failures.*

Timing

DMSS-155 *The DMS shall maintain the spacecraft Elapsed Time, SCET, and shall distribute Spacecraft Elapsed Time (SCET) on the OBDH bus to all on-board users.*

DMSS-160 *All time reference in telemetry, telecommand and on-board the spacecraft shall be in SCET.*

DMSS-165 *The time format shall be Consultative Committee for Space Data Systems (CCSDS) Unsegmented Code. Four octets of coarse time and 16 bits of fine time shall be implemented.*

DMSS-170 *It shall be possible to set the time (SCET) by telecommand.*

DMSS-175 *The total on-board time error shall not exceed the value required as resolution for the time line.*

DMSS-180 *The timing system error budget shall have a positive margin.*

DMSS-185 *The CDMU shall support Standard Time Source Packet generation at rates selectable to avoid ambiguity.*

DMSS-186 *The spacecraft time reference shall be triple redundant.*

4.11.3 Performance requirements

DMSS-190 *The telemetry rate shall be switchable between 8 bps and 65536 bps in steps with the factor of 2. Some additional bit rates shall be available to optimise the data return during near comet operations.*

DMSS-192 *The accuracy of the SCET (absolute setting of reference oscillator) shall be better than*
- short term; 3.10^{-7} /day under nominal operations
- long term; 4.10^{-6} /yr (1st year), 2.10^{-6} /yr (2nd yr), 1.10^{-6} /yr (after 2nd yr)
- setting error less than $2\text{-}10^{-7}$

DMSS-195 *The bus throughput shall be minimal 131 kbps.*

DMSS-200 *The SSMM shall support an input data rate of up to 5 Mbps useable data (physical data rate excluding IEEE-1355 protocol overhead) over the collective direct links from the high rate instruments.*

DMSS-205 *The telecommand rate shall be switchable between at least 4 bit rates, i.e., 7.81 ($4000/2^9$), 250, 1000 and 2000 bps. Some Additional bit rates shall be available to optimise command traffic during near comet operations.*

DMSS-210 *The usable memory size shall be at least 1MWord RAM and 512 KWords EEPROM for each of 4 processors, and 512KWords PROM (redundant) accessible from each processor.*

DMSS-226 *The SSMM shall have a size of at least 25 Gbit at end of life taking into account failures in memory cells.*

The size of each storage media shall be sufficient to store the scientific and housekeeping data, programme data and Software as required for the Rosetta mission and specified in the other chapters of this and other related documents.

DMSS-230 *The PROM storage shall provide a bit failure rate of less than 1 in 10^{13} after ten years.*

DMSS-231 *The RAM storage shall provide a bit error rate of less than 1 in 10^{13} after ten years.*

DMSS-232 *Rosetta SSMM shall support a bit error rate of less than 1 bits in 3 x 10^{11} after one year.*

4.11.4 Design Requirements

DMSS-235 *Testability and failure indication shall be taken into account during the design.*

DMSS-236 *All processor modules shall incorporate built in test routines used for regular evaluation as health checks.*

DMSS-240 *The hot redundant functions including failure correction functions (eg., EDAC facility for memories) shall be testable.*

4.12 Software Requirements

Existing software

SWRE-005 *Software already produced and embedded in a pre-designed unit that does not require any modification to the software for Rosetta, shall be the subject of a formal review of it's Documentation(design, test and maintenance), coding standards together with a code inspection.*

SWRE-010 *Should this review identify any additional actions required for the software to be made acceptable they shall be performed by the contractor supplying the unit or the embedded software* .

SWRE-015 *Should this review identify any modification required to the software then the modification and validation of the whole software shall be performed using the same requirements as* *for software to be produced.*

Software to be produced

The following requirements shall cover all software that will be implemented in an on-board sub-system unit, including on-board procedures for executing the mission time line.

SWRE-025 *All software production and test shall follow the ESA standard PSS-05 Issue 2* as applicable or other software standards agreed by the Agency for Rosetta.

