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SOLAR ORBITER EXPERIMENT INTERFACE DOCUMENT - PART A

prepared by/*préparé par*	Solar Orbiter Team
book captain	Danielle Renton
approved by/*approuvé par*	Philippe Kletzkine
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European Space Agency
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ESTEC
Keplerlaan 1 - 2201 AZ Noordwijk - The Netherlands
Tel. (31) 71 5656565 - Fax (31) 71 5656040

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1 GENERAL

1.1 Background

In May 2002, the new ESA Science Programme was defined and presented to SPC. This programme contains groups of scientific missions related either technically or programmatically. One such mission group comprises:

- The Bepi Colombo mission to Mercury which includes two orbiting spacecraft,
- The Solar Orbiter mission which will permit close-up and high-latitude studies of the Sun.

At its 105th meeting held on 5-6 November 2003, in response to the SSAC proposal concerning the reconstruction of the ESA Science Programme, the SPC decided to maintain Solar Orbiter in the Programme 2003-2013.

The 107th meeting of ESA's Science Programme Committee on 7-8 June 2004 endorsed the recommendations of the SSWG and SSAC and confirmed the place of Solar Orbiter in the programme. This was reconfirmed at the SPC meeting on 8-9 February, 2006 with the assumption of the implementation of a May 2015 launch.

At its meeting in May 2006, the SPC directed the Science programme Directorate to proceed with a Call for Letters of Intent for the payload complement, to optimize the mission in terms of science requirements and industrial implementation and to seek international cooperation, so as to bring the mission back into an acceptable cost envelope to ESA and maintain the possibility for a launch in 2015.

These activities were carried out as directed by the SPC from June 2006 to October 2007. The Call for Letters of Intent elicited 23 responses, covering all scientific measurements and providing inputs to instrument accommodation studies that were performed with Industry. Industrial studies were also carried out to optimize the spacecraft design and advance critical technologies. ESA and NASA cooperated to define a joint mission composed of ESA’s Solar Orbiter and NASA’s Inner Heliospheric Sentinels projects which will have coordinated instrument Announcement of Opportunities. The science objectives of the joint mission were established by a Joint Science and Technology Definition Team. . The AO for Solar Orbiter was issued in October 2007. Following postponement of NASA’s Inner Heliospheric Sentinels, ESA and NASA further cooperated to re-define the joint endeavour as composed of ESA’s Solar Orbiter and NASA’s Solar Probe Plus projects and science objectives.

Due to continuing programmatic issues at its 123rd meeting on 18 & 19 November 2008, ESA’s Science Programme Committee decided that the Solar Orbiter mission would be placed into the Cosmic Vision programme for a 2017 launch and, as such, would compete with other mission candidates for full development funding. The Solar Orbiter payload selection was announced in March 2009. This selection is contingent upon selection of the mission.

1.2 Scope

The purpose of the document is to ensure that:

- The Principal Investigators (PIs) design, build and verify their instruments within the technical constraints imposed by the Solar Orbiter spacecraft and compatible with the Solar Orbiter programme constraints.

- The Solar Orbiter Prime Contractor designs, builds and verifies the spacecraft such that the instruments can be successfully integrated into the system.
- The spacecraft can be successfully launched and operated to achieve the scientific objectives of the Solar Orbiter mission.

The EID consists of two parts; A and B.

The EID-A contains the interface specifications that are applied to the design of the instrument as defined in the EID Part B written by the Principal Investigator. Part A defines the Solar Orbiter technical and programmatic requirements all Solar Orbiter PI's have to comply with.

The EID-B defines the PI response to the technical requirements in part A specifying in detail the interface information applicable to a particular experiment. Part B will form the sole formal and binding document for all technical and programmatic agreements between the ESA Solar Orbiter Project Office and each Solar Orbiter Principal Investigator.

The EID A and B shall be placed under formal configuration and change control once signed and thus any change requires formal agreement between ESA and the PI.

The EID A and B will become applicable documents to the Solar Orbiter prime contractor.

1.3 Document Concept & Architecture

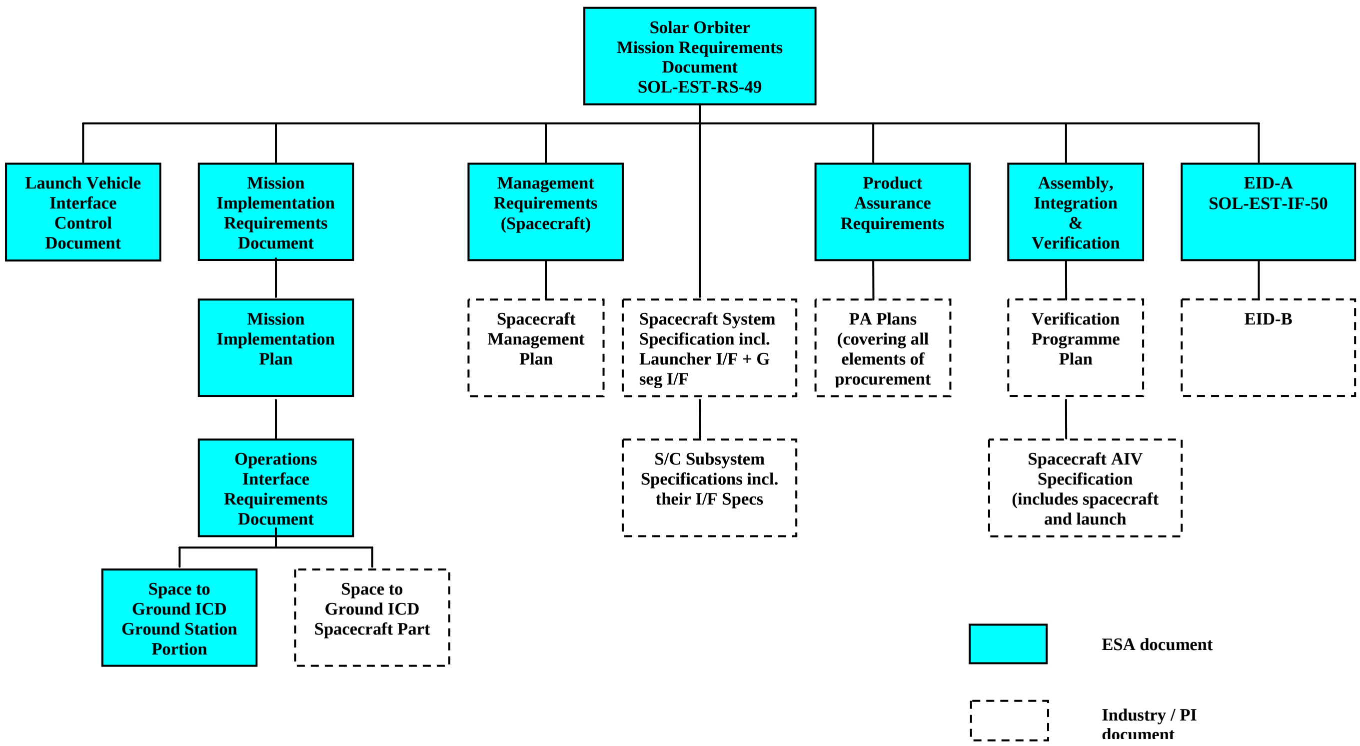


Figure 1 Preliminary Solar Orbiter Project Document Structure

2 KEY PERSONNEL AND RESPONSIBILITIES

2.1 ESA Personnel

Address
ESA/ESTEC
PO Box 299
2200 AG Noordwijk
The Netherlands

Switchboard: +31 71 565 6555
Project Fax: +31 71 565 5950
Project Administrator: +31 71 565 5192
Generic Email: FirstName.FamilyName@esa.int

Name	Responsibility	Contact Information
Philippe Kletzkine	Solar Orbiter Study Manager	Tel: +31 71 565 3761 Email: Philippe.Kletzkine@esa.int
Danielle Renton	Solar Orbiter Instrument System Engineer	Tel: +31 71 565 6586 Email: Danielle.Renton@esa.int

2.2 Contractor Personnel

Address
EADS Astrium Ltd.
Gunnelswood Road
Stevenage
SG1 2AS
United Kingdom

Switchboard: +44 (0) 1438 3134 56

Name	Responsibility	Contact Information
Ivan Ferrario	Project Manager	Tel: +44 (0) 14 3877 3662 Email: Ivan.Ferrario@astrium.eads.net
Paolo D'Arrigo	Engineering Manager	Tel: +44 (0)14 3877 4399 Email: paolo.darrigo@astrium.eads.net
Mark Ayre	Systems Engineer	Tel: +44 (0)14 3877 4258 Email: mark.ayre@astrium.eads.net
TBD	Payload Interface Manager	

3 SPACECRAFT DESCRIPTION

This section provides an overview of the spacecraft design and related instrument interfaces.

3.1 Mission Objectives

The Solar Orbiter mission will provide the next major step forward in the exploration of the Sun and the heliosphere to investigate many of the fundamental problems remaining in solar and heliospheric science. It includes both a near-Sun and a high solar latitude phase.

The near-Sun phase of the mission enables the spacecraft to approach the Sun as close as 0.22 AU during part of its orbit and thereby permitting observations from a quasi heliosynchronous vantage point. The satellite will co-rotate with the Sun, as at these distances the angular speed of a spacecraft near its perihelion approximately matches the rotation rate of the Sun. This characteristic enables the instruments to track a given point on the Sun surface for several days.

During the out of ecliptic phase of the mission, the Orbiter will reach higher solar latitudes (up to 35° close to the end of the mission), making possible detailed studies of the Sun's polar caps by the remote sensing instruments.

The main scientific objectives of the Solar Orbiter mission are to:

- Determine the properties, dynamics and interactions of plasma, fields and particles in the near-Sun heliosphere
- Investigate the links between the solar surface, corona and inner heliosphere
- Explore, at all latitudes, the energetics, dynamics and fine-scale structure of the Sun's magnetized atmosphere
- Probe the solar dynamo by observing the Sun's high-latitude field, flows and seismic waves

The ESA and NASA joint mission is referred to as Heliophysical Explorers (HELEX). Heliophysical Explorers (HELEX) brings together and augments the unique capabilities of ESA's Solar Orbiter mission (near-Sun and out-of-ecliptic in-situ plus remote-sensing observations) with those of NASA's Solar Probe.

This joint ESA-NASA science program offers a unique opportunity for coordinated, correlative measurements, resulting in a combined observational capability and science return that far outweighs that of either mission alone. Building on the knowledge gained from ground-breaking missions like Helios and Ulysses, and more recently STEREO, HELEX will bring to bear the power of multipoint, in-situ measurements using previously unavailable instrumental capabilities in combination with remote-sensing observations from a new, inner–heliospheric perspective to answer fundamental questions about the Sun-heliosphere linkage.

The three overarching questions to be addressed by the HELEX program are:

- What are the origins of the solar wind streams and the heliospheric magnetic field? I
- What are the sources, acceleration mechanisms, and transport processes of solar energetic particles?
- How do coronal mass ejections evolve in the inner heliosphere?

3.2 Mission Overview

The mission baseline is for Solar Orbiter to be launched by NASA on an Atlas V launch vehicle (with Delta IV as a possible back-up). However, compatibility with a Soyuz-Fregat launch will be maintained and the trajectory for such a launch is shown in Table 1 and Figure 2 as well as in [IR6]. The trajectory with an Atlas launch will be similar to the one currently outlined.

The mission will rely on a chemical propulsion system for maneuver performance. The “in-situ” set of instruments will be commissioned shortly after LEOP and, nominally, from then on will be continuously switched on, while the “remote-sensing” instruments are to be commissioned after the first Earth GAM and operated pre-defined science windows. Science operations during the initial part of the trajectory (i.e. before Venus GAM 2) will be accommodated if possible. After Venus GAM 2 the spacecraft is injected in a resonant orbit (the period of revolution of the Solar Orbiter around the Sun is 2/3 the period of Venus) in order to increase the inclination of the orbit with respect to the ecliptic. The following timelines are indicative of the baseline and backup mission scenarios:

Mission Launch	Absolute Minimum Sun Distance (AU)	Absolute Maximum Sun Distance (AU)	Absolute Maximum Earth Distance (AU)	Maximum Ecliptic Inclination (deg)	Maximum Solar Latitude (deg)
2017	0.23429 2023/04/04	1.48387 2017/11/21	1.981021 2020/02/26	30.246 achieved at EEM	35.89 achieved at EEM

Mission Phase Duration (days)	Event	Date (Calendar)	Flight Time (days)	Ecliptic Inclination (deg)	Aphelion (AU)	Perihelion (AU)
LEOP (7)	Launch	2017/01/04	0	2.27	0.98331	0.65238
	End of LEOP	2017/01/11	7	2.27	0.98331	0.65238
CVP (90)	End of CVP[1]	2017/04/11	97	2.27	0.98331	0.65238
Cruise (1400)	GAM V1	2017/04/15	101	2.224	1.48387	0.71648
	GAM E1	2018/08/25	597	3.706	1.11051	0.41579
	GAM E2	2020/08/25	1328	3.461	1.05532	0.32801
	GAM V2	2021/02/09	1497	2.681	0.93756	0.25663
	End of Cruise	2021/02/09	1497	2.681	0.93756	0.25663
Full Science Nominal Mission (1229)	GAM V3	2022/12/15	2171	11.135	0.86979	0.23422
	GAM V4	2024/03/09	2620	19.759	0.83438	0.26963
	ENM (1st perihelion after GAMV4)	2024/06/23	2726	19.759	0.83438	0.26963
Full Science Extended Mission (881)	GAM V5	2025/06/01	3069	25.842	0.78782	0.31619
	GAM V6	2026/08/24	3519	29.311	0.74485	0.35916
	EEM (1st perihelion after GAMV6)	2026/11/20	3607	29.311	0.74485	0.35916

Table 1 Mission Trajectory for a 2017 Launch

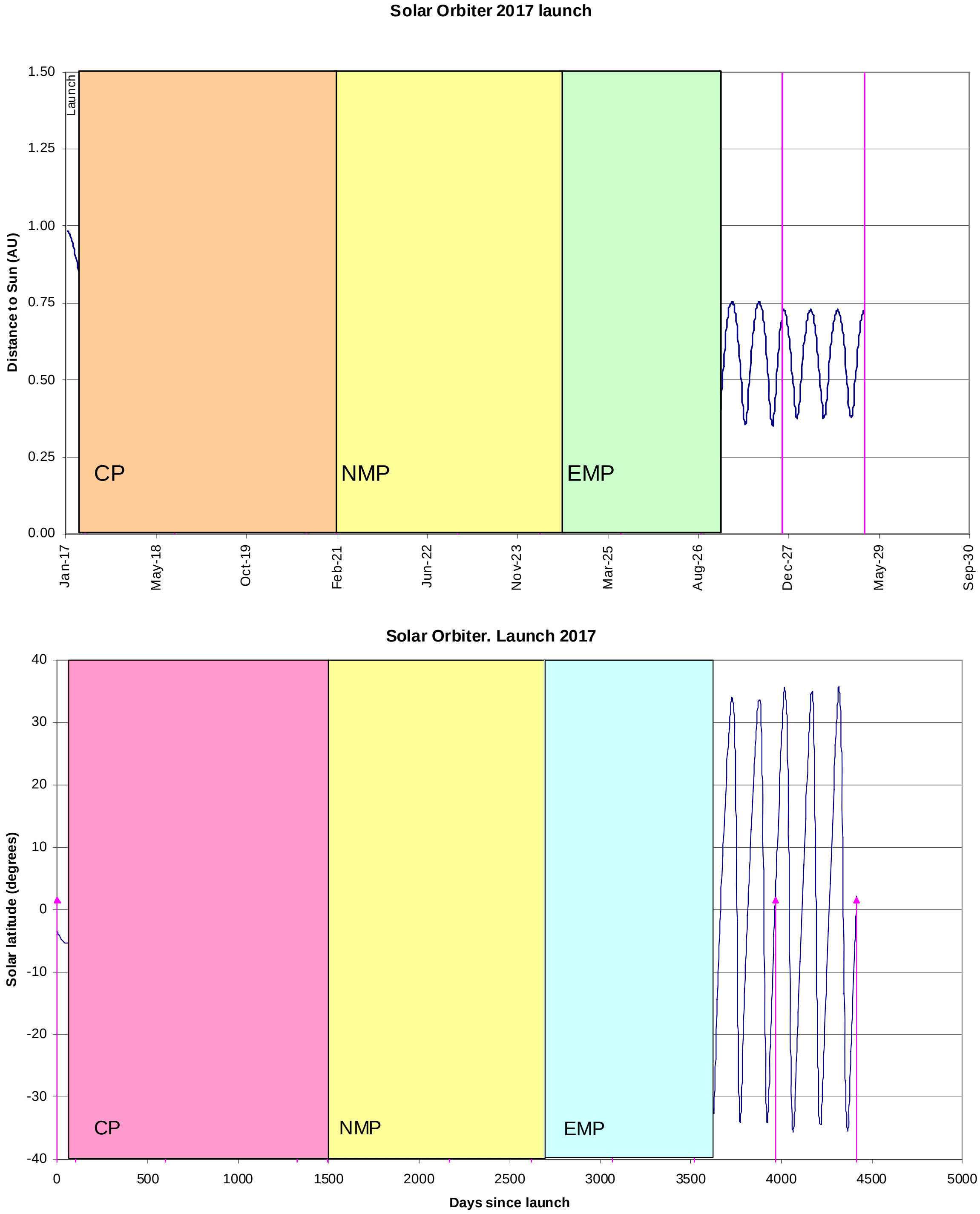


Figure 2 Mission Trajectory for a 2017 Launch (Top: evolution of Sun distance over the mission, Bottom: evolution of helio-latitude over the mission)

Mission Launch	Absolute Minimum Sun Distance (AU)	Absolute Maximum Sun Distance (AU)	Absolute Maximum Earth Distance (AU)	Maximum Ecliptic Inclination (deg)	Maximum Solar Latitude (deg)
2018	0.2434 2022/01/27	1.35266 2019/07/30	1.892128 2021/03/26	26.299 achieved at EEM	33.54 achieved at EEM

Mission Phase Duration (days)	Event	Date (Calendar)	Flight Time (days)	Ecliptic Inclination (deg)	Aphelion (AU)	Perihelion (AU)
LEOP (7)	Launch	2018/07/30	0	4.421	1.01862	0.69281
	End of LEOP	2018/08/06	7	4.421	1.01862	0.69281
CVP (90)	End of CVP	2018/11/04	97	4.421	1.01862	0.69281
Cruise (1139)	GAM V1	2019/01/23	177	3.501	1.35138	0.71832
	GAM E1	2019/12/01	489	0	1.08291	0.47564
	GAM E2	2021/09/30	1158	0.289	1.01668	0.34637
	GAM V2	2021/12/17	1236	3.003	0.86003	0.24378
	End of Cruise	2021/12/17	1236	3.003	0.86003	0.24378
Full Science Nominal Mission (948)	GAM V3	2023/03/12	1686	13.129	0.83329	0.27071
	GAM V4	2024/06/03	2135	20.612	0.78831	0.31546
	ENM (1st perihelion after GAMV4)	2024/07/22	2184	20.612	0.78831	0.31546
Full Science Extended Mission (925)	GAM V5	2025/08/27	2585	24.981	0.74157	0.36226
	GAM V6	2026/11/20	3035	26.299	0.72081	0.38773
	EEM (1st perihelion after GAMV6)	2027/02/02	3109	26.299	0.72081	0.38773

Table 2 Mission Trajectory for a 2018 Launch

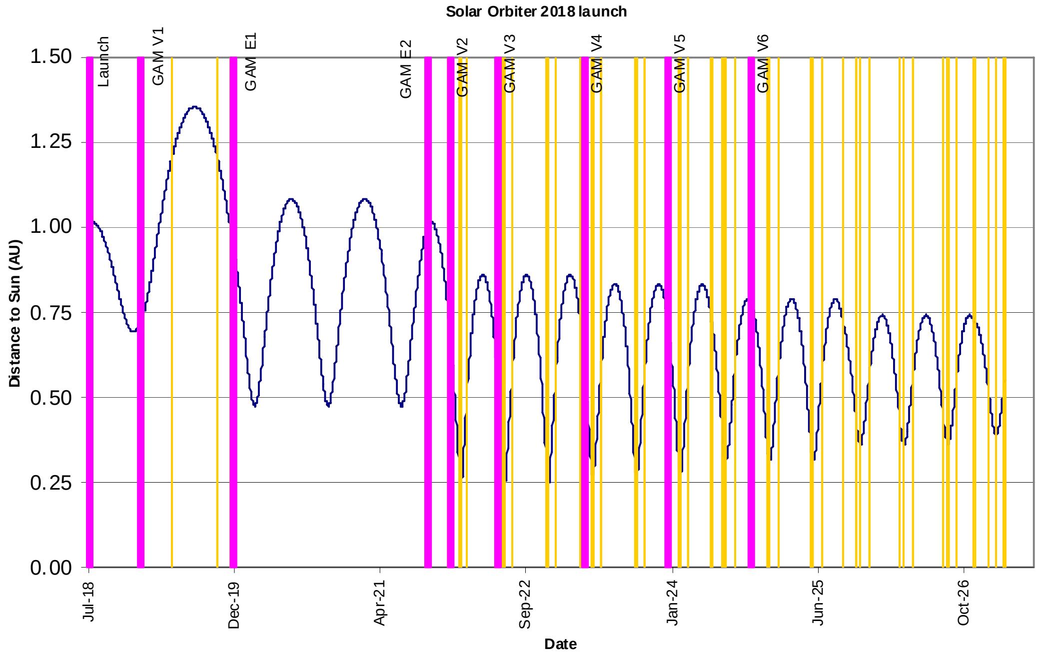


Figure 3 Mission Trajectory for a 2018 Launch (Top: evolution of Sun distance over the mission, Bottom: evolution of helio-latitude over the mission)

During the nominal mission, the Solar Orbiter performance requirements shall be fully met with all specified margins according to the Mission Requirements Document [NR1] for the Solar Orbiter mission.

The duration of the Solar Orbiter extended operational lifetime would be about 2.5 years. Approval of an extension of the mission is dependent on the spacecraft health and status of expendables such as fuel.

3.2.1 GROUND SEGMENT

The Mission Operations Centre (MOC) and the communications infrastructure shall be under ESA/ESOC responsibility. Satellite housekeeping data and payload raw data are archived at the MOC.

The Science Operations Center (SOC) generates payload operations requests that are implemented in the MOC. The SOC receives payload raw data and auxiliary data from the MOC and distributes them to the PIs.

3.3 Spacecraft Main Characteristics

The Solar Orbiter Spacecraft is designed to be compatible with both a launch by an Atlas V (or Delta IV) launch vehicle from Kennedy Space Centre (baseline) and a launch by a Soyuz-ST /Fregat launch vehicle from Kourou, French Guiana (backup).

Solar Orbiter is a three axis stabilized spacecraft. The design includes a heat shield extending on all sides of the front face as a means to protect the spacecraft from the intense solar flux at perihelion.

The remote sensing instruments requiring a solar view shall have an optical feedthrough in the heat shield. The in-situ instruments shall mainly be considered to be in the shade of the sunshield except for short operational exceptions.

Some general spacecraft characteristics can be listed here, although they may be subject to modifications:

- 3-axis stabilized spacecraft.
- Sun pointing.
- Reaction Control System based on reaction wheels and chemical propulsion thrusters.
- Deployable solar arrays with single axis articulation.
- RF subsystem, with one HGA and one MGA for science telemetry downlink (TBC).

3.3.1 COMMUNICATIONS

The spacecraft will be equipped with a combined X/Ka-band transponder providing telemetry in parallel. The LEOP communications are via the X-band LGA. The HGA link will not be available close to the perihelion of the Solar Orbiter orbit and during planetary alignments, eclipses, etc.

The baseline ground station is New Norcia with Cebreros considered as a backup.

3.3.1.1 Data Acquisition

During science operations the in situ instruments will be operating continuously while the remote sensing instruments will have three 10 day operational periods during each orbit. These three periods will take place +/- 5 days around perihelion, and near the positions of highest northern latitude and at highest southern latitude.

To accommodate synoptic observations with NASA's Solar Probe, limited remote sensing observations may be accommodated, outside the three nominal science windows, during the joint periods of operation of the two missions. The additional data acquisition will be defined by the mass memory and telemetry capabilities.

3.3.2 PAYLOAD COMPLEMENT, RESOURCES AND BUDGET CONTROL

From the initial instrument design up to launch, the spacecraft resources allocated to the payload will be controlled according to strict rules in order to show adequate margins, commensurate with the programme milestones.

Such margins will ensure that technical, schedule and financial risks are limited in the interest of all participants of the Solar Orbiter programme. The main resources submitted to margin control are: Mass, Power, Data Rate, thermal and electrical interfaces. A margin philosophy shall be applied, with a contingency depletion scheme under control of the ESA project office.

Mass [kg]	Power [W]	Data Rate [kbps]
180	180	100

Table 3 Spacecraft Payload Allocations

The above table outlines the overall payload resources on Solar Orbiter for mass, power and data rate.

The payload complement consisting of six remote-sensing and four in-situ experiments is shown in the following table.

Instrument	Acronym	Unit	Acronym	Number of Units
Remote-Sensing				
Polarimetric and Helioseismic Imager	PHI	Optics Unit Electronics Box		1 1
Spectral Imaging of the Coronal Environment	SPICE	Optics Unit Electronics Box		1 1
Extreme Ultra-violet Imager	EUI	Optics Unit Electronics Box		1 1
Coronagraph	METIS	Optics Unit Electronics Box		1 1
Spectrometer/Telescope for Imaging X-rays	STIX	Optics Unit Electronics Box		1 1
Solar Orbiter Heliospheric Imager	SoloHI	Optics Unit Electronics Box		1 1
In-Situ				
Solar Wind Analyzer	SWA	Electron Analyzer System	EAS	2
		Proton-Alpha Sensor	PAS	1
		Heavy Ion Sensor	HIS	1
		Electronics Box		1
Radio and Plasma Wave Analyzer	RPW	Antenna	ANT	3
		Search Coil Magnetometer	SCM	1
		Electronics Box		1
Magnetometer	MAG	Out Board Sensor In Board Sensor	MAGOBS MAGIBS	1 1
		Electronics Box		1
Energetic Particle Detector	EPD	SupraThermal Electron sensor	STE	1
		Suprathermal Ion Spectrograph	SIS	1
		Electron Proton Telescope	EPT	2
		Low Energy Telescope	LET	2
		High Energy Telescope	HETn	1
		Electronics Box		1

Table 4 Solar Orbiter Baseline Payload Complement

3.3.3 INSTRUMENT ACCOMMODATION

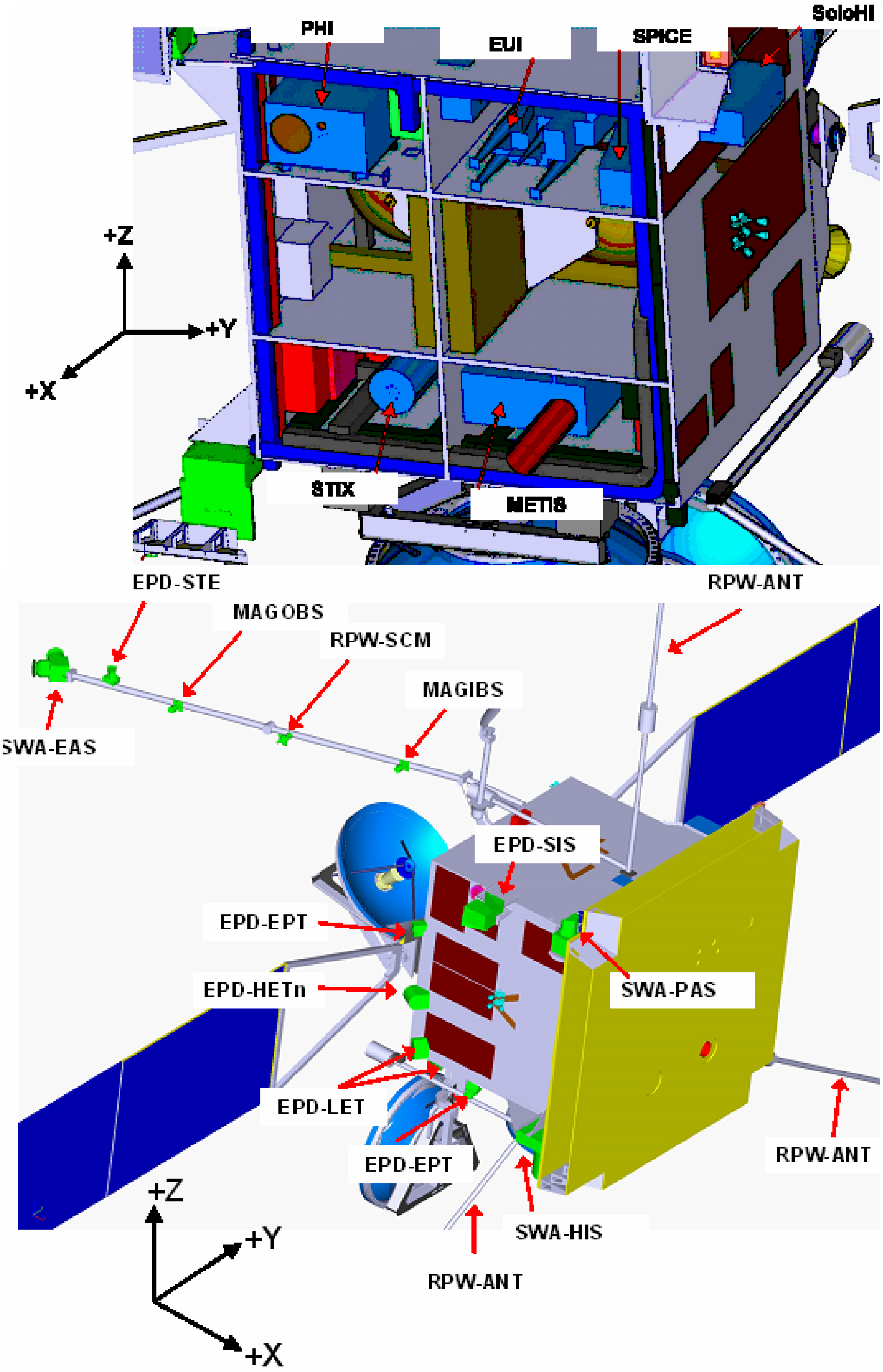


Figure 4 Baseline Accommodation for Remote Sensing (top) and In-Situ (bottom) Instruments

All six of the remote-sensing instruments will be accommodated behind the heat shield.
Five remote-sensing instrument are accommodated inside the spacecraft:

- PHI
- EUI
- SPICE
- STIX
- METIS

The sixth remote-sensing instrument, SoloHI, is accommodated externally on the +Y spacecraft panel.

The in-situ instruments are externally accommodated on the spacecraft Z and Y walls within the protective shadow of the heatshield.

A deployable instrument boom which extends on the anti-sun side of the spacecraft accommodates the following instruments:

- MAGOBS and MAGIBS
- RPW-SCM
- EPD-STE
- SWA-EAS

This boom is to be spacecraft provided.

3.4 Solar Orbiter Thermal Control System

The Solar Orbiter thermal control is based on using a sun pointed, flat heat shield to limit the Sun flux on the spacecraft structure. By using this approach the elements behind the heat shield will be in a more benign thermal environment. The heat shield temperature will be dictated by the final material selection of the front layer and will potentially be as high as 700 °C.

All external components shall be shielded from direct solar illumination by the heat shield except for the instruments requiring direct view of the sun and the spacecraft appendages (i.e. the solar arrays, the RPW antennas and the HGA). The heat shield is sized to prevent direct solar illumination on any of the shaded components during nominal pointing and for safe mode events of spacecraft off-pointing up to 6.5 degrees. Note that the sun radius is ~2.5 degrees and since the spacecraft may be pointed at the solar limb during nominal operations the sunshield is designed to prevent illumination up to an angle of 8 degrees (including margin) onto the spacecraft. Nevertheless, the externally mounted instruments shall need to withstand direct sun illumination for short durations.

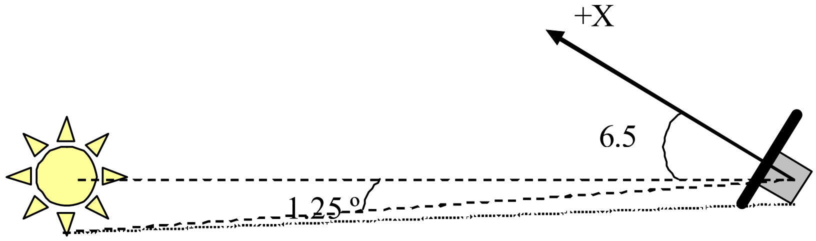


Figure 5 Spacecraft Off-Pointing During FDIR

3.4.1 PAYLOAD UNIT THERMAL CONTROL CATEGORIES

Internally mounted units
These units are mounted inside the spacecraft body and therefore thermally controlled by the spacecraft TCS. They are conductively and radiatively coupled to the spacecraft structure.

It shall be noted that there are internally mounted units, which have protrusions / apertures outside the Spacecraft body. They therefore experience external environmental heat loads. The thermal interface between the protruding elements and the unit is under the design responsibility of the PI.

The following units are accommodated internally:

- PHI
- EUI
- SPICE
- METIS
- STIX
- EPD Electronics Box
- SWA Electronics Box
- RPW Electronics Box
- MAG Electronics Box
- SolOHI Electronics Box

Externally mounted units
Units requiring special exposure to the space environment are mounted outside the spacecraft body. They are thermally decoupled from the spacecraft structure and therefore thermally controlled by the unit itself.

The following units are accommodated externally:

- RPW-ANT
- RPW-SCM

- MAGOBS and MAGIBS
- EPD-STE
- EPD-SIS
- EPD-EPT
- EPD-LET
- EPD-HETn
- SWA-HIS
- SWA-EAS
- SWA-PAS
- SolOHI

Units requiring a heatshield feedthrough interface

The sun pointing remote-sensing instruments that are accommodated within the spacecraft require a feedthrough within the heatshield (see figure below).

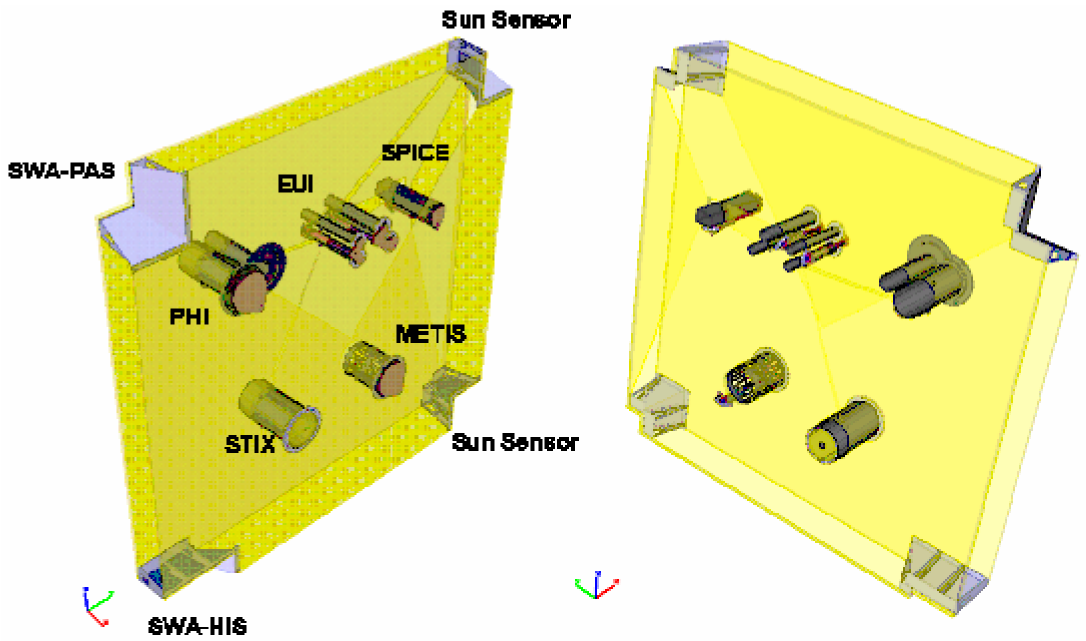


Figure 6 Spacecraft Heatshield with Feedthroughs

The sun-pointing in-situ instruments that require a solar view (and are accommodated at the +Z/-Y and -Z/-Y corners of the heatshield) will also be provided with heatshield feedthroughs.

The following instruments require a feedthrough:

- EUI (x3)
- PHI (X2)
- METIS
- SPICE

•STIX

•SWA-HIS

•SWA-PAS

3.4.2 HEAT SHIELD

The heat shield consists of a high temperature heat barrier and support panel structure arranged to provide two gaps between the spacecraft and the front shield. The front layer is supported by star brackets. The feedthroughs, doors and mechanisms required for the payloads are arranged within the heatshield. The feedthrough is attached to the spacecraft through a series of blades designed to provide isostatic mounting.

The distance between the spacecraft +X panel and the front shield is 400 mm (TBC). The heat shield feedthroughs will provide an unobstructed field of view for solar observation. This feedthrough will limit the out-of-field direct illumination entering the instrument aperture, but will provide only limited stray light reduction (limiting the contribution of feedthrough).

The heat shield and associated feedthroughs and integrated doors will be spacecraft delivered items under spacecraft prime responsibility, while any further baffles, occulters, heat rejecting filters, internal instrument doors, etc., for optical, contamination or thermal reasons will be instrument provided.

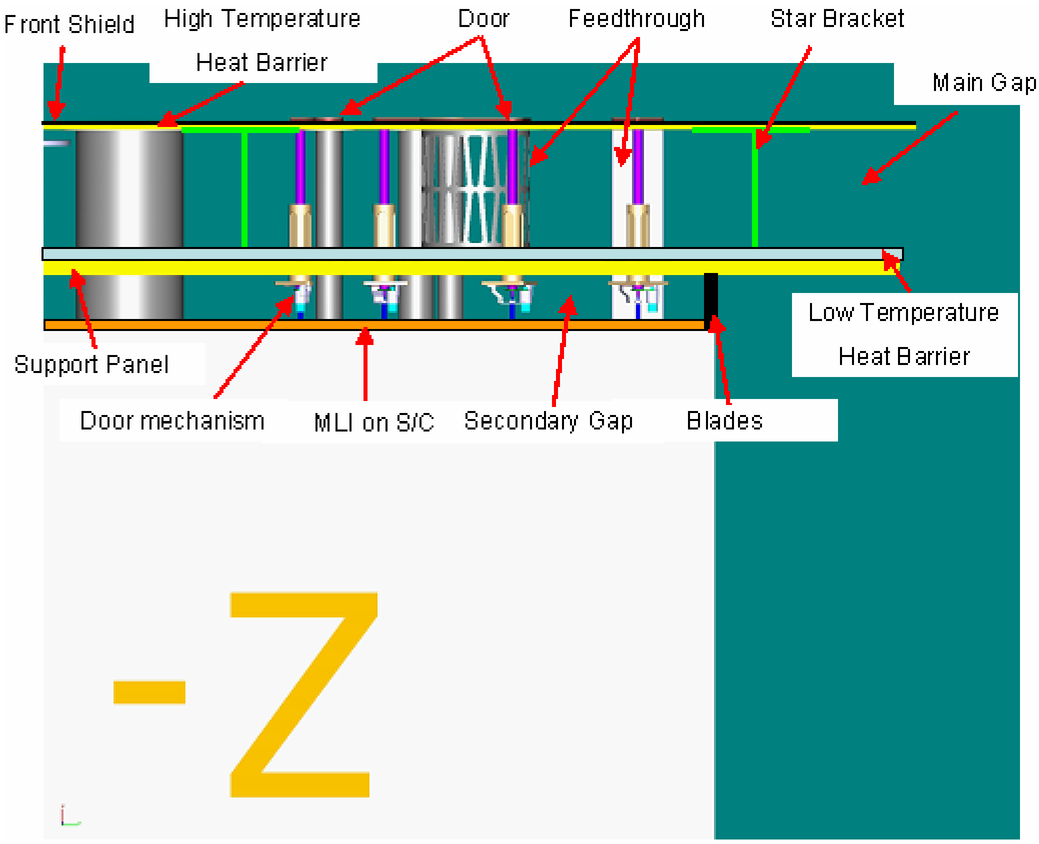


Figure 7 Baseline Heatshield Configuration

3.4.2.1 Heat Shield Door Design

For Sun pointing instruments, a door shall be provided by the spacecraft where required. The door may act as a sunshade to the instrument aperture; however it may not be sufficiently effective for contamination purposes. The door will be opened and closed by the spacecraft operating system. The system shall only support a minimum number of door operations due to the complexity of enabling operations with the high expected heat flux to be encountered by the doors (TBC).

The door (if required) will be made of a material which withstands extreme temperatures and will be compatible with the heat shield. When the door is closed it will reduce the heat flux towards the instrument.

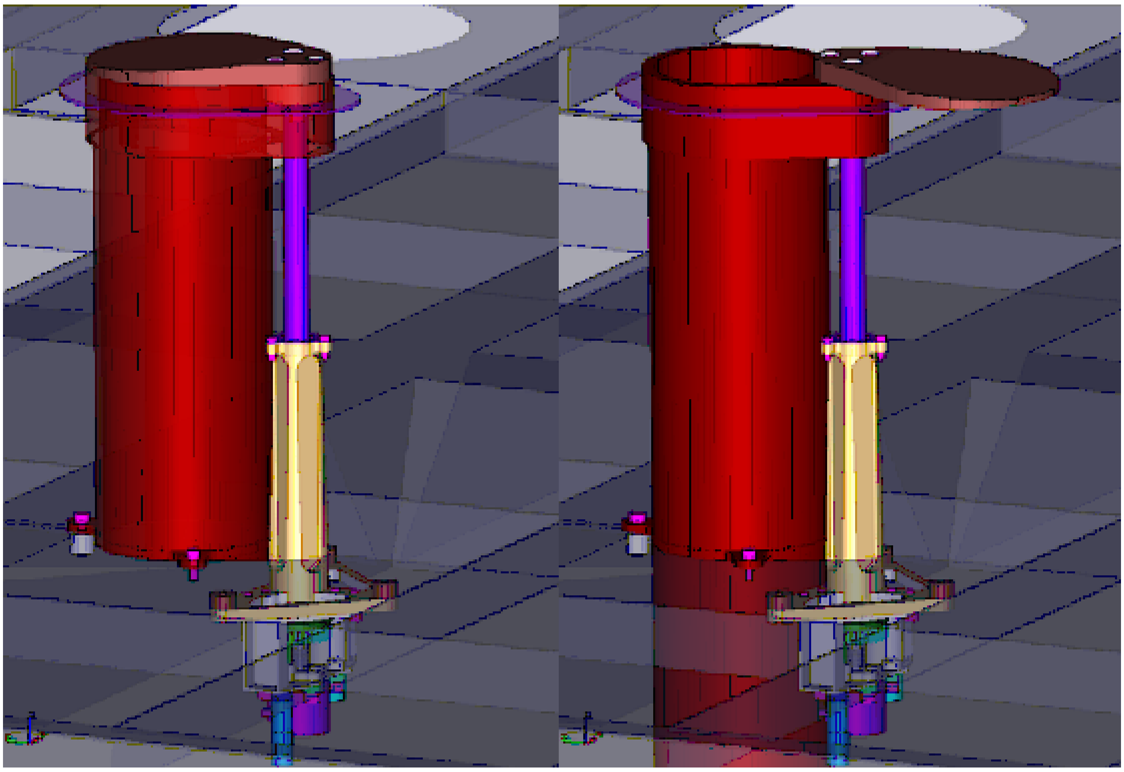


Figure 8 Conceptual Design of Feedthrough in the Heat Shield with External Door

The objective of the overall design is to minimize the number of mechanisms and the complication of the heat shield. Therefore, if a door is required, each instrument shall have a minimal number of mechanisms and a single door may be used to cover multiple apertures in a single instrument. The door must be placed at the sun entrance side of the feedthrough due to thermal constraints. The door mechanism is accommodated in the secondary gap through mounting on the spacecraft side of the support panel.

Although located behind the heat shield, some instruments could be exposed to solar illumination. In the event of a Deep Space Maneuver, during the transfer phase at distances > 0.8 AU the shielded side of the spacecraft could be exposed to the Sun. Of more importance, due to pointing errors or critical failures the shielded side could be exposed to the Sun at a much closer distance to the Sun. This would result in solar illumination impinging on panels that were previously shaded as well as the illumination of internal instrument components that are nominally protected from illumination (for sun pointed instruments). Spacecraft operated doors will not be able to mitigate the impact of this off pointing.

3.4.2.2 Feedthroughs

3.4.2.2.1 Remote Sensing Instruments

A feedthrough is provided for the remote sensing instruments which require an unobstructed view of the sun through the heat shield. The conceptual design of these feedthroughs is shown below.

The feedthrough element performs two functions:

- a thermal baffling function to protect the inside of the heat shield from sun illumination.
- an optical function to provide the necessary unobstructed field of view to the instrument while minimizing the additional straylight generated by the feedthrough itself (if required by the instrument).

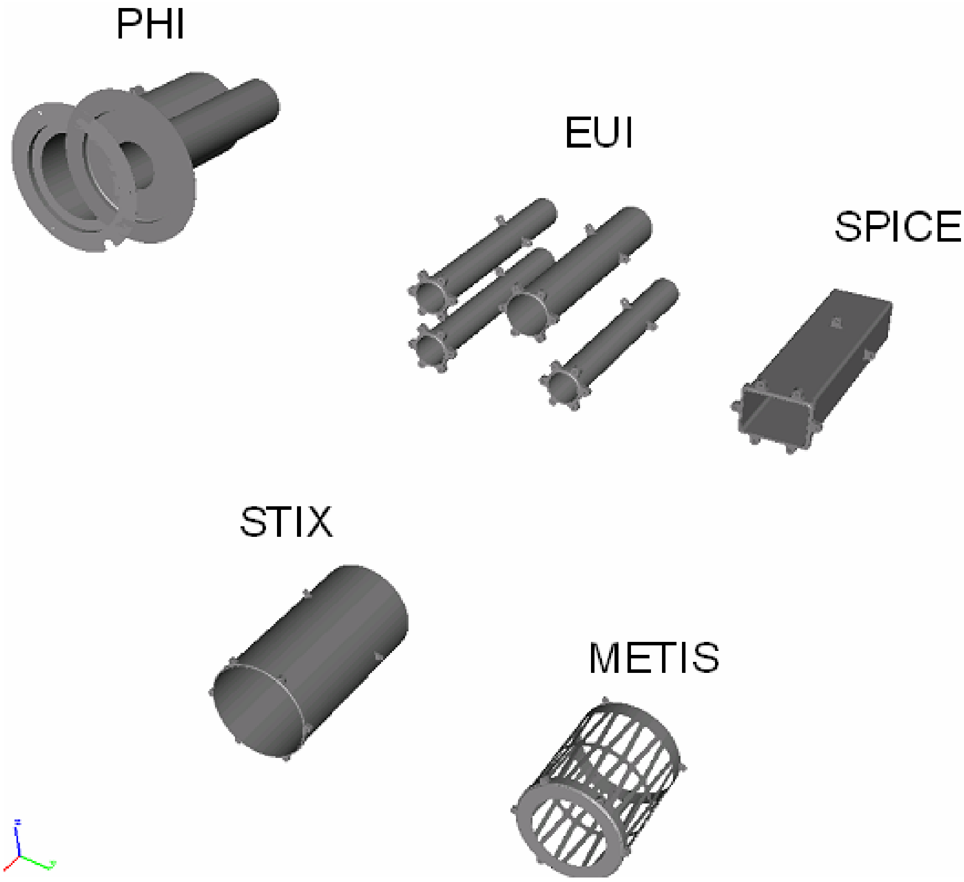


Figure 9 Conceptual Design of Remote Sensing Instrument Feedthroughs

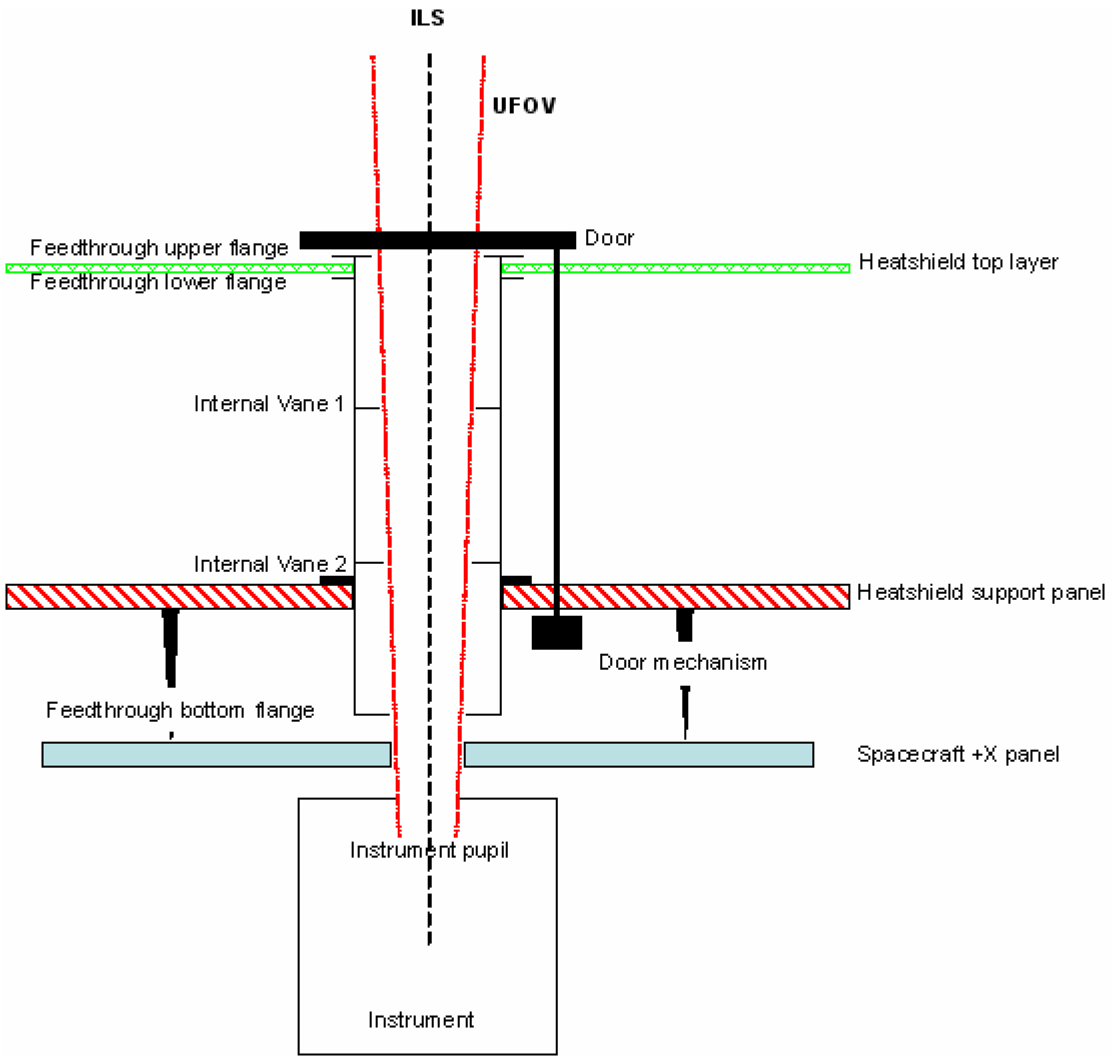


Figure 10 Generic Feedthrough Layout for Remote Sensing Instruments

Out-of-field stray light should be attenuated through two reflections from a diffuse blackened surface before entrance into the instrument pupil. The vane design produced is based on preventing unwanted reflections from the feedthrough walls entering the instrument pupil for light incident up to 2.5° from the spacecraft $+X_{opt}$ axis. This baseline places two vanes inside the feedthrough which, when coupled with the top and bottom feedthrough flanges, provides stray light attenuation for the most intense stray light source, the solar disc. The placement of two vanes within the feedthrough is in line with the basic optical function which the feedthrough will provide.
Vane designs have been produced for SPICE, PHI and EUI.

The feedthrough will endure high temperatures, potentially as high as 350˚C (The equivalent temperature is an average of the temperature gradient along the feedthrough.) These high temperatures have important consequences for the thermal behaviour of the instrument due to the additional IR heat flux.

An important aspect of this interface is that the aperture at the bottom of the feedthrough (i.e. instrument interface) will need to be oversized in order to ensure that the unobstructed field of view (UFOV) is provided at all times. This oversizing is necessary due to the following reasons:

- The translation and rotation of the instrument, and thus instrument pupil, due to the thermoelastic deformation of the spacecraft structure
- The translation and rotation of the feedthrough due to the thermoelastic deformation of the heat shield support panel
- The thermal expansion of the feedthrough
- The mounting/integration tolerances of instrument and feedthrough

The consequence of this oversizing is that solar flux will impinge directly onto the instrument outside of the instrument pupil. This is illustrated in the figure below.

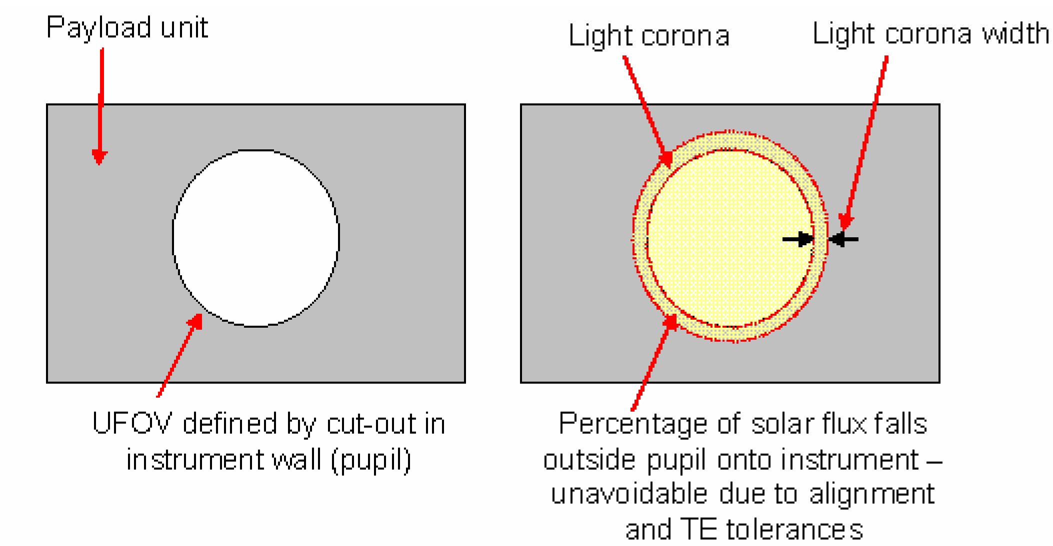


Figure 11 Solar Flux Impinging on an Instrument Outside of the Instrument Pupil

3.4.2.2.2 In-Situ Instruments

In-situ instruments that require a FOV towards the Sun will also have feedthroughs in the heat shield. The SWA PAS and HIS instruments require large FOVs towards the Sun. Therefore, both instruments have been placed at the -Y corners of the heat shield, to minimise the heat shield cut out. These corner feedthroughs are only partially enclosed with two sides open to space. This enables a reduction in the feedthrough temperatures. With this partial enclosure of the feedthrough the spacecraft heat shield will not fully protect the instrument during nominal pointing. As such, it is the responsibility of the PI to provide additional thermal protection of the area of the instrument that is exposed to direct solar flux.

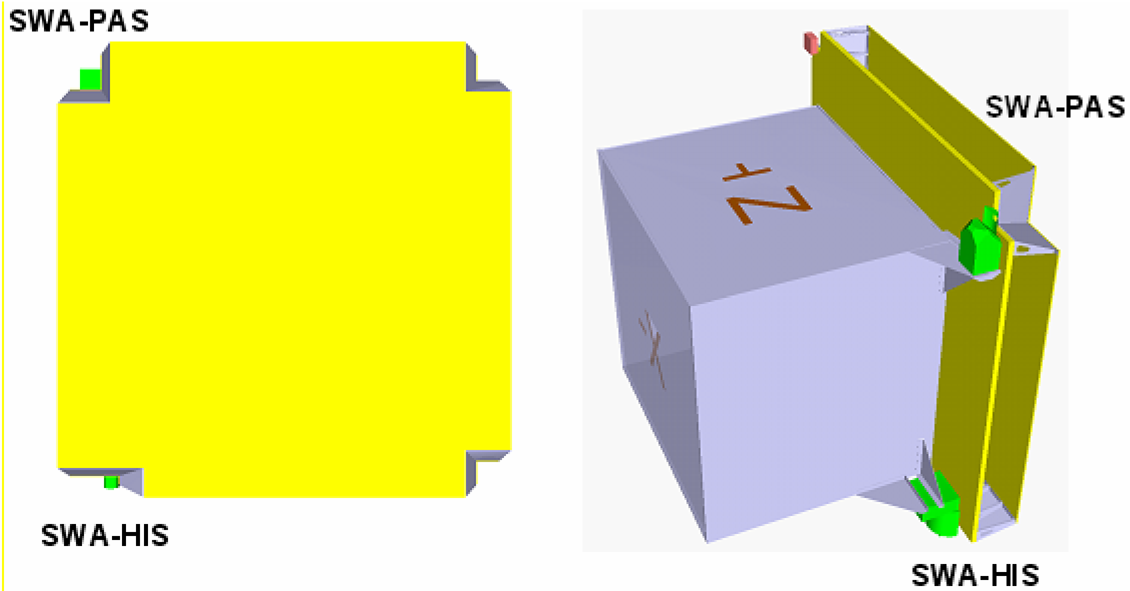


Figure 12 Corner Feedthrough Concept for In-Situ Instruments

3.4.2.3 Occulter

If an instrument has the need for an occulter, such as the coronagraph, it will be instrument provided and directly attached to the instrument to provide the necessary alignment. An occulter in the centre of an instrument's FOV will receive a large direct thermal flux (circa 80W) at perihelion (0.22AU).

Due to the strict straylight requirements of a coronagraph instrument, the vanes for straylight protection will also be instrument provided. Specific interfaces will be provided by the spacecraft, through a heat shield thermal baffle or by other means. The alignment requirements for both the vanes and occulter shall therefore be maintained under PI responsibility. The spacecraft will provide a feedthrough ({Figure 3.5-9}) which will support the heat shield layers such that the occulter alignment with respect to instrument optical axis is not affected.

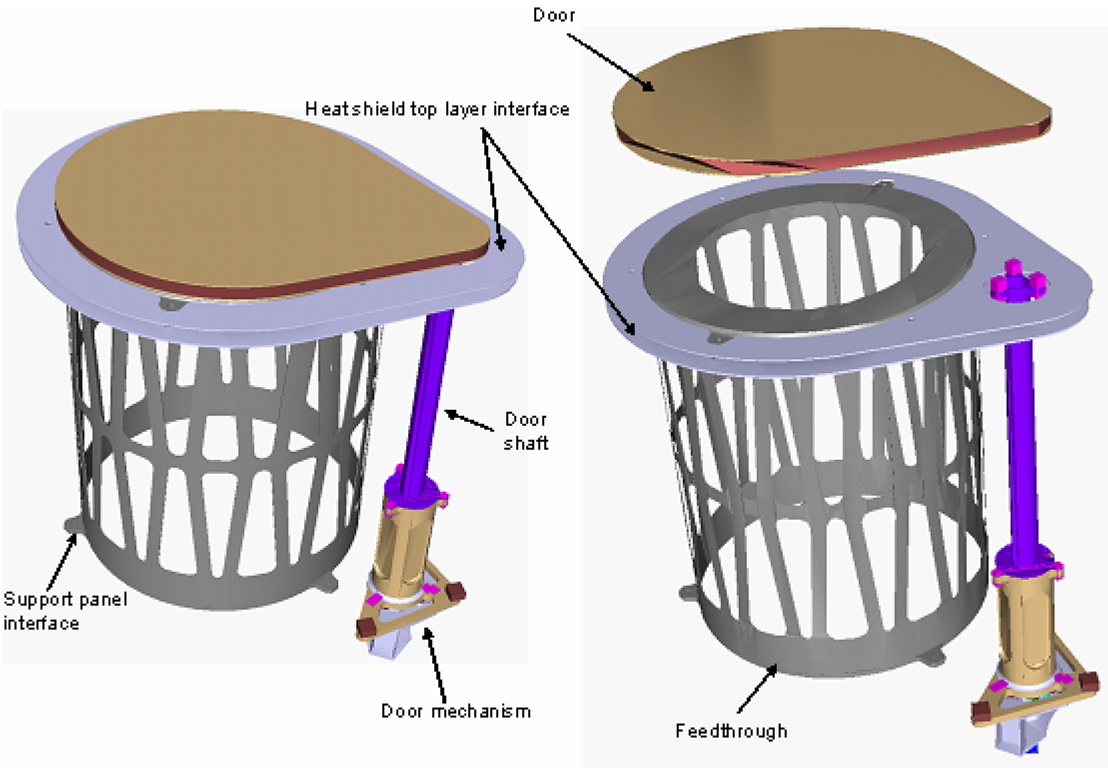


Figure 13 Conceptual METIS Feedthrough Design

3.4.2.4 Filters, Sun Shades, Windows, etc.

If a heat rejecting filter (ref. PHI), sun shade (ref. STIX) or any other type of window are required to be mounted in the heat shield for thermal reasons, it will be provided by the PI. The interface shall be defined by the spacecraft.

Conceptual interface designs are shown in the figure below.

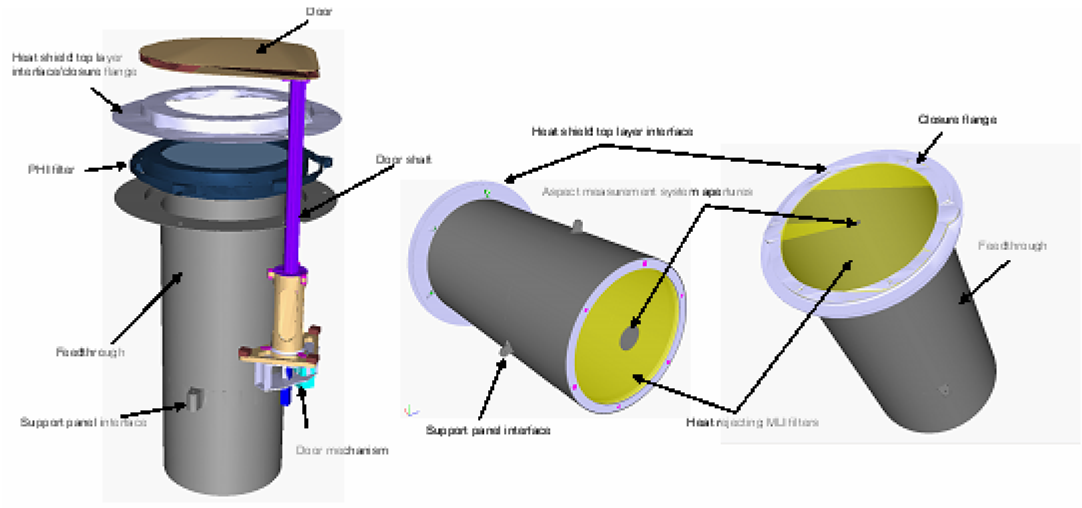


Figure 14 Conceptual Design of PHI Filter and STIX Sun Shade Mounting

3.4.2.5 Integration of Instrument with Heat Shield

Any element, such as the doors and baffles that will be located in the heat shield will be considered as part of the main spacecraft. Therefore, the instruments will be without these elements prior to integration with the spacecraft. Careful consideration of this problem is required such that the instrument is designed to be developed and tested by the PI taking into account these elements.

The PI shall consider all aspects related to the design, cleanliness, AIV, etc. that are affected by this late integration with the heat shield.

3.4.3 REMOTE SENSING INSTRUMENT THERMAL CONTROL

The remote sensing instruments will be accommodated behind the heat shield with feedthrough provided to allow the required FOV to each instrument. The thermal control for these instruments has a separate hot element interface connected to the hot elements such as heat stop, primary mirror etc. These hot element interfaces are connected to dedicated radiators using heat straps, heat pipes, etc. This interface shall be located outside the instrument box.

Some detectors require low temperatures and would have a cold finger interface, also with a separate dedicated radiator. This cold finger interface shall be located outside the instrument structure. The spacecraft would provide the required interface temperature using similar hardware as for the hot element interface (TBC).

The assumptions for the current remote sensing thermal design philosophy are shown in the figure below.

The three dedicated radiators;

- Hot element radiator (high temperature)
- Instrument box radiator (medium temperature)
- Cold finger radiator (low temperature)

will run at different temperatures to provide the required interface temperatures for the units.

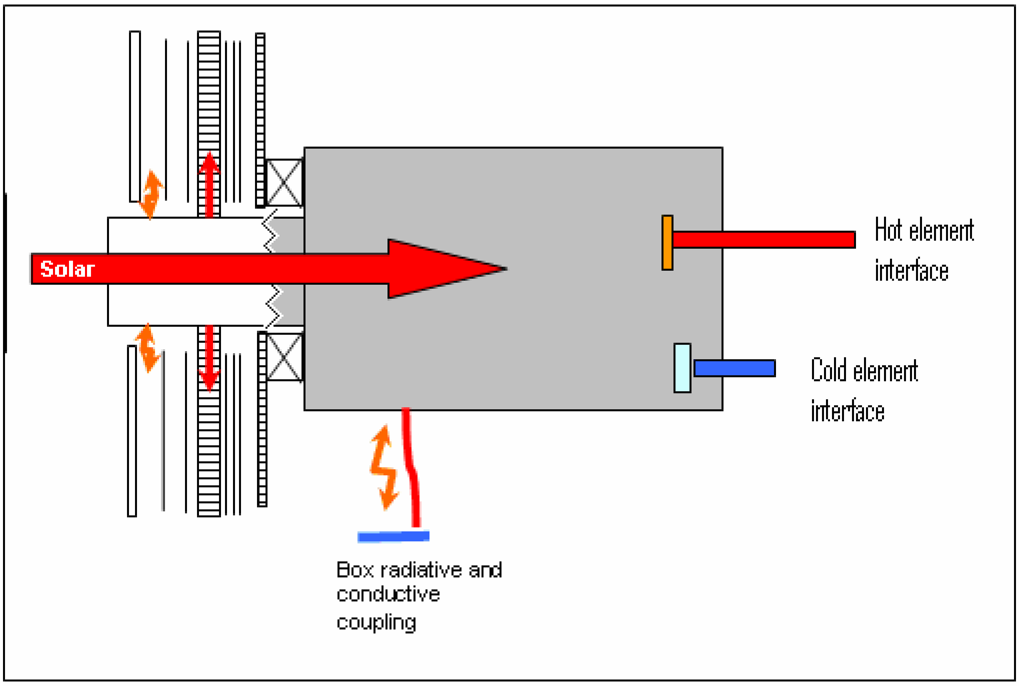


Figure 15 Remote Sensing Instruments General Thermal Design Philosophy

Any heating internal to the instrument will have to be performed by the instrument itself. Instruments shall size their maximum internal heating power taking into account the maximum spacecraft to sun distance (in the order of 1.5 AU) and the instrument requirements regarding door operation (if required), that could result in negligible external heat input.

3.4.4 THERMAL CONTROL OF IN-SITU INSTRUMENTATION

Any externally mounted instrument shall be thermally decoupled from the spacecraft structure and the thermal control of these instruments shall be done under the responsibility of the PI. This implies that they would need to be shielded from reflected radiation from the spacecraft appendages (i.e. antenna, solar array etc.) and from potential transient cases of direct Sun illumination when off-pointing the spacecraft. In addition, any required heaters shall be under PI responsibility.

3.5 Satellite Data Handling Architecture

The onboard data handling (OBDH) system, also called Data Management System (DMS) fulfills the traditional Command and Data Management Unit (CDMU) and the Attitude Control Computer (ACC) functions. In Figure 16 the Payload Data Management unit of the DMS is shown with the interfaces to the instruments. With respect to the instruments the DMS represents the gateway to ground. In detail the DMS provides:

- Instrument Command and Control
- Command Timeline Management
- Packetized Science Data Acquisition
- Intermediate (Mass Memory) Storage and Transmission to Ground
- Health and Safety Monitoring when instruments are off

The OBDH bus architecture is based on the Spacewire standard providing services to instruments as defined in the ECSS-E-50-12 standard [NR2].

The instruments shall interface with the S/C data handling system using a dual cold redundant Spacewire Interface for transmission of TM and reception of TC packets.

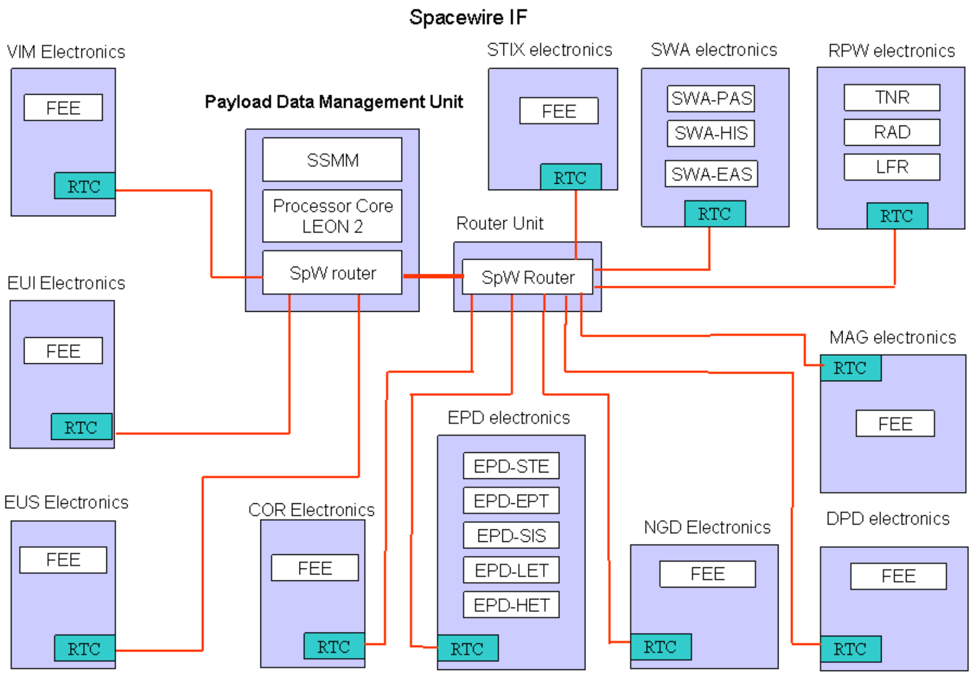


Figure 16 Potential Interface between Instruments and Platform DHS through a Data Management Unit and the Adoption of SpaceWire Interface

3.5.1 REMOTE TERMINAL

The Remote Terminal Unit could be implemented by using the SpaceWire RTC ASIC, which is currently under ESA development. However, the RTC is considered to be a PI provided item.

The performance of the RTC is targeted to be in the order of 50 MIPS. Power consumption should be less than 500mW and the mass less than 100g. The RTC contain one FPU. The floating point processing power will be in the order of 10 MFLOPS.

Figure 17 below shows a block diagram of the SpaceWire RTC.

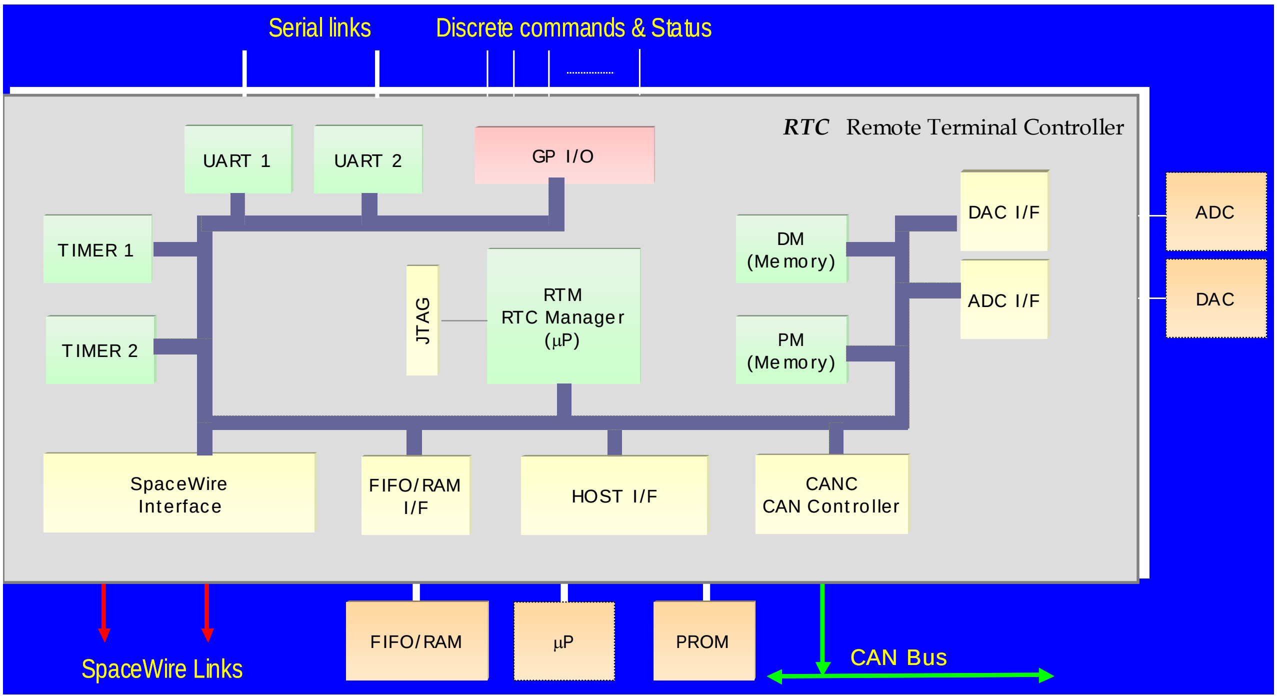


Figure 17 SpaceWire RTC Block Diagram

3.6 Description of Spacecraft Dynamic Environment

The pointing accuracy (error) and stability (drift) of the spacecraft are outlined in section 4.4.2.

3.6.1 SPACECRAFT DISTURBANCES

The Solar Orbiter Relative Pointing Error (RPE) will have several disturbances that will affect its performance. This is mainly due to moving components in the spacecraft.

At these fine-pointing levels, the AOCS wheels will produce vibrations, especially at high speeds (> a few thousand rpm). Depending on the spacecraft design, this may degrade the pointing stability performance (RPE), exceeding the specification. Operational procedures limiting the maximum wheel speed will consequently need more frequent momentum off loadings, during which times the RPE would be briefly (TBC minutes) exceeded.

An additional moving component that will contribute to pointing disturbances is the operation of the HGA, which, during movement, will create disturbing torques on the spacecraft.

The solar arrays have to be adjusted to maintain their temperature to below the maximum specified operating temperature, whilst still generating sufficient power. Presently the arrays do not require moving more than once per day (TBC).

4 INSTRUMENT INTERFACES

4.1 Identification and Labelling

1. Each instrument unit is required to bear a unit identification label containing the following information:
   - Project code
   - Unit identification code
   - Model (e.g. STM, PFM, FM)

2. The identification label shall be attached to each instrument unit at a location that guarantees maximum visibility.

3. The location and content of the instrument unit's identification label shall be shown on the external configuration drawing(s) of the respective unit.

4. The identification label shall be clearly legible.

4.1.1 PROJECT CODE

1. For each instrument the Project code, which is the normal reference used for routine identification in correspondence and technical descriptive material, is defined in chapter 4.1 of the EID-B.

4.1.2 UNIT IDENTIFICATION CODE

1. The Unit Identification Code is allocated in accordance with a computerised configuration control system and also for connector and harness identification purposes. The first 6 characters of this code are the allocated Project code.

2. The unit identification code is composed of 3 parts:
   - 3 characters for instrument identification, (e.g. VIM, EUS, EUI, etc.)
   - 3 characters for unit identification, (e.g. DPU)
   - 2 characters for model identification, i.e. ST for Structural Thermal Model, QM for Qualification Model, FS for Flight Spare Model, FM for Flight Model, etc.

3. For connector and harness this code is limited to the characters up to unit identification, followed by the connector identification (see below).

4.1.3 CONNECTOR IDENTIFICATION

1. Each equipment box is required to bear visible connector identification labels closely adjacent to the appropriate connector. Spacecraft philosophy is to locate a “J” character to all units fixed (hard mounted) connectors and a “P” character to all harness mounted connectors, followed by a 2 digit number. After this number, an additional character identifies the type of contact, “P” for male and “S” for female contact. Each unit is treated individually in this respect, starting at “J01"for unit fixed connectors.

2. For full connector identification these three alphanumeric characters are preceded by the S/C identification code of the instrument unit, e.g. connector “J03S"on box the VIM DPU will have the full reference “VIMDPUJ03S", the mating harness connector will have the reference “VIMDPUP03P".

3. Since the S/C identification code already appears on the unit identification label however, unit fixed connectors are not required to bear the full connector identification code; in the example above “J03S" would suffice. The same rules apply for supplied instrument interconnect harnesses, and harness from an instrument EGSE if it requires connection to test connectors on an instrument unit.

4. The location and content of the above described identification labels shall be included in the external configuration drawing.

4.2 General Design Requirements

4.2.1 STANDARD METRIC SYSTEM

Drawings, specifications and engineering data shall use the International System (SI) Metric Standard, with the exceptions allowed in the ECSS-E-30 Part 1A - Table E-3 and 5 [NR3]. The key and derived units shall be specified in:

- Dimensions in Millimetres [mm]
- Angles in degrees
- Temperatures in degrees Celsius
- Power / Heat in Watts [W]
- Energy in Joules [J]
- Mass in Kilogram [kg]
- Magnetic Field in Tesla [T]
- Time in seconds [s]
- Electric Current in Ampere [A]
- Amount of substances in moles
- Luminous Intensity in candelas

4.2.2 LIFETIME REQUIREMENTS

1. Design lifetime requirements shall be applied (if not specified differently elsewhere in the documentation for mechanical, thermal and electrical design) with respect to environmental influences and use conditions.

2. Where the design margin is required for demonstration of resistance to failure modes, a factor of two times the nominal life time shall be included as a minimum.

3. In the frame of the instrument design the following life time requirements shall be made applicable to all parties involved in the instruments:

   - The shelf-life time shall be compatible with a launch delay of 2 years from the nominal launch date (Ground Environmental Influence)
   - The overall instrument life time shall be compatible with the nominal mission duration of 9 years in space.
   - An extended operational life time of 2 years beyond the nominal mission life is desirable. Provided technically and financially feasible the payload design shall be compatible with this goal (Space Environmental Influence and Use Conditions).
   - For items which degrade with usage the life time shall be two times the nominal operational life time. Exceptions are the mechanisms where specific requirements apply.

4.2.3 MAINTAINABILITY

1. The equipment shall be designed to require a minimum of special tools and test equipment to maintain calibration, perform adjustment and accomplish fault identification.

2. Items to be removed before flight (red tag) shall be visible after integration with the spacecraft.

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

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

4.2.4 FAULT TOLERANCE

1. The instrument design shall use redundancy as the main means of designing fault tolerance.

2. In the event of a failure an instrument shall not automatically reconfigure in order to continue operations.

3. The instrument shall go to a safe mode in the event of a failure and TM records of the fault shall be transferred to the spacecraft DMS.

4. Recovery from safe mode shall be done through ground telecommand.

5. Other requirements related to Fault Tolerance are TBW.

4.3 Co-ordinate System

4.3.1 S/C REFERENCE COORDINATE SYSTEMS

4.3.1.1 Solar Orbiter Physical Reference Frame

The Spacecraft Co-ordinate Systems are axis reference frames physically attached to the respective spacecraft.