SWRE-030 *All software produced shall be verified by the producer and a separate (from another company) Independent Software Verification Team. Both shall be responsible for delivery of the software on schedule.*

SWRE-035 *All reviews shall be formal and be supported by ESA, and Higher Level Contractors.*

SWRE-045 *All software shall be validated against, not only the real hardware but by using a Software Validation Facility (see SDE and SVF section) in addition.*

General Requirements

SWRE-050 *Starting or stopping processes shall not affect the execution or performance of running processes.*

SWRE-055 *Rosetta shall be loaded with the full software necessary for the complete mission in PROM.*

This means the basic software, defined application programs, flight control procedures and monitor tables.

SWRE-065 *Software engineering parameters shall be available in telemetry house keeping to enable ground to FULLY diagnose the status of the software.*

SWRE-067 *Software shall protect itself against infinite loops, computational errors and possible lock ups resulting from an undetected hardware failure.*

By using watch dogs and other techniques

On-Board Memory

SWRE-070 Fixed areas of on-board memory shall be dedicated to:

a) code
b) fixed constants
c) variable parameters

This requirement will simplify the mechanism supporting the patching of software in flight.

SWRE-075 The software shall normally run in RAM.

This is to save power consumption during the mission

SWRE-080 The size of the non-volatile memory shall allow for a 20% free area for use during the mission, for the storage of code patches and new software.

The goal is to launch with the memory empty of any patches to the on-board software.

SWRE-085 RAM shall contain at least 30% free (unused) memory after coding and testing has been completed, it shall be possible to use this free area for code and / or data areas.

SWRE-090 Rosetta shall support the dumping to ground of any elements of the on-board memory, initiated by tele-command.

The dump request will specify the name of the memory to be dumped, the start address and length of the dump, multiple selections shall be allowed in one command. Even if several packets are required to convey the dump data to ground, only a single tele-command will be required.

Processors

SWRE-095 The code required to boot up a processor shall be shown to be optimised for reliability, and the time taken to initialise.

SWRE-100 The processor shall boot to a stage that provides the basic critical functions (HK telemetry, Tele-command execution and critical autonomous functions). From this point it shall be possible to select a further software load from PROM, RAM, SSMM or any combination.

This also implies the capability to copy information (code and data) between PROM, RAM and SSMM, as follows:- PROM to RAM; RAM to SSMM; SSMM to RAM.

SWRE-105 The predicted CPU load should not exceed:

a) 50% capacity at the completion of Architectural Design phase

b) 75% at the completion of the unit level testing of the software.

Tele-Commands

SWRE-110 Commands shall be forwarded to the addressed unit conforming to the packet standard , in exceptional cases where a unit can not support the packet standard only data will be forwarded to the unit.

SWRE-115 Lower level validation of received telecommands shall be performed by a dedicated device, not done in the main onboard software.

Telemetry Collection by the DMS

SWRE-120 All data transfer from and to the DMS shall be polled by the DMS and not interrupt driven.

SWRE-125 The data collection algorithm shall be automatically optimised in a configurable way according to available sources and on-board resources.

SWRE-130 The DMS shall be able to pause a data transfer to perform a higher level function without loosing data from the transfer.

Scheduler

SWRE-150 The Rosetta scheduler shall maintain the current execution status of processes such that at switch over to another processor, the executing processes continue execution from the current execution point.

SWRE-155 The Rosetta scheduler shall be cyclic and provide deterministic scheduling of processes.

SWRE-170 The Rosetta scheduler shall initiate software processes in accordance with a pre-defined strategy which shall be reconfigurable by Telecommand.

The scheduled order in which processes will be performed will form an intrinsic part of the software design.

Software Updating and Maintenance

SWRE-185 *Rosetta shall support post-launch modification of all on-board software, except for boot software contained in PROM.*

SWRE-195 *Rosetta on-board software shall be structured such that modifications to any individual code module is possible without affecting any other modules.*

This requirement will simplify the mechanism supporting the patching of software in flight, and will help ensure that all patches are deterministic.

SWRE-205 *Rosetta shall provide a mechanism to allow / inhibit the automatic application of software modifications at initialisation.*

This will be necessary to protect against faulty modifications being applied.

SWRE-210 *All RAM, EEPROM, SSMM memory shall support direct patching, including the areas where memory management is used.*

Software Development

SWRE-220 *Rosetta on-board software shall be developed using design methodology using a CASE tool to be approved by ESA.*

SWRE-225 *Use of formal methods to define elements of the Rosetta on-board software shall be shown to be beneficial to the development of those software elements and agreed by ESA.*

SWRE-230 *Rosetta on-board software shall be developed using a high level language, except where explicitly exempted.*

SWRE-235 *The use of assembler code in Rosetta software shall be shown to be absolutely necessary to achieve the required performance, to the satisfaction of ESA before it is implemented. Submission of a justification by the contractor shall be made on a case by case basis and in good time such that the agency has time to review the submission (minimum 40 working days) without affecting the project schedule.*

The use of assembler will be restricted to optimisation of small sections of code where the measured performance cannot meet the requirements using optimisation of the Ada alone.

SWRE-245 All Rosetta on-board software running in sub-systems shall be developed using the same development environment.

Standardising the development environment for all subsystems will significantly reduce program cost. The development environment may be duplicated in each development site.
The standard development environment comprises standard processor type, standard development language, standard software development tools, and a single development team per system development.

SWRE-255 Test tools used in the development testing of Rosetta software shall support the same telemetry and tele-command data base as used in the system level testing.

Software Test During Development

SWRE-270 Rosetta Software shall be tested to the agreed (ESA / prime contractor) project standards.

The standards should set targets for boolean, statement and branch test coverage.

SWRE-275 Software units which are changed following formal verification shall undergo full regression testing.

This may require the update of the tests themselves.

Static Analysis

SWRE-290 All software code shall be statically analysed by a standard tool.

The static analysis tool will be mechanised to improve the consistency of the analysis.

SWRE-295 The selection of the static analysis tool shall be a prime contractor task, and agreed by ESA.

As selection criteria the following list should guide:-

-identify language features prohibited by the Rosetta programming standard.
-identify operating system features prohibited by the Rosetta programming standard.
-identify unreachable code
-identify endless loops.
-identify multiple entry points in any single unit of code.

-identify multiple exit point in any single unit.
-identify variables used before they are initialised.
-identify unused data.
-identify incompatibility in the interface between calling and called units, except where this is implicitly tested by the adopted compiler.
-identify the volume of code in each unit.
-measure the complexity of code in each unit, in accordance with the project standard.
-compare the metrics it generates for each tested unit against pre-defined project standards.
-verify that code meets the project coding standard

Dynamic Analysis

SWRE-300 All software code shall be dynamically analysed by a standard tool.

The dynamic analysis tool will be mechanised to improve consistency of the analysis.

SWRE-305 The selection of the dynamic analysis tool shall be a prime contractor task and agreed by ESA.

As selection criteria the following list should guide:-
-instrument the source code of the unit under test without manual intervention.
-instrument source code without functional effect on that code.
-measure source code statement coverage achieved by the test being performed.
-measure source code branch coverage achieved by the test being performed.
-collate measured statement and branch coverage in a summary for each unit after integration.
-support batched testing
-identify untested branches in source code.
-identify unexecuted lines of source code.
-measure Boolean expression coverage achieved by a test.
-measure path coverage achieved by a test.

Software Maintenance Environment

SWRE-310 The Rosetta software maintenance environment shall fully adopt the configuration procedures and tools of the development phase

SWRE-330 *The Rosetta software maintenance environment shall provide the means to generate and prepare software patches or full images for uplink to the spacecraft as telecommands.*

The ability to uplink changes to the software will be mission critical.

SWRE-335 *The Rosetta software maintenance environment shall be integrated with the SVF.*

New software, or software patches to be uplinked to the spacecraft will need to be formally tested and verified prior to their release and use.

Software Validation Facility

SWRE-350 *The SVF shall support Rosetta software verification at the following levels during development:*
- *a)* *unit level;* i.e., procedure or function level
- *b)* *integration level;* i.e., module level
- *c)* *subsystem level;* i.e., SW subsystem level.