All reference frames shall be right-handed orthogonal triads.

The Solar Orbiter Physical Reference Frame shall be as defined in Table 5 (TBC).

Item	Definition
Origin	Point of intersection of the launcher longitudinal axis ($+X_{LV}$) with the separation plane between the launcher and the composite.
$+X_{SO}$ (Roll Axis)	Longitudinal axis of Solar Orbiter, pointing from the Origin towards Solar Orbiter, positive upwards (launcher in vertical position), coinciding with the +X axis ($+X_{LV}$) of the launcher.
$+Y_{SO}$ (Pitch Axis) $+Z_{SO}$ (Yaw Axis)	Transverse axes, completing the right-handed orthogonal triad ($Z_{SO} = Y_{SO} * X_{SO}$).

Table 5 Solar Orbiter Physical Reference Frame

4.3.1.2 Solar Orbiter Optical Reference Frame

The Solar Orbiter Optical Reference Frame shall be as defined in Table 6.

Item	Definition
Origin	Geometrical centre of the Optical Plane. The Optical Plane is the face of Solar Orbiter, supporting the instruments looking at the Sun.
$+X_{Opt}$	Longitudinal axis of Solar Orbiter, pointing from the Origin towards the Sun
$+Y_{Opt}$ $+Z_{Opt}$	Transverse axes completing the right-handed orthogonal triad ($Z_{Opt} = Y_{Opt} * X_{Opt}$). The y-axis shall be nominally along the velocity direction (-Y being in the velocity direction), the z-axis is normal to the orbit plane.

Table 6 Solar Orbiter Optical Reference Frame

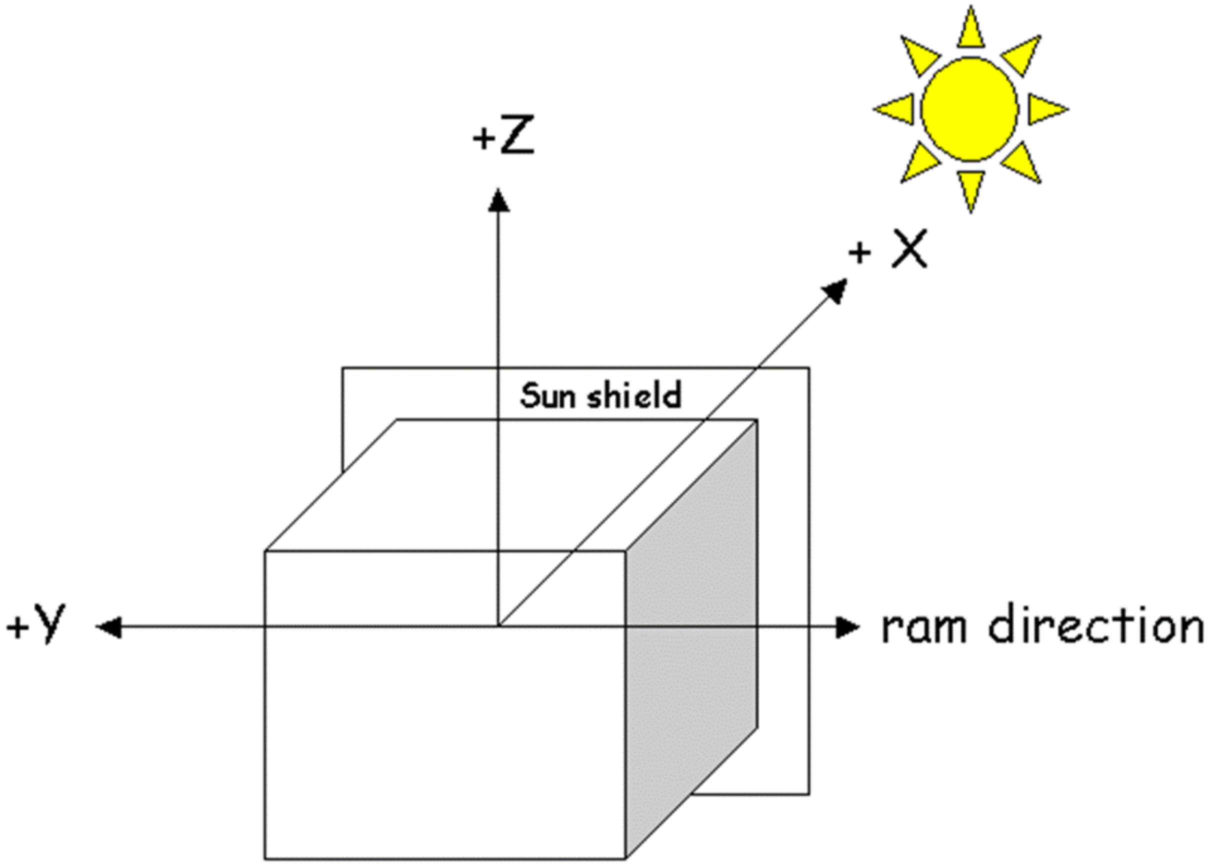


Figure 18 Optical Reference Frame

Note:

The correspondence of the –Y axis with RAM (velocity) direction is only appropriate when the spacecraft is orbiting at the minimum and maximum heliocentric radii (and in absence of any de-pointing with respect to the Sun centre). During science operations the +X-axis may be off-pointed from the Sun centre by up to ±1.25 ^{o} (TBC). This implies spacecraft slewing to ensure that offset pointing is maintained during offset observations up to the solar limb.

4.3.1.3 Unit co-ordinate System

Unit reference Frame (URF) [Xu, Yu, Zu]

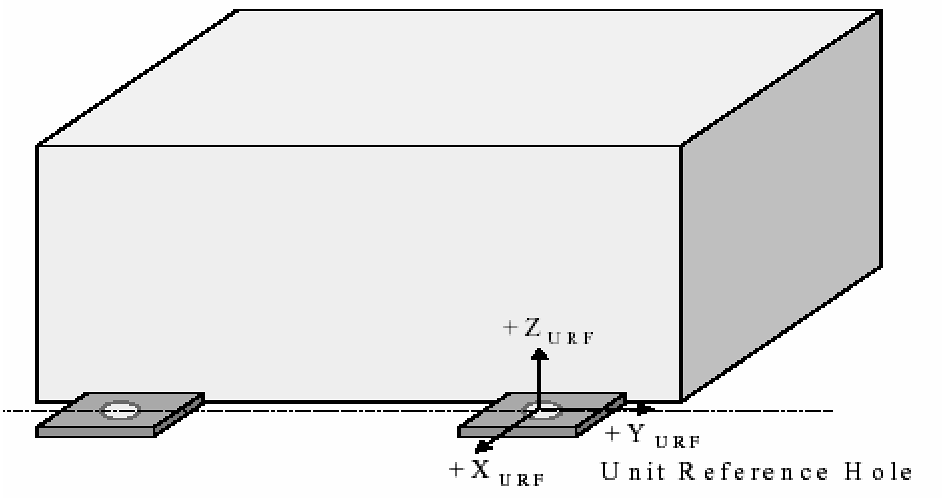


Figure 19 Unit Reference Frame (URF)

1. In order to provide a local reference system for describing the unit physical properties each instrument unit shall have a right-handed cartesian coordinate system see Figure 19.

One option for the URF system is defined as follows:

- the origin is located in the centre of the reference hole on the interface plane. (The reference hole can be freely selected as best suited for the instrument)
- the mounting plane is the plane of the unit that is to be attached to the spacecraft and it shall contain the Xu and Yu axis.
- the +Zu axis is normal to the mounting plane in the direction from the mounting plane to the unit.
- the Yu axis should pass through the centre of at least one other mounting hole in addition to the one located at the origin.

2. The definition of the unit co-ordinate system shall be contained in the unit configuration drawings.

4.4 Instrument Location and Alignment

4.4.1 INSTRUMENT LOCATION

The baseline instrument locations are summarised in the table below.

Instrument	Unit	Location	S/C panel	Notes
PHI	Optics Unit Electronics Box	Internal Internal	+Z/-Y shear panel +Z/-Y shear panel	
SPICE	Optics Unit Electronics Box	Internal Internal	+Z/+Y shear panel +Z/+Y shear panel	
EUI	Optics Unit Electronics Box	Internal Internal	+Z/+Y shear panel +Z/+Y shear panel	
METIS	Optics Unit Electronics Box	Internal Internal	-Z/+Y shear panel -Z/+Y shear panel	
STIX	Optics Unit Electronics Box	Internal Internal	-Z/-Y shear panel -Z/-Y shear panel	
SoloHI	Optics Unit Electronics Box	External Internal	+Y wall +Y wall	S/C provided bracket
SWA	EAS	External	Boom	S/C provided instrument boom
	PAS	External	-Y wall	S/C provided bracket
	HIS	External	-Y wall	S/C provided bracket
	Electronics Box	Internal	+Z/-Y shear panel	
RPW	ANT	External	+Z/-Z walls	1 ANT on +Z, 2 ANT on -Z
	SCM	External	Boom	S/C provided instrument boom
	Electronics Box	Internal	Center panel	
MAG	Out Board Sensor In Board Sensor	External External	Boom Boom	S/C provided instrument boom
	Electronics Box	Internal	Center panel	
EPD	STE	External	Boom	S/C provided instrument boom
	SIS	External	-Y wall	S/C provided bracket
	EPT	External	-Y wall	S/C provided bracket
	LET	External	-Y/-Z walls	S/C provided bracket
	HETn	External	-Y wall	S/C provided bracket
	Electronics Box	Internal	Center panel	

Table 7 Baseline Instrument Unit Positions

4.4.2 INSTRUMENT ALIGNMENT

4.4.2.1 Definitions

Unit Reference Frame (URF) [X_U,Y_U,Z_U]
The Unit Reference Frame is defined in section 4.3.1.3.

Unit Optical Alignment Frame (UOAF) [X_{UO},Y_{UO}F,Z_{UO}]
The Unit Optical Alignment Frame (UOAF) is defined as a right handed, orthogonal coordinate system used for alignment purpose as follows:

- the OUAF origin is located at the centre of the alignment cube
- the X, Y, Z axes are identified by the normal to the mirror faces of the optical alignment cube.

Unit Vertex
The Unit Vertex is defined as the Origin of the field of View in the URF coordinate system.

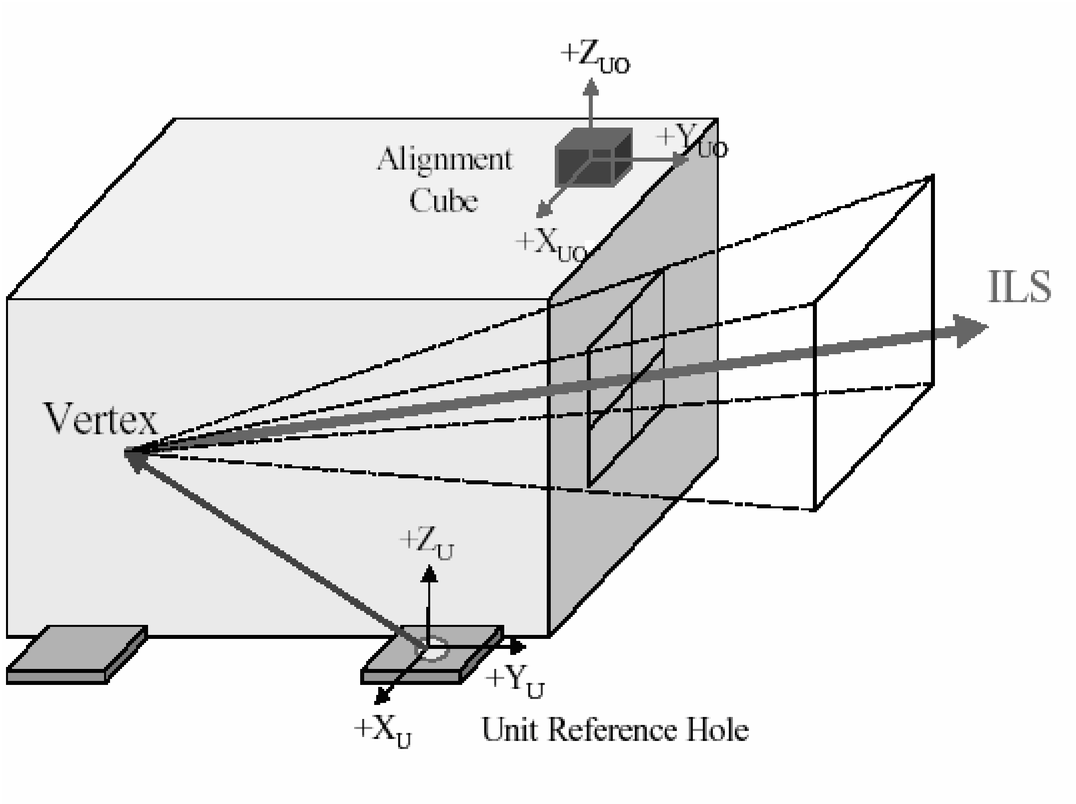


Figure 20 Unit Vertex

Unit Field of View
The Field of view is defined as being the angular opening of the instrument's viewing field used for scientific observations, measured from the Unit Vertex.

Unit Unobstructed Field of View
The unobstructed field of view is defined as being the angular opening of the instrument's viewing field, in which no material (especially reflecting materials) shall be located during the accommodation. It is at the discretion of the PI to define a different Vertex for the UFOV.

Instrument Line of Sight (ILS)

The Instrument Line of Sight (ILS) is defined as being the centreline of the instrument Field of View and described in terms of Azimuth (Azi (ILS)) and Elevation (Ele (ILS)) such that:

- Azimuth is the angle between the ILS projection in the Xu-Yu plane and the +Xu-axis. This angle is positive from +Xu to +Yu within the range 0 up to 360 degrees.
- Elevation is the angle between the ILS direction and its projection on the +Zu direction. This angle is counted positive from the +Zu axis towards the Xu-Yu plane being positive in direction +X and negative in direction -X. The angle varies in the range ± 90 deg.

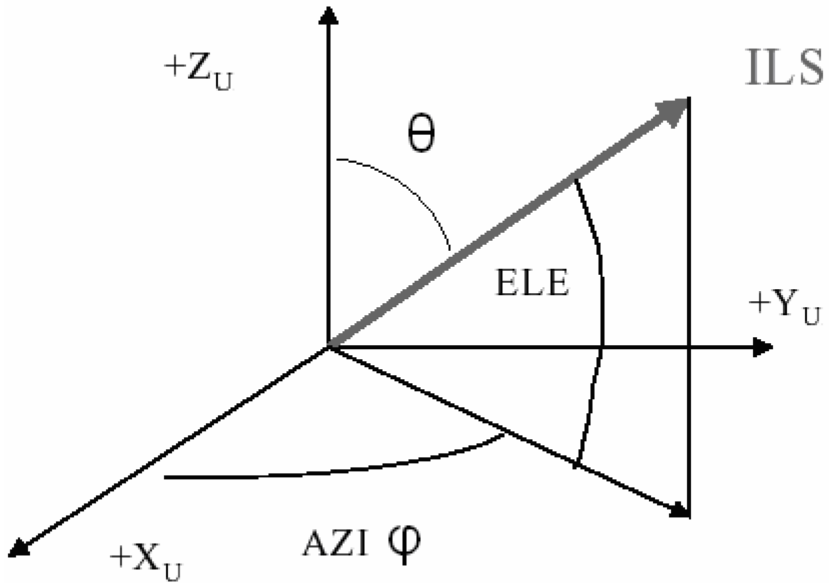


Figure 21 Instrument Line of Sight (ILS)

4.4.2.2 Pointing Definitions

1. The following pointing terminology shall be applied:
   - PRB: Pointing requirement of ILS with respect to the target (may be related to s/c reference frame later in the Project, this is the commanded viewing direction).
   - APE: is the absolute pointing error as a difference APE(t) = Actual Pointing(t) - PRB for the instantaneous value, the requirement should define the envelope as the maximum allowed deviation.
   - RPE: is the allowed relative pointing error during a given integration time. It is calculated as the allowed deviation envelope from the median value during the integration time.
   - PDE: is the allowed drift, defined for an interval .t between two observation points. It is calculated from RPE2 - RPE1 (median values, for RPE see below).
   - AME: This is the absolute measurement accuracy (knowledge) for the boresight axis, derived from attitude reconstitution data.

Note: that all the above parameters (except PRB) shall be defined at a 95% (2s) confidence level.

4.4.2.3 Pointing and Alignment Approach

Absolute Pointing Error APE (and similarly APD, RPE, AME)

This section shall provide an overview about the pointing and alignment philosophy applied for the instruments, giving justification for the derived requirements in the following sections.

Figure 22 gives a graphical display of the terminology used.

The scientifically relevant error to be quantified is the error between the commanded (intended) pointing attitude (PRB) and the actual pointing of the Instrument Line of Sight (ILS). At the first glance the error budget can be divided in two major continuants:

- the misalignment contribution between the ILS and the AOCS Reference Frame which is the $+X_{OPT}$
- the system related pointing error (consequently indexed with system)

The AOCS Reference Frame is currently assumed to be determined by a star tracker (STR).

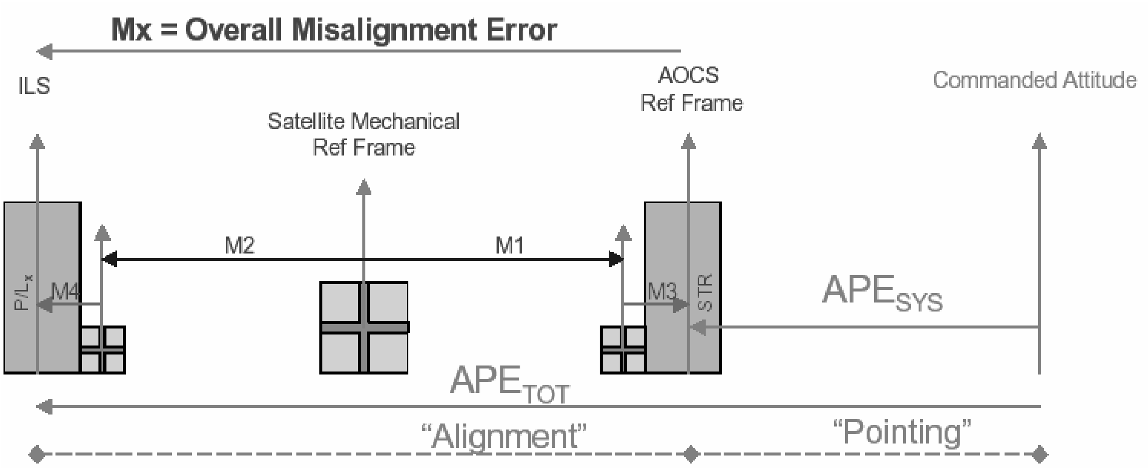


Figure 22 Overview of Misalignment Terminology

The first contribution is measured at system level and described (in principal) by the following rotational matrices:

- M1…Rotation Matrix to pass from S/C MRC to STR Cube as derived from the alignment data (system AIV).
- M2…Rotation Matrix to pass from S/C MRC to instrument Unit Cube as derived from the alignment data (system AIV).
- M3…Rotation Matrix to pass from STR Cube to STR Functional Frame as provided by the STR supplier.

- M4…Rotation Matrix to pass from Unit Cube to Unit Functional Frame as provided by the PI.

The S/C MRF is the Mechanical Reference Frame is defined by the Satellite Master Reference Cube (MRC) mounted in a specific position on the satellite and representing the three satellite axes.

The total misalignment will result in:

$Mx = M2 \times M4 \times M1^{-1} \times M3^{-1}$

MRC/UC x UC/UF = MRC/UF
inverse(MRC/STR C x STRC/STR F)= inverse(MRC/STR F) = STR F/MRC

Mx = M2 x M4 x inverse (M1 x M3)

In addition, in flight errors will be analytically be included in the overall error determination. They are mainly derived by Systematic (S), Long Term (LT), Short Term Errors (ST), such as:

- (S) Alignment Measurement Errors, Accuracy, Launch Settings, Gravity Release, Moisture Release, Outgassing, S/C thermoelastic Distortion, STR Thermoelastic Distortion
- (LT), (ST) are basically thermoelastic distortion related errors.

The second contribution is derived from the AOCS by the prime contractor. In principal this contribution describes the error between the commanded attitude and the actual AOCS reference frame. The contributors are APE SYS includes Systematic, Long Term, Short Term, and Random Errors

- (S) Controller Offset
- (S) Operational, i.e. Star catalogue, Guide Star change, Orbit Determination
- (S) STR: Aging, Bias, internal alignment knowledge, launch impact on internal alignment
- (S) Gyros related errors
- (S) Calibration errors
- (LT) Internal Thermoelastic Distortion (Hot/Cold)
- (ST) Noise / Quantisation effects
- (R) Moving Parts

The complete error APE total will be the sum of APE misalignment and APE sys.

As far as the PI is concerned the error source under his control is the misalignment and in flight stability of the ILS with respect to the instrument alignment cube. The remaining error contributors are under system control and will be managed by the Prime. Following are the derived requirements applicable to the instrument and the total performance requirements to be implemented by the system in order to achieve the overall required scientific performance.

Co-Alignment

Similar to the absolute misalignment between ILS and AOCS Reference Frame, the co-alignment (and stability) is defined. The Figure 22 shows the principle.

As for the alignment measurement between instrument and STR, the co-alignment between two instruments is measured via the Spacecraft Mechanical Reference Frame.

The co-alignment of the Remote Sensing instruments will be determined on the ground and a thorough analysis of the effective co-alignment in space will be performed.

Derived Alignment and Field of View Requirements

FOV and Instrument Line of Sight (ILS)

1. The PI shall, where relevant, specify with respect to the unit coordinate system URF and in accordance to the pointing definitions:
   - the Instrument line of Sight (ILS)
   - the FOV
   - the UFOV
   - the Vertex

   The information shall be contained in the EID-B / Mechanical ICD.

2. The PI shall, where relevant, provide the transformation matrix defining the ILS with respect to the instrument alignment references (alignment cube) for further inclusion in the alignment and pointing budgets at system level.

3. The PI shall further define the acceptable accuracy, knowledge and stability at 95% confidence level (2-sigma) in terms of azimuth and elevation of the ILS with respect to the Spacecraft Mechanical Build Axes or other Payload units (co-alignment).

4. The PI shall define the misalignment and in flight stability of the ILS with respect to the instrument alignment cube.

These values, in correlation with the instrument internal (including mounting error) alignment accuracy*, knowledge and stability will be used to compute the mechanical mounting accuracy knowledge requirement of the unit. For stability the appertaining time period must always be given.

**** Note:*** *The internal alignment is under the PI’s responsibility. It shall be compatible with the acceptable uncertainty specified.*

Optical Reference Requirements

Whenever a precise angular or positional alignment or co-alignment is required for an instrument, an optical reference system shall be employed to determine and verify the alignment.

5. For any unit equipped with an optical reference cube, the normal to the faces of that cube shall define the Unit Optical Reference Frame (UORF).

6. The instrument reference cubes shall be mounted on a fixed part of the instrument structure.

7. Their positions shall be agreed with ESA / prime contractor in order to account for System level integration constraints.

Depending on the alignment requirements, an instrument which might have this reference system will be aligned ("active alignment") or its alignment will be measured ("alignment knowledge") with respect to the spacecraft optical axes.

8. Where "active alignment" is required, the means to adjust the unit (i.e. shims, screws, eccentrics...) shall be considered as part of the mechanical interface.

9. In addition a description of the adjustment method (including the value of the minimum and maximum tilt angle achievable) and of the adjustment hardware used for that purpose shall be submitted to the ESA Project Office / Prime for approval.

10. It shall be demonstrated that the adjustment activities will not introduce stresses in the instrument and in the spacecraft structure (or that the stresses are quantified and stay below an acceptable level).

Pointing and Alignment Budget

11. The PI shall determine the position of the alignment cube with an accuracy of ±0.1 mm with respect to the position of the internal scientific relevant detectors.

12. The PI shall ensure that the faces of the alignment cube are representative for the ILS.

13. The maximum allowed deviation shall be less than TBD arcsec.

14. The PI shall declare the scientifically relevant instrument pointing error in terms of APE, RPE, PDE, AME and co-alignment requirements with any further instrument.

The system will implement the following pointing performance (TBC). Errors given refer to 2σ probability.

Pointing Parameter	Line of Sight (X_{Op})	Around Line of Sight
APE : Absolute Pointing Error	< 2 arcmin	< 20 arcmin
PDE : Pointing Drift Error	< 1 arcmin / 10 days	< 10 arcmin / 10 days
RPE : Relative Pointing Error	< 1 arcsec / 10 secs	< 2 arcsec / 10 secs
AME : Absolute Measurement Error	No req	< 3 arcmin in 10 secs

Table 8 Pointing Performance*

* Note: The pointing performance may be relaxed outside of remote sensing observation windows in order to reduce wheel off-loading requirements.

In-Flight Alignment Calibration
The in-flight efficiency will be limited by the in-flight evolutions of alignments (thermoelastic effects for instance).

15. The PI shall define the need for in-flight alignment calibration, if necessary, with respect to the AOCS reference sensors or with respect to other instruments.

Co-Alignment Requirements
The spacecraft shall provide 2 arcminutes co-alignment between the alignment cubes of PHI, METIS, SPICE, EUI and STIX (TBC).

16. If multiple apertures are required within an instrument the PI shall define the co-alignment accuracy of the apertures.

4.5 External Configuration Drawings

For each instrument unit, a configuration drawing is required to establish the mechanical interfaces with the spacecraft structure, harnesses and thermal hardware.

1. These drawings shall contain the following information:
   - Dimensions and associated tolerances (at ambient temperatures), including feet, internal connectors and their dedicated clearance
   - Focus position w.r.t. instrument coordinate system (dimensions and tolerances at operational temperatures)
   - Identification of a reference hole
   - Mounting hole pattern dimensions and hole patterns
   - Dimensions of mounting feet and contact area (base-plate and mounting feet)
   - Spot-faced area for seating of the mounting screw washers (if and where applicable)
   - Dimensions and location of dowel pins (where applicable)
   - Mass and associated tolerances (precise if estimated, calculated or weighted)
   - Location, naming, type and function of all connectors
   - Connector key shape orientation, the identification of connector contact “1”, showing connector in front view and the connector center line
   - Information about connector fixation
   - Identification of bonding studs
   - Identification of non-flight items
   - Location of unit and connector identification labels
   - Details of instrument provided mounting hardware, thermal/electrical isolation provisions

- Location and routing of any harness interconnecting modules of a “stacked” box configuration
- Identification of free areas for harness fixation
- Calculated Centre of Gravity location in instrument unit co-ordinate system and Moments of Inertia and its co-ordinate system if different from instrument unit co-ordinate system
- Location of transport/storage purging connections (if applicable)
- Material of housing and surface finish
- Flatness and roughness of contact area
- Base plate material and surface treatment
- Surface coating (IR Emissivity and Solar absorptance if external location)
- Specific heat (J/Kg/K) (calculated or measured)
- Design and location of handling points

2. Drawings shall clearly specify the unit they represent and the responsible design authority; they shall be subject to a properly controlled numbering and revision updating system. Each revision of a drawing shall be accompanied by a list detailing all changes that have been incorporated since the previous revision on the drawing itself.

3. 2D Drawings shall be submitted to the Project as computer readable and editable files, preferably in a vectorial file format (.hgl, .drw or .cgm (compatible MS word) , or pdf avoid definition loss) together with one hard copy of each file.

4. The Metric Standard (SI-SYSTEM INTERNATIONAL) shall be used for design and manufacturing of all instruments. For components and equipment, the dimensions shall be given in millimetres and the angles in degrees.

4.6 Mechanical Interfaces

4.6.1 MECHANICAL INTERFACE CONTROL DOCUMENT

The Spacecraft Mechanical Interface Control Document (MICD) will complement the EID-B data defined by the PI.

The MICD shall contain the following key information with the accuracy required below:

4.6.1.1 Mass and Mass Tolerance

1. The instruments shall be designed taking into account the allocated mass with adequate contingencies according to the contingency scheme as given below.

2. At equipment level, the following design maturity mass contingencies shall be applied:
   - > 5 % for "Off-The-Shelf" items (ECSS Category: A / B)
   - > 10 % for "Off-The-Shelf" items requiring minor modifications (ECSS Category: C)
   - > 20 % for newly designed / developed items, or items requiring major modifications or re-design (ECSS Category: D).

3. The unit Accommodation Mass shall include the total instrument hardware that is intended for flight. The mass budget shall include at least (as applicable):
   - Structure, mechanisms and optics;
   - electronics up to the interfaces with the spacecraft power and data systems;
   - thermal control hardware, including any necessary thermal straps or heaters/thermistor, instrument blankets, cold fingers defined by the instrument (i.e. not part of the spacecraft TCS)
   - pigtail and interconnecting harness (if instrument consists of more than one unit)
   - electrical connectors, but not the mating harness connector
   - attachment hardware but excluding standard fixation bolts to the spacecraft structure and washer
   - potting compounds used in the units
   - alignment references, e.g. mirrors, that are not removed before flight
   - internal balance mass (applicable for periodically operating mechanisms)
   - electrostatic screens and/or magnetic shielding
   - in-flight covers, purge ports, purging pigtails

4. The difference between the measured mass of each STM/QM and FM unit and the respective estimated mass, specified on the Interface Control Drawing, current at the time of the STM/QM and FM unit delivery to the Project, shall be less than 1%.

4.6.1.2 Centre of Mass

1. The estimated Centre of Mass (CoM) coordinates of each deliverable unit shall be computed and specified with respect to the Unit Coordinate System.

2. Any variation of CoM coordinates due to consumables or appendages deployment shall be specified.

3. In computing the CoM values, non-flying items (e.g. temporary installation items, etc.) shall not be taken into account.

4. The tolerance between the measured CoM coordinates of each STM, FM and FS unit and the respective estimated coordinates, specified by the PI on the relative Interface Control

Drawing, current at the time of the STM, FM and FS delivery to the Project, shall be within a sphere of 3 mm radius.

4.6.1.3 Moments of Inertia

1. The Moments of Inertia and Cross Products of Inertia shall be computed for each instrument, referred to a reference system parallel to the URF axes and with its origin at the CoM for all in-flight configuration (with and without cover and movable parts).

2. The MoI shall be measured with the accuracy defined in Design Verification Requirements (Vol. IV).

3. Unit MoI variations due to mechanisms shall be defined in the EID-B and shall be less than 0.1 Kgm2 (TBC) (applicable to in flight configuration excluding one shot mechanisms).

4. The difference between the measured MoIs of each STM, FM and FS unit and the MoIs, specified on the Interface Control Drawing, current at the time of the STM, FM and FS delivery to the Project, shall be less than 10%.

4.6.1.3.1 Unit Dimensions

1. The dimension, d, of each unit shall be specified in the EID-B to a tolerance smaller than:
+ 0.5/-0.0 mm for d < 500 mm
+ 1.0/-0.0 mm for 500 mm < d

4.6.2 SIZE AND MASS

1. The length of the remote sensing instruments shall be no more than 1m class such that there is sufficient space to accommodate them in the spacecraft.

2. The total instrument mass allocation for Solar Orbiter is 180 kg. The mass allocation for each instrument is outlined in the table below.

Instrument	**Basic Mass [kg]**	**Maturity margin (%)**	**Nominal Mass [kg]**
EPD		15	13.8
MAG	1.898	10	2.088
RPW			11.33 (TBC)
SWA		~15	15.9
EUI w/ 2 HRI			18.2
METIS-COR only		25	20.6
PHI	23.3	25	29.1

STIX	4	10	4.4
SOLOHI	8.64	30	11.2
SPICE	14.7	25	18.4

Table 9 Instrument Mass Allocations

4.6.3 INSTRUMENT MOUNTING

4.6.3.1 Boom Mounted Instruments

1. The mechanical interface of the instruments mounted on the boom shall be TBD.

4.6.3.2 Spacecraft Mounted Instruments

2. The attachment points of the equipment/subsystem shall be designed to guarantee the compliance to the following general functional requirements:
   - Ease of accessibility with standard tools to the attachment bolts during (de)integration of the equipment/subsystem to the spacecraft.
   - The position of the connectors and grounding studs shall provide sufficient accessibility to enable the mounting and removal.
   - The mechanical design of the mounting attachments shall contribute to a proper thermal control of the equipment/subsystem, by taking into account the thermal loads encountered throughout the mission lifetime.

3. The attachment points shall provide a controlled surface contact between the units and the structure to allow control of thermal conditions on the units as well as electrical bonding. This contact shall be maintained under all operating conditions, taking into account loading resulting from the different thermal coefficients of expansion between dissimilar materials.

4. The mechanical mounting interface shall be consistent with the thermal and EMC design requirements. In particular, the contact area shall be free of paint.

5. The number of attachment bolts for any piece of equipment shall ensure that the tensile load per interface bolt shall not exceed 10N under a 1g environment in any direction for the standard M5 bolts and standard mounting inserts.
   Note: In order to limit the total number of bolts, larger bolts, carrying higher loads are possible.

6. The interface plane flatness of an equipment/subsystem shall be better than 0.1mm, i.e. all attachment points shall be in a common plane within +/-0.05mm.

7. The type and number of the equipment/subsystem bolts shall be defined to withstand the worst-case environmental conditions.
   For the attachment on the support panel, the equipments/subsystems shall preferably use M5 bolts. The use of other bolt types may be acceptable, but shall be reviewed and agreed on a case by case basis.

8. All equipments/subsystems shall be through bolted into threaded inserts. The interface bolts clearances specified below shall be respected:

Interface Clearance	**M4 bolts**	**M5 bolts**	**M6 bolts**
Hole size [mm]	4.5+-0.1	5.5+-0.1	6.5+-0.1
Positional tolerance of hole [mm]	0.1	0.1	0.1
Positional tolerance of insert [mm]	0.1	0.1	0.1

Attachment bolt	M5	(1)
Attachment bolt material	Titanium; Stainless Steel (TBC)	
Washer dimensions	Typical Diameter: 10 mm Typical thickness: 1 mm	(1)
Attachment hole diameter	5.3 (+0.1, -0.0) mm	
Distance (d) between attachment holes (lugs)	Typically 100 mm	
Tolerance distance between centre of attachment holes (w.r.t. reference hole, R)	⊕ ∅ 0.2 | R	
Diameter of Spot Face Area for bolt head and washer	12 mm	(2)
Attachment lugs:		
- dimensions	see figure below	(2)
- roughness	≤ 3.2 microns	
- flatness	≤ 0.1/100 mm	
- edge radius	≤ 0.5 mm	

Table 10 Equipment Interface Requirements

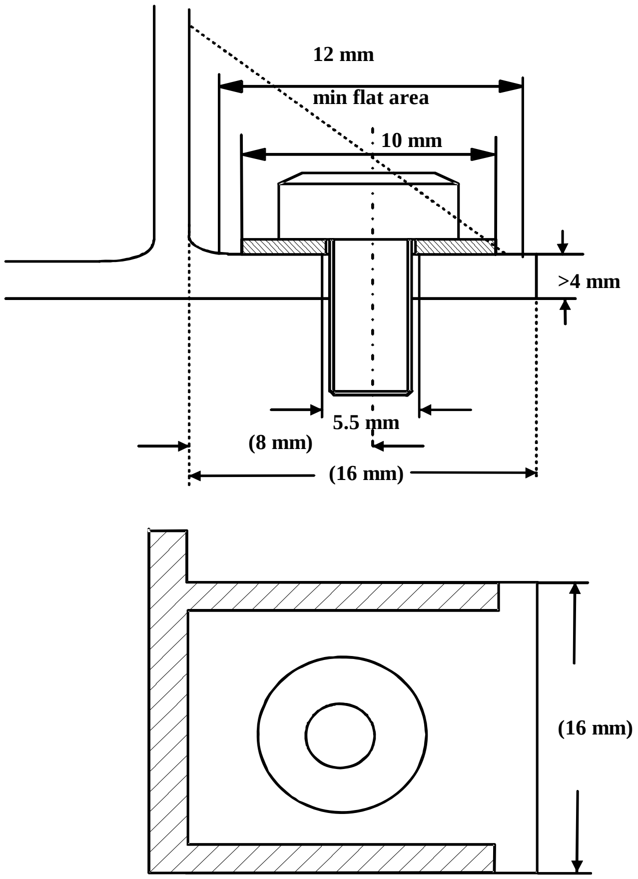


Figure 23 Standard Interface Geometry

9. The thickness of the equipment/subsystem mounting feet shall be at minimum 3.0 mm.

10. Minimum clearance between mechanical parts shall cover design, manufacturing, assembly tolerances, alignment translation/rotation ranges and environmental displacements.

11. All equipment/subsystems shall be designed allowing for the failure of any single attachment bolt.

12. The minimum load carrying capability of the inserts shall be:

Based on the minimum properties of the equipment panels				
with 20mm core (32kg/m^3) and 0.25mm facings				
the minimum insert load capability [N] should be:				
type	Diameter[mm]	Height[mm]	shear	tensile
M4 insert	14.2	9.5	1400	900
M5 insert	14.2	12.7	1400	1200
M6 insert	17.4	15.9	1700	1700

Table 11 Minimum Load Carrying Capability of Inserts

13. The following typical insert distances along the mounting edges shall be considered:

    For cubic shaped units (with ρ'=1000kg/m^3) typically:
    a. 1 insert / 80 mm along the edge shall be applied in case of M4 inserts; this is safe for unit masses < 15 kg corresponding to = 1 insert/(0.5 to 1 kg)
    b. 1 insert / 100 mm along the edge shall be applied in case of M5 inserts; this is safe for unit masses < 30 kg corresponding to = 1 insert/(1 to 2 kg)
    c. 1 insert / 120 mm along the edge shall be applied in case of M6 inserts; this is safe for unit masses < 45 kg corresponding to = 1 insert/(2 to 4 kg)

4.6.3.3 Instrument Connectors

14. For internally accommodated remote sensing instruments all connectors will be located on the –X (i.e. anti-sun) side of the instrument.

4.6.4 FEEDTHROUGHS

1. The mounting tolerance of the feedthrough with respect to the instrument line of sight shall be less than 1mm (TBC).

2. The thermo-elastic distortions between the feedthrough and the instrument line of sight shall be less than 1 mm (TBC). This leads to a total maximum misalignment of 2 mm in the lateral direction.

3. The distance between the aperture of the internally accommodated remote sensing instruments and the front panel of the heat shield is 425.5mm, as shown in the figure below. The 425.5 mm consists of 400 mm for the heatshield, 22.5mm thickness of the spacecraft panel and 5 mm clearance between instrument box and spacecraft panel. To accommodate this distance no element of the instrument shall protrude beyond the optical aperture(s).