SWRE-355 *The SVF shall allow Software Verification to be performed before and after the software is integrated at subsystem level.*

SWRE-360 *The SVF shall support both development and maintenance phases of the Rosetta project.*

SWRE-365 *The SVF shall support full white box testing of the software at source code and machine level.*

SWRE-370 *The SVF shall allow software testing without instrumentation of the target code*

Software Development Environment and Software Verification Facility (SDE & SVF)

The following requirements address both the SDE & SVF. It is recommended that the SDE and SVF be implemented using the same hardware, and the SDE software be available on the SVF.

SWRE-375 *All software licenses for Any software used to develop and test the on-board software shall be maintainable at the same version and issue over the full life of the mission at a freeze point in the schedule and be deliverable to the agency. The freeze point for the version and issues of the software licenses shall take place at an identified point during the phase CD schedule. Only one freeze point for the whole project shall be made.*

SWRE-380 It shall be possible migrate/move ***All*** *software to another hardware platform as old platforms become not supported by the manufacturers,* ***OR*** *, The hardware platform shall be supported over the full life of the mission by the manufacturers.*

SWRE-385 The SVF shall use the content of the Rosetta System Data Base (RSDB) for all parameters that correspond to real or derived Command and Telemetry parameters on the ground-space and space-ground link.

Independent Software Validation

SWRE-390 Rosetta software development shall be subject to review by an independent software validation (ISV) team.

Independent Software Validation (ISV) will provide invaluable feedback on the development process of these mission critical functions.

SWRE-395 The ISV team shall review the URDs and SRDs with minimal visibility of development documentation or support from the development team.

The purpose of this review will be to identify errors in the SRD of these Rosetta functions.

SWRE-400 The ISV team shall generate and execute a system test specification from the SRDs only.

The high level system test specification will identify the means by which the integrated system implementation will be qualified, as a black box.

SWRE-405 Following completion of the system test specification, the ISV team shall review code generated by the software development team, against the SRDs.

The purpose of the review will be to provide feedback on the quality of the low level implementation of requirements.

4.13 Harness Requirements

4.13.1 General Requirements

HARN-001 *The harness performs the electrical connection between all electrical and electronic equipment in the ROSETTA spacecraft.*
It shall provide adequate distribution and separation of all power supplies, signals, scientific data lines and pyrotechnic firing pulses and all connections to the umbilical, safe/arm brackets/connectors and test connectors. Requirements of ESA PSS-02-10 shall be applicable unless otherwise specified.

HARN-002 *The industrial contractor shall be responsible to specify the experiment harness length, shape and routing optimized for minimum length/mass.*

4.13.1.1 Functional requirements

HARN-003 *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 spacecraft subsystems and those lines to the payload interface, test connectors, safe/arm brackets/connectors and umbilical connectors.*

4.13.1.2 Performance requirements

HARN-004 *The harness shall transmit all electrical currents in a manner compatible with the requirements of the source and destination unit/interface.*

HARN-005 *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 ESA PSS-02-10.*

HARN-006 *The inductance of the power distribution harness for frequencies up to 100 kHz shall not exceed the value required by the power standard ESA PSS-02-10.*

HARN-007 *The isolation requirements between leads within one connector, which are not connected together and between shields and centre conductor and shield to shield shall be at least 10 MOhms at 500 V DC at both polarities.*

HARN-008 *Where it is necessary to have a shield connection through a connector, separate pins shall be used.*

HARN-009 The pyrotechnic harness shall satisfy the applicable EMC and safety requirements and shall have a cable route which is physically separate from other harness classes.

HARN-010 The pyro harness shall consist of twisted pairs of wires with an overall shield being continuos and connected to the conductive connector shells at all interfaces and grounded to structure at all intermediate attachment points.

HARN-011 Connections to the initiators shall be capable of being mechanically broken during ground handling by safe/arm connectors accessible from the outside of the S/C.

4.13.1.3 Design requirements

HARN-012 All equipment shall use a separate connector dedicated to its functional interface, as applicable, with the power lines, the data handling interface, the pyro lines, the external spacecraft interface and the payload interface.

HARN-013 Wiring of redundant systems, subsystems or elements of subsystems shall be routed through separate connectors, through separate wire bundles (as far as practical).

HARN-014 Redundant wire bundles shall be routed differently, wherever practical.

HARN-015 Cross strapping of redundant paths and circuits shall not be carried out in the harness.

HARN-016 All harness and all box and bracket mounted connectors supplying power shall have socket contacts.

HARN-017 The shields of cables shall not be used as return lines.

HARN-018 All hot and return lines shall be twisted together.

The initial design shall take care that sufficient (including spare) pins are available for all foreseeable subsystem and experiment functions.

4.13.1.4 Mechanical requirements

HARN-020 The mechanical construction of the spacecraft harness shall assure the reliable operation of the S/C under all specified 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.*

Weight and volume of the harness shall be kept as low as possible.

HARN-021 *Wire gauges <u>smaller than</u> AWG 28 shall not be used.*

HARN-022 *All individual wire-to-pin interfaces shall be covered with transparent heat shrink sleeves.*

HARN-023 *The possibility of incorrect mating of connectors shall be excluded by design as far as practical.*

HARN-024 *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.*

HARN-025 *Where harness is used to connect to deployable appendices the relevant mechanical properties shall be specified, characterized and tested.*

5. PAYLOAD INTERFACES

It should be noted that in case of conflict between statements written in the following chapters and statements written in the EID documents for the orbit payload and the LID document for the SSP, the information contained in the EID/LID shall govern.

PLIS-001 *The baseline payload/SSP designs are identified in the EID-Bs/LID-B and are the current development status without contingency. The Contractor shall be prepared to be responsive both technically and programmatically when further evolutionary changes take place.*

PLIS-005 *The Contractor shall maximise the resources available to the payload/SSP.*

PLIS-030 *The total scientific telemetry bit rate dedicated to the payload shall be at least 5 kbps during coverage from the Southern Hemisphere ground station during scientific operations at less than 3.25 AU (comet to Sun distance). The apportionment to individual experiments will be a Rosetta Science Working Team decision.*

PLIS-035 *The on-board solid state recorder shall be able to receive data rates from each payload unit as defined in the EID-Bs/LID-B.*

PLIS-040 *The spacecraft shall support periodic (not more than once per 6 month) maintenance of life time limited elements of payload experiments. The electrical power required for such maintenance shall not drive the overall spacecraft electric power/thermal design.*

5.1 Orbiter Payload

PLOS-001 *The spacecraft shall provide a 165 kg mass allocation for the baseline payload accommodation as defined in EID-A including contingency for Phase C/D.*

PLOS-005 *The structural design of the spacecraft including payload interfaces shall be able to meet its requirements if the mass of any particular payload unit is increased by 30% above the values agreed in the EID-Bs at the beginning of Phase B.*

PLOS-010 *The spacecraft shall provide 208W mean power for the payload during nominal scientific operations phases starting at 3.25 AU including comet orbiting and asteroid fly-bys.*

PLOS-015 *The spacecraft shall provide peak power for individual experiments as defined in the EID-Bs.*

PLOS-020 The spacecraft shall provide 25W mean heater power for the individually controlled payload during the hibernation phase.

PLOS-025 The thermal design of the spacecraft shall be able to meet its requirements when the power dissipation of payload units is within their range of minimim to maximum values agreed in the EID-Bs and the total orbiter payload power dissipation is up to 300W.

PLOS-030 The thermal design of the spacecraft shall support near comet operations:- at 1.4 AU with total orbiter payload dissipation; - at 1.0 AU with 205W payload dissipation.

5.2 Surface Science Package

PLSS-001 The spacecraft shall provide a mass allocation of 100 kg (including contingency margin for phase C/D) for the SSP comprising:
- lander, that is all equipment to be ejected from the orbiter
- the Mechanical Separation System (MSS) i.e., all support structures, the separation mechanism, its electronics and relevant harness, which all remain on the orbiter
- the Electrical Support System (ESS) i.e., all electrical/electronic support equipment, including relevant harness, antennae, which will also remain on the orbiter.

PLSS-005 The structural design of the spacecraft including SSP interfaces shall be able to meet its requirements if the mass of the SSP lander or of any unit on the orbiter is increased by 30% above the values agreed in the LID-B at the beginning of Phase B.

PLSS-010 The spacecraft shall provide the following maximum power to the SSP/ESS for any check-out period, Lander delivery preparation, Lander delivery and telecommunication during the relay phase: 40 W.