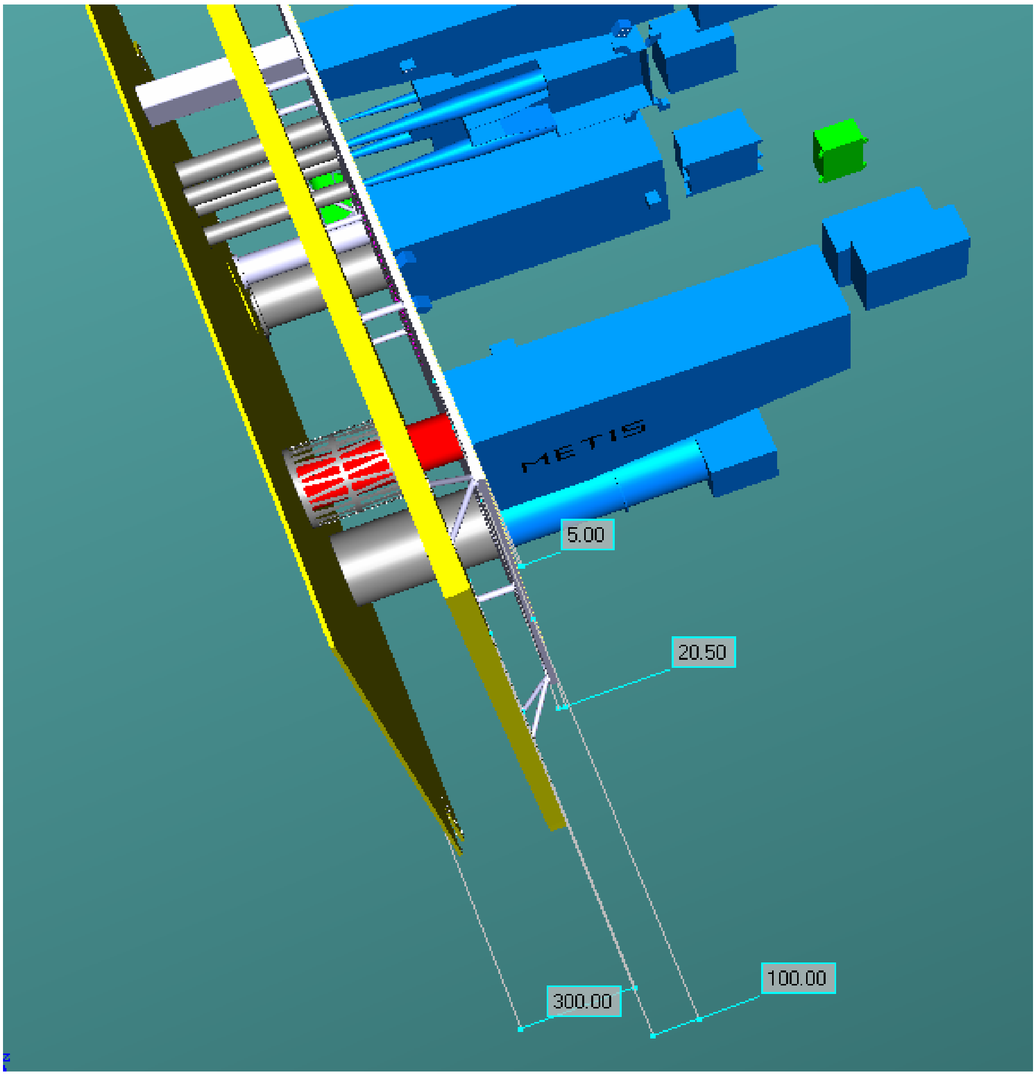


Figure 24 Outer Heatshield Distance with respect to Instrument Apertures

4. There shall be 5 mm of static clearance between the +X spacecraft panel and the front of the internally accommodated remote sensing instruments (with the exception of METIS).

5. Any protrusions of the instruments (i.e. METIS) within the feedthroughs shall have a minimum of 5 mm clearance with the inner wall of the feedthrough.

The location of the feedthroughs in the current spacecraft design is as shown in the figure below.

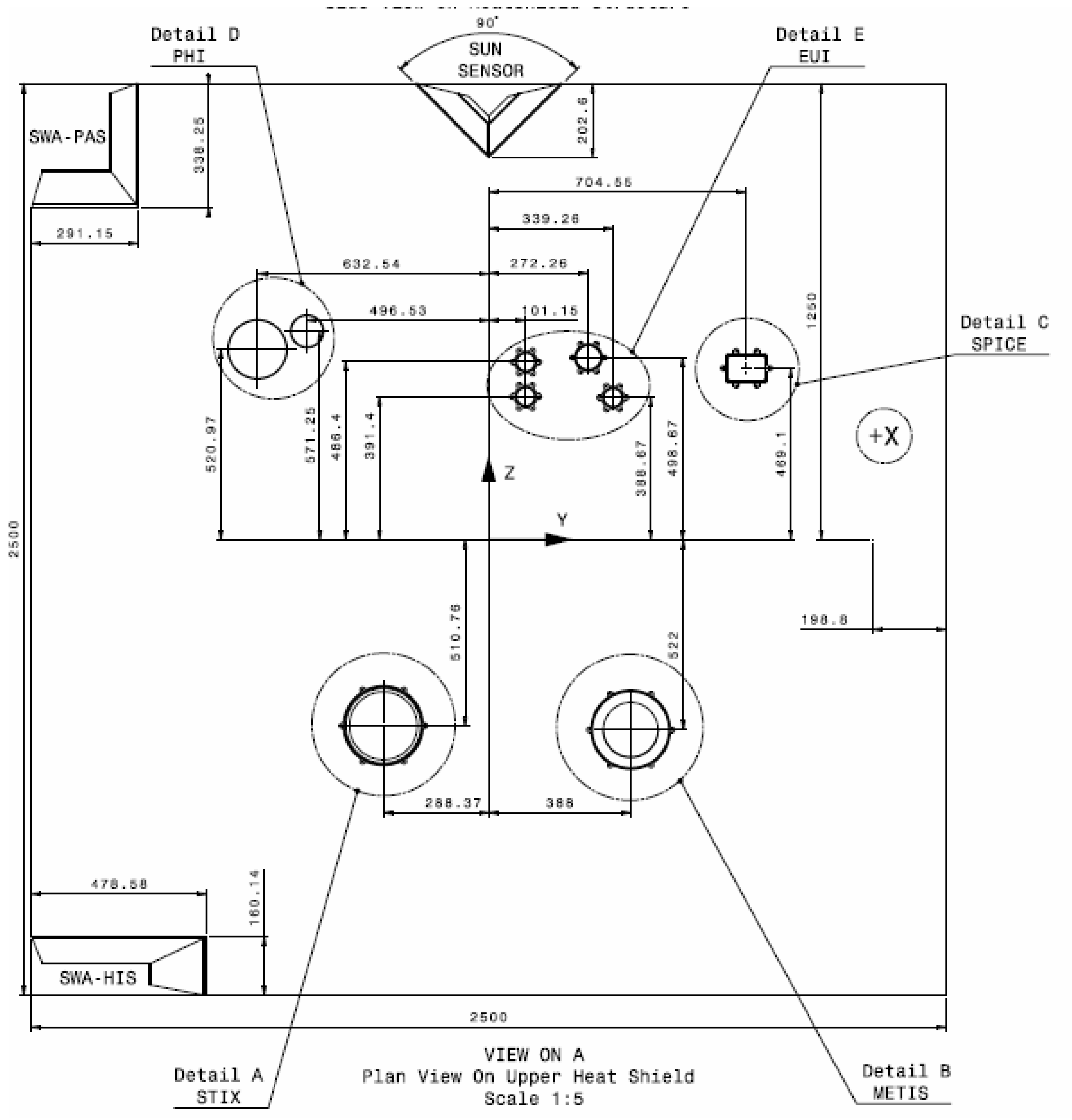


Figure 25 Locations of Feedthroughs in Outer Heatshield Layer
Note: 3 HRI feedthroughs are depicted but only 2 will be provided.

4.6.4.1 Feedthrough Mounted Items (Filters, Sun Shades, etc.)

6. The PHI Filters and STIX Sun Shades shall be provided by the respective PIs. The interfaces are TBD.

7. The spacecraft shall provide, for feedthrough mounted items, an alignment of:
   - Alignment of window along X: ± 10 mm (TBC)

- Alignment of window axis with respect to ILS, along Y: ± 3 mm (TBC)
- Alignment of window axis with respect to ILS, along Z: ± 3 mm (TBC)
- Window tilt our of Y-Z plane: ± 30'(TBC)

8. The dynamic loads on the feedthrough mounted items are:
    - PHI HRT Filter: TBD
    - PHI FST Filter: TBD
    - STIX upper sun shade: TBD
    - STIX lower sun shade: TBD

4.6.5 DOORS

1. The spacecraft may provide doors, accommodated in the heat shield if required. The PI shall justify the need for a door accommodated inside the heat shield as well as the required number of cycles. All opening and closing modes shall be defined.

2. For internally accommodated remote sensing instruments, the PI shall justify the need of an internal instrument door, if required, and define all of its opening and closing modes.

3. If an internally accommodated remote sensing instrument requires an internal instrument door, the spacecraft will supply a heat shield door to protect the spacecraft in the case the instrument door fails in the closed position.

4. The spacecraft provided door shall not be designed to close rapidly or autonomously during off-pointing between the spacecraft optical axis and the Sun centre.

5. The spacecraft provided doors shall be designed for a minimum (TBD) number of cycles.

6. The spacecraft provided doors shall be designed to prevent direct sunlight entering the feedthroughs of the instruments for off-pointing between the spacecraft optical axis and the Sun centre of up to 15 degrees (TBC) when closed.

7. The spacecraft provided doors shall not be sealed and will therefore only provide limited contamination protection. TBD

4.6.6 MECHANICAL ENVIRONMENT

1. Instruments shall be designed to withstand the mechanical environment during all phases of AIT/AIV, qualification and acceptance testing (see Section 6.4.5) and that produced at launch.

4.6.7 STRUCTURAL DESIGN

4.6.7.1 Margins of Safety

1. Instruments shall be designed to comply with the factors of safety defined in the following table:

Structure type and sizing case	**FOSY**	**FOSU**	**FOSY for verification by analysis only**	**FOSU for verification by analysis only**	**Additional factors**
Metallic structures	1.1	1.25	1.25	2.0	
Composite structures, uniform material, brittle		1.25		2.0	
Sandwich structures: - Face wrinking - Intracell buckling - Honeycomb shear		 1.25 1.25 1.25		 2.0 2.0 2.0	 1.2 1.2 1.2
Glass structures		2.5		5.0	
Composite structures discontinuities		1.5		2.0	
Joints and inserts		1.25		2.0	1.2
Global buckling		2.0		2.0	

Table 12 Factors of Safety extracted from ECSS-E30-Part2 [NR8]

4.6.7.2 Design Loads

2. The PI shall take into account the design loads provided below in Figure 26, factored by the corresponding safety factor, for the design of bolts, feet and adjacent structure. It shall be used for static sizing of the structure, but the actual unit internal dynamic behaviour is not taken into account.

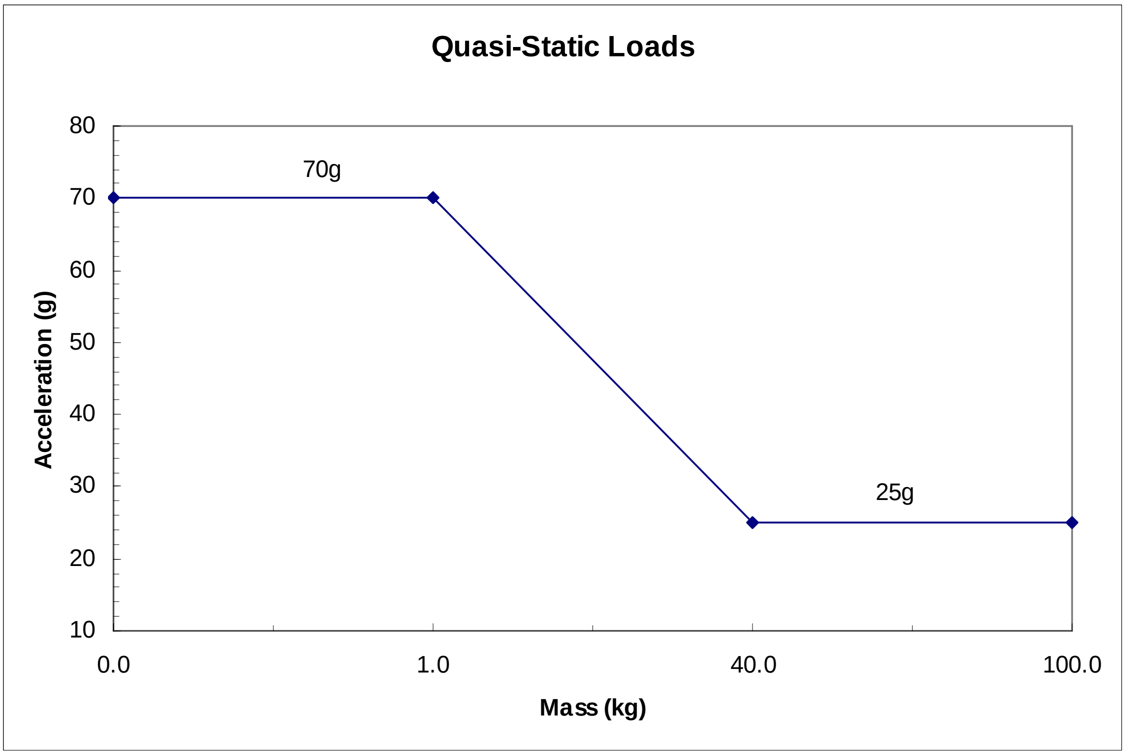


Figure 26 Quasi Static Loads

4.6.7.3 Stiffness Requirements

3. The instrument units shall have all fundamental resonance frequencies above 140 Hz.

4.6.8 PAYLOAD GENERATED DISTURBANCES

1. The PI shall define the instrument generated disturbances (i.e. internal mechanisms, etc.). First level estimates shall include the moving mass and the movement frequencies and characteristics.

4.7 Thermal Design Requirements

4.7.1 THERMAL CONTROL DEFINITIONS AND RESPONSIBILITIES

4.7.1.1 Thermal Control Definitions

This section provides a list of terms used in the context of thermal control. It excludes the ground phases (test, transport, storage, etc.).

1. The definition of the interface temperatures and associated margins shall be according to the following definitions:

Mission
The word "mission" covers the launch and flight phases.

Spacecraft Thermal Control sub-System (TCS)
The Spacecraft TCS includes all the means, hardware (heaters, thermostats, temperature sensors…) and software, to control the spacecraft heat flows and temperatures.

Unit
The word "unit" refers to a payload instrument under the responsibility of the PI. It is electronics, a sensor e.g. a telescope or a part of this sensor e.g. cooler, optical bench… or even the complete instrument.

Unit Reference Point (URP)
The URP is a physical point located on the unit close to the mechanical interface to the spacecraft. Its temperature provides a simplified representation of the unit thermal behaviour. Its temperature is controlled by the TCS for internally mounted units or by the PI for externally mounted units.

System Reference Point (SRP)
The SRP is a physical point located on the spacecraft structure close to a unit mechanical interface. It is used to evaluate the thermal interaction between an externally mounted unit and the spacecraft. Its temperature is controlled by the TCS.

Cold Finger Interface Temperature (T_{CF})
Temperature of the cold finger structure used to define the conductive heat exchange to the spacecraft.

Hot Element Interface Temperature (T_{HE})
Temperature of the hot element interface used to define the conductive heat exchange to the spacecraft.

Radiative Sink Temperature (T_R)

This is a virtual black body radiation temperature used to define the equivalent radiative thermal load on a unit.

Unit Temperatures
All unit temperatures recalled in this section shall be defined at the URP or the SRP as appropriate and given for the following conditions:

- at switch-on
- when operating
- when non-operating

They are the design temperatures specified for the ground and mission phases.

(TCS) Design Temperature Range
This is the maximum range of temperature experienced in flight by a unit throughout the mission (see definition in section 4.7.1.1) and during ground phases. In absence of specification, the ground range is assumed to coincide with the flight range.

(TCS) Calculated Temperature Range
This is the unit temperature range obtained by analysis excluding prediction uncertainties.

(TCS) Predicted Temperature Range
This is the temperature range obtained by adding the prediction uncertainties to the calculated temperature range.

Switch-on Temperature
This is the lowest temperature at which a unit can safely be switched-on throughout the mission and during ground phases. In absence of specification, the ground range is assumed to coincide with the flight range.

(TCS) Acceptance Temperature Range
It is an extension of the design temperature range by the acceptance margin at both ends. Partial deviation from the performance requirements may be accepted within unit qualification margins provided they do not affect the interfaces with the spacecraft and they are reversible when the unit is brought back within its acceptance test temperature range.

(TCS) Acceptance Test Temperature Range
All flight units shall be tested prior to delivery to the spacecraft at this extreme temperature range. It is an extension of the acceptance range by the test uncertainties.

(TCS) Qualification Temperature Range
It is an extension of the acceptance temperature range by the qualification margin at both ends.

(TCS) Qualification Test Temperature Range

This is the extreme test temperature range at which a unit shall be tested to qualify its design. It is an extension of the qualification range by the test uncertainties.

(TCS) Internal Design Temperature Range
This is the extreme temperature for which unit components or parts are selected.

The required URP temperatures and margin logic are described in the Figure 27. For externally mounted units, the qualification margin shall be increased to 30˚C.

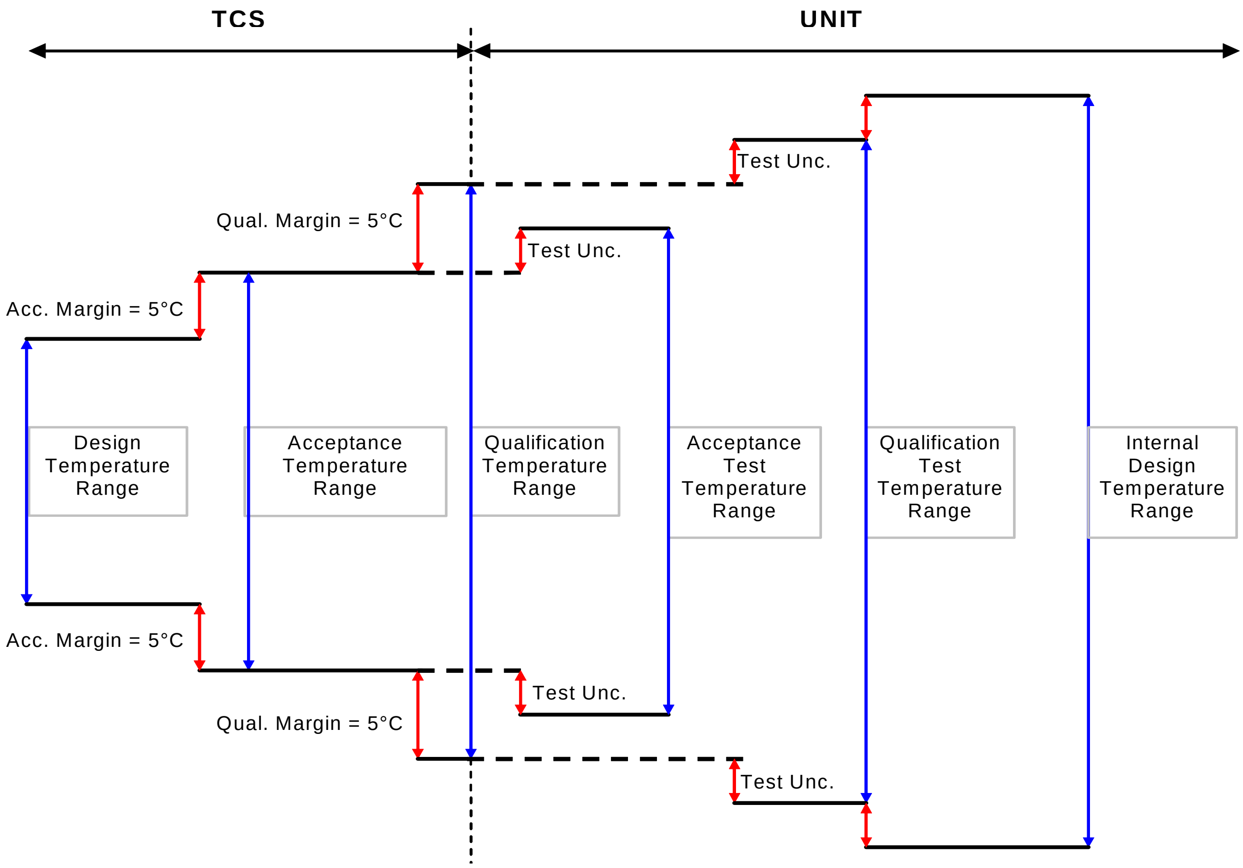


Figure 27 URP Temperatures and Margins

4.7.1.2 Spacecraft Thermal Control System Responsibilities

The spacecraft will:

- design the spacecraft TCS
- define (if required) the SRP location for externally mounted units

- maintain the internally mounted unit URP and externally mounted unit SRP temperatures within their allowable range at any time of the mission (design limits) and during ground operations (ground limits)
- monitor the unit URP or the SRP temperatures as relevant
- define, procure and install the necessary thermal control H/W (heaters, thermostats, temperature sensors ...) and control S/W that is necessary to provide the relevant interfaces
- demonstrate the performance of the TCS by analysis and test incl. uncertainties

4.7.1.3 Payload Responsibilities

The PI shall:

1. Define and describe the unit internal thermal design with particular attention to:
   - the thermal control principles,
   - the baffle (if required), the cold finger and the hot elements of the unit,
   - the thermal interfaces to the spacecraft and outer space.

2. Define the URP location.

3. Define the URP temperature and the temperature requirements of critical internal parts.

4. Maintain the internal parts within their allowed temperature limits during:
   - the mission i.e. launch and flight,
   - ground phases,
   - unit level acceptance and qualification tests.

5. Provide an Interface Geometric Mathematical Model (IGMM) and an Interface Thermal Mathematical Model (ITMM) for coupled thermal analysis with the spacecraft as specified in section 6.3.2.4 on Thermal Mathematical Models.

6. Procure the necessary instrument thermal H/W such as heaters, etc. to maintain the payload unit within the specified temperature limits.

7. Provide the figures on the heat dissipated by the unit and report the interface heat flux in all relevant environments.

8. Demonstrate the performance of the unit internal thermal design by analysis and test including uncertainties.

4.7.2 THERMAL ENVIRONMENT

4.7.2.1 AIV Clean Room Environment

1. The instruments shall be designed to operate during the ground operations under the following maintained environmental conditions (TBC):
   - Ambient temperature 21°C ± 3°C
   - Relative humidity 50% ± 10%
   - Cleanliness class 100.000
   - Pressure atmospheric conditions

During the system functional tests in air, the units dissipating inside the S/C can reach considerably higher temperatures than the ambient temperature.

2. In operation, all units shall be able to withstand at least 40°C at the URP.

4.7.2.2 Launch Thermal and Pressure Environment

1. During launch and ascent, the P/L units shall be designed to cope with the thermal fluxes (TBC) given below. For most of the P/L units this should not be a design driver, since internally mounted. Externally mounted units will be provided with more details in the course of the project.

   - under fairing

Duration	3 min
Flux	< 800 W/m^2
Direction	on any surface of the satellite

   - after fairing jettison (aerothermal)

Duration	20 s
Flux	< 1135 W/m^2
Direction	TBD
Duration	up to 4 hours (TBC)
Flux	Solar + Earth albedo + Earth infrared radiations

2. During ascent and still under the fairing, the units shall be designed to cope with a peak pressure decay rate of 4500 Pa/s (TBC).

4.7.2.3 Cruise and In-Orbit Thermal Environment

1. For externally mounted units, environmental heat loads and heat exchanged with the spacecraft surfaces shall be evaluated by considering:
   - Deep space temperature -270°C
   - Solar intensity $SC=1370 \pm 10$ W/m^2 at 1 AU
     $\frac{SC}{d_S^2}$ at distance d_S of the Sun
   - Sun collimation Half-cone angle = $\tan^{-1}\frac{R_S}{d_S}$
   - Absorbed heat loads and heat exchanged by radiation computed by means of spacecraft interface mathematical models provided by the Spacecraft TCS

2. The instruments shall survive:
   - Illumination from direct sunlight at distances at 0.8AU for long duration
   - Illumination from direct sunlight at 0.22 AU for short duration (15 degree off pointing with respect to the Sun centre line for up to 20 seconds, TBC)

4.7.2.4 Fly-Bys Thermal Environment

The thermal environment for the fly-by of Earth is:

Solar Constant (W/m²)		Albedo Coefficient (-)		Earth Temperature (K)	
Min	Max	Min	Max	Min	Max
1320	1420	0.2	0.4	245	265

The Spacecraft attitude is TBD.

For Venus fly-by, it is:

Solar Constant (W/m²)		Albedo Coefficient (-)		Venus Temperature (K)	
Min	Max	Min	Max	Min	Max
2570	2655	0.72	0.78	229	229

The Spacecraft attitude is TBD.

4.7.3 THERMAL INTERFACES - DEFINITIONS

1. The PI shall use the following definitions in defining the instrument interface with the spacecraft.

4.7.3.1 Nomenclature

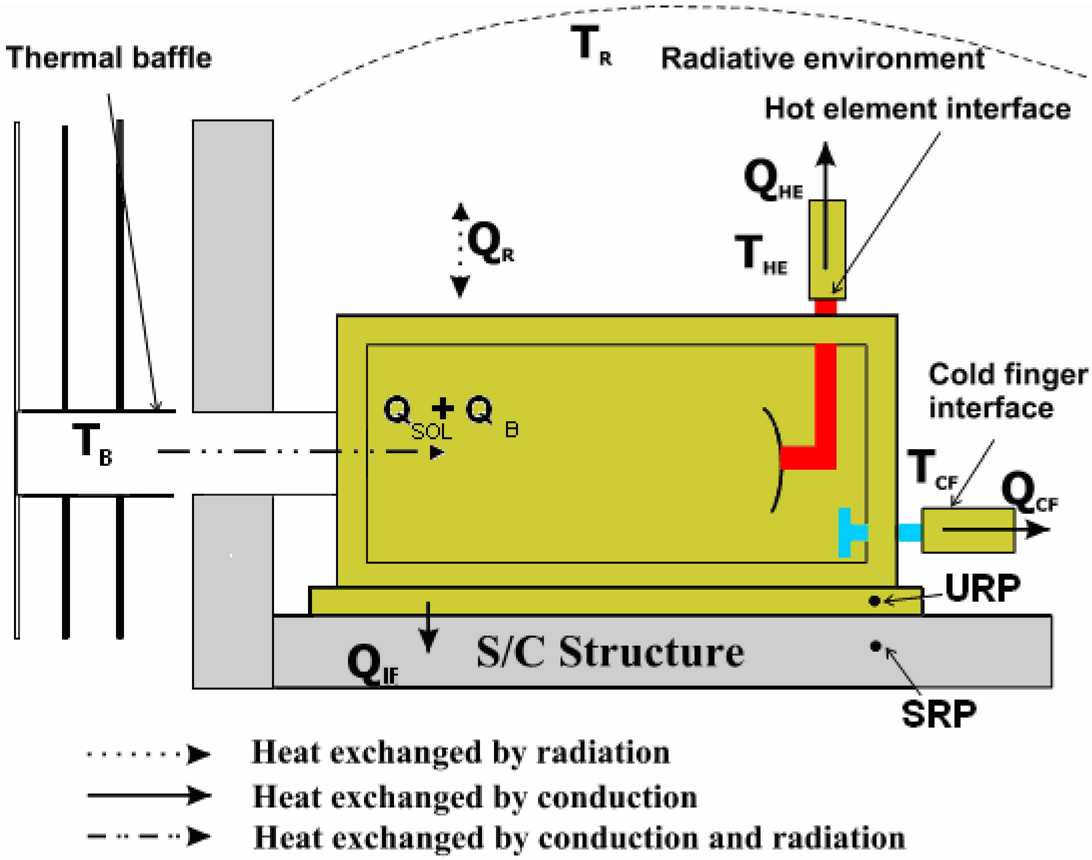


Figure 28 Temperature and Heat Load Definitions

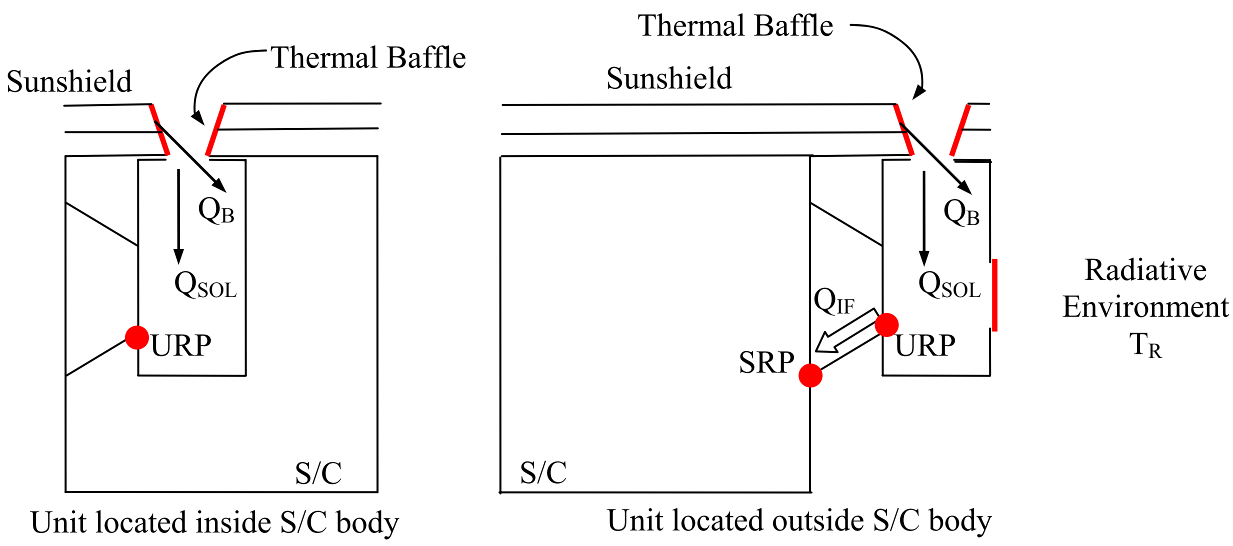


Figure 29 Thermal Interfaces

The thermal interface between the feedthrough and the instrument, aiming to reduce the heat flow into the instrument, is TBD.

The above mentioned thermal interfaces can be summarized as follows:

- Conductive Interface Temperature: $\mathbf{T_{URP}}$ or $\mathbf{T_{SRP}}$
  Net Conductive Heat Flux: $\mathbf{Q_{IF}}$
- Radiative Interface Temperature: $\mathbf{T_R}$
- Cold Finger Interface Temperature: $\mathbf{T_{CF}}$
  Net Conductive Heat Flux: $\mathbf{Q_{CF}}$
- Hot Element Interface Temperature: $\mathbf{T_{HE}}$
  Net Conductive Heat Flux: $\mathbf{Q_{HE}}$

Furthermore, the solar energy passing through the apertures shall be taken into account. This energy also accounts for reflection, for example on the baffle.

- Solar Heat Load: $\mathbf{Q_{SOL}}$

Finally, the infrared heat load from the thermal baffle shall be taken into account:

- Baffle Temperature: $\mathbf{T_B}$

Net Radiative Heat Flux: $\mathbf{Q_B}$

4.7.4 THERMAL INTERFACES – REQUIREMENTS

1. The interface heat flows provided in **Table 13** define the maximum allowable heat rejection from the payload interfaces.

These heat flows represent the current best understanding of the heat loads and thermal interface loads for the instruments based on analysis and data extracted from the instrument proposals and are calculated for the closest perihelion of 0.234 AU.

Payload	Max. Allowable Payload I/F Heat Rejection				
	Q_{HE} (W)	Q_{CF} (W)	$Q_R + Q_{IF}$ (W)	Total (W)	Notes:
PHI Optics	30	4	0	34	De-coupled from the S/C internal environment
PHI electronics	-	-	24	24	Via baseplate coupling
SPICE Optics	7	5	1	13	Heat rejected by payload mirror not part of this budget
SPICE electronics	-	-	9	9	Via baseplate coupling
EUI Optics	42	4	24	70	
EUI electronics	-	-	9	9	Via baseplate coupling
METIS Optics	90	5	7	102	
METIS electronics	-	-	26	26	Via baseplate coupling
STIX	0	4	6*	10	No hot finger interface required * Via conductive interface only. Radiative heat rejection not possible due to payload operating temperature
SWA-PAS	-	-	10	10	Via conductive I/F with S/C
SWA-HIS	-	-	10	10	Via conductive I/F with S/C
SWA-EAS	-	-	TBD	TBD	Thermaly de-coupled from Boom
SWA-Electronics	-	-	3	3	Via baseplate coupling
EPD-SIS	-	-	10	10	Via conductive I/F with S/C
EPD-STE	-	-	TBD	TBD	Thermaly de-coupled from Boom
EPD-HETn	-	-	10	10	Via conductive I/F with S/C
EPD-EPT	-	-	10	10	Via conductive I/F with S/C
EPD-LET	-	-	10	10	Via conductive I/F with S/C
EPD-electronics	-	-	5	5	Via baseplate coupling
RPW-SCM sensor unit	-	-	TBD	TBD	Thermaly de-coupled from Boom
RPW SCM Electronics	-	-	TBD	TBD	Via baseplate coupling
RPW Antenna	-	-	TBD	TBD	Majority of heat rejected to space, some exchange with S/C sides
RPW Electronics	-	-	12	12	Via baseplate coupling
MAG Sensor	-	-	TBD	TBD	Thermaly de-coupled from Boom
MAG electronics	-	-	1	1	Via baseplate coupling

Table 13 Instrument Maximum Allowable Heat Rejection to Spacecraft (TBD)

2. The instrument interfaces design temperature range shall be as presented in **Table 14 (TBC)**.

These interface temperatures represent the current best understanding of the thermal interface for the instruments based on analysis and data extracted from the instrument proposals. The non-operating temperature range is applicable to the unit survival temperature.

Payload	App. To	T_{HE}, T_{URP}, T_{SRP}(°C)					T_{CF}(°C)	T_B(°C)		T_R (°C)	
		Min Non-Op	Min Op	Max Op	Max Non-Op	Start Up	Max Op	Min Op	Max Op	Max	Min
PHI Optics	URP	-30	-20	50	60	-30	-10	-80	280	-20	40
PHI Electronics	URP	-30	-20	50	60	-30	-	-	-	-20	40
SPICE Optics	URP	-30	-20	50	60	-30	-60	-70	460	-20	40
SPICE Electronics	URP	-30	-20	50	60	-30	-	-	-	-20	40
EUI Optics	URP	-40	-30	60	70	-40	-60	-60	300	-20	40
EUI Electronics	URP	-40	-30	60	70	-40	-	-	-	-20	40
METIS Optics	URP	-30	-20	50	60	-30	-60	-40	295	-20	40
METIS Electronics	URP	-30	-20	50	60	-30	-	-	-	-20	40
STIX Optics	URP	-30	-30	25	30	-30	-30	-80	255	-20	40
STIX Electronics	URP	-30	-30	30	60	-30	-	-	-	-20	40
SoloHI Optics	SRP	-30	-30	-10	60	-30	-	-	-	TBD	-270
SoloHI Electronics	URP	-30	-20	50	60	-30	-	-	-	-20	40
SWA EAS	SRP	-190	-160	-130	-30	-190	-	-	-	TBD	-270
SWA PAS	SRP	-30	-20	50	50	-10	-	-110	95	TBD	-270
SWA HIS	SRP	-30	-20	40	50	-10	-	-	-	TBD	-270
SWA Electronics	URP	-25	-20	50	55	-20	-	-	-	-20	40
RPW Antenna	SRP	-30	-20	50	60	-30	-	-	-	TBD	-270
RPW SCM Sensor Unit	SRP	-190	-160	-130	-30	-190	-	-	-	TBD	-270
RPW SCM Electronics	URP	-30	-20	50	60	-30	-	-	-	-20	40
RPW Electronics	URP	-30	-20	50	60	-30	-	-	-	-20	40
MAG Sensor	SRP	-190	-160	-130	-30	-190	-	-	-	TBD	-270
MAG electronics	URP	-30	-20	50	60	-30	-	-	-	-20	40
EPD STE	SRP	-190	-160	-130	-30	-190	-	-	-	TBD	-270
EPD SIS	SRP	-40	-30	30	40	-40	-	-	-	TBD	-270
EPD EPT	SRP	-40	-30	30	40	-40	-	-	-	TBD	-270
EPD LET	SRP	-40	-30	30	40	-40	-	-	-	TBD	-270
EPD HETn	SRP	-40	-30	30	40	-40	-	-	-	TBD	-270
EPD Electronics	URP	-55	-45	50	65	-55	-	-	-	-20	40

Table 14 Thermal Interfaces: In-Situ Solar Viewing Instruments

3. The interface range for the boom mounted payloads (SWA-EAS, RPW SCM Sensor, MAG sensors, EPD STE) relates to the temperature range of the boom. The boom mounted payloads shall be thermally decoupled from the boom structure.

All externally located units will have a view of both the space skin environment and the spacecraft body. The heat exchange with the spacecraft will be dependent on the location of the unit (distance from spacecraft). The instrument thermal design shall consider as a worst case a spacecraft body temperature of 50°C (maximum temperature of the spacecraft radiators).

The payload filter design temperature range is presented in **Table 15.**

4. The filters shall be compatible with the filter temperatures represented in the current best understanding of the thermal design of the instrument filters on analysis and data extracted from the instrument proposals. These temperatures represent the unheated temperature range over the entire mission.
Note: In the situation that the filter design temperature range exceeds that possible for the filter design heating of the filter interface is possible. The heater power required for an individual filter shall not exceed 15W (TBC).

Filter	Max Sun Pointing (°C)	Min Sun Pointing (°C)	Min Anti-Sun Pointing (°C)
PHI HRT	300	-20	-130
PHI FDT	325	-25	-120
STIX	285	-20	-210

Table 15 PHI Heat Rejection Window Temperature Range (TBC)

5. Nominal operations of the instruments shall be performed with the external spacecraft provided feed through doors open. However certain operations require the door to be closed. When applicable the PI shall consider the environment of the instrument with the door closed based on door and feedthrough temperatures as provided in the table below. Naturally when the door is closed the solar flux environment incident on the instrument is not applicable.

S/C Orientation	Door		Baffle (T_B) °C	
	Max. (°C)	Min. (°C)	Max. (°C)	Min. (°C)
Sun Pointing	610	120	300	-80
Anti-Sun Pointing	-70	-150	20	-130

Table 16 Instrument Feedthrough and Baffle Temperatures with Closed Doors (TBC)

4.7.4.1 Conductive Interface

URP Location

1. For electronics, it shall be located on the unit mounting baseplate or one of the feet.

2. The URP(s) shall be unequivocally described in the MICD.

URP Temperature Range (T_{URP})

3. Internally mounted units shall be designed internally against the following temperatures, which are to be understood as the design temperatures (all values TBC). In the unit thermal

analyses or during tests, the unit shall be assumed to be thermally coupled to a conductive sink at the SRP temperatures. The SRP temperature level shall be adjusted in the analyses or during tests that the unit URP is equal to:

Operational	-20 °C / + 50 °C
Non-Operational	-30 °C / + 60 °C
Switch-On	> -30 °C

NOTE: The internal design of internally mounted units shall be done against the unit qualification test temperature range at the URP.

SRP Temperature Range (T_{SRP})

4. It shall be assumed that externally mounted units are coupled to a conductive sink at the SRP design temperatures (all values TBC), equal to:

Operational	-20 °C / + 50 °C
Non-Operational	-30 °C / + 60 °C
Switch-On	> -30 °C

Maximum Net Conductive Heat Flux (Q_{IF})

5. This flux shall be limited to **500 W/m2** over the unit total contact area.

4.7.4.2 Radiative Interface

6. The radiative environment of internally mounted units shall be assumed as a black-body cavity at a temperature $\mathbf{T_R}$ (all TBC) equal to:
   - Cold case -20°C
   - Hot case +40°C

NOTE: The internal design of these units shall be done against the value above plus/minus a margin of 10°C for qualification.

7. The radiative environment of externally mounted units shall be assumed as a perfect (ε=1) black-body at a temperature $\mathbf{T_R}$, as defined in Table 14 and Table 15. This temperature is highly dependent on the details of the individual unit accommodation on the spacecraft.

8. The radiative flux from the RPW antennas shall be limited to 4500W/m^2 in all directions.

4.7.4.3 Cold Finger Interface

Cold Finger Design Temperature Range (T_{CF})

1. The CF interface temperature range ($\mathbf{T_{CF}}$) of the relevant unit is TBD.

Cold Finger Heat Rejection (Q_{CF})

2. The heat extracted through the cold finger interface, $\mathbf{Q_{CF}}$, shall be limited to the values TBD.

4.7.4.4 Hot Element Interface

Hot Element Design Temperature Range (T_{HE})

1. The HE interface temperature range ($\mathbf{T_{HE}}$) of the relevant unit is TBD.

Hot Element Heat Rejection (Q_{HE})

2. The heat extracted through the hot element interface, $\mathbf{Q_{HE}}$, shall be limited to the values TBD.

4.7.4.5 External Heat Loads

1. For units with apertures, the total solar loads (direct or indirect), $\mathbf{Q_{SOL}}$, through the apertures shall be used when designing the units.