PLSS-015 The spacecraft shall provide the following mean heater power for the SSP during all phases including hibernation: 15W.

PLSS-020 The spacecraft shall provide the following maximum power for the ejection device of the Lander: 145W.

PLSS-025 The thermal design of the spacecraft shall be able to meet its requirements when the power dissipation of SSP units on the orbiter is within the range of minimum to maximum values agreed in the LID-B and the total SSP power dissipation is up to 42W.

PLSS-030 *During the Lander relay phase, the SSP data collection shall take priority over the nominal payload operations according to the requirements defined in LID-B.*

5.3 Standard Radiation Environment Monitor

PLRS-001 *A Standard Radiation Environment Monitor (SREM) will be flown on the Rosetta orbiter. The SREM will be supplied by the Agency and the interfaces are defined in SREM/ROS/RN/001 (AD-33).*

PLRS-005 *The spacecraft shall allocate a mass for the SREM of 2 kg.*

PLRS-010 *The spacecraft shall allocate a mean power for the SREM of*

2W

PLRS-015 *The data from the SREM shall be part of the spacecraft housekeeping data stream.*

5.4 Active Spacecraft Potential Control

PLAS-001 *Deleted*

PLAS-002 *Deleted*

PLAS-003 *Deleted*

PLAS-004 Deleted

6. LAUNCH VEHICLE INTERFACE

LVIS-001 *The spacecraft shall be compatible with a launch on Ariane 5.*

LVIS-005 *The spacecraft design and operations shall comply with all performances, requirements, interfaces and operations specified in the Ariane 5 Users Manual (AD-1).*

LVIS-010 *The flight adapter together with the separation mechanism (including clamp-band) will be provided by the Launch Vehicle Authority via the Project.*

LVIS-015 *The test adapters which need to be representative of the launch vehicle adapter shall be provided by the Rosetta contractor.*

LVIS-020 *The spacecraft shall be designed to satisfy the launcher safety requirements as defined in AD-1 and AD-2.*

7. OPERATIONS INTERFACE

The detailed operations requirements are subject of a separate document, the Rosetta Operations Interface Requirements Document (ROIRD), RO-ESC-RS-5001.
The interfaces between the spacecraft and the ground system are specified in the Rosetta Space/Ground Interface Control Document (SGICD), RO-ESC-IF-5002.

OPSI-001 *The spacecraft shall comply to the requirements specified in the Rosetta Operations Interface Requirements Document (AD-3)*

OPSI-002 *The spacecraft shall be compatible with the interfaces defined in the Space/Ground Interface Control Document (AD-4).*

8. ACRONYMS AND ABBREVIATIONS

AD	Applicable Document
AIV	Assembly, Integration and Verification
ALICE	Orbiter payload instrument
AME	Absolute Measurement Error
AOCMS	Attitude and Orbit Control and Measurement System
AOCS	Attitude and Orbit Control System
APE	Absolute Pointing Error
APXS	Lander payload instrument
ASF	Additional Safety Factors
ASI	Agenzia Spaziale Italiana
ATP	Approach Transition Point
AU	Astronomical Unit
AWG	American Wire Gauge
BER	Bit Error Rate
BERENICE	Orbiter payload instrument
BIT	Build In Test
BMOS	Buckling Margin Of Safety
BRU	Battery Regulator Unit
C/C	Collectively Controlled
CAP	Comet Acquisition Point
CCD	Charge Coupled Device
CCSDS	Consultative Committee for Space Data Systems
CEPHAG	Centre d'Etude des Phénomènes Aléatoires et Géophysiques
CHAMPAGNE	Lander payload instrument
CHARGE	Lander payload instrument
CIRCLE	Lander payload instrument
ÇIVA	Lander payload instrument
CDMU	Central Data Management Unit
CNES	Centre National Etude Spatial
COG	Centre Of Gravity
CONSERT	Orbiter/lander payload instrument
COSAC	Lander payload instrument
COSIMA	Orbiter payload instrument
CPPP	Lander payload instrument
CSG	Centre Spatiale Guyanaise
DC	Direct Current
DLR	Deutsche Forschungsanstalt für Luft- und Raumfahrt e.V.
DM	Dynamic Model
DMA	Direct Memory Access