2. The infrared loads, $\mathbf{Q_B}$ from the thermal baffle and reaching the inner part of the instrument through this aperture shall be computed with the temperature of the baffle, $\mathbf{T_B}$. The baffle shall be simulated by a perfectly black (ε=1) surface closing the aperture and maintained at temperature $\mathbf{T_B}$.

4.7.4.6 Unit Dissipated Heat

1. The power dissipated by a unit shall be specified, for all mission phases, in terms of:
    - (1) Steady levels correspondent to unit modes and BOL/EOL conditions;
    - (2) Timelines of variable power correspondent to unit modes and BOL/EOL conditions.

4.7.4.7 Instrument Heat Shield Aperture Environment

1. The instrument thermal design shall be compatible with the absorption of solar flux and IR flux from the baffle that falls outside of the edge of the payload pupil.

The width of the area surrounding the instrument pupil that is exposed to solar flux is defined in the table below. Due to thermo-elastic distortion the solar flux area may shift from the centre of the payload boresight by ±1mm (TBC) in either the spacecraft Y and Z direction. The worst case analysis of the incident heat flux on the area outside the payload pupil shall consider this.

The solar flux environment and the baffle temperatures used to determine the IR flux is as defined above.

	STIX	PHI-HRT	PHI-FDT	EUI-HRI alpha	EUI-HRI 174	EUI-HRI 195	EUI-FSI	SPICE
Corona width (mm)	1.21	1.36	1.36	1.32	1.3	1.34	1.36	1.53

Table 17 Solar flux Impinging on Instrument Outside of the Instrument Pupil (TBC)

4.7.5 THERMAL HARDWARE INTERFACES

4.7.5.1 Spacecraft Temperature Sensors Interfaces

Spacecraft Temperature Sensing Concept

Temperature sensors are classified according to the nomenclature of Figure 30.

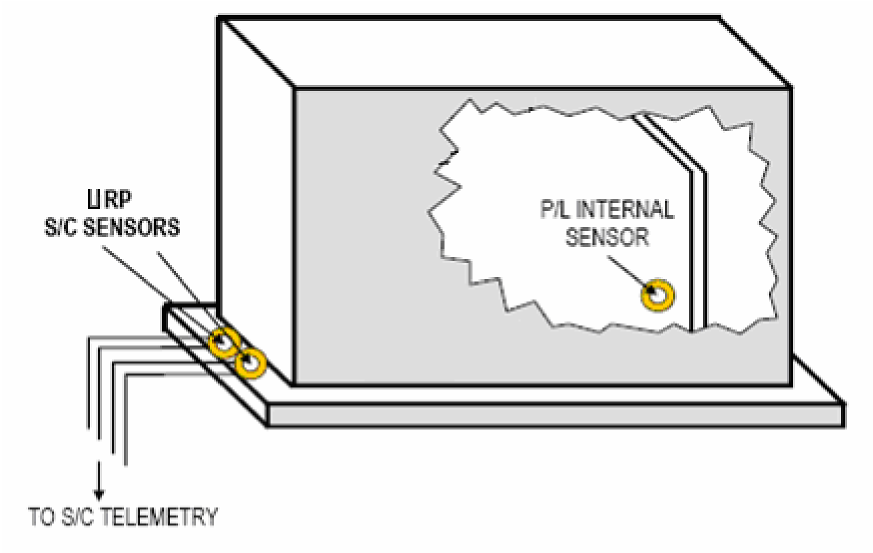


Figure 30 Temperature Sensors Classification

P/L Internal Sensors

Temperature sensors referred as P/L internal sensors are under PI responsibility. They are part of the unit design.

S/C URP Sensors

Temperature sensors referred as URP sensors are under spacecraft TCS responsibility. They are part of the S/C thermal design.

Each URP temperature will be monitored by a sensor and its reading will be available in the spacecraft telemetry at any time.

Each unit URP location will be equipped with a nominal and a redundant URP temperature sensor, i.e. 2 temperature sensors.

4.7.5.2 Spacecraft Heaters Interfaces

Heaters are classified according to the nomenclature of Figure 31.

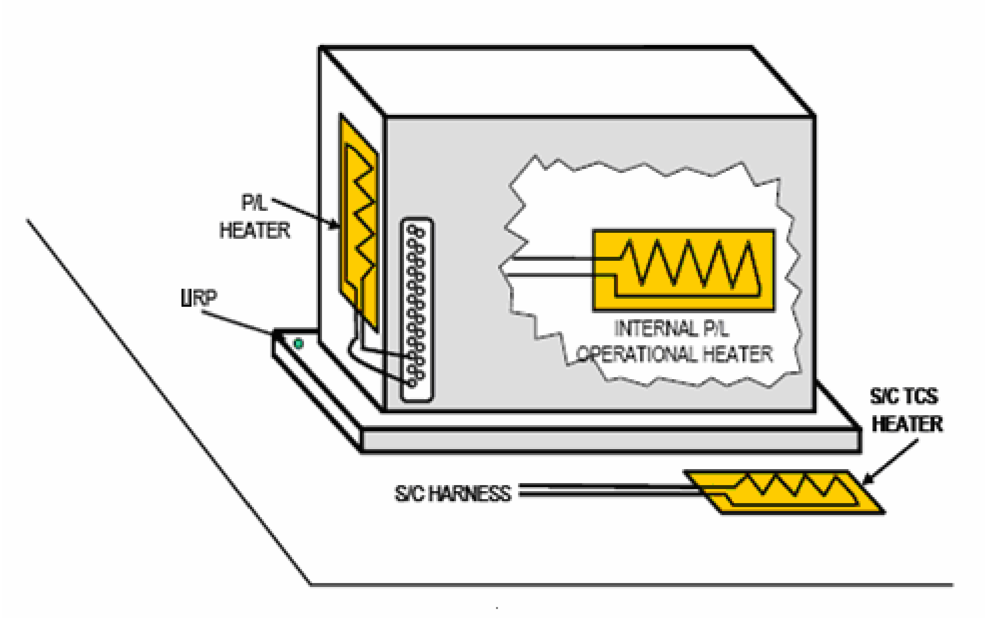


Figure 31 Heaters Classification

<u>Spacecraft TCS Heaters</u>
Under the spacecraft TCS responsibility, they are intended to maintain the URP temperature within the specified range. The PI may have to accommodate spacecraft TCS heaters directly on the unit structure.

1. To ensure failure tolerance, any instrument internal heaters shall provide redundancy (i.e. prime and redundant heaters). The prime and redundant heaters shall also be in separate mats, i.e. two heater mats per each heater line.

<u>Internal Instrument Operational Heaters</u>
Under the PI responsibility, they are intended to support the unit operations profile. They are part of the unit design.

4.8 Electrical Design Requirements

4.8.1 ELECTRICAL POWER DESIGN AND INTERFACE REQUIREMENTS

4.8.1.1 Definitions

LCL - Latching Current Limiter
PCU - Power Converter Unit
PDU - Power Distribution Unit
EPS - Electrical Power Sub-system
DNEL - Disconnect Non Essential Loads

4.8.1.2 Power Generation and Distribution Architecture

The satellite EPS will generate, condition, control, monitor, and distribute electrical power to the spacecraft users from the **regulated bus**, and manage battery charge and discharge to fulfill the power demands throughout all mission phases.

Independently of the mission phase instrument units will receive regulated 28V D.C. (TBC) electrical power from the solar array and/or the batteries through the Power Distribution Unit (PDU). The PDU will provide the following types of power interfaces normally in cold redundancy:

- Latching Current Limiters (LCL)
  1) for nominal instrument power supply purposes;
  2) for Actuators of Doors and other mechanical devices
  3) for special cooling / heating devices
- Non-Explosive Actuators Interfaces
  - Max No-Fire Current: TBD
  - All Fire Current: TBD
  - Input Resistance: TBD
  - Isolation Resistance > TBD
- Pyro Actuator Interfaces (TBD)
- Solid State Switches (TS) (TBC)
  1) as Thermal Control Power Interfaces for Heaters
  2) as Thermal Control Power Interfaces for Peltier Elements

4.8.1.3 Instrument Power Supply

Main and Redundant Power Voltage:

1. The instruments shall be designed to operate with nominal performance within the following steady state voltage limits (TBC) provided by the PDU:
   - Power Bus Voltage = 28 V:
     - Min: 26 V
     - Max: 29 V

   This applies for both Main and Redundant Lines.

Voltage Fluctuations:

2. All users of these power lines shall safely survive any standing or fluctuating voltage in the full range 0 V to 32 V. In case of failure in the power sub-system a transient of 1 ms and 33 V may be generated and shall be survived by the experiment without failure or performance degradation.

3. Instrument computers shall reset and resume operations from a defined mode in case of power on/off cycling.

4.8.1.4 Power Interface Requirements

Redundancy:

4. The instrument will be provided with two independent power lines routed via two dedicated connectors. The instrument shall be designed accordingly.

Note: *In case of failure, both the nominal and the redundant power lines may be applied simultaneously therefore isolation shall be included in the instrument to avoid loss of one power source by a failure in the other power source.*

Short Circuit Protection:

5. Equipment shall survive an instantaneous short circuit occurring on the external power line.

Inrush Current:

6. The start-up power characteristics shall be compatible with the current limiter characteristics given in Figure TBS. The instrument shall limit its inrush current to TBD A/μS.

Initial Electrical Status:

7. The equipment shall survive an intentional or unintentional switch-off in any configuration without degradation of nominal performance.

8. When powered up, equipments shall have an initial electrical status (except for latching relays if used), which is reproducible and identified in the EID Part B, which is independent of the switch-off configuration.

9. This status shall be safe, i.e. no degradation of nominal performance shall be caused if this initial status is kept for an unlimited time.

Latching Current Limiters (LCL)

The following LCL classes are provided by the satellite PDU:
- Class 1: Trip-Off Current = TBD
- Class 2: Trip-Off Current = TBD
- Class 3: Trip-Off Current = TBD

Solid State Switches

The design and limitations of solid state switches is TBD.

Actuators

10. The instruments shall not use explosive devices (pyros), but use Non-Explosive Actuators (NEA) instead. The performance interface characteristic of the used NEA shall comply with the relevant interface requirements.

Bus Impedance

TBD

Power Line Budgets

TBD

Power Quality

TBD

Isolation between Primary and Secondary Power Lines

11. The isolation between primary and secondary power lines in the instrument shall be TBD.

4.8.2 DATA HANDLING ELECTRICAL INTERFACE DESIGN

4.8.2.1 SpaceWire Software Interface Protocol

At instrument level, a SpaceWire Link Interface building block must be implemented to interface with the network / packet router. The implementation must be compliant up to the packet & network levels of SpaceWire Std. The electrical specifications of a SpaceWire Link Interface are defined by the ECSS-E-50-12 document [NR6].

As reminder, a SpaceWire Link Interface consists of:

- Link assembly based on cables and connectors
- LVDS drivers
- SpaceWire Codec
- FIFO instrument applicable dependant

1. The instruments shall be able to receive SpaceWire time codes according to ECSS-E-50-12 [NR6].

4.8.2.2 Monitoring and Synchronization Interfaces

2. Sharing of data between instruments shall only be allowed within the capabilities of the Spacewire link (Time stamping accuracy better than ~1 millisecond (TBC); Instrument to Instrument package delivery time better than ~10 milliseconds (TBC)).

4.8.2.3 Allocation of Lines and Redundancy

3. For redundancy, each instrument shall support 2 independent SpaceWire lines, which will be addressed and routed separately to the spacecraft DHS.

4.8.2.4 Digital Telemetry

TBW

4.9 Software Design and Interface Requirements

4.9.1 SOFTWARE DESIGN REQUIREMENTS

1. All on-board software shall comply with the software standard ECSS-E-40 [NR20].

2. In view of the in-flight software maintenance the instrument software shall support the following requirements:

   Functionally distinct areas of memory shall be assigned to
   - code;
   - fixed constants;
   - variable parameters.

3. A minimum boot software shall reside in PROM. All functions of the instrument shall be accessible from the minimum boot state including EEPROM updates, or any direct hardware test functions.

4. On-board S/W shall be structured such that modifications can be made to a software module without affecting other module positions in the memory.

5. On-board S/W maintenance activities shall not cause a blockage of the instrument and can be cleared by a power cycling of the instrument.

6. The instrument software design shall ensure that erroneous operation cannot cause a safety hazard.

7. As a goal, the resources utilized by on-board software shall be telemetered (e.g., memory usage, central processor unit (CPU) usage and I/O usage).

8. The capability shall be provided to check that on-board software has been correctly uploaded before enabling it.

9. Enabling of on-board software should use only a single telecommand.

10. Any communication between the ground and an on-board software function or software task shall be effected by means of telecommand and telemetry source packets specifically designed for the purpose.

11. Whenever a condition that forces a processor reset is detected by software, an event report shall be generated prior to enforcement of the reset.

12. Whenever a processor overload condition is detected, an event report shall be generated.

13. Whenever an unexpected arithmetic overflow condition is detected, an event report shall be generated.

14. Whenever an illegal program instruction is encountered during execution of a program code, an event report shall be generated.

15. Whenever a data bus error is detected by software, an event report shall be generated.

16. Whenever a memory corruption is detected by an error detection and correction mechanism, an event report shall be generated.

17. Whenever a checksum error is detected, an event report shall be generated.

18. The software shall feature a modularized structure, i.e. a split of the software into controllable and exchangeable units.

4.10 Electromagnetic Design and Interface Requirements

4.10.1 GENERAL CONCEPT

The EMC requirements are to ensure the proper system functions which are characterized by low electrical sensor signals and extensive use of computer high rate digital equipment.

The spacecraft/payload EMC requirements described in the following sections cover the system-imposed aspects:

- Design requirements, which ensure a coherent satellite system design
- Performance requirements, at system and unit levels, which quantify the applicable emission and susceptibility levels to ensure the required safety margin at system level.

4.10.2 DESIGN REQUIREMENTS

4.10.2.1 Grounding and Isolation

The satellite system applies a modified distributed single point ground scheme based on the following principles applicable also to instruments:

Power:
1. The primary DC power is grounded to structure in one point within the power subsystem only.
2. All return lines shall be isolated from the structure.
3. Each unit shall generate his own secondary power isolated from the input power.
4. The return of each secondary power shall be connected to the structure in one point ONLY by an external removable connection which serves as the signal reference ground, for all the circuits fed by that secondary power.
5. The signal reference ground shall be connected to structure via a unit-external and removable grounding strap.
6. Power Lines Isolation shall be in accordance with section 4.10.2.2.

Signal grounding and isolation:
1. Between electrical units, signal driver outputs shall be referenced to ground and signal receiver inputs shall be isolated from ground.
2. The connection to ground shall be anyway made only on one side of electrical connections between units.
3. Isolating receivers shall provide common mode rejection capability. Balanced differential signals are preferred.

4.10.2.2 Electrical Bonding and Case shielding

1. Each unit shall be housed in a non-magnetic metallic case which shall form an electromagnetic shield.

2. The case shall not contain any apertures other than those essential for sensor viewing or outgassing vents. If outgassing vents are required they should be as small as possible (less than 5 mm in diameter) and should be located in the case surface which is closest to the spacecraft $–X_{OPT}$ face.

3. Electrical connectors for pyros and RF are to be considered as part of the case; all connectors shall include a metallic outer shell such that when the mating cable harness connector is inserted in the box mounted part, the whole connector is completely shielded. The shell of the box mounted part shall be bonded to the equipment case as required by this specification.

4. The case of each unit shall be grounded to the spacecraft structure, with a low impedance bonding strap.

5. The DC resistance across the electrical bond between any two adjacent parts, including connector shells, shall not exceed the following limit for both test polarities: 2.5 (TBC) mOhm (test at 100 mA and with both directions of polarity using a 4-wire measurement).

6. Non-metallic conductive structure parts shall be electrically bonded to the metallic reference in order to avoid differential charge build-up. This applies also to external thermal blankets and baffles.

4.10.2.3 Cable Shielding and Separation

Power and signal lines shall be grouped into the following EMC classes:

- Class 1: Power lines and Heater Lines
- Class 2: Digital lines (TM/TC)
  Non-sensitive analog lines (except RF)
- Class 3: Pyro / Mechanisms
- Class 4: Low level sensitive lines
- Class 5: RF

1. Lines of different EMC classes shall be separated by at least 5 cm.

2. Lines of different EMC classes shall not be routed parallel (as a goal)

3. Lines of different EMC classes shall be routed through separate connectors. Where this is not possible, separation shall be implemented by a row of grounded pins.

4. Redundant lines shall be routed through separate connectors.

5. Lines of Class 2 to Class 5 shall be shielded.
6. All line bundles of classes 1 to 5 shall have an overshield grounded to the structure at intervals of no more than 15 cm.
7. These rules will also be preferably applied within electrical units.
8. The active wire(s) shall be twisted with the return wire. The twisted wires shall be routed through a connector on adjacent pins to minimize the wire loop.
9. Cable shield shall not be used as the return path for signal or power.
10. Harness and connector layout shall permit the termination of cable shields at both ends.
11. For category 3 connectors shields shall be terminated on metallic shell all-over 360 deg.
12. For less sensitive signal lines only (i.e. class 2) the pig-tail connection to connector metallic shell is allowed.
13. The pig-tail length shall be less than 5 cm.
14. The ground connection of the shield via a connector pin is forbidden.
15. The unshielded length of any single cable shall not exceed 2.5 cm.
16. The resistance between harness shield and unit shall be less than 7.5 mOhm.

4.10.3 PERFORMANCE REQUIREMENTS

4.10.3.1 General

The requirements shall be met for any operating mode of the instruments. The assumed set-ups are as indicated in section 5 of this document.

Intentional signal emissions are not subject of the emission requirements below.

4.10.3.2 Conducted Emissions

Power Lines:

• Differential Narrow Band emissions: TBD
• Common mode emissions: TBD

Signal Lines:
• Differential Narrow Band emissions: 30 Hz ... 50 MHz: ≤ 50 mVpp (*) (TBC)
• Common Mode Narrow Band emissions: TBD

Note: The actual values may be circuit type dependent; in any case a 6 dB margin applies to ensure self compatibility.

4.10.3.3 Conducted Susceptibility

Conducted Susceptibility Time Domain:
The instrument units shall not exhibit any failures, malfunctions or unintended responses when the following voltages are superimposed on the primary power bus inputs (as per figure below). The voltages to be applied on the units connected on the regulated bus should be the ones here defined (TBC):

- Injection mode: DM (slow)
  - Max Voltage: ± 3 Vp
  - Duration: 700 µsec
  - Repetition frequency 10 Hz
  - Applied time 5 min

- Injection mode: DM (fast)
  - Max Voltage: ± 14 Vp
  - Duration: 10 µsec
  - Repetition frequency 10 Hz
  - Applied time 5 min

- Injection mode: CM (slow)
  - Max Voltage: ± 12 Vp
  - Duration: 10 µsec
  - Repetition frequency 10 Hz
  - Applied time 5 min

- Injection mode: CM (fast)
  - Max Voltage: ± 5 Vp
  - Duration: 0.15 µsec
  - Repetition frequency 10 Hz
  - Applied time 3 min

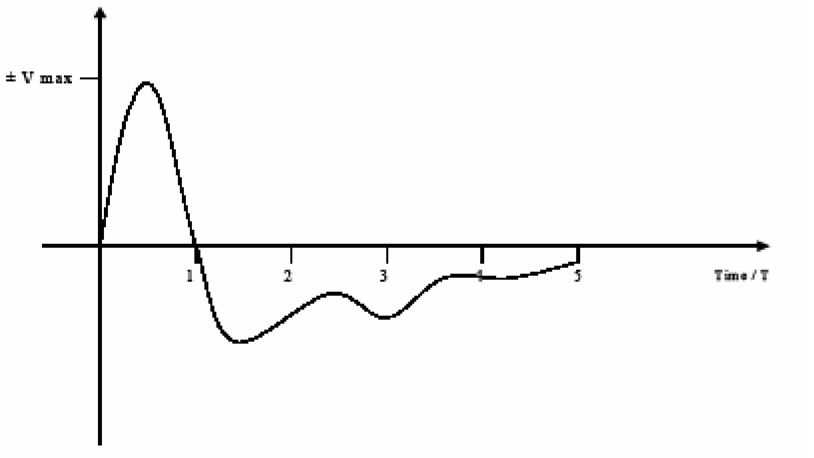


Figure 32 Primary Power Bus Inputs

Conducted Susceptibility Frequency Domain:

Signal Lines:
- Differential sinusoidal signal: TBD
- Common mode sinusoidal signal: TBD

Power Lines:
- Differential sinusoidal signal: 30 Hz ... 50 MHz: ≥ 0.5 Vrms (TBC)
- Common Mode: TBD

4.10.3.4 Radiated Emissions

E-field:
- Narrow Band emission limits are: TBD
- Broad Band emission limits are: TBD

H-field:
- Static field, 1 m distance: ≤ 90 dBpT (TBC)
- Narrow Band emission, 1 m distance: 30 Hz ... 50 KHz: ≤ 60 dBpT (r.m.s) (TBC)

4.10.3.5 Radiated Susceptibility

E-field:
1. Unit shall operate with nominal performance when exposed to an electrical field of:
   - TBD V/m rms in the frequency range from TBD KHz – TBD GHz
   - TBD V/m rms from TBD to TBD GHz

2. The sweep shall be less than one octave per minute and the signal shall be amplitude modulated to 30% by a 1 KHz square wave (TBC).

H-field:

- Sinusoidal magnetic field, 30 Hz ... 50 KHz: ≥ 130 dBpT (r.m.s) (TBC)
- Static magnetic field: ≥ 160 dBpT (TBC)

4.10.3.6 Electrostatic Discharge Immunity

1. Each instrument unit shall operate with nominal performance under Electrostatic Discharges (ESD) with the following characteristics:
   - Conducted ESD (current injected in the structure of the equipment):
     Imax: ≤ 25 A
     Rise time: < 5 ns (10-90%)
     Duration: 30 nsec. (at half amplitude)
     Repetition rate: 10 Hz
     Min. duration: 1 min

   - Radiated ESD:
     Spark gap discharge at 30 cm of the unit and harness
     Energy: 15 mJoules
     Voltage: ≥ 10 kV
     Repetition rate: 10 Hz
     Min. duration: 1 min

2. Materials selection shall take into account internal spacecraft charging, as specified in 6.3.3.3d of ECSS-E-20A [NR7].

4.10.3.7 Electrostatic Cleanliness (of space exposed surfaces)

1. Any payload space-exposed surface or surface coating shall be electrically conductive with a surface resistivity less than 100kOhm / square, and bonded to structure with a resistance less than 10 (TBC) Ohm.

2. Exposed, insulating dielectrics should be avoided.

3. Higher surface resistivity is acceptable only if it can be demonstrated that the differential charge of the surface when exposed to space plasma is less than 10 V.

4. There shall be no harness dielectric exposed to space plasma environment.

5. Electrostatic discharge at high voltage units (more than 200V) shall be avoided by applying par. 5.9 of ECSS-E-20A [NR7].

6. To control the differential charging potential of the spacecraft exposed to the plasma environment the following design requirements shall apply for all surface exposed to the plasma environment:
   - In order to achieve a conductive outer surface of the spacecraft, the sheet resistance of materials of any outside surface shall be less than 100 kOhm / square.
   - All external/internal metallic parts without area consideration (such as metallic labels, baseplates, tyraps, insulated electrical circuits ....) and intrinsically conductive parts (like carbon) that do not perform any electrical function shall be grounded to the main structure by a DC resistance lower than 1kOhm. Floating metallic parts are strictly prohibited without any area consideration.

4.10.3.8 DC Magnetic Cleanliness

Magnetic cleanliness practices will be applied wherever possible in the spacecraft and payload.

A system level magnetic budget will be prepared in the course of the definition and implementation phase (TBC).

1. The PI shall list the magnetic elements of the instrument and provide an estimate of the residual magnetic moment.

2. The spacecraft shall provide the following magnetic cleanliness: TBD at TBD meters

3. The instruments shall not produce more than TBD at TBD m.

4.11 Instrument Handling

4.11.1 TRANSPORT CONTAINER

1. The PI shall provide an adequate transport container for the instrument units.

4.11.2 CLEANLINESS

TBW

4.11.3 PHYSICAL HANDLING

TBW

4.11.4 PURGING

During the system AIV Programme, purging can be provided (continuously or regularly) to the instruments with stringent cleanliness requirements with dry nitrogen. The purging system may be interrupted during the spacecraft environmental testing and will be disconnected from the supply shortly before integration on the launcher.

1. Individual purge rates for each instrument shall be specified by the PI and will be controlled by means of pipe throttling.

2. The exact location of the purging interface shall be agreed with the ESA Project Office and the selected Prime and defined in the relevant EID-B.

3. The PI shall specify the purging requirements for the instrument during spacecraft integration and testing, transportation and the launch campaign.

4.12 Environment Requirements

The Solar Orbiter Space Segment shall be designed to withstand the environment defined in the Mission Environment Specification [ID2], predicted for the worst case extended mission duration.

4.12.1 CLEANLINESS

Contamination is addressed in the Solar Orbiter Environment Specification Document [IR2] and a Solar Orbiter Cleanliness Plan will be prepared based on inputs from PI instrument cleanliness plans.

1. The PI shall provide an instrument contamination plan.

4.12.1.1 Particulate and molecular Cleanliness

A potentially major risk for the programme is the UV fixation of out-gassing materials on the instrument optics.

To estimate the AIT contamination budget, it can be assumed that purging of the instrument apertures will be available throughout the AIT sequence except during TB/TV tests (TBC).

To estimate the molecular contamination level at the end of the S/C AIT sequence, the following inputs are considered:

- Typical contamination 10^{-6} g/cm^2/year in a clean room environment
- Typical contamination 5E10^{-8} g/cm^2 for the thermal vacuum facilities

According to these assumptions, at the end of the AIV/T activities a molecular contamination level of 7E10^{-8} g/cm^2 (TBC) is expected on the entrance windows.

1. The PI shall define the particulate and molecular cleanliness levels for the instrument with justification.

4.12.1.2 Spacecraft Charging

Both absolute charging (with respect to the plasma wind) and differential charging (i.e. non-equipotential surfaces) need to be addressed. Sun illuminated surfaces are expected to charge positively, due to the emission of photoelectrons while surfaces in shade will have the tendency to charge negatively due to the impact of the ambient ions and electrons.

The following points will be elaborated by the Prime Contractors during the Formulation Phase:

- Realistic spacecraft charging requirements need to be established in the future project phases, including specific analysis.
- It will be difficult to impose additional electrical charging requirements on the already challenging development of Sun illuminated surfaces such as the Heat shield and solar arrays.
- Differential charging requirements < 10 V will be technically very difficult to achieve and is likely to result in a large cost impact of the overall mission.
- A more realistic approach is to provide specific spacecraft areas, in proximity of the charged particle sensors (e.g. EAS), with more stringent charging requirements, with particular reference to negative potentials. This option also includes placing the relevant sensor on the boom, although the impact of the solar arrays would still have to be considered.

4.12.1.3 Magnetic Cleanliness

The electromagnetic environment must be stable in time as well as contained. The maximum stray DC magnetic fields at the location of the magnetometer sensor shall be 10 nT.

The possibility to guarantee a DC field stability of order 10% of the nominal field over 1 day is TBC.

1. The PI shall list all magnetic materials in use in their instrument with an estimated magnetic dipole moment for the instrument.

4.12.2 RADIATION

The Solar Orbiter Space Segment shall be designed to withstand the radiation environment, defined in the Mission Environment Specification [ID2], predicted for the worst case extended mission duration. The Mission Environmental Specification provides mission-integrated data for solar energetic particles at the 90% and 95% confidence intervals. This is a statistical risk analysis and therefore does not include the worst case of a single large solar particle event when below 0.3AU.

Proton Energy [MeV]	Solar Orbiter - Integrated Solar Proton Fluence [#/cm2]	
	Nominal Science Phase	Mission Total
0.1	2.84E+12	3.17E+12
0.5	1.75E+12	2.15E+12
1	1.22E+12	1.50E+12
2	7.31E+11	8.82E+11
3	4.87E+11	5.98E+11
4	3.61E+11	4.41E+11
5	2.92E+11	3.49E+11
6	2.44E+11	2.87E+11
8	1.75E+11	2.06E+11
10	1.30E+11	1.58E+11
12	1.10E+11	1.31E+11
15	8.93E+10	1.04E+11
17	7.71E+10	9.24E+10
20	6.50E+10	7.76E+10
25	4.87E+10	5.98E+10
30	3.74E+10	4.77E+10
35	3.17E+10	3.97E+10
40	2.72E+10	3.37E+10
45	2.40E+10	2.93E+10
50	2.07E+10	2.55E+10
60	1.62E+10	1.98E+10
70	1.30E+10	1.56E+10
80	1.06E+10	1.25E+10
90	8.53E+09	1.01E+10
100	6.90E+09	8.26E+09
120	4.87E+09	5.65E+09
140	3.41E+09	3.97E+09
160	2.48E+09	2.82E+09
180	1.83E+09	2.05E+09
200	1.34E+09	1.50E+09

Figure 33 Solar Proton Fluence as a function of Energy for a 90% Risk Level (5% probability of numbers being exceeded) and Employing the Inverse Square Scaling

1. Electronic equipment design shall be based on components and sensors which have been proven to withstand the expected radiation environment. The ECSS-E-10-12 Standard on Methods for Calculation of Radiation Effects [NR4] shall be applicable.

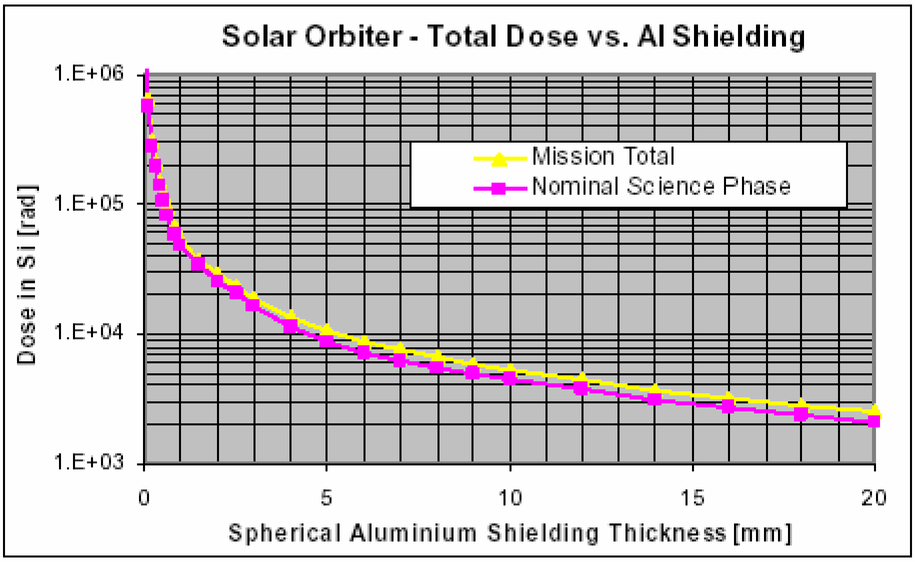


Figure 34 Dose in Silicon as a function of spherical aluminium shielding

Aluminium shielding thickness [mm]	Total ionising radiation dose in Si [rad]	
	Nominal Science Phase	Mission Total
4.00E-01	1.40E+05	1.51E+05

Table 18 Dose in Silicon for spherical aluminium shielding

Internal unit and detector layouts and the optimization of the local shielding shall be the responsibility of the PI, assuming a radiation shielding provided by the spacecraft of 0.4 mm Al (TBC).

Detector damage and damage to some classes of electronic components is due to “displacement damage”. The parameter used to quantify the effect and characterize the environment is non-ionizing energy loss (NIEL), rather than the ionizing dose [NR4].

Figure 35 provides the NIEL “dose” as a function of shielding depth [ID2].

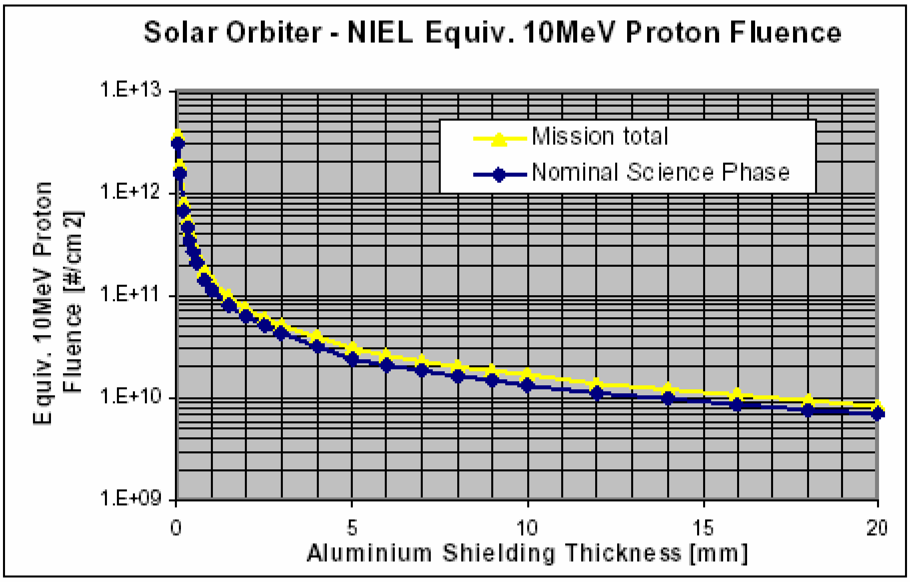


Figure 35 Non-Ionising Energy Loss equivalent 10 MeV Proton Fluence as a function of Shielding Thickness for the Mission

Aluminium shielding thickness [mm]	Non ionising energy loss equivalent 10MeV proton fluence [#/cm2]	
	Nominal Science Phase	Mission Total
4.00E-01	3.44E+11	3.97E+11

Table 19 Non Ionizing Dose (in 10MeV equivalent proton fluence) for spherical aluminium shielding

4.12.2.1 Single Event Upset (SEU)

A single event upset is a soft error consisting of bit-flips with no preference for 1 to 0 versus 0 to 1 transitions. The bit-flips can cause loss of a pointer, resulting in a micro processor stopping, or corruption of a data register which could cause a control parameter to be altered.

The other important parameter with respect to SEU is the critical charge which is the lowest ionization-generated charge required to produce upset in the device. For normal incidence this is directly related to Linear Energy Transfer (LET) and LET spectra form part of the analysis method as defined in ECSS-E-10-4 [NR5] and ECSS-E-10-12 [NR4].

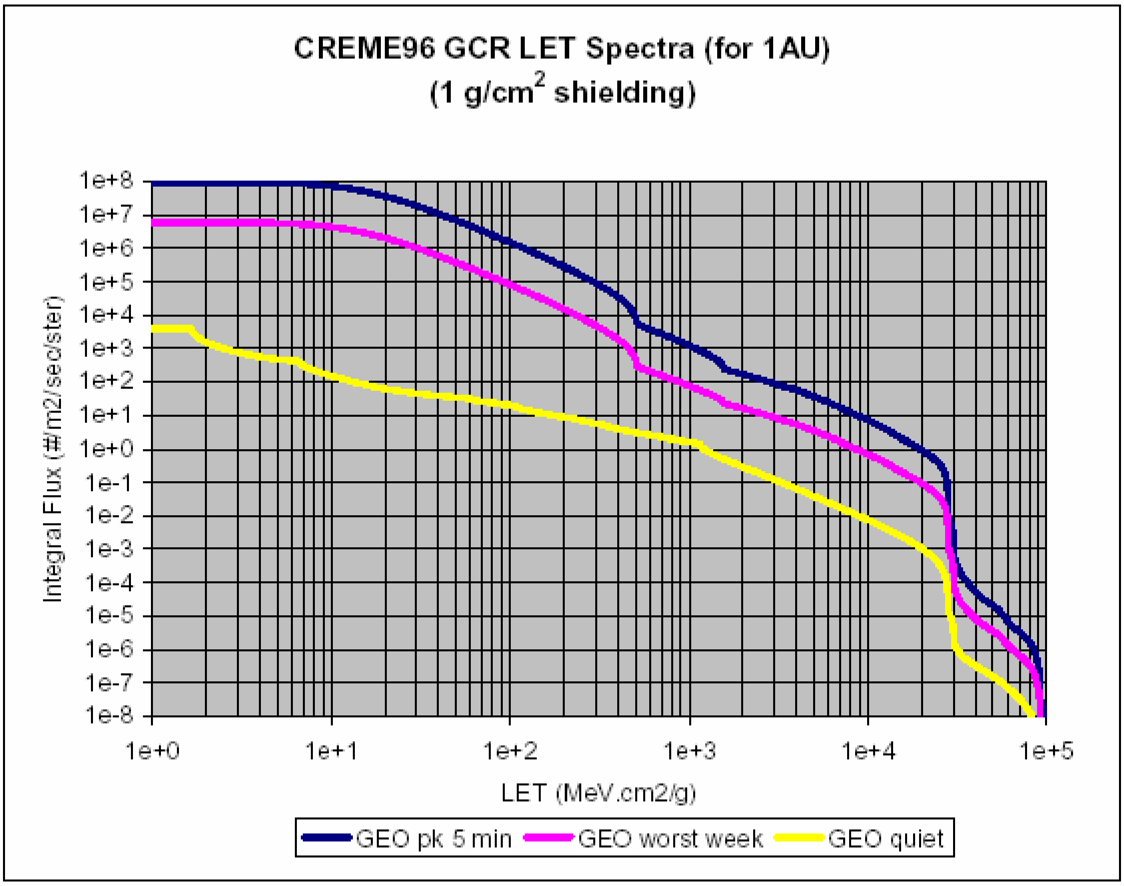


Figure 36 CREME96 Galactic Cosmic Ray Linear Energy Transfer (LET) Spectra for the three levels of activity, nominal (Geostationary Earth Orbit (GEO) quiet), worst week/worst case, and peak 5 minute for a component shielded by 1 g/cm2. The predictions are for 1AU.

4.12.2.2 Requirements

1. Instrument design shall be based on existing RAD HARD components as much as possible. Parts having a LET threshold for SEU of less than 25 MeV cm^2/mg shall not be used in critical circuits. Parts sensitive to SEL (Single Event Latchup) with a LET threshold of less than 100 MeV cm^2/mg shall not be used.

2. Devices not currently available as RAD HARD shall be characterized for radiation tolerance before incorporation into the design. The ESA Project Office shall be consulted in such cases.

3. The instrument shall be designed to cope with the environments defined in [ID2].

4. The effect of radiation on detectors, shall take full account location and shielding, and of <u>potential scattering of low energy particles in detector enclosures</u> and temporary directed fluxes of SEPs.

5. The PI shall define any parts or locations that require special shielding.

4.12.3 MICROMETEORITE ENVIRONMENT

The Solar Orbiter Space Segment shall be designed to withstand the micro-meteorite environment defined in the Mission Environment Specification [ID2], predicted for the worst case extended mission duration.

5 OPERATIONAL REQUIREMENTS

The purpose of this section is to identify the requirements on the Solar Orbiter instruments for the conduct of all mission operations.

5.1 Ground Segment Description

5.1.1 OPERATIONAL GROUND SEGMENT

1. The Solar Orbiter Mission Operations Centre (SMOC) at ESOC (Darmstadt, Germany) will be responsible for the operation and control of the spacecraft during the cruise phase as well as after the second Venus Gravity Assist (VGA-2), which will inject the spacecraft in the desired Sun observation orbit.

2. The Solar Orbiter Ground Segment will provide capabilities for monitoring and control of the spacecraft and payload during all mission phases, as well as for the reception, archiving and distribution of payload instrument data.

3. The ground segment is composed of:
    a. A Ground Station and Communication Network performing telemetry, telecommand and tracking operations within the X/Ka-band frequencies. The ground station used throughout all mission phases will be the ESA New Norcia (NNO) deep space terminal, complemented by the ESA Cebreros 35m station during near-Earth mission phases and other critical mission phases.
    b. Solar Orbiter Mission Operations Centre (SMOC) located at ESOC, Darmstadt, Germany including:
        i. The Solar Orbiter Mission Control System (SMCS), to support with both hardware and software, the data processing tasks essential for controlling the mission, as well as spacecraft performance evaluation and software validation.
        ii. The Solar Orbiter Data Disposition System, supporting the acquisition and interim storage of raw scientific data, to be accessible together with raw housekeeping and auxiliary data at remote locations.
        iii. The Solar Orbiter Mission Planning System, supporting command request handling and the planning and scheduling of spacecraft and payload operations.
        iv. The Flight Dynamics System, supporting all activities related to attitude and orbit determination and prediction, preparation of slew and orbit manoeuvres, spacecraft dynamics evaluation and navigation in general.
        v. The System Simulator, a software simulator of the ground stations and space segment, to support procedure validation, operator training and the simulation campaign before each major event of the mission.

4. A Solar Orbiter Science Ground Segment (SO-SGS), to provide pipeline processing of all scientific mission data in support of the PI teams and prepare the ingestion of the scientific data products into the planetary Science Archive of ESA. For Solar Orbiter science operations, the SO-SGS will include a Science Operations Centre (SSOC), responsible for scientific mission planning and experiment command request preparation for consolidated submittal to the SO Mission Operations Centre (SMOC).In case of a catastrophic event at ESOC, preventing spacecraft control over periods beyond the maximum survival capabilities of the Solar Orbiter spacecraft, a Back-Up Control Centre will be available in another location, currently baselined to be the Cebreros ground station. This centre will include all basic ground segment equipment to safe the spacecraft until the nominal control centre returns operational.

5.1.2 SCIENCE GROUND SEGMENT

1. The Solar Orbiter science operations shall be conducted from a Science Operations Centre (SOC) as part of the Science Ground Segment (SGS), in close collaboration with the PI Teams and the Mission Operations Centre (MOC). The SGS shall be responsible for the SO scientific operations coordination and planning, and archiving the SO data. The SGS shall also be responsible for establishing the SO Archive.

2. The format for data to be delivered to the SO Archive shall be compatible with the one defined for the ESA science data archive. The ESA science data archive shall be the repository of all mission products and will be based on the ESA's Planetary Science Archive.

3. The SGS Data Processing System is the core of the data handling. It processes the input telemetry stream from SMOC into the various data levels. 4 different data levels are distinguished. The telemetry stream from SMOC (Level 0b) is processed into scientific data units (Level 1b). These data units are still uncalibrated, but in a scientific known format as e.g. FITS, HDF, CDF, and PDS-labeled.

4. The Level 1b processor is a single piece of software with one configuration file per instrument and one or several configuration files for the S/C housekeeping data needed.

5. The Level 2 processor takes these data and calibrates them into the calibration level that is needed and agreed on with the instruments teams. One software module is needed for each instrument and the instruments teams shall deliver this module to the SGS before launch and maintain it afterwards in cooperation with the SGS.

6. The Level 3 processor is computing higher-level data, derived data or combined data from several instruments. This processing step shall be the task of the instrument teams and only executed in particular, well-defined, situations at the SGS.

7. The derived and higher level products that are produced within the instrument teams shall be delivered to the SGS at intervals TBD for each individual data product.

8. Level 1b and Level 2 data shall be access controlled in agreement with the PIs. Level 1b data shall be made accessible to the corresponding instrument teams in case of analysis problems and instrument malfunctioning. Typically, interested team members shall be prepared to be collocated to the SGS whenever necessary.

9. The details of the data delivery level and mechanism shall be worked out by SGS together with the PI teams at a later stage.

10. The PI shall deliver software tools to SGS for Level 1b to Level 2 (calibration software).

11. The PI shall support installation of level 0 to 1a/1b and calibration software at the SGS by collocating experts at the SGS for a period of typically several months.

12. The PI shall provide inputs for the Experiment modelling at the SGS, format to be defined.

5.2 Mission Operations

5.2.1 BASIC PRINCIPLES

5.2.1.1 General

1. Operations for both, spacecraft and scientific payload, will only be conducted in strict compliance with validated event sequences and procedures documented in the Flight Operations Plan. This encompasses all operations i.e. special operations and contingency operations as well as routine operations during the different mission operation phases.

2. The SMOC will switch-off any instrument which is deemed to be interfering with or endangering the mission objectives, using agreed and validated contingency procedures.

3. Science TM packets will not be processed at SMOC, so all information relevant to the health and safety of the payload and in general required for engineering activities on the instrument (monitoring and control, troubleshooting, software maintenance, etc.) shall be contained in non-science TM packets and follow the requirements specified in the Operation Interface Requirements Document (OIRD).

5.2.1.2 Off-Line Operations

1. Due to the one-way propagation delay of up to 16 minutes (maximum Earth distance of 2 AU), the spacecraft will be mainly controlled via off-line operations. After the initial spacecraft commissioning, all telecommands required to carry out the mission will normally be loaded in advance on the Mission Timeline for later execution. All telemetry generated on-board will be stored for later retrieval by ground.

2. Telemetry evaluation will also be mainly off-line, with limited possibility of quasi real-time intervention in selected critical phases and in major contingency cases.

3. In order to support the off-line operations approach required for a deep-space mission, the following autonomy capabilities shall be provided by the spacecraft:
   a. On-board Control Procedures, as a way to autonomously execute complex procedures including decision loops which ground cannot support due to propagation delay. On-board Control Procedures are modifiable in flight.
   b. Detection and autonomous recovery of any single failure, and reconfiguration to a safe back-up mode in case the detected failure is not recoverable.
   c. The spacecraft will be able to continue nominal operations (and generation of mission products) without ground contact during the longest non-communication period. In case of a non-recoverable failure, the spacecraft shall be able to survive for 7 days during cruise (between the end of the commissioning and up to the 2nd Venus flyby. These 7 days represent three times the longest expected non-coverage period) and at least the duration of the longest non visibility period by the ground station after VGA-2.
   d. Anomalies will only be detected by SMOC with a delay, corresponding at least to the light travel time, but typically rather of the order of one day in the case of daily passes, and two to four days in the case of cruise when 3 passes per week are planned. Reaction to on-board failures from the SMOC within these typical reaction times will require unambiguous identification of the failure in telemetry, and the related contingency procedures being contained in the instrument user manual (and translated in the FOP).
   e. The PI teams shall support the investigation and resolution of Instrument-related anomalies in-flight. This may include provision of technical consultancy, and presence of PI team technical experts at ESOC if required.

5.2.1.3 Ground Contact

1. The contacts between the Solar Orbiter Mission Operations Centre (SMOC) at ESOC and the spacecraft will not be continuous and will be primarily used for pre-programming of autonomous operations functions on the spacecraft, and for data collection for subsequent off-line status assessment. The ground contact frequency will vary between once per day and three times per week, depending on the mission phase.

2. **Science Cruise and Planet Swing-bys.** The period around a planetary swing-by is dedicated to navigation operations. In such arcs the frequency of the ground station passes

varies from 3 per week to one per day. Such phases usually start a few months before the event (typically 2 months before a planetary swing-by, 1 month before a deep space manoeuvre) and finish about one month later. Payload operations in these phases are conducted on a best effort basis. The term "science cruise" is used for the periods where the spacecraft activity is low and dedicated to the generation and downlink of science by the "in-situ" instruments. In such periods passes are taken 3 times per week for health checks and telemetry recovery. The platform operations are limited to routine maintenance activities such as reaction wheel off-loadings.

5.2.1.4 In Flight Thermal Characterization

1. Due to the difficulty to accurately predict the thermal behaviour of a spacecraft in all possible conditions during the deep space cruise and the full science orbit after VGA-2, thermal characterization campaigns will be performed during the Near-Earth commissioning and after arrival at the full science orbit, with the aim to define the operational thermal envelope of the spacecraft.

2. No on-line thermal modeling will be utilized in any phase of the mission. While in cruise the above thermal characterization approach is deemed sufficient, in the full science orbit a stable baseline plan is required and short term mission redefinitions based on updated thermal constraints are not affordable.

3. The entire operations planning for the full science orbit after VGA-2 will be produced on the basis of a robust, conservative but realistic set of spacecraft operations constraints that will ensure safe thermal operations during the observation windows.

5.2.1.5 Solar Conjunction Operations

1. The nominal RF link to-/from- the spacecraft will be degraded when the Spacecraft-Sun-Earth angle becomes lower than 5 degrees (based on X-band experience). Degradation of the Ka-band signal will have also to be taken into account in a similar angular range.

2. Degradation of the signal also affects tracking measurements. For this reason the mission shall be designed such that critical navigation activities (e.g. manoeuvres, planet swing-bys) do not take place within 5 (TBC) degrees angular separation from the Sun as seen from Earth.

3. The spacecraft shall be able to operate autonomously during the solar conjunctions, but it shall not be able to continue mission product generation continuously, as this would have driven the size of major onboard resources such as the Mission Timeline and the Mass

Memory. Therefore, science operations in this period shall have to be adjusted to the available on-board resources.

5.2.1.6 Reporting

1. The SMOC shall regularly report on the mission and spacecraft status with a frequency depending on the criticality of the mission operation:
    a. LEOP: Operations reports will be issued daily.
    b. Critical event: Operations report will be issued ad-hoc.
    c. Routine phases: Operations reports will be issued weekly to monthly depending on the level of activity. Contents and distribution lists of these reports will be agreed with the Mission Manager.

2. Anomalies shall be reported within one working day from their detection by SMOC to the Mission Manager, the Flight Operations Director, Industry (if still providing support to the mission). In case of anomalies affecting the payload, the Project Scientist, SGS, and affected PI will be added to the distribution list.

3. The PI teams shall issue instrument operations reports after each active non-science in-flight phase, i.e. one for the Near-Earth Commissioning phase, and one for each periodic check-out during cruise.

5.2.2 MISSION PHASES SUPPORT

The following mission phases have been defined:

Phase	Sub-Phase	Phase mnemonic	Duration	Start Criteria	End Criteria	Remark
Launch and Early Orbit Phase		LEOP	< 7 days	8 hrs before Launch	TCM to correct Injection errors	Launch date and associated LW to be defined by CReMA. Launch date 2017/01/04
Near Earth Commisioning Phase		NECP	90 days	TCM to correct injection errors	Start of NECP + 90 days	Activation and functional check-out of S/C and P/L
Cruise Phase	GAM V1	VGA1	90 days	VGA1 - 60 days	VGA1 +30	This phase overlaps with NECP. Most

					days	probably part of NECP to be done after VGA-1. Additional ground station support VGA1 date 2017/04/15
	Cruise 1	CR1	406 days	VGA1+ 30 days	EGA1- 60 days	In-situ instruments and periodic check out of Remote sensing instruments
	GAM E1	EGA1	90 days	EGA1- 60 days	EGA1+ 30 days	Additional ground station support EGA1 2018/08/25
	Cruise 2	CR2	641 days	EGA1+ 30 days	EGA2- 60 days	In-situ instruments and periodic check out of Remote sensing instruments
	GAM E2	EGA2	90 days	EGA2- 60 days	EGA2+ 30 days	Additional ground station support EGA2 date 2020/08/25
	Cruise 3	CR3	79 days	EGA2+ 30 days	VGA2- 60 days	In-situ instruments and periodic check out of Remote sensing instruments
	GAM V2	VGA2	90 days	VGA2- 60 days	VGA2+ 30 days	Additional ground station support VGA2 date 2021/02/09
Full Science Nominal Mission	1st resonant set of orbits	ORB1	584 days	VGA2+ 30 days	VGA3- 60 days	Full science with both set of instruments
	GAM V3	VGA3	90 days	VGA3- 60 days	VGA3+ 30 days	Additional ground station support. VGA3 date 2022/12/15
	2nd resonant set of orbits	ORB2	359 days	VGA3+ 30 days	VGA4- 60 days	Full science with both set of instruments

	GAM V4	VGA4	90 days	VGA4-60 days	VGA4+ 30 days	Additional ground station support. VGA3 date 2024/03/09
	Arc to 1st perihelion after VGA4	ORB3	76 days	VGA4+ 30 days	ENM	Full science with both set of instruments
End of Nominal Mission		ENM	1 day	ENM	ENM	ENM date 2024/06/23

Table 20 Mission Phases

The characteristic support required during the different operational phases is summarized in the following table:

Mission Phase	Operations Support
Launch and LEOP	• During LEOP the Mission Control Team covers 24 hours daily operations in two shifts of about 12 hours each. • The LEOP operations will be carried out from the Main Control Room (MCR), supported by the ESTRACK Control Centre (ECC), the Flight Dynamics Room (FDR), the Software Support Room (SSR) and the Project Support Room (PSR). • Launch support will start 8 hrs before launch and includes a final readiness test with the stations. After spacecraft separation from the launch vehicle, a series of configuration activities will be performed automatically by the spacecraft. The post-launch spacecraft operations will start immediately following Acquisition of Signal (AOS), when the control centre takes over control of the spacecraft and completes the initial configuration activities. • Execution of a Trajectory Correction Manoeuvre is planned during LEOP in order to cancel any possible orbit injection error by the launcher. • Ground station network is assumed to be New Norcia and Cebreros (use of a 15 m station for initial acquisition is TBC). • On site support from Project and Industry teams • Duration of the phase: less than 7 days;
Commissioning Phase	• Any remaining subsystem initialisation/switch on will be performed.

Mission Phase	**Operations Support**
	• Activation and functional checkout of the SolO spacecraft and payload; in particular, all RF links will be tested during this phase, and both sets of instruments will be commissioned. • The control centre operations will be carried out from the Dedicated Control Room (DCR) with the support of the Project Support Room (PSR). • The New Norcia 35 m ground station will be used over the full visibility through the CP. • The FCT will reduce the support to 1 shift. • Flight Dynamics will provide off-line support. • On-site support by Project, Industry and PI Teams for selected operations. • Duration: 3 months;
Cruise	• During this phase monitoring and maintenance activities in the spacecraft platform will be performed off-line. The size of the FCT is reduced with respect to LEOP and CP. • Reduced science operations are performed with the "in-situ" instruments offline through the normal mission planning cycle. • Periodic (typically twice per year) non-interactive check-out of the remote sensing instruments are planned. • New Norcia is used 3 times a week for a pass duration sufficient to provide 4 hour of science dump. • The support for the planetary swing-bys will typically start two months before the swing-by and finish 1 month after, and will mainly consist in trajectory correction manoeuvres and if required spacecraft configuration changes. Additional support from the 35 m antenna in Cebreros is required for Orbit Determination purposes (both conventional and delta-DOR tracking activities). • Operations during this period will be conducted from the DCR. • This phase starts after end of NECP and finishes at the end of VGA2.
Full Science Phase after 2nd Venus Flyby	• This phase is supported by a FCT with increased size. • Full science operations are performed with the entire payload complement offline through the normal mission planning cycle. • New Norcia is used daily for a pass duration sufficient to provide 4 hour of science dump. • The support for the planetary swing-bys will be the same as for the cruise phase (see above).

Mission Phase	Operations Support
	• Operations are conducted from the DCR.

Table 21 Support Required during Operational Phases

5.2.3 PAYLOAD OPERATIONS SUPPORT

Solar Orbiter payload operations will be governed by the rules and guidelines established and periodically discussed by the Science Working Team (SWT). The preparation, coordination and execution of science operations will be carried out differently in the various phases of the mission.

5.2.3.1 Near Earth Commissioning Operations

Payload operations during the Near-Earth Commissioning Phase are conducted in a near-real time manner, to support the critical post-launch initial activation and checkout activities, and taking advantage of the relatively short distance to Earth.

1. During Payload commissioning Near-Earth, all experiment operations shall be executed at the SMOC using a detailed phase timeline and related procedures established before to the start of the phase. Timelines and procedures will be defined by the SWT and the experiments' teams, produced by the ESOC Flight Control Team, reviewed and agreed by the PIs. After validation via the system simulator, they shall be included in the Flight Operations Plan.

2. In the Near-Earth Commissioning Phase it will be possible for the experiments' teams to submit change requests to procedures and/or timelines until very close to the execution time. These requests shall be discussed with the Flight Control Team in daily operations review meetings under the supervision of the Project Scientist and the Spacecraft Operations Manager.

3. A Principal Investigator Support Area (PISA) shall be provided at ESOC to accommodate PI provided EGSE to be used during Near-Earth Commissioning Phase, when critical payload operations will be conducted that might require near-real time interaction between the Flight Control Team and the Instrument Team for decision making.

4. The PISA shall enable data access and commanding capabilities, as well as communication with remote locations. At the PISA an interface with the DDS (identical to the remote interface) will be available to support both the telemetry delivery services to the experiment EGSE and special command requests from PIs to the SMOC.

5. In this phase the presence of PI team experts and installation of instrument EGSE equipment at the SMOC shall be required to monitor the operations execution in near-real

time (compatible with the availability of data at the SMOC) and to support GO/NOGO decisions at predefined steps in the procedures.

5.2.3.2 Cruise Operations

During cruise only the "in-situ" set of instruments will be operated on a best effort basis and the "remote sensing" instruments will go through periodic (twice per year approximately) check-outs.

1. The SSOC shall be responsible for planning all payload operations, while SMOC remains responsible for the overall mission planning and mission operations.

2. In the three months around "special cruise events" like planet swing-bys payload operations shall be supported on a best effort and non-interference basis, giving priority to the critical spacecraft navigation activities.

3. The PI teams shall submit their operations requests to the Solar Orbiter SOC, who will coordinate and prepare the necessary science plans in order to deliver to SMOC the list of needed payload operations requests (PORs). The SMOC shall process and merge the operations requests into a timeline to be uplinked to the spacecraft.

4. All activities in this phase shall be carried out off-line, according to the planning and deadlines established in the mission planning concept. The final instruments' checkout timeline generated at the SMOC will be checked against the mission rules and constraints and the available spacecraft environmental resources, iterated if necessary with the SSOC/PI and finally implemented in the mission timeline to be uplinked to the spacecraft.

5.2.3.3 Full Science Operations

After VGA2 the mission will enter in a full science operations phase, in which all instruments on board the spacecraft will perform scientific measurements.

1. The SSOC shall be responsible for planning all payload operations, while SMOC remains responsible for the overall mission planning and mission operations.

2. The SSOC shall be responsible for submitting consolidated payload operations requests to the SMOC at the level of command sequences. It is the responsibility of the SMOC to convert the submitted operations requests into commands and to ensure timely uplink to the spacecraft for execution. The interface between SMOC and SSOC will include a list of command sequences authorized for scheduling by the SSOC.

3. All activities in this phase shall be carried out off-line according to the planning periods and deadlines established in the mission planning concept. The inputs from the SSOC will be checked by the Flight Control Team at the SMOC against the mission rules and constraints and the available spacecraft and environmental resources, iterated if necessary

with the SSOC/PI and finally implemented in the mission timeline to be uplinked to the spacecraft.

5.2.3.4 Payload On-Board Software Maintenance

1. Responsibility for maintaining the instrument on-board software shall remain with the PI team throughout the mission.

2. ESOC shall provide the facilities and services required to safely uplink and install onto the instrument during flight the required software modifications, as developed by the PI team and delivered through an agreed interface and format.

3. The on-board software maintenance support service provided by ESA/ESOC for Solar Orbiter shall therefore be as follows:
    a. Pre-launch, the PI team provides in the user Manual a generic software maintenance procedure, which contains the detailed steps to configure the instrument in its maintenance mode and the constraints related to any in-flight software maintenance activity.
    b. In flight, when an instrument software change is required, the PI team will develop, check and validate at instrument level the required change.

4. The PI team shall then submit memory maintenance requests in form of text files in an agreed format. Such requests include Memory Patch Requests, Memory Dump Requests, Memory Check Request. As part of the request, the PI team indicates a time window where the memory maintenance request has to be executed.

5. ESOC shall be responsible for converting the text files input into Memory Maintenance commands (Service 6). These commands are uplinked to the mass memory as a TC file for delayed execution.

6. ESOC shall be responsible for scheduling and executing the maintenance activity. Instrument pre- and post-maintenance operations are executed as specified in the instrument User Manual, normally from the Mission Timeline, unless requested otherwise by the PI. When the instrument is ready to receive the maintenance commands, the execution of the corresponding TC file is started and the on-board system issues the maintenance commands to the instrument.

7. The PI shall be responsible for the verification of correct loading of the experiment software updates, since science telemetry processing is not performed at ESOC.

8. If requested by the PI, telemetry generated by the maintenance commands (dump / check) can be compared by ESOC against the contents expected by the PI. These telemetry packets shall also be available to the PI via the DDS.

9. Changes affecting the functioning of the operation of experiments shall be implemented only with explicit approval of both, the ESA Mission Manager and the ESA Spacecraft Operations Manager (SOM). In addition, before the implementation of software changes, any effects related to the ESOC ground software shall be determined and, if required, modifications shall be initiated by the SOM.

10. Though the responsibility for experiment on-board software validation is with the respective principal investigator (PI), system-level operational validation of instrument software updates shall be supported by ESOC upon PI request provided a representative instrument model is mounted on the Engineering Test Bed (ETB), if available, to support the activity.

5.2.3.5 Mission Planning

Solar Orbiter payload operations will be governed by the rules and guidelines established and periodically discussed by the SWT. While in the Near-Earth Commissioning Phase and for special engineering activities like contingency recovery, anomaly troubleshooting and on-board software maintenance operations are executed following dedicated procedures and timelines defined in the FOP, for all other mission phases, the preparation, coordination and execution of instrument operations will be carried out via an automated cyclic mission planning and execution approach, as described below:

1. The Mission Planning approach for all the routine science operations phases shall be built on the experience of the precursor planetary missions Mars Express and Venus Express. The development approach is based on a common system to support BepiColombo, Rosetta (comet phases) and Solar Orbiter.

2. In a typical Mission Planning scenario the PI teams shall provide, at fixed deadlines and with a fixed periodicity, inputs to the SSOC for the requested science operations, the SSOC passes a consolidated request to the SMOC which checks the requests against mission, environmental and resource constraints.

3. The planning concept is traditionally based on an iterative process during which operations are iteratively refined and the required level of checking is performed. The planning concept shall allow to pre-plan instrument and spacecraft operations evolving from coarse to more detailed planning while being able to freeze spacecraft resources, like pointing, as early as possible, to give SMOC enough time to evaluate the requests at plan level and resolve conflicts if needed.

4. For the routine science operations phase the timeline of spacecraft attitude and the season (eclipses, occultations, Earth distance, etc.) will play a major role in establishing the constraints scenario against which the payload operations plan shall have to be checked. This means that the Mission Planning System shall utilize information coming from the Flight Dynamics System defining the evolution of the S/C orbit and attitude and the epoch.

5. The set of constraints applicable to the payload operations around the Sun indicates that a baseline science plan, which already takes into account the major constraints, shall have to be established long before submitting the final science operations requests to the Mission Planning for release to the front-end Mission Control System.

6. The mission planning scenario for the routine science operations phase shall be divided into different levels:
   a. *long term planning* shall deal with the establishment of the baseline science plans; in this cycle there will be an input from the SWT and SSOC to the SMOC to define the operations timeline based on the scientific objectives to be achieved; typically one long term plan will be defined for each major payload operations phase of the mission (e.g. one for cruise science and one for the full science phase), and the final iteration shall take place around 6 to 12 months in advance of the actual operations of each phase. Output of this cycle is a high level payload operations plan (priorities assigned and conflicts considered across instruments) and a finalized ground stations coverage for the entire phase
   b. *medium term planning* shall deal with the definition and refinement of an attitude strategy for the next planning cycle (typically of the duration of one month). The baseline plan shall be translated into an attitude timeline (by Flight Dynamics) and P/L command requests (by SSOC) thus allowing the SMOC to allocate resources and identify eventual conflicts on instruments operations; output of this cycle is a finalized attitude and resource consumption profile.
   c. *short term planning* shall work on shorter planning cycles. The SMOC, basing on the refined final command requests checked against rules in the Mission Planning System, will freeze resources and produce the sequences of commands to be up linked to the spacecraft, plus the related operation schedule inputs for the ground stations. Deadlines for submission of requests in this phase shall be in the order of one week before the event. The planning period shall also be in the order of a week. In this cycle, detailed pass instructions are also prepared for the on-console personnel.

7. A set of routine mission operational rules and constraints shall be identified by the Flight Control Team based on the spacecraft and payload user manuals as laid down by the manufacturer/instrument teams. These rules and constraints will drive the checks and modeling used during the planning process to validate a particular plan of operations.

5.2.4 DOWNLINK BITRATE

TBW

5.2.5 NAVIGATION

TBW

5.2.6 ATTITUDE POINTING ACCURACY AND STABILITY

TBW

5.2.7 INTERPLANETARY CRUISE TRAJECTORY AND ATTITUDE CONSTRAINTS

TBW

5.2.8 FULL SCIENCE RESONANT ORBIT AND ATTITUDE CONSTRAINTS

TBW

5.3 Mission Products

Mission products will be made available to the SGS and to the Solar Orbiter PIs in parallel, and will include all spacecraft and experiment raw telemetry data plus auxiliary data as defined in this section.

5.3.1 TELEMETRY PROCESSING AT SMOC

5.3.1.1 Generic

1. All telemetry packets received at the SMOC shall be stored as raw data and made available to all mission users. Upon delivery of raw data to external users, additional information such as quality data and packet timing are provided to enable the users to time correlate the data with UTC.

2. Decompression of data compressed by the instrument itself is not supported by ESOC. These packets shall be delivered as received by the on-board data handling system.

3. Non-science telemetry packets shall be further processed by ESOC in near real time for spacecraft control and monitoring purposes. In particular telemetry parameters shall be extracted from packets, and can be calibrated, displayed and checked against predefined

limits. A subset of telemetry packets shall be systematically processed for command verification, performance assessment, trouble shooting and on-board software maintenance as required.

4. ESOC shall not perform any processing of science telemetry packets beyond archiving, neither for calibration nor for instrument monitoring purposes. For this reason, it is essential that any information required at ESOC for health and safety monitoring is included in the instrument non-science telemetry.

5. Information to drive the processing of payload non-science data shall be provided in the Instrument user manual and database.

5.3.1.2 Auxiliary Data

1. Auxiliary data are non-telemetry data required to support mission planning and science data analysis. They shall be stored and made available to external users in the same way as telemetry data, and will be correlated with UTC. It is foreseen to typically include:
    a. Spacecraft ephemeris with respect to Sun, Earth and planet (swing-by).
    b. Spacecraft attitude prediction/reconstitution.
    c. Event files.
    d. Command history data.
    e. Time relation history (OBT/UTC).
    f. Mission planning information.

2. Auxiliary data shall be provided in a format and within coordinate systems to be jointly defined between ESA and the PIs through the relevant SWT.

5.3.2 DATA DISPOSITION SYSTEM

1. The SMOC system that provides access to the mission telemetry and auxiliary data described above shall be called Data Disposition System (DDS).

2. The DDS shall allow the authorized user to:
    a. Request a catalogues of available TM packets (per APID and time range of generation and/or reception on ground).
    b. Request a set of TM packets per APID and time range (only for those APIDs for which he is authorised to submit request).
    c. Request a specific file in the set of available auxiliary information files
    d. Specify off-line to ESOC that selected auxiliary information files are automatically transmitted to the user's institute when a new version becomes available in the archive.

3. The DDS is meant to be used as a temporary repository of fresh telemetry and auxiliary information, which must be requested as soon as possible by the relevant user and transferred to the user's private archive. In order to avoid unnecessary overload of the DDS, the users will be discouraged from using the DDS as a remote archive, thereby requesting repetitively the same data more than once. To this aim data transfer quotas per user shall be introduced and possibly a limit of availability of past data shall be defined (typically the last 2 weeks of data will be retrievable at any time). This interface shall be governed by the Data Disposition Interface Document (DDID).

5.3.2.1 Long Term Raw Data Archiving

1. Raw telemetry and auxiliary data shall be kept by SMOC throughout all post launch mission phases on the Long Term Archive (LTA). This archive shall be accessible remotely via the DDS throughout the mission, up to the end of the rundown phase (typically 3 to 6 months after end of the science mission). SMOC shall ensure completeness and integrity of the LTA during its active lifetime through back-up tapes.

2. There shall be no delivery of data on Raw Data Media during or after the mission.

3. Processed scientific and auxiliary data shall be archived by the SGS according to the Solar Orbiter Archive Plan.

5.3.2.2 Delivery Formats

1. Each data delivery request to the DDS shall result in a transfer of a block of data containing three main areas:
   a. An acknowledgment, including request details and status.
   b. A catalogue entry giving identification details of the requested data actually supplied (e.g. experiment, date, time).
   c. The requested data itself.

2. A simple packaging within Standard Formatted Data Units (SFDUs) shall be applied, following a recommendation of the Consultative Committee for Space Data Systems (CCSDS). Apart from providing a convenient mechanism for handling the variable length of requested data, this standard shall also provide administrative support for description of application data. Both the formatting of data delivered through the DDS and for data long term archiving.

5.3.2.3 Command Request Handling

In addition to the data access capability, the DDS allows for transfer of consolidated command requests to the SMOC as inputs to the mission planning system. The SMOC will support approval, authentication and authorisation of command requests. After validation the SMOC will incorporate the command requests into the mission planning system, which generates the final command schedule for uplink to the spacecraft. This interface will be governed by the Planning ICD (PLID).

5.4 Testing, Training and Simulation

5.4.1 GENERAL

The ground system test and validation activities begin around 2 years before launch. Activities will be mostly performed as part of the ESOC ground segment Satellite Interface Tests (SIT) and System Operations Validation (SOV) programme, and will include tests involving the payload as described in the following sections.

5.4.2 SATELLITE INTERFACE TESTS

The purpose of the Satellite Interface Tests (SITs) is to test and validate the external interfaces to the satellite and the basic TM and TC database definition. They are performed with the actual satellite linked to the SMOC via a communications network for TM, TC and voice connections. The SMOC mission control software will be validated as far as possible early in the programme, with the aid of a dedicated spacecraft software simulator, using telemetry tapes or equivalent, generated during satellite check-out tests. The PI shall support the satellite interface tests outlined below through preparation of related inputs, review of test plans and procedures, and if required, through actual participation in the tests itself.

5.4.3 MISSION SEQUENCE TEST

1. To verify the feasibility of selected mission scenarios a set of Mission Sequence Tests (MST) will be performed as part of the Satellite Interface Tests. Mission scenarios for the full science phase, in particular during the dedicated observation windows shall be prepared.

2. Each MST scenario covering representative spacecraft and payload operations for a typical mission slice, time-tag command sequences shall be defined for the payload and the spacecraft subsystems, loaded on-board and executed in conjunction with typical ground station passes activities. The MST should be performed as soon as possible in the overall project schedule, as part of the spacecraft/payload functional tests, and/or be conducted during integrated satellite tests in the thermal vacuum under the responsibility of the project. The MST shall consist of a number of tests of approximately one day each and accumulated overall duration of TBD days.

3. PI teams support for MST is TBD.

5.4.4 SYSTEM VALIDATION TESTS

1. The Project shall provide for on-line access to the Solar Orbiter Flight Model for closed loop testing (System Validation Test) with the ground segment and the flight control software. The SVTs will comprise:
    a. Spacecraft commanding from the SMOC
    b. Telemetry flow between satellite and SMOC. Real time non-science TM data processing in the SMOC in parallel to the TM processing in the check-out equipment.

2. A series of SVTs shall be performed with the satellite, starting at around L-18 months. Typically SVT0, SVT1 and SVT2 slots will be scheduled and executed in this period. SVT0 shall extend over a longer time period and mainly acquire satellite telemetry to verify databases and to perform some basic commanding; SVT1 emphasises “software” validation activities which include all mission control software facilities and databases. SVT2 is intended for re-validation of outstanding software facilities as well as for exercising and validating FOP sequences with the actual spacecraft.

3. The PI teams shall provide support to SVTs as follows:
    a. Provide test procedure inputs.
    b. Review/approve procedures defined by the SMOC.
    c. Provide real-time support at test site and/or in ESOC during SVT execution.
    d. Evaluate test results.
    e. Support anomaly investigation and resolution.

5.4.5 SYSTEM OPERATIONS VALIDATION (SOV)

1. The System Operations Validation (SOV) programme aims to execute a series of end-to-end operational scenarios to verify readiness of the ground segment as a whole to support the mission. As such, a number of standard and mission unique test are executed. It should be noted that some of the test involving the end-to-end science operations systems will be differed to the post launch phases. Details about the overall system testing activities shall be defined in the Ground Segment System Test Plan.

2. For all SOVs defined below, the PI teams shall provide support for procedure definition, procedure approval, results analysis and anomaly investigation/resolution.

5.4.6 DATA DISPOSITION SYSTEM (DDS) INTERFACE TESTS

1. At around L-10 months the DDS interface to the remote PIs and SGS shall be tested to demonstrate compatibility in terms of physical/logical connectivity and application interfaces (file request/transfer mechanism, command request capability). This test may be

performed applying an operational scenario with multiple users, and may include measurements of the turn around times.

2. Note that the DDS interfaces shall have to be tested both in remote configuration and with the payload support systems installed at the SMOC in the configuration required for critical operations.

5.4.7 SMOC/SGS END-TO-END TEST

1. The objective of the SMOC/SGS Interface test is to verify the interface functions and procedures required to generate a consolidated operation request schedule, ready for subsequent up-link to the spacecraft. Furthermore, all operational interfaces defined in the PLID and in the Science Operations Implementation Agreement (SOIA) shall be exercised.

2. This test shall be performed not later than 6 months prior to launch. Furthermore, the cruise after NECP phase shall be used as SGS commissioning.

5.4.8 TRAINING AND SIMULATION

1. Pre-launch operations support shall start approximately 6 months before the launch. During this period the SMOC at ESOC performs its final simulation programme including the validation of the Flight Operations Plan (FOP) and the mission control system. Principal Investigators with experiment specialist participation is required for the simulations related to the first experiment switch on and other critical operations (NECP simulations).

5.5 Instrument Documentation and Data Inputs

5.5.1 DOCUMENTATION

1. The experiment shall be operated and controlled in-flight according to the requirements defined in a set of documents. These will be mainly the documents which shall be used to prepare the ESOC Flight Operations Plan (FOP), which governs all flight operations. These documents are:
    a. Instrument On-board Software ICD
    This document is the formal ICD between the instrument software and the on-board software. It is an essential input to operations since it describes in detail the services provided by the on-board central software to the instrument, including operationally relevant aspects like data transfer and autonomy functions.

b. Instrument User Manual

This document shall contain:

- Detailed description of the instrument
- Operational constraints related to all instrument operations (including constraints on spacecraft activities affecting the instrument)
- Systematic description of all operations required to conduct its flight operations (including periodic maintenance activities, operational modes and transitions).
- Operational procedures required to perform all nominal and contingency activities
- Requirements for health and safety telemetry monitoring at the SMOC and the relevant recovery actions where applicable
- Requirements for trend analysis and overall performance monitoring where applicable.

2. The PI teams shall review/approve the FOP for the aspect/sections relevant to Instrument operations.

5.5.2 INSTRUMENT DATABASE

1. A single, Project-wide spacecraft TM/TC database shall be specified, using the structure and detailed definition of the SCOS-2000 MIB (Mission Information Base) compliant to SCOS-2000 Database ICD, EGOS-MCS-S2K-ICD-0001. This shall ensure compatibility of the spacecraft database required by the multi-mission control system which is part of the ESOC infrastructure.

2. The ESOC operations team shall be formally part of the review and approval process for all change requests produced on the MIB during the pre-launch population and maintenance phase.

3. ESOC shall contribute to the population work pre-launch with direct inputs in areas agreed with the Project, such as payload TM/TC, displays, etc.

4. Responsibility for database maintenance shall be transferred to ESOC at the Flight Acceptance Review.

5. An Instrument Data Base (IDB) shall be established, maintained and delivered by the Instrument PI to the Project to become part of the MIB. The IDB shall contain a complete definition of telemetry and telecommand data required for the detailed design of the flight control software, for the design of the software simulator and for setting up the operational telemetry and telecommand data files. The IDB shall comply to the Database Definition Document (DBDD), and shall be delivered according to TBD schedule.

6. Delivery format is TBD.

6 VERIFICATION REQUIREMENTS

6.1 General

6.1.1 INTRODUCTION

The objective of the instrument Verification Programme is to demonstrate to ESA and its selected Prime that the instrument design is fully compliant

- with the instrument scientific goals;
- with the mission environment;
- with the spacecraft performance;
- with the spacecraft interface requirements;
- with the operational requirements;
- with the provided operational documentation;

hence capable to contribute to the overall scientific goals.

This section establishes the verification requirements for qualification and flight certification of the instrument units giving specific test levels, durations and describing acceptance and qualification tests and analytical methods for implementing the requirements.

6.1.2 RESPONSIBILITIES

1. The PI shall, in a systematic manner, verify the instrument design and build against each requirement specified in the EID-A and B [Qualification].

2. The PI shall, in a systematic manner, verify the FM instrument certification for flight against each requirement specified in the EID-A and B [Acceptance].

3. The PI shall include in Instrument Development Plan (IDP) the tests and analyses that collectively demonstrate that hardware and software complies with the requirements.

Note: *The verification shall follow the classical methods approach for the review of design either by testing, or analysis, or similarity (through an already verified design).*

6.1.3 DEFINITIONS

The following definitions (in alphabetical order) are recommended:

- **Acceptance Verification / Certification**:

Tests intended to demonstrate that hardware is acceptable for flight. It also serves as a quality control screen to detect deficiencies, and normally to provide the basis for delivery of an item under terms of a contract or agreement.

- **Acoustics/Random Vibration**:

An environment induced by high-intensity acoustic noise associated with various segments of the flight profile: it manifests itself throughout the instrument in the form of directly transmitted acoustic excitation and as structure-borne random vibration excitation.

- **Design Qualification Verification**:

Tests and analyses intended to demonstrate that the item will function within performance specifications under simulated conditions more severe than those expected from ground handling, launch and orbital operations. The purpose is to uncover deficiencies in design and method of manufacture and is not intended to exceed design safety margins or to introduce unrealistic modes of failure.

- **Electromagnetic Compatibility (EMC)**:

The prevailing condition when various electronic devices are performing their functions according to design in a common electromagnetic environment.

- **Electromagnetic Interference (EMI)**:

Electromagnetic energy which interrupts, obstructs, or otherwise degrades or limits the effective performance of electrical equipment.

- **Electromagnetic Susceptibility**:

Undesired response by a component, instrument or system to conducted or radiated electromagnetic emissions.

- **Environmental Tests**

Environmental tests shall be conducted on the flight or flight configured hardware to assure that the flight hardware will perform satisfactorily in one or more of its flight environments. To this class of test belong: Acoustic, Thermal Vacuum and EMC tests.
They are normally combined with functional testing providing it is compatible with test objectives.

- **Functional Tests**:

Testing of the operation of a unit in accordance with defined operational procedures to determine that the (mechanical or electrical or similar) performance is within the specified requirements. According to the needs of verification it can have several degrees complication and depth. The Full Functional Test (FFT) and the Abbreviated Functional Test (AFT) are subsets of this type.

- **Incoming / Receiving Inspection:**

Inspection and / or functional tests to declare that the item is ready for integration on the spacecraft.

- **Modal Survey Test**:

A series of mechanical investigations to determine the natural frequencies and associated modes of a structure.

• **Performance Verification**:
Determination by test, analysis, or a combination of the two that the complete instrument or instrument unit can operate as intended in a particular mission: this includes proof that the design of the complete instrument or instrument unit has been qualified and that the particular item has been accepted as compliant to the design and ready for flight operations.

• **Protoflight Verification:**
The protoflight concept replaces the classical approach of design qualification and flight acceptance on dedicated models by a combination of qualification and acceptance on the flight hardware. A protoflight item is designated in advance to serve both as qualification and flight model and has to be designed for such purpose.

The protoflight model is subject to tests at qualification levels but with flight acceptance duration.

• **Thermal Balance Test**:
A test conducted to verify the adequacy of the Thermal Model, the adequacy of the thermal design, and the capability of the thermal control system to maintain thermal conditions within established mission limits.