DMS	Data Management System
DoD	Depth of Discharge
DOF	Degree Of Freedom
DSN	Deep Space Network
DTMM	Detailed Thermal Mathematical Model
EDAC	Error Detection And Correction
EEPROM	Electrically Erasable Programmable Read Only Memories
EGSE	Electrical Ground Support Equipment
EID	Experiment Interface Document
EM	Engineering Model
EMC	Electromagnetic Compatibility
ESA	European Space Agency
ESARAD	ESA Radiation
ESATAN	ESA Thermal Analyser
ESOC	European Space Operations Centre
FD	Flight Dynamics
FEM	Finite Element Model
FMECA	Failure Modes, Effects and Criticality Analysis
FMI	Finnish Meteorological Institute
FMS	Failure Management System
FOP	Flight Operations Plan
FOV	Field Of View
GIADA	Orbiter payload instrument
GMI	Global Mapping Injection point
H/W	Hardware
HGA	High Gain Antenna
HK	House Keeping
HOOD	Hierarchical Object Oriented Design
I/C	Individually Controlled
I/F	Interface
I-BOB	Intelligent Break Out Box
IAS-CNR	Instituto di Astrofisica Spaziale/Consiglio Nazionale delle Ricerche
IFEM	Interface Finite Element Model
IR	Infra Red
ISIS	Lander payload instrument
ISV	Independant Software Validation
IUE	International Ultraviolet Explorer
JPL	Jet Propulsion Laboratory
KFKI	Hungarian Research Institute for Particle and Nuclear

	Physics
LAS	Laboratoire d'Astronomie Spatiale
LCDA	Launcher Coupled Dynamic Analysis
LCL	Latching Current Limiter
LGA	Low Gain Antenna
LID	Lander Interface Document
LOS	Line Of Sight
LPCE	Laboratoire de Physique et Chimie de l'Environnement
LW	Launch Window
MAC	Modal Assurance Criterion
MGA	Medium Gain Antenna
MGSE	Mechanical Ground Support Equipment
MIDAS	Orbiter payload instrument
MIRO	Orbiter payload instrument
MLI	Multi Layer Insulation
MMH	Mono Methyl Hydrazine
MODULUS	Orbiter/lander payload instrument
MOS	Margin Of Safety
MPI	Max Planck Institut
MPIK	Max Planck Institut für Kernphysik
MPP	Multiple Phase Pinning
MPPT	Maximum Power Point Tracking
MUPUS	Lander payload instrument
NAC	Narrow Angle Camera
NASA	National Astronautic and Space Administration
NASTRAN	NASA Structural Analysis Tool
NTO	Nitrogen Tetroxide
OBDH	On-Board Data Handling
OIP	Orbit Injection Point
P/L	Payload
PCU	Power Control Unit
PFM	Proto Flight Model
PI	Principal Investigator
PMD	Propellant Management Device
PROM	Programmable Read Only Memory
PSR	Processor Status Registers
PTOLEMY	Lander payload instrument
RD	Reference Document
RAL	Rutterford Appleton Laboratory
RAM	Random Access Memory
RF	Radio Frequency

ROIRD	Rosetta Operations Interface Requirements Document
ROLIS	Lander payload instrument
ROMAP	Lander payload instrument
ROSINA	Orbiter payload instrument
RPC	Orbiter payload instrument
RPE	Relative Pointing Error
RSS	Root Sum Square
RTMM	Reduced Thermal Model
S/W	Software
S/A	Solar Array
SADM	Solar Array Drive Mechanism
SCET	Spacecraft Elapsed Time
SCL	Spacecraft Control Language
SDE	Software Development Environment
SESAME	Lander payload instrument
SF	Safety Factor
SGICD	Space Ground Interface Control Document
SM	Structural Model
SPC	Science Program Committee
SREM	Standard Radiation Environment Monitor
SSP	Surface Science Package
SSMM	Solid State Mass Memory
SVF	Software Validation Facility
TBC	To Be Confirmed
TBD	To Be Defined
TCS	Thermal Control Subsystem
TMM	Thermal Mathematical Model
TT&C	Tracking, Telemetry & Commanding
UMOS	Ultimate Margin Of Safety
USO	Ultra Stable Oscillator
UV	Ultra Violet
VIRTIS	Orbiter payload instrument
VIS	Visual
WAC	Wide Angle Camera
YMOS	Yield Margin Of Safety