• **Thermal Cycling Test**
A test to demonstrate the ability of the instrument to fulfill all functional and performance requirements over the qualification temperature range.

• **Thermal Vacuum Test**:
A test to demonstrate the validity of the design to meet its functional and performance requirements under vacuum and in a thermal environment equivalent to the worst conditions expected for the mission. The test can also uncover latent defects in design, parts and workmanship.

• **Shock Tests:**
A test conducted to verify the design under the environment induced by shocks produced by the launcher during events such as stage and satellite separation and by the S/C during events such as pyro firings.

• **Sinus Vibration Test**:
A test to demonstrate that the instrument can withstand the mechanical environment of the low frequency (less than 100 Hz) sinusoidal and transient vibrations. This test can also be used to demonstrate compatibility with the static loads.

• **Static Loads**:
The maximum combination (longitudinal and lateral) of static loads which acts on an instrument during the various segments of the flight profile. It consists of steady state accelerations (e.g. due to engine constant thrust or lateral wind loads) and quasi-static loads which are structure borne loads generated by the launch vehicle in the low frequency (less than 100 Hz) range (e.g. engine cut-off loads or wind gusts).

• **Verification by Analysis:**
The compliance to a requirement is verified analytically. The typical method used is using mathematical models. They may be supplemented or supported by hardware simulation.

6.1.4 DOCUMENTATION

6.1.4.1 Instrument Development Plan

1. The PI shall prepare a Verification Plan (called Instrument Development Plan) defining the tests and analysis that collectively demonstrates that hardware and software complies with the mission, design, scientific requirements laid out in this EID-A.

Note: *The Instrument Development Plan shall highlight the overall approach which will be undertaken by the instrument consortium to accomplish the instrument qualification and acceptance. When appropriate the interaction of the tests and analysis shall be described.*

2. The Instrument Development Plan shall be complemented by analysis reports, test procedures and upon test completion by test reports.

6.1.4.2 Verification Control Matrix

1. The PI shall provide a verification matrix that summarizes all the tests that will be performed on each instrument unit and on instrument system level.

The purpose of the matrix is to provide, in a synthetic manner, a reference to the test programme in order to prevent the deletion of a portion of the test programme without an alternative of accomplishing the verification objectives. It further ensures that all flight hardware has seen environmental exposures that are sufficient to demonstrate acceptable workmanship.

2. The matrix shall provide traceability of the qualification heritage of the instrument units hard- and software.

3. The matrix shall provide traceability of the verification of the design and test requirements contained in the EID-A.

4. All flight hardware, spares and prototypes (EM/QM) shall be included.

5. The matrix shall be included as annex to the Instrument Development Plan and provided to ESA and its selected Prime at the major reviews (and updated as changes occur).

6.1.4.3 Analysis Reports

1. For each analysis verification activity the PI shall submit a formal report, describing the mathematical model and the relevant outputs and interpretations.

6.1.4.4 Test Related Documentation

• **Test Specification:**

1. For each test defined in the Instrument Development plan (e.g. EMC, vibration, electrical, thermal, etc.), the PI shall provide a test specification describing the relevant configuration, test setup, facility, test goals, success criteria etc.

• **Test Procedures:**

2. For each test defined in the Instrument Development Plan (e.g. EMC, vibration, electrical, thermal, etc.), the PI shall provide a detailed step-by-step procedure.

• **Test Report:**

3. For each test defined in the Instrument Development Plan (e.g. EMC, vibration, electrical, thermal, etc.), the PI shall provide a test report containing the objectives, a description of test setup, a result summary result summary and the as-run procedure.

The test related documentation will be subject of review during the project lifetime by ESA and its selected Prime.

6.2 Verification Concept

The instrument units belong in general to the category of newly designed equipment with performances to be fully demonstrated by qualification and acceptance programmes.
These programmes will be reviewed by ESA and its selected Prime upon compatibility with the overall system verification concept.

The verification objectives are primarily:

- to qualify the design;
- to ensure that the product is in agreement with the qualified design, is free from workmanship defects and acceptable for use in its spacecraft environment;
- to verify that the space system (including tools, procedures and resources) will be able to fulfill mission requirements;
- to confirm product integrity and performance after particular steps of the project life cycle (e.g. pre-launch, in-orbit, post-landing).

1. Verification shall be accomplished preferably by testing, but in certain cases when testing is not possible, one or more of the following **verification methods shall be applied as described below**:

Assessments (see section 6.3):
i.e. Analysis (Structural, Thermal), when verification is achieved by performing theoretical or empirical evaluation by accepted techniques.

Tests (see section 6.4):
i.e. Functional Tests (FFT, AFT) or Environmental Tests (Vibration, TB/TV, EMC), when requirements have to be verified by measuring product performance and function under various simulated environments.

Inspection (see section 6.5):
Verification is achieved by visual determination of physical characteristics (such as construction features, hardware conformance to document drawing or workmanship requirements).

Review-of-design (Similarity Assessment):
Verification is achieved by validation of records or by evidence of validated design documents or when approved design reports, technical descriptions, engineering drawings unambiguously show the requirement is met.

6.3 Analysis

A number of analyses are required by the Project to ensure the design integrity of the instrument. All analyses are deliverable and are listed below. This list may be only partial since specific requests for analysis may occur following need during the course of the project development.

6.3.1 STRUCTURAL MATHEMATICAL ANALYSIS

6.3.1.1 General

1. The mechanical performances of the instrument shall be calculated by means of Structural Mathematical Models (SMMs).

2. The PI shall use models for his own design and shall also provide model(s) to the Agency for use during spacecraft design and test results predictions. The PI shall update the models according to instrument and system test results.

3. The instruments SMMs shall be delivered according to the dates TBD.

The detailed requirements for each model / analysis are listed in the following sections.

6.3.1.2 Detailed Stress Analysis

1. The PI shall perform and deliver a detailed Stress Analysis. This shall include at least:

- A description of the configuration analyzed with reference to interface controlled drawings
- A description of the mathematical finite element model and/or of the assumptions taken to verify the structure
- A description of all possible loading cases and an identification of the design driving load cases or load combinations
- Detailed description of the most loaded elements listed with relevant stresses, and the loading cases that generated them
- A list of the materials and structural components with characteristics data sheets (including long-life effects under space environment)
- A set of tables showing, for each structural element, the maximum value on each type of stress or combination of them with the allowable value, and the load case that determines it, together with its margin of safety.

6.3.1.3 Mechanism Functional Analysis

1. Each mechanism shall be analyzed functionally and the following information shall be at least supplied:

   - A detailed description of the mechanisms, with particular reference to its discrete components (bearings, actuators, switches) and to its operational/safety features
   - A detailed description of the operating modes with reference to ground and orbital activations
   - A definition of operating loads in various configurations with a clear definition of analysis assumptions. In particular, the functional analysis shall include the effects of the worst environmental conditions that could produce distortions or changes in clearance between movable parts (e.g. thermal gradient through bearings)
   - A Failure Modes, Effects and Criticality Analysis (FMECA) defining the failure modes and the functional margins of safety against each of them
   - A performance description of the control system that the mechanisms form a part of.

6.3.1.4 Dynamic Model

1. The structural mathematical model of the instrument shall be detailed enough to predict the dynamic loads to size the structure elements, and the interface loads in particular, with sufficient accuracy.

2. This means that it shall be able to reproduce the low frequency modes with an upper limit to the frequency range to be defined on a case-by-case basis.

3. The model shall fulfill the requirements of the Design Verification Requirements, when compared to test results.

4. A finite element model shall be accompanied by a clear description of the model itself and of the assumption made in the model, particularly concerning the boundary conditions at the spacecraft interfaces (i.e. hard mounted I/F). For mechanisms, two or more models (stowed, deployed; general position), may be required.

5. All mathematical models shall be maintained in current configuration.

6. The mathematical models to be delivered to ESA shall be compiled in accordance with the Solar Orbiter Prime Requirement Specification for Structural FEM Models (TBW - during Implementation Phase).

6.3.1.5 Dynamic Analysis

1. The PI shall perform a structural dynamic analysis and include at least:

   • A description of the configuration analyzed with reference to interface controlled drawings

   • A description of the mathematical finite element model and/or of the assumptions/reductions introduced in the analysis

   • A description of the checks performed on the model to verify its quality (e.g. rigid body modes, residual forces)

   • A list of eigen-frequencies with relevant mode type and associated modal effective

   • Plots and listings of eigen-vectors.

2. The PI shall perform, where necessary (large exposed areas, e.g. mask):
   - frequency analysis and response
   - acoustic response analysis.

6.3.2 THERMAL ANALYSIS

6.3.2.1 General

1. A thermal analysis of a payload unit shall be performed by the unit responsible with the following objectives:

   - Verify that internal parts and materials are below their maximum allowed temperatures under acceptance/qualification testing;
   - Verify the ability of the thermal design to maintain the internal required temperatures and intended heat flow pattern that ensure performance requirements under the worst flight cases;
   - Verify the compliance with the spacecraft interface requirements under the worst flight cases.

6.3.2.2 Units

The S.I. units are mandatory for all documentation that has to be exchanged with ESA. Temperatures can be presented either in Kelvin or degree Celsius.

6.3.2.3 Thermal Design Cases

A number of thermal conditions given in the table below can be taken into consideration during the analysis campaign. This list is not exhaustive especially for the unit internal design that might require more cases to prove the feasibility of the thermal design.

Case	Type	Properties	Dissipations	Environmental Heat Fluxes
Flight Hot Op.	steady-state	EOL	max (EOL)	max (0.22 AU)
Flight Cold Op.	steady-state	BOL	min (BOL)	min (1.5 AU)
Flight Cold Non-Op.	steady-state	BOL	min *	min (1.5 AU)
Acceptance Test	steady-state	BOL	min/max (BOL)	T_{URP} and T_R at acceptance level
Qualification Test	steady-state	BOL	min/max (BOL)	T_{URP} and T_R at qualification level
Transient Solar Illumination	transient	EOL	max (EOL)	max (0.22 AU) + 15° half-cone angle depointing over 20 s

Table 22 Thermal Design Cases

* during non-operating phases, a heating power inside the unit might be necessary

6.3.2.4 Thermal Mathematical Models

1. Unit thermal analyses shall be performed by the unit responsible using a Detailed Thermal Mathematical Model (DTMM) and a Detailed Geometrical Mathematical Model (DGMM).

2. A unit Interface Thermal Mathematical Model (ITMM) and Interface Geometrical Model (IGMM) for coupled thermal analysis with the spacecraft shall be derived from the DTMM and the DGMM, respectively.

3. Requirements to insure compatibility of the interface models with the spacecraft will be defined during the Definition Phase (TBC).

6.3.2.5 Software Codes

For unit detailed thermal analysis, the following codes are recommended:

- Thermal network solver: ESATAN v 9.4 or higher;
- Radiation coupling computation: ESARAD 5.6.1 or higher.

For ITMM exchange, the following codes are required:

- Thermal network solver: ESATAN;
- Radiation coupling computation: ESARAD.

6.3.2.6 Deliverable Models

1. The ITMM shall be regularly updated and delivered according to the unit design maturity. As a minimum, the following model updates shall be delivered for each unit:
   - Preliminary model of the flight unit;
   - Updated models of the flight unit after any major design modification;
   - Updated models of the flight unit after the thermal verification tests;
   - Model of the STM unit if this is required by the S/C verification programme;
   - Final model of the flight unit with measured dissipations.

6.3.2.7 Thermal Analysis Uncertainties

1. The temperature of the unit internal parts shall be predicted by adding the thermal analysis uncertainty to the computed temperatures. In the hot cases, the absolute value of the uncertainty will be added while, in the cold cases, the absolute value of the uncertainty will be subtracted. The uncertainty shall be assessed with a 99% confidence level according to what indicated in ECSS-E-30 Part 1A section A.1 [NR3].

6.3.2.8 Thermal Control Design Documentation

1. The PI shall deliver to ESA and its selected Prime the **Unit Thermal Design Description Report** in the format described in TBD - to be provided later.

2. The PI shall deliver to ESA and its selected Prime the **Unit Thermal Analysis Report** in the format described in TBD - to be provided later.

3. The PI shall deliver to ESA and its selected Prime the **Unit Thermal Tests Reports** in the format described in TBD - to be provided later.

4. The PI shall deliver to ESA and its selected Prime the **Unit Thermal Model Correlation Report** (following thermal tests) in the format described in TBD - to be provided later.

5. The PI shall deliver to ESA and its selected Prime the **Unit ITMM/IGMM Description Report** in the format described in the TBD - to be provided later.

6.4 Testing

6.4.1 GENERAL

6.4.1.1 Test Sequences

The verification activities can be divided in
- Qualification Programme
- Acceptance Programme
- Recertification
- Incoming Inspection

No specific environmental test sequence is required, but the test programme should be arranged in a way to best disclose problems and failures associated with the characteristics of the hardware and the mission objectives.

It is strongly recommended that the vibration/acoustic test precede the thermal vacuum test unless there is an overriding reason to reverse that sequence.

Qualification Programme:

1. The qualification programme shall demonstrate that the item will function within performance specifications under simulated conditions more severe than those expected from ground handling, launch and orbital operations.

As a guideline for the PI the following sequence of tests is highly recommended:

a. Visual Inspection	6.5.1
b. Dimensions Verification	6.5.2
c. Physical Properties	6.5.2
d. Functional Test	6.4.2
e. Low Level Sine	6.4.4.2
f. Strength Load	
g. Shock	6.4.4.5
h. Sine Vibration	6.4.4.2
i. Low Level Sine	6.4.4.2
j. Random Vibration	6.4.4.3
k. Low Level Sine	6.4.4.2
l. Functional Test	6.4.2.
m. Acoustic Noise (when applicable)	6.4.4.4
n. Functional Test	6.4.2.
o. Thermal Vacuum	6.4.6
p. Functional Test	6.4.2.
q. Grounding / Bonding / Isolation	6.4.3.3
r. EMC Conducted Emission / Susceptibility	6.4.3.4 / 5
s. EMC Radiated Emission / Susceptibility	6.4.3.6 / 7
t. DC Magnetic Properties	
u. Purging Rate Verification	
v. Visual Inspection	6.5.1

2. Limited Lifetime demonstration of elements concerned shall be incorporated in the qualification test programme or performed separately.

Acceptance Programme:

3. The acceptance shall demonstrate that the hardware is acceptable for flight and shall serve as a quality control screen to detect deficiencies.

As a guideline for the PI the following sequence of tests is highly recommended:

a. Visual Inspection	6.5.1
b. Dimensions Verification	6.5.2
c. Physical Properties	6.5.2
d. Functional Test	6.4.2
e. Sine Vibration	6.4.4.2
f. Low Level Sine	6.4.4.3
g. Random Vibration	6.4.4.3
h. Low Level Sine	6.4.4.2
i. Functional Test	6.4.2
j. Thermal Vacuum	6.4.6

k. Functional Test	6.4.2
l. Grounding / Bonding / Isolation	6.4.3.3
m. EMC Conducted Emission / Susceptibility	6.4.3.4 / 5
n. DC Magnetic Properties	
n. Visual Inspection	6.5.1

Recertification:

4. The recertification shall certify that modified / repaired units are acceptable for flight. It is applicable for any unit which has been disassembled from the S/C after the system environmental testing and refurbished / repaired and then supposed to be re-integrated.

Note: *The recertification is a limited acceptance certification and serves also as a quality control.*

As a guideline for the PI the following sequence of tests is highly recommended:

a. Visual Inspection	6.5.1
b. Dimensions Verification	6.5.2
c. Physical Properties	6.5.2
d. Functional Test	6.4.2
e. Low Level Sine	6.4.4.2
f. Random Vibration - 1 axis	6.4.4.3
g. Low Level Sine	6.4.4.2
h. Functional Test	6.4.2
i. Thermal Vacuum (2 cycles)	6.4.6
j. Functional Test	6.4.2
k. Grounding / Bonding / Isolation	6.4.3.3
l. EMC Conducted Emission / Susceptibility	6.4.3.4 / 5
m. DC Magnetic Properties	
n. Visual Inspection	6.5.1

Depending on the kind of refurbishment the programme of Recertification can be reduced in agreement with ESA and its selected Prime.

Incoming Inspections:

5. The incoming inspection at the Prime Contractor site shall verify that the Instrument is ready for integration into the S/C.

As a guideline the following sequence of tests will be performed:

a. Visual Inspection	6.5.1
b. Dimensions Verification	6.5.2
c. Physical Properties	6.5.2
d. DC Magnetic Properties (TBC)	

e. Functional Test	6.4.2
f. Grounding / Bonding / Isolation	6.4.3.3
g. EMC Conducted Emission / Susceptibility	6.4.3.4 / 5
h. Review of completeness of documentation	

6.4.1.2 Test Level Tolerances

The test tolerances, unless otherwise specified are:

Temperature:
• -55°C to +150°C Tmax: 0 to +3°C,
Tmin: 0 to -3°C

Environmental heat fluxes:
• solar fluxes: +/- 3%
• infrared fluxes: +/- 3%

Pressure:
• Equal or above 0.1 mbar 10%
• Below 0.1 mbar 50%

Relative humidity: ± 5%

Sinusoidal vibration:
• Acceleration, amplitude ± 10%
• Frequency above 50 Hz ± 2%

Random vibration:
• Power spectrum density (50 Hz or narrower)
20 to 500 Hz ± 1.5 dB
500 to 2000 Hz ± 3.0 dB
• Overall g rms ± 1.5 dB

Static force: ± 5.0%

Acoustic: ± 1 dB

Electromagnetic Compatibility
• Voltage Amplitude: ± 5% of the peak value
• Current Amplitude: ± 5% of the peak value
• RF Amplitudes: ± 2 dB
• Frequency: ± 2%
• Distance: ± 5% of specified distance or ± 5 cm, whichever is greater

Magnetic Properties

• Mapping distance measurement:	± 1 cm
• Displacement of assembly Centre of Gravity (CoG) from rotation axis:	± 5 cm
• Vertical displacement of single probe centre line from CoG assembly:	± 5 cm
• Mapping turntable angular displacement:	± 3 degrees
• Magnetic field strength:	± 1 nT
• Repeatability of magnetic measurements (short term):	± 5% of ± 2 nT, whichever is greater
• De-magnetizing and magnetizing field level:	± 5% of nominal

Mass Properties

• Weight:	± 1 %
• Centre of Gravity:	± 5 mm
• Moments of Inertia:	± 10%

6.4.2 FUNCTIONAL TEST REQUIREMENTS AT INSTRUMENT LEVEL

6.4.2.1 Full Performance Test

The Full Functional (Performance) Test (FFT) shall be a detailed demonstration that the hardware and software meet their performance requirements within allowed tolerances.

1. The FFT shall demonstrate operations of all nominal and redundant circuitry.

2. The FFT shall demonstrate satisfactory performance in all operational modes. The test shall also demonstrate that, when provided with appropriate stimuli, performance is satisfactory and outputs are within allowed limits.

3. The initial FFT shall serve as a baseline against which the results of all later FFTs can be readily compared.

4. The FFT shall be exercised in ambient as well as in hot / cold environmental conditions.

6.4.2.2 Abbreviated Functional (Performance Tests)

The Abbreviated Functional (Performance) Tests (AFT) is normally a subset of the FFT, however it also tests both the nominal and redundant branches.

Note: *The Abbreviated Performance or Functional Test can also be called Limited*

Performance of Functional Test.

1. The AFT shall be performed before, during and after environmental tests, as appropriate, in order to demonstrate that functional capability has not been degraded by the environmental tests.

2. Specific items on which it is intended that AFTs will be performed shall be listed in the Verification Plan.

3. The AFT shall demonstrate that the performance of selected hardware and software functions is within allowed limits.

The limited tests may also be used in cases where comprehensive performance testing is unwarranted or impracticable.

6.4.3 FUNCTIONAL TEST REQUIREMENTS AT SYSTEM LEVEL

The requirements for tests to be defined when the instruments have been integrated into the spacecraft are TBD.

6.4.4 EMC TEST REQUIREMENTS

6.4.4.1 General Set-Up Requirements

1. The tests shall be performed in an ambient electromagnetic environment which is at least 6 dB below the performance levels required in section 3.6. Included in the ambient level are also emissions from test equipment, including unit-testers (EGSE) with its harness.

2. Measuring antenna ends shall not be closer than 1.0 metre from any electrically conductive elements during the test.

3. The tests shall be performed with test samples, unit-testers (EGSE) and harness placed on a conductive ground plane with a length greater than 2.5 metres and a width of more than 1 metre.

4. If a shielded room is used the ground plane shall be bonded to the room with low inductive bonds separated by less than 0.5 metre.

5. This connection shall be verified by a resistance test.

This connection of the ground plane is very important when the EGSE has to be located outside the shielded room because of emission or susceptibility excess.

6. In the cases where real electrical/electronic loads cannot be used these shall be simulated by dummy loads with similar characteristics.

It is forbidden to take the interface wires to ground if not done in the actual installation.

7. The power sources used for the tests shall have a well defined impedance below 10 MHZ.

8. The test harness shall be flight representative.

No shielding between the test set-up and measurement antennae is allowed.

9. Grounding of interfaces shall be in accordance with flight installation.

10. Bonding of units - unit tester etc. to the ground plane shall be verified by a bonding test.

11. The unit bond shall be similar to that specified for the actual installation except for conducted common mode emission/susceptibility tests when a ground strap between the grounding lug and the ground plane shall be used.

12. Radiated susceptibility tests shall be performed such that regulations and laws at the test location are met.

13. Reflection effects shall be minimized by means of absorber materials.

14. All equipment used for emission and susceptibility tests shall be calibrated and wear calibration certificates.

15. Passive equipment, such as antennae, current probes etc. shall have calibration curves from the manufacturer.

16. In order to reproduce the power bus impedance seen by the users and to standardize the measurement conditions used in different test sites, a Line Impedance Stabilization Network (LISN) shall be inserted between the EGSE power supply and the unit under test when performing emissions and susceptibility measurements on primary power lines. The LISN schematic and the relevant impedance versus frequency are chosen in accordance with the bus impedance mask and harness.

17. The LISN schematic and the relevant impedance versus frequency given in Figures 37 and 38 shall be used.

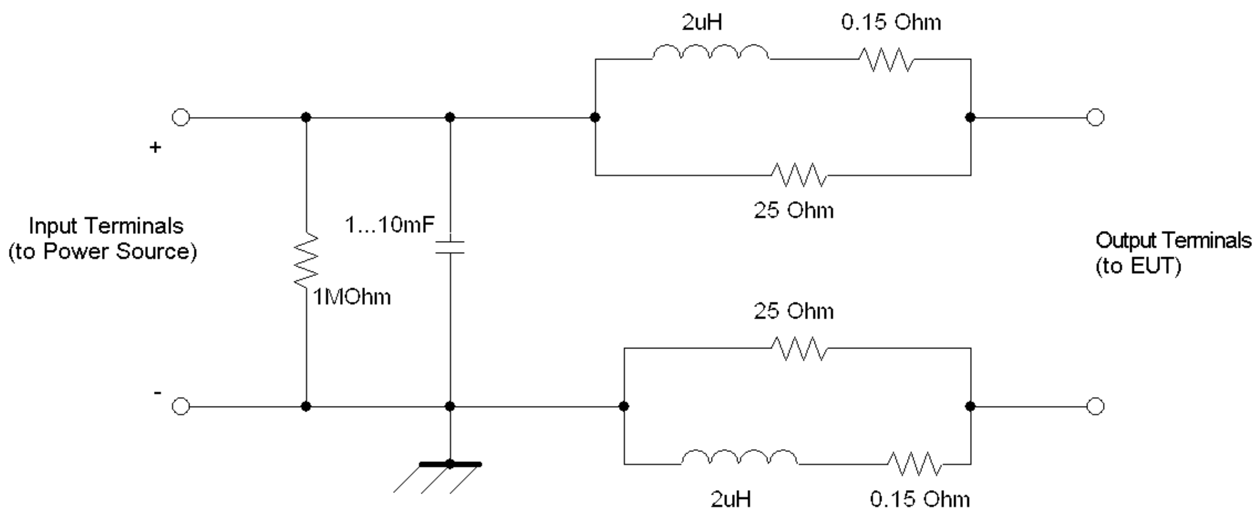


Figure 37 LISN Schematic

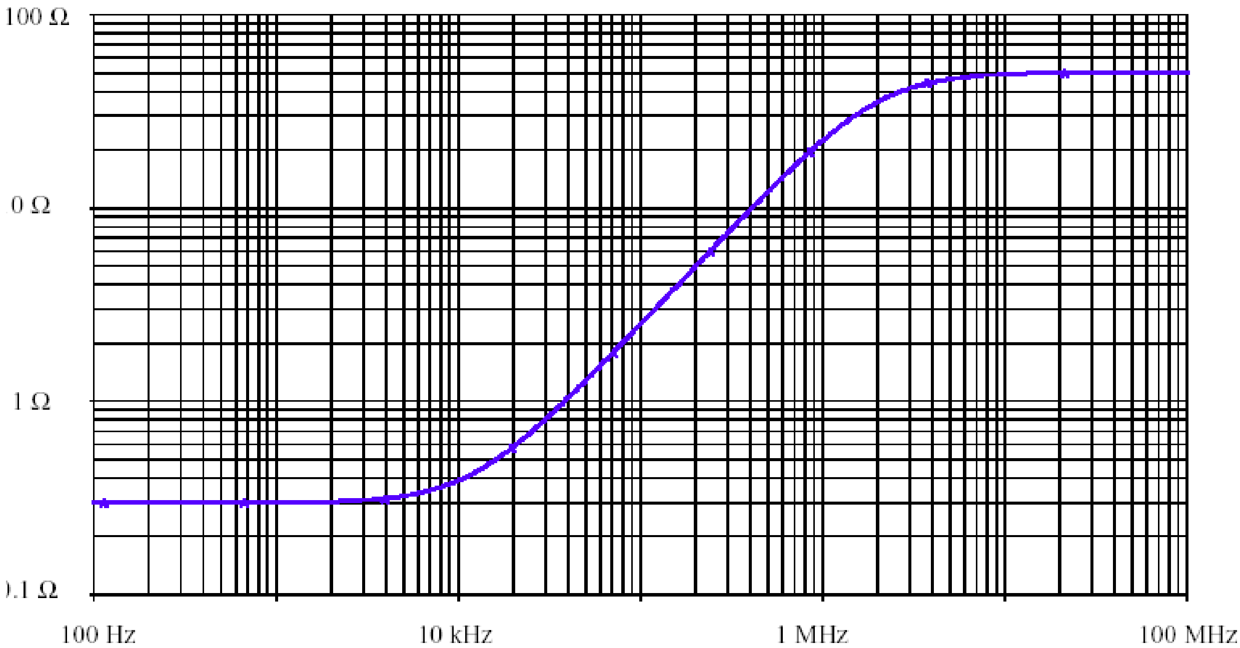


Figure 38 Output Impedance of the LISN with Shorted Input Terminals

6.4.4.2 EMC Test Categories

Development Test

These tests should be performed at an early stage of the programme to evaluate the design approach, indicate critical areas where design improvement is required and assure design compliance with EMC requirement and support analytical methods or generate essential design data.

Qualification Test

1. For the qualification the instrument EQM shall be subjected to a full EMC test sequence outlines below:
    - Bonding
    - Isolation
    - Grounding and conductivity test of space exposed surfaces
    - Conducted emission
    - Conducted susceptibility
    - Radiated emission
    - Radiated susceptibility
    - Electrostatic discharge susceptibility
    - DC magnetic field characterization

Note: for DC magnetic field characterization a special facility is required (for example IABG, Braunschweig) the effort shall be considered and discussed with ESA and its selected prime.

Acceptance Test

2. This test shall be accomplished on all FM hardware. Acceptance level testing shall comprise of the verification of
    - Bonding
    - Isolation
    - Grounding and conductivity test of space exposed surfaces
    - Conducted emission
    - Conducted susceptibility
    - Electrostatic discharge susceptibility

Note: *Further details will be elaborated by the ESA selected Prime in the course of Phase B- C/D. The EMC design and test specification shall be consulted.*

6.4.4.3 Bonding, Isolation and Grounding/Conductivity Tests

1. These tests shall be carried out to demonstrate compliance with the required Instrument performance as defined in section 3.6.

6.4.4.4 Conducted Emission Test

The suggested test set-up is as shown in Figure 39. The tests are applicable at each signal and power input/output. Any switch for ON/OFF test will be positioned between the LISN and the unit under test. The transients are then measured on the power lines between the switch and the unit under test.

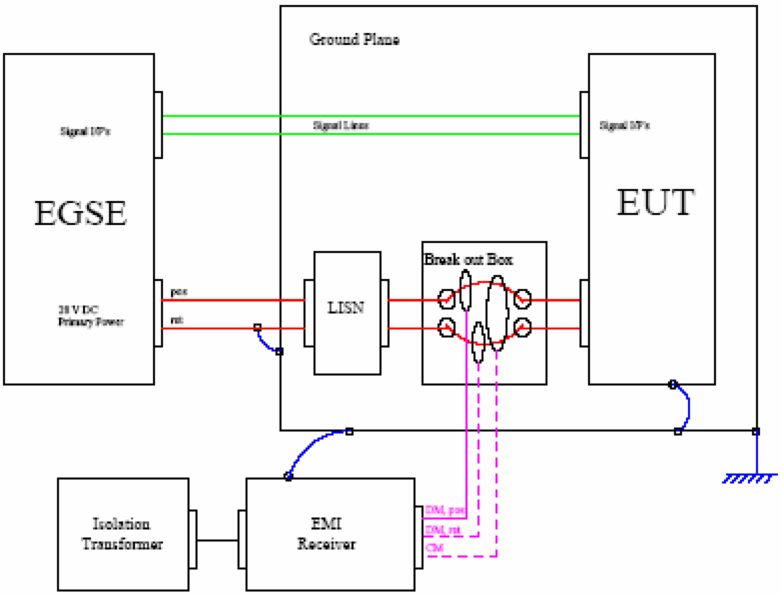


Figure 39 Conducted Emission – Test Set-up

6.4.4.5 Conducted Susceptibility Test

The test set-up for power lines is shown in Figure 37.

1. The injected voltage relevant to the susceptibility threshold shall be monitored and recorded.

2. The injected current shall be limited to 1 Ampere peak on the input power lines.

3. The test set-up for signal lines, differential mode, shall be similar to the test set-up for power. The signal lines shall be loaded with electrical simulators of the interfacing circuits.

The test set-up for signal lines, common mode, is given in 40 and Figure 41.

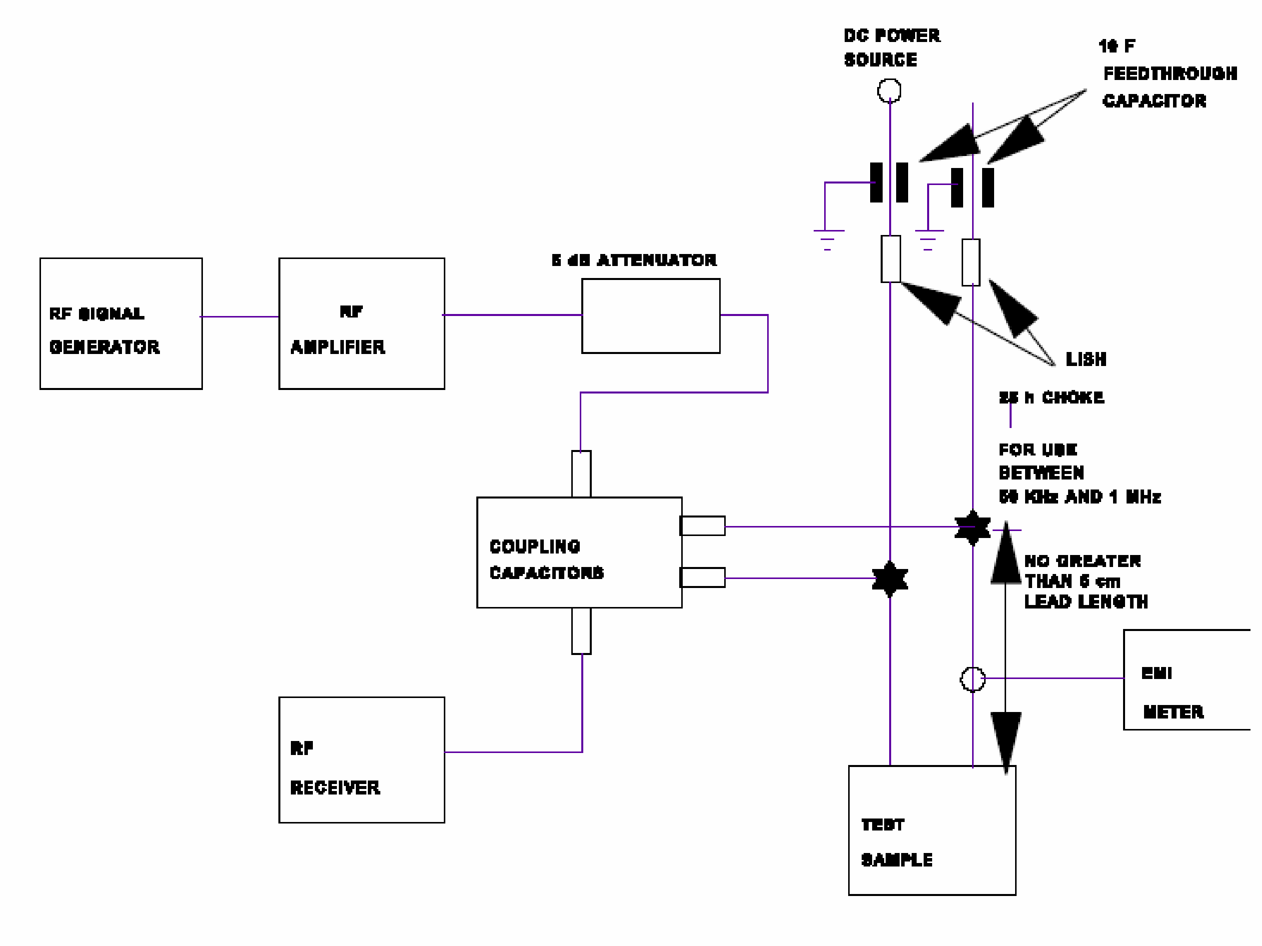
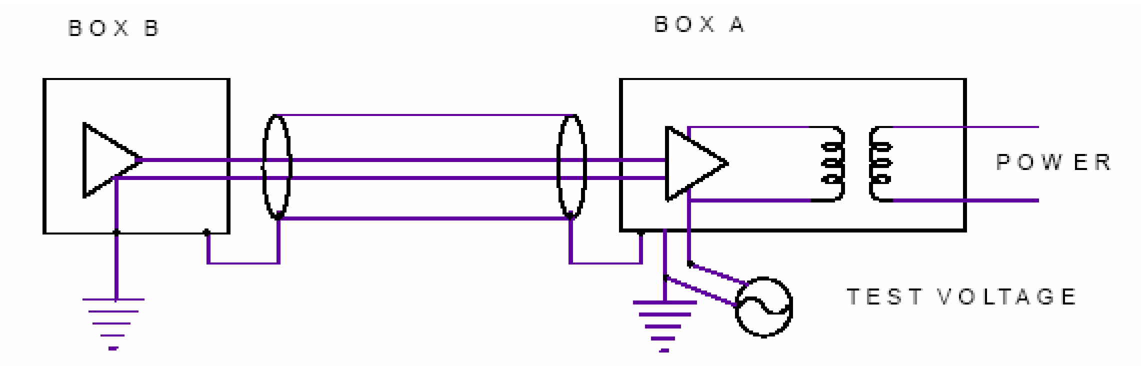
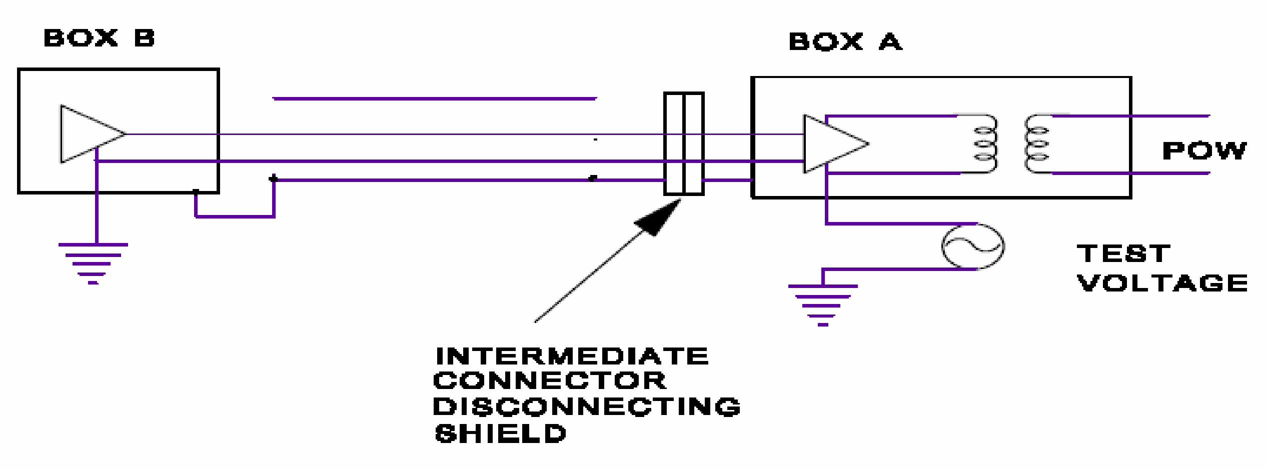


Figure 40 Conducted Susceptibility – Power Set-up



a) Test set-up for externally accessible ground wire



b) Test set-up when ground wire is not accessible. Box fitted from structure and shield disconnected

Figure 41 Common Mode Rejection Test Set-up

6.4.4.6 Radiated Emission

The suggested test set-up is as shown in Figure 42. The emission at the antenna at 1 metre distance from the test object which gives the highest reading shall be the Radiated Electric Field Emission (REE).

1. Above 25 MHZ, the requirement shall be met for both horizontally and vertically polarized waves.

Radiated E-Field Frequency Range for Emission Test (TBC)

2. The upper frequency range of the measurement shall be in accordance with the following values:
   - For an Highest Operating Frequency of Equipment < 1 GHz the required upper limit is "To tenth harmonic or 1 GHz whichever less";
   - For an Highest Operating Frequency of Equipment 1-10 GHz the required upper limit is "To fifth harmonic or 10 GHz whichever less";

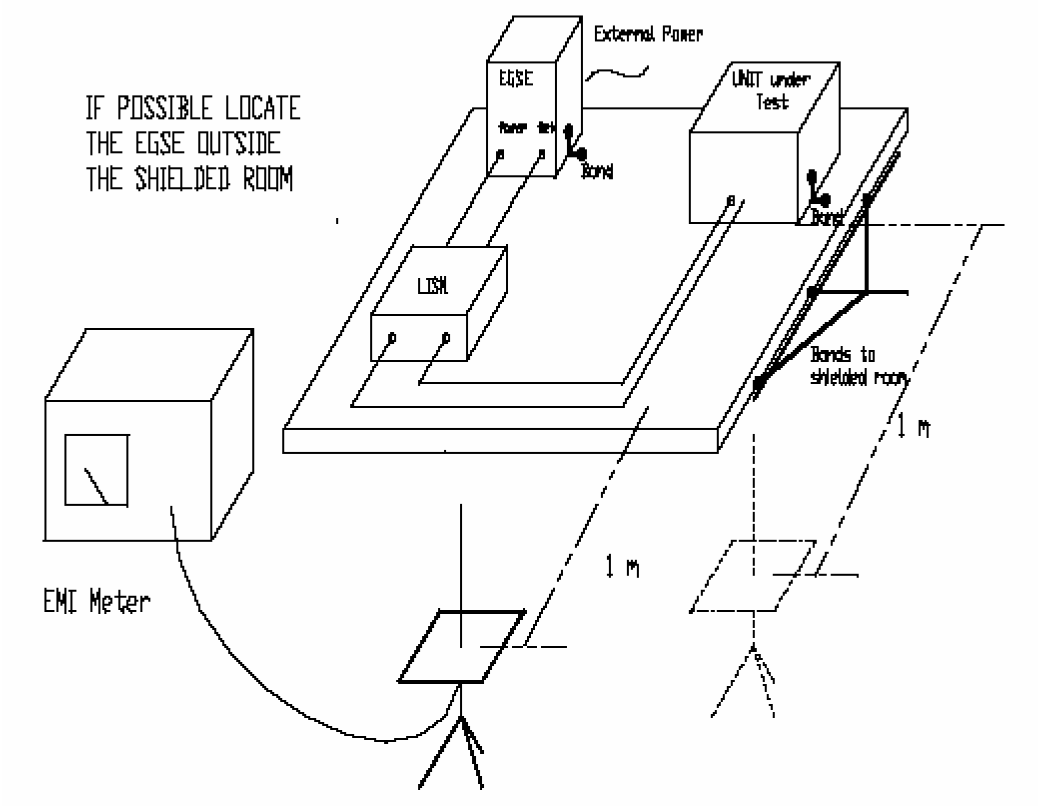


Figure 42 Radiated E-Field Emission Test Set-up

6.4.4.7 Radiated Susceptibility

Radiated Magnetic Susceptibility (RMS)

1. The test set-up shall be as in Figure 43. The distance between the radiating antenna and the UUT shall be the most suitable to achieve the specified level of field strength in the test region.