9. LIST OF REQUIREMENT ABBREVIATIONS

The table below contains an alphabetically sorted list of the requirement abbreviations used in this document, and the chapters in which they are used. Please note that table shows the highest level chapter numbers for the abbreviations. For example, the abbreviation POWR is shown to appear in chapter 4.9. This means that the same abbreviation is also used in the lower level chapters such as chapter 4.9.1, 4.9.2 etc.

In some cases, where the subject area of the chapter is not clear from the title, an indication of this is given in parenthesis. For example, the abbreviation MRDE, is used in chapter 4.4.5 Design Requirements. The subject area of chapter 4.4.5 is Mechanical Requirements, and this is indicated next to the chapter title in parenthesis.

Requirement Abbreviation	Chapter Number	Chapter Title
AOCS	4.6	Attitude, Orbit Control and Measurement Requirements
AUTO	4.1.2	Autonomy
CLEN	4.2.3	Cleanliness Requirements
DMSS	4.11	Data-Handling Requirements
EMCR	4.2.5	Electromagnetic Compatibility Requirements
ERME	4.2.1	Mechanical Environment
ERTE	4.2.2	Thermal Environment
GERE	3.3	General Requirements
HARN	4.13	Harness Requirements
IDMA	4.1.7	Identification and Marking
LIFE	4.1.1	Lifetime requirements
LVIS	6	LAUNCH VEHICLE INTERFACE
MART	4.1.6	Maintenance, Accessability, Reparability, Testability
MATE	4.1.5	Materials and Processes
MISS	3.2.4	Mission Design Requirements
MRDE	4.4.5	Design Requirements (Mechanical Requirements)
MRDF	4.4.1	Definitions (Mechanical Requirements)
MRFR	4.4.2	Functional Requirements (Mechanical Requirements)
MRPR	4.4.3	Performance Requirements (Mechanical Requirements)
MRVA	4.4.4.2	Analyses (Mechanical Requirements)
MRVM	4.4.4.1	Modelling (Mechanical Requirements)
MRVT	4.4.4.3	Testing (Mechanical Requirements)

Requirement Abbreviation	Chapter Number	Chapter Title
NAVI	4.7	Optical Navigation Camera Requirements
OPSI	7	Operations Interface
PINT	4.1.3	Pointing
PLIS	5	PAYLOAD INTERFACES
PLOS	5.1	Orbiter Payload
PLRS	5.3	Standard Radiation Environment Monitor
PLSS	5.2	Surface Science Package
POWR	4.9	Power Supply, Storage and Conditioning Requirements
PROP	4.8	Propulsion Requirements
RADI	4.2.4	Radiation Environment
RELI	4.1.4	Reliability, Fault Tolerance/Single Point Failures
SCIR	3.1.3	Scientific Requirements
SRDE	4.3.5	Design Requirements (Structural Requirements)
SRDF	4.3.1	Definitions (Structural Requirements)
SRFR	4.3.2	Functional Requirements (Structural Requirements)
SRPR	4.3.3	Performance Requirements (Structural Requirements)
SRVA	4.3.4.2	Analyses (Structural Requirements)
SRVM	4.3.4.1	Modelling (Structural Requirements)
SRVT	4.3.4.3	Testing (Structural Requirements)
SWRE	4.12	Software Requirements
TRDE	4.5.5	Design Requirements (Thermal Requirements)
TRDF	4.5.1	Definitions (Thermal Requirements)
TRFR	4.5.2	Functional Requirements (Thermal Requirements)
TRPR	4.5.3	Performance Requirements (Thermal Requirements)
TRVA	4.5.4.2	Analyses (Thermal Requirements)
TRVM	4.5.4.1	Modelling (Thermal Requirements)
TRVT	4.5.4.3	Testing (Thermal Requirements)
TTCS	4.10	Telecommunication Requirements