2. For radiated susceptibility tests loop antennas together with the signal source shall be capable of supplying sufficient current to produce magnetic flux densities 10 - 20 dB greater than the applicable limit at the test frequencies.

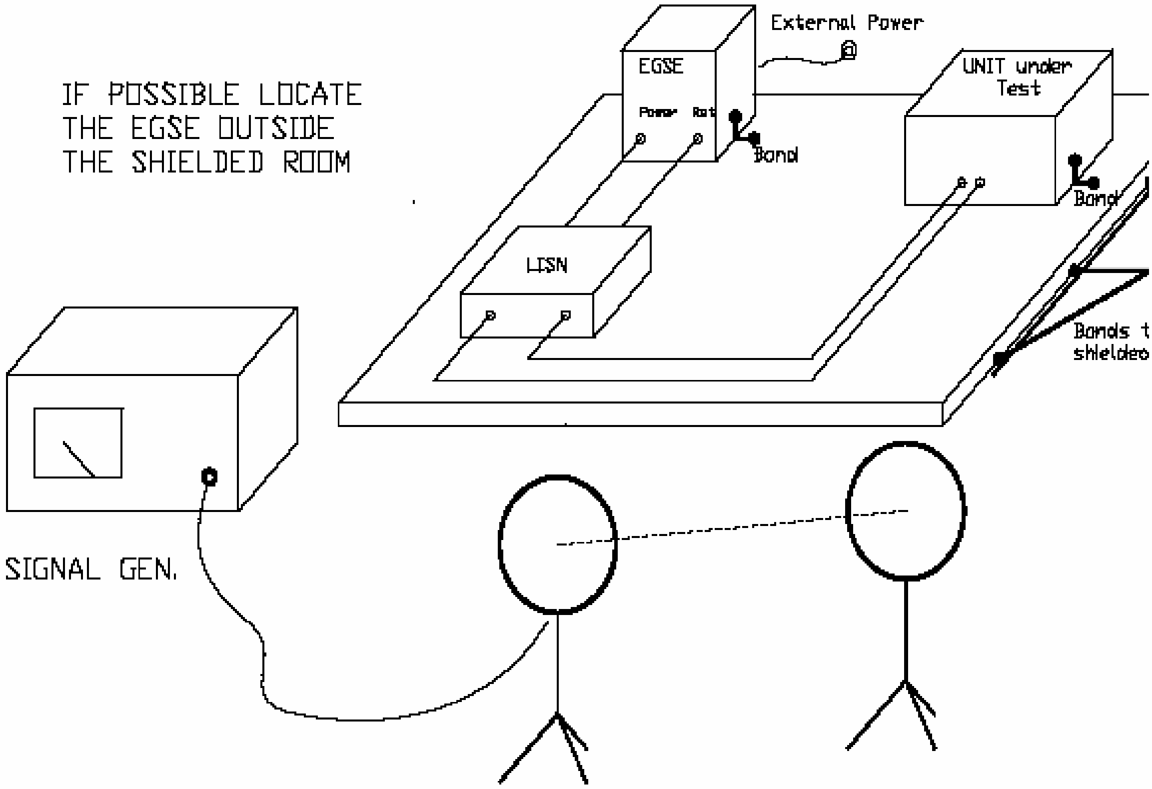


Figure 43 Radiated Magnetic Susceptibility Test Set-up

Radiated Electric Field Susceptibility (RES)

3. The test set-up shall be as in Figure 44. The distance between the radiating antenna and the unit under test shall be not less than 1 metre.

In case the specified field strength cannot be achieved a shorter distance is permitted as long as the test region against the field strength is measured and specified.

4. The sweep speed for the test shall not be faster than 1 octave/minute and the sine wave signal shall be 30% amplitude modulated by 1 KHz square wave.

5. Above 25 MHZ, the requirement shall be met for both horizontally and vertically polarized waves.

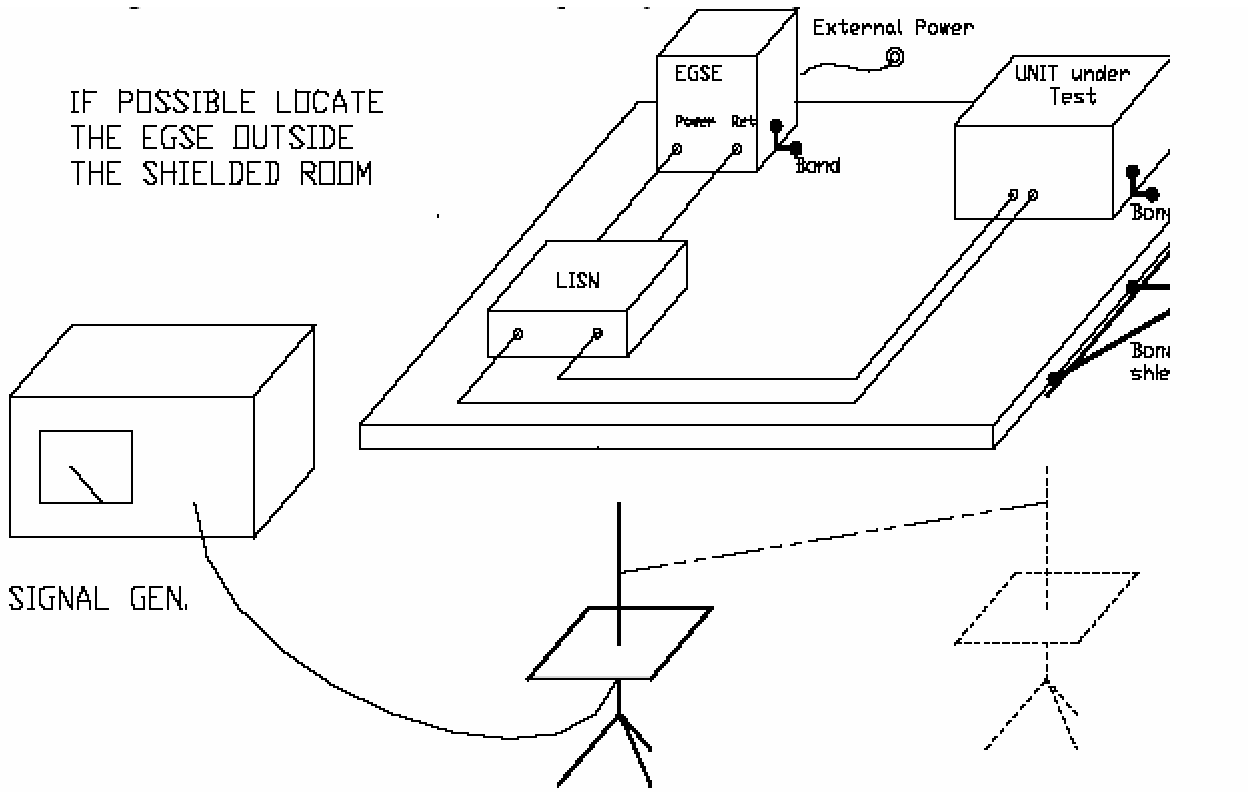


Figure 44 Radiated E-Field Susceptibility Test Set-up

6.4.4.8 Electrostatic Discharge Tests (ESD)

Conducted ESD Susceptibility

Figure 45 contains a suggested arc source schematic capable of establishing the required radiated discharge.

1. The discharge circuit must be adjusted in order to get:
    - a current rise time lower than 15 nS
    - a current duration higher than 40 nS.

2. Any other equivalent type of circuitry can be used and shall be fully described in the relevant plan.

3. A minimum of 10 discharges shall be performed.

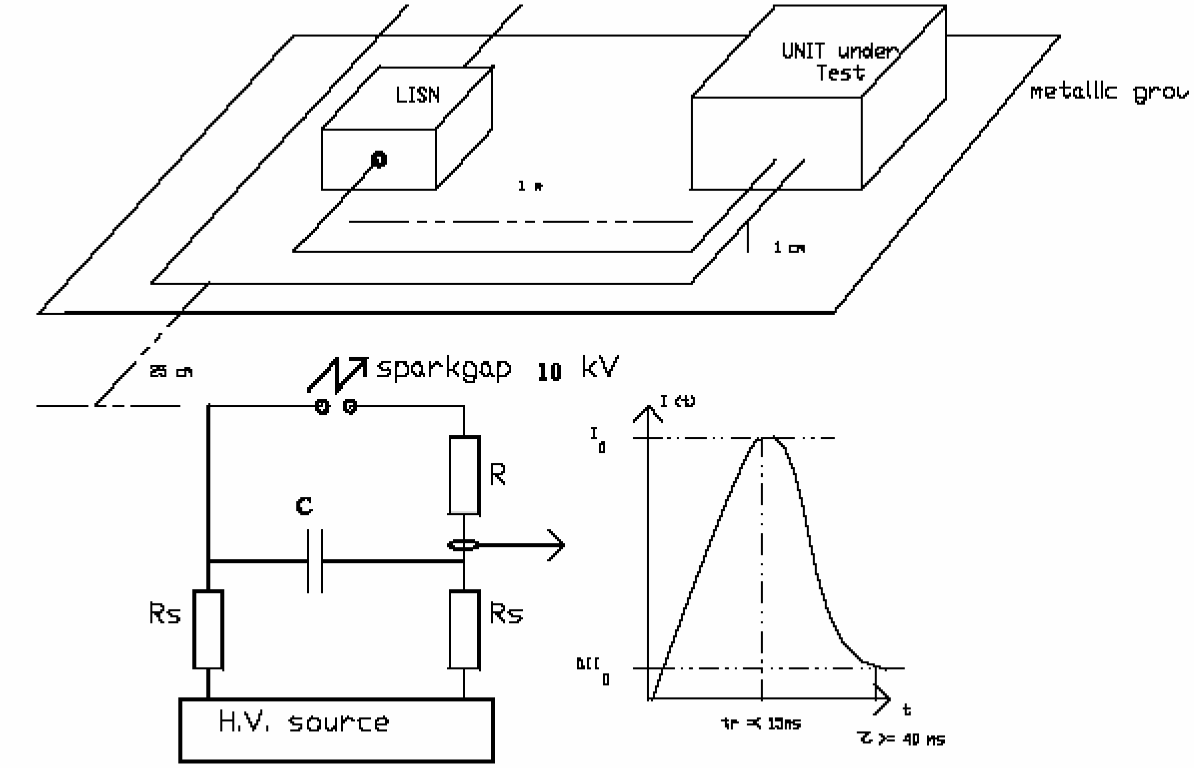


Figure 45 ESD Radiated Discharge Test Set-up

Conducted ESD Susceptibility

Figure 46 contains a suggested discharge generator capable of establishing the required current (length of injection circuit shall be minimized in order to get the proper rise time).

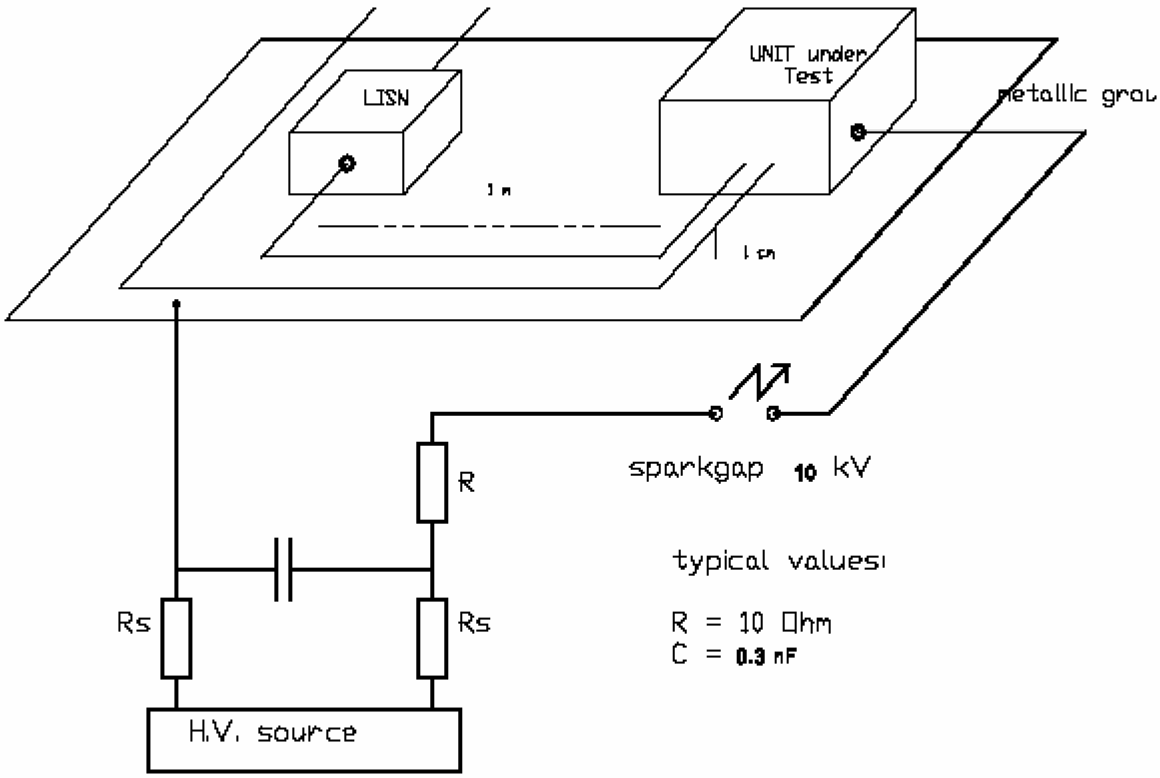


Figure 46 ESD Conducted Discharge Test Set-up

6.4.5 STRUCTURAL TEST REQUIREMENTS

6.4.5.1 Structural Test Setup

1. The instrument unit shall be tested in Launch configuration.

2. Test adaptors and / or non flight items shall be removed before test.

3. The instrument shall be vibrated in hard mounted configuration through the designated S/C interface points.

4. The PI shall provide any special test adapter required for the test.

5. The adaptor shall have a high first resonance frequency (above 2 kHz) in order not to influence the test. Any amplification from the fixture shall not contribute more than 1% to the G rms value during the random test.

6. Standard Instrumentation and procedural guidelines shall apply and be reflected in the procedure.

6.4.5.2 Sine Vibration Test Levels

The qualification and acceptance test levels during sine vibration tests for units for both in-plane and out of plane axes are given below:

Band	qualification level (g)	acceptance level (g)
5-21 Hz	+/- 11mm	+/- 9mm
21-100 Hz	25	20
Duration		
	2 oct/min	4 oct/min

Table 23 Qualification and Acceptance Levels for Sine Vibration Tests

6.4.5.3 Random Vibration Test Levels

Qualification levels during random vibration tests for *units* or assemblies interfacing with the S/C for each axis are defined as follows:

Band	**Out of Plane**	**unit**
20 - 100 Hz	+3 dB/oct	
100 - 300 Hz	PSD(M) = 0.12 x (M+20)/(M+1)	g^2/Hz
300 - 2000 Hz	- 5dB/oct	
Band	**In-Plane**	**unit**
20 - 100 Hz	+3 dB/oct	
100 - 300 Hz	PSD(M) = 0.05 x (M+20)/(M+1)	g^2/Hz
300 - 2000 Hz	-5 dB/oct	
Duration	2 min	

Table 24 Qualification Levels for Random Vibration Tests

Acceptance levels shall be applied during 1 minute and shall be calculated using the formula below:

PSD Acceptance Level = PSD Qualification levels / 1.5625

6.4.5.4 Acoustic Test Levels

The instrument units shall survive the acoustic levels shown in the table below.

Octave band centre frequency (Hz)	Qualification level (dB)	Acceptance level (dB)
31.5	128	124
63	135	131
125	137	133
250	139	135
500	137	133
1000	128	124
2000	124	120
4000	115	111
8000	109	105

Table 25 Acoustic Test Levels (TBC)

6.4.5.5 Shock Test Levels

The instrument units shall be verified against the shock environment defined in the Figure below.

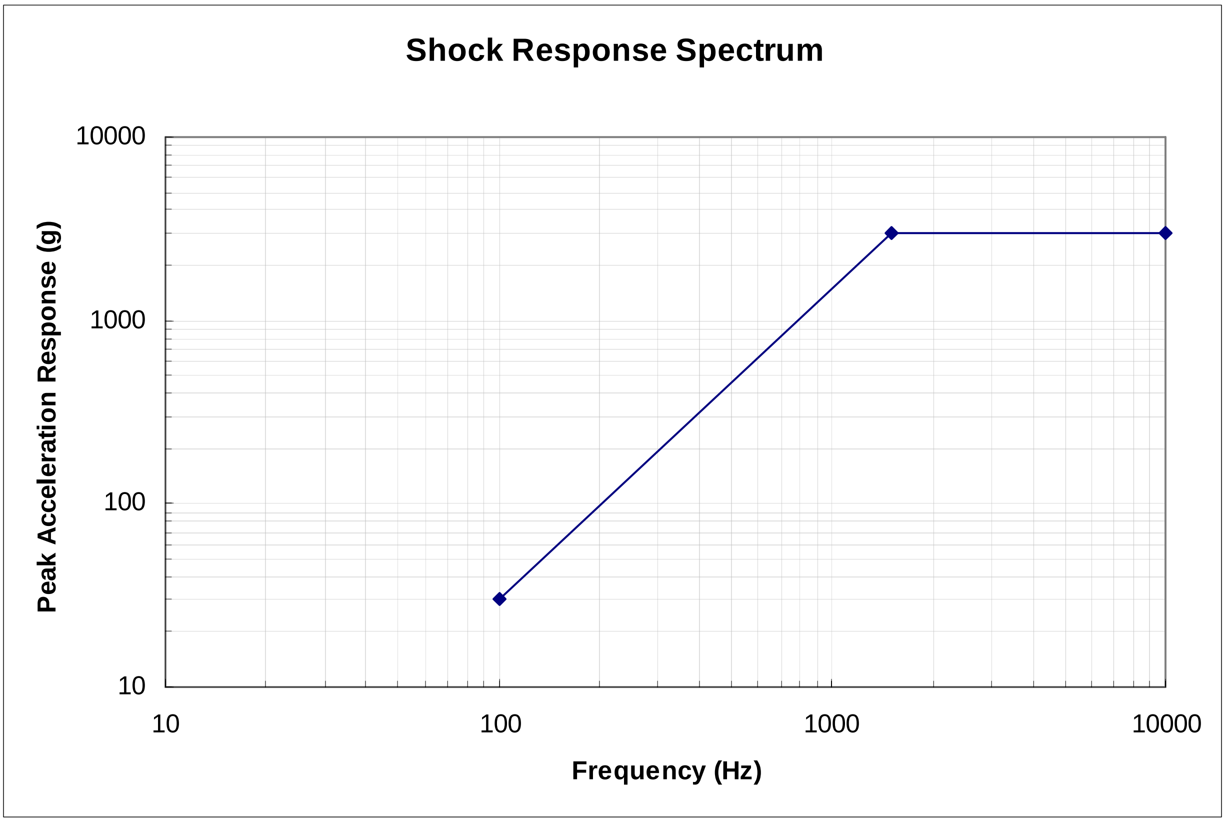


Figure 47 Shock Environment

6.4.5.6 Pressurized Items Test Requirements

1. The guidelines and requirements listed in ECSS-E-30-01a [NR19] and ECSS-E-30-02 [NR8] (draft issue) shall apply.

6.4.6 MECHANISM TEST REQUIREMENTS

6.4.6.1 Mechanisms Verification

1. The mechanisms verification test programme shall ensure that the hardware conforms to the design, construction and performance requirements as specified in the relevant applicable documents.

2. Tests shall be performed to check mechanisms performance in both launch and operational configurations.

3. Mechanisms can be considered as structures as far as strength and stiffness tests are concerned, and their design shall be verified against the same requirements as other structural components.

As a reference, the following tests' sequences are applicable:

- Functional tests (before and after thermal vacuum exposure)
- Mechanical environment tests
- Thermal vacuum functional test

6.4.6.2 Mechanism Lifetime Tests

1. The lifetime of a mechanism shall be demonstrated by test in the appropriate environment, using the sum of the predicted nominal ground test cycles and the in-orbit operation cycles.

2. For the test demonstration, the number of predicted cycles shall be multiplied by the following factors:

Type/Number of Predicted Cycles

Ground Testing • number of on-ground test cycles (the minimum number to be used is 10)	x4
In-orbit predicted cycles:	

• 1 to 10 actuations	x10
• 11 to 1,000 actuations	x4
• 1001 to 100,000 actuations	x2
• > 100,000 actuations	x1.25

As actuation, a full output cycle or full revolution of the mechanism is defined.

3. In order to determine the lifetime to be demonstrated by test, an accumulation of actuations multiplied by their individual factors shall be used.

6.4.7 THERMAL TESTS REQUIREMENTS

6.4.7.1 Thermal Design Verification

1. The thermal design of a payload unit shall be verified by a dedicated thermal balance test according to the guidelines and requirements laid down in section 6.4.7.2.

2. The thermal balance test will consist of at least a hot and a cold steady-state and several transient phases that simulate boundary conditions experienced during the mission, **including actual Sun exposure.**

Figure TBS depicting Thermal Balance Test.

3. The validity of the unit design to meet its functional goals and to operate satisfactorily in vacuum in the temperature range expected during the mission shall be verified by a combined thermal vacuum and thermal cycling test.

4. The tests shall be designed on a case-by-case basis by the unit responsible and agreed with ESA.

5. The Acceptance Temperature Range shall be equal to the Design Temperature Range "+" or "-" an acceptance margin of 5 deg C.

6. The Qualification Temperature Range shall be equal to the Acceptance Temperature Range "+" or "-" a qualification margin of 5 deg C.

For an ordinary electronic box, the thermal verification can be derived from the unit qualification test, if the unit is adequately internally equipped with thermal sensors and proper steady-state phases are included in the test.

7. Instrument specific thermal testing requirements at unit and system levels shall be defined by the PI.

6.4.7.2 Test Methods

1. The equipment shall be mounted in a vacuum chamber in a thermally controlled environment (See Figure 48).

2. Temperatures shall be controlled, measured and selected such that it can be guaranteed that the test item experiences actual temperatures equal to or beyond the minimum and maximum qualification/acceptance temperatures in the test environment.

3. The instrument shall be qualified using the type of fixations and mountings as designed in the instrument specification.

This is achieved by adopting one of the following test methods as appropriate.

Non Special Equipment, Internally Mounted.

4. The equipment shall be bolted to a mounting panel, using the correct bolts and bolt torques as specified in the equipment interface specification.
5. The mounting panel shall be black-painted (except for the mounting contact area) and have the following dimensions as a guideline:
    - thickness representing standard platforms/sidewalls,
    - length and breadth approximately equal at least to twice the nominal base dimensions of the equipment.
6. The mounting panel is temperature controlled.
7. During the test, the shroud and/or the panel temperatures shall be controlled to a fixed temperature to provide the spacecraft internal environment to give the qualification temperature level on the equipment itself.
8. The temperature reference point should be located at the outer surface of the instrument on its baseplate or near to its mounting feet. The number of reference points should be kept to a minimum - only one point whenever possible. A temperature sensor shall be located at the reference point, as an integral part of the equipment.

Special Equipment, Internally Mounted

Certain internally mounted equipment will require special test provisions. Examples of such equipment would be:

- sensors having viewing apertures seeing space and / or the planet
- highly dissipating, directly radiator cooled equipment.

9. In such cases, the required approach is to modify the test method given for internally mounted equipment, to the extent needed to give a reasonable representative test environment.

Equipment, Externally Mounted

Any instrument mounted outside the main spacecraft body will have special test requirements.

10. The test arrangement shall be designed to give the required qualification temperatures on the equipment, with approximately representative heat flows to and from the environment.

11. The following minimum test requirements shall be satisfied:
    - Equipment shall be tested in a thermal vacuum environment, having a pressure of 0.0013 Pa (10-5 Torr) or less. The test may be commenced when the pressure falls below 0.013 Pa (10-4 Torr), and a pressure of 0.0013 Pa or less shall be achieved prior to start up of the units not operating during first ascent.

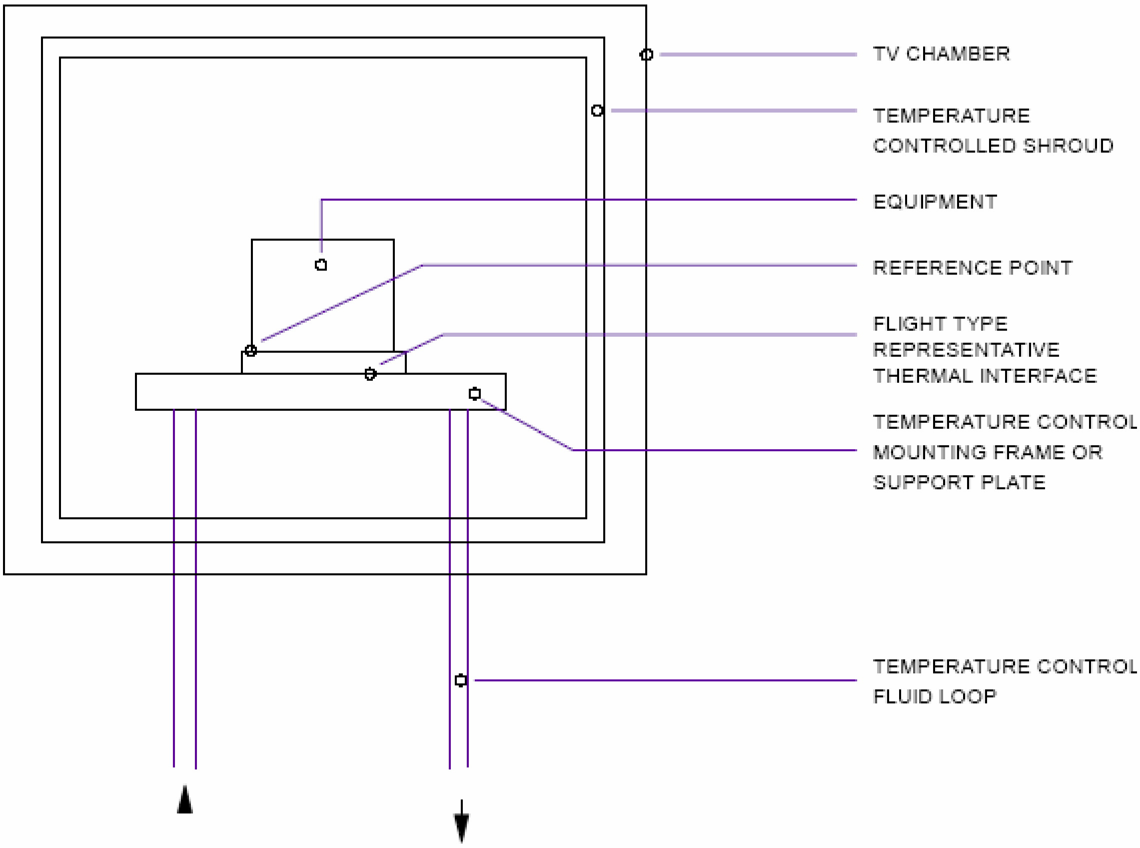


Figure 48 Equipment Thermal Vacuum Test Arrangement

6.4.7.3 Temperatures and Cycles

1. The equipment shall be tested in the thermal vacuum test sequence, as shown in Figure 49.

The Temperature cycle begins with the initial functional test with the chamber at ambient temperature. At a pressure of 0.013 Pa, the temperature is increased first, for better outgassing, up to the high non-operating level (TNO-MAX). After a dwell time tE, the temperature is decreased to the hot start-up level (TSU-MAX), then the instrument switched ON and thereafter the temperature stabilized at the high operating temperature (TOP-MAX) during a time tE. After the time tE, the functional test is performed.

The equipment is switched off and the temperature is decreased and stabilized at the low non-operating minimum temperature (TNO-MIN) during the time tE. The temperature is increased to the cold start-up to switch the equipment ON. After stabilization at the low operating level (TOP-MIN), after a time tE, the functional test is performed. This constitutes one complete cycle.

Then at the high operating level after a time tE, the functional test is repeated, followed by a low operating level with a functional test after the time tE. This is the second cycle (without the hot and cold start-up and non-operating levels). The second cycle is repeated for the number of cycles required. The number of cycles, the temperature levels and rate of change and the dwell time are specified in Table 26.

The equilibrium temperature is reached when the temperature changes less than 1°C/hr.

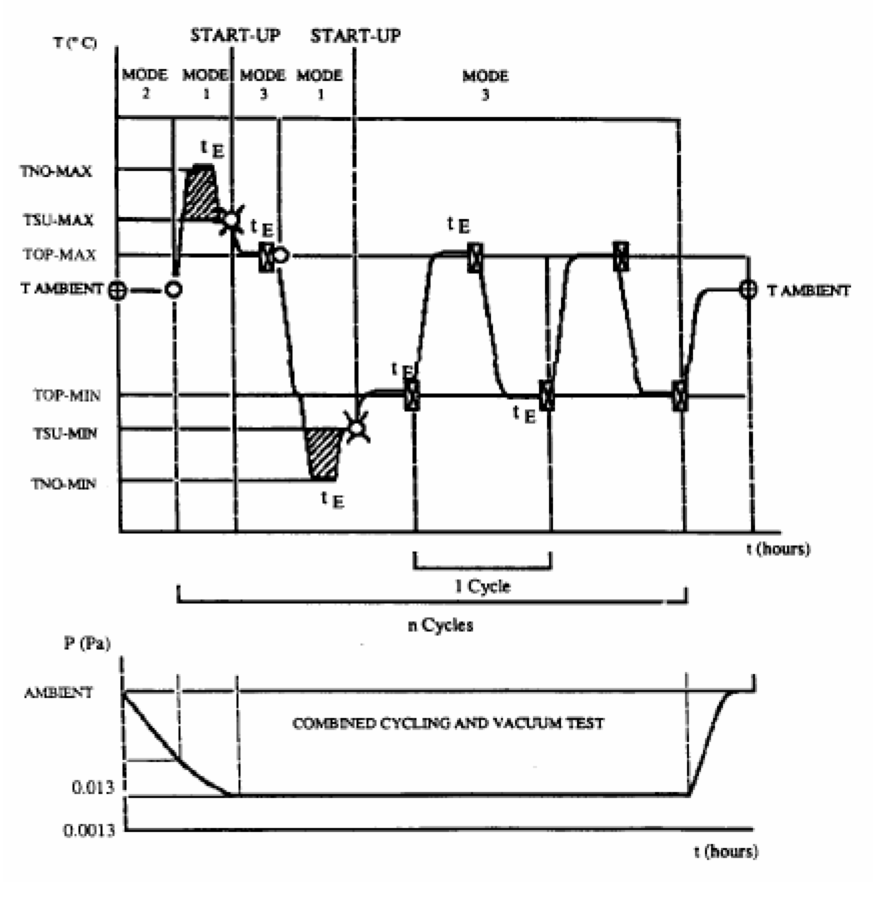


Figure 49 Equipment Thermal Cycling and Thermal Vacuum Combined Test Sequence

NOMENCLATURE TO Figure 49

• T	Temperature
• TNO-MAX	Maximum Non-Operating Temperature (the highest design temperature the equipment has to survive not powered)
• TNO-MIN	Minimum Non-Operating Temperature (the lowest design temperature the equipment has to survive not powered)
• TAMBIENT	Ambient temperature
• TOP-MAX	Maximum Operating temperature (the highest design temperature at which the equipment has to demonstrate full design ability)
• TOP-MIN	Minimum Operating temperature (the lowest design temperature at which the equipment has to demonstrate full design ability)
• TSU-MAX	Maximum Start-up Temperature (the highest design temperature of the equipment, at which the equipment may be switched on)
• TSU-MIN	Minimum Start-up Temperature (the lowest design temperature of the equipment, at which the equipment may be switched on)
• P	Pressure
• MODE 1	Functionally inert (test item not energized normally applicable to the non-operating condition).
• MODE 2	Partially functioning. Conditions as detailed in applicable design
• STAND-BY	specifications, but normally applicable to conditions during launch.
• MODE 3	Fully functioning (test item fully energized and fully stimulated). Normally applicable to conditions during orbit.
• Å	Initial and Final Performance Test
• X	Intermediate Reduced Performance Test
• tE	Intermediate Equilibrium Temperature Time, dwell time
• 0	Switch-on (Start-up)
• 0	Switch-off

Temperatures	For qualification test the qualification temperatures shall be used. For acceptance test the acceptance temperatures shall be used
Temperature rate of change	dT/dt = 5 ... 20°C/min
Dwell time	$t_E \geq 2$ h
Stabilisation criterion	$\Delta T / dt \leq 1$°C/h
Number of cycles	n = 8 for qualification n = 4 for acceptance n = 4 for recertification

Table 26 Test Parameters Values

6.5 Inspections

6.5.1 VISUAL INSPECTION

1. Visual Inspections shall be performed at the beginning and end of acceptance and qualification testing.

2. The inspection shall include as a minimum:
   - Completeness of hardware
   - Identification of hardware
   - Connectors
   - Grounding Points
   - Attachment Surfaces
   - Thermal Surfaces (any visible changes)
   - Inspection of transport conditions
   - Inspection for damage
   - Inspection of Interfaces
   - Completeness of documentation

6.5.2 PHYSICAL PROPERTIES

The purpose of physical properties measurements is to determine the equipment physical characteristics, i.e. dimensions, mass, centre of gravity and momentum of inertia.

1. The measurement of physical properties shall include:
   - Mass

- CoG
- Momentum of Inertia

2. The following dimensions as a minimum shall be verified:
    - interface dimensions
    - envelope dimensions (including envelope of separate electronics box, filter, etc. if applicable)

6.6 Calibration

1. The PI shall provide a calibration plan adapted to the scientific requirements and the overall development plan of the instrument and of the satellite.

2. The instrument shall be delivered fully calibrated.

3. Calibration activities at system level shall be only considered when scientifically justified, i.e. when for example the flight configuration is reached only after integration on the satellite. This type of activity is subject of agreement with ESA and its selected Prime.

4. The calibration plan shall be part of the Instrument Development Plan.

6.7 Final Acceptance

6.7.1 GENERAL REQUIREMENTS

1. The acceptance process shall demonstrate that the Instrument has been fully verified in terms of:
    - scientific performances (including calibration and characterization)
    - behavior versus environmental conditions (including EMC)
    - all functional interfaces

2. The acceptance of the Instrument shall follow the sequence hereafter:
    - completion of acceptance tests, including calibration / characterization at the Instrument supplier premises, in order to verify that the Instrument together with its ground support equipment meet all interface specifications and that the Instrument is ready, for integration onto the satellite
    - acceptance review of the tests results and of the completeness of the acceptance data package at instrument manager premises and release of a consent to ship if the acceptability is stated by the review board

- delivery to the satellite AIT site of the Instrument together with the ground support equipment (including test software and documentation) and the acceptance data package
- performance - by the Instrument supplier - of a post shipment inspection and at an incoming test at the AIT site
- after successful completion of the incoming verifications by the Principal Investigator and formal incoming inspection by system level QA, the Instrument will be released for integration onto the satellite
- notwithstanding the mandatory Instrument level tests, the Instrument software will only, be accepted after successful S/C level tests.

6.7.2 ACCEPTANCE REVIEW

1. The acceptance review will check and ascertain the following topics:
   - visual inspection and completeness of the hardware to he delivered
   - compliance of the interfaces measurements (Spacecraft interfaces)
   - availability of a complete set of functional performances data (using both the Limited Performances Test and Full Performances Test procedures)
   - availability of calibration and characterization data
   - ground support equipment relevant characteristics and documentation
   - verification of the S/W configuration
   - verification of the built standard
   - completeness of the Acceptance Data Package

6.8 System Level AIT

6.8.1 MODEL PHILOSOPHY

6.8.1.1 Satellite Model and Test Philosophy

1. The Satellite Model Philosophy to be applied for Solar Orbiter is the classical 3 Model Approach consisting of:
   - STM: Structural Thermal Model;
   - EM: Engineering Model
   - FM: Flight Model

The different models will undergo their dedicated qualification / acceptance test programme according to agreed system level requirements.

The instrument model will be integrated on the satellite and tested as an integral part of the satellite system.

Test	STM	EM	PFM
• Physical Properties	(+)		(+)
• Static Load			
• Fit-check Elements	(+)		(+)
• Fit-check Launcher	(+)		(+)
• Deployment + Separation Shock	(+)		(+)
• Launcher Separation Shock	(+)		(+)
• Low Level Sine	(+)		(+)
• Sine Vibration	(+)		(+)
• Acoustic Noise	(+)		(+)
• Modal Survey	(+)		(+)
• Mechanisms	(+)		(+)
• Alignment	(+)		(+)
• Pressure / leak	(+)		(+)
• Thermal Balance	(+)		(+)
• Thermal Vacuum	(+)		(+)
• Electrical Interfaces		(+)	(+)
• HW/SW Compatibility		(+)	(+)
• Conducted EMC		(+)	(+)
• Radiated EMC		(+)	(+)
• RF compatibility		(+)	(+)
• Ground Segment Compatibility			(+)

The STM system test objectives are:

- qualification of primary structure by test
- verification of mechanisms function in system environment
- verification of structural mathematical model
- qualification of thermal design by test
- verification of thermal mathematical model.

From this the instrument STM built standards described in the following section are derived.

The EM system test objectives are:

- verification of all electrical and software interfaces
- verification of subsystem and instrument performance within system environment
- qualification of on-board software
- verification of system performance
- verification of operational procedures
- verification of electro-magnetic compatibility by emission, susceptibility and ESD tests.

From this the instrument EM build standards described in the following section are derived.

The FM system test objectives are:

- completion of qualification programme where necessary the a PFM approach may be needed
- acceptance of spacecraft system by functional and environmental tests

The Flight Spares (FS) objectives are:

1. replacement of failed or damaged equipment at integration and launch site.

6.8.1.2 Derived Deliverable Instrument Models

1. The instrument STM units therefore shall have the following build standard:
    - structure flight standard
    - mechanisms flight standard
    - pyrotechnics flight standard (including electrical interfaces)
    - thermal control hardware flight standard
    - internal units flight representative for mass, CoG, stiffness, mounting, shape and internal power dissipation.
    - harness flight representative for mass, shape

2. The instrument EM units therefore shall have the following minimum build standard:
    - electronics flight standard except for parts.
    - commercial parts have to be of same technology, same supplier as FM parts
    - mechanisms flight representative for electrical actuators
    - structure flight representative for mounting and shape
    - software flight standard
    - harness flight representative.

In order to save cost the EM hardware contents may be reduced by reducing redundancy (to be agreed with ESA):

a. cold redundant units may be deleted if no automatic switch-over function is involved
b. multiple redundancy of hot redundant units or modules may be reduced by electrical dummies (to e.g. dual redundancy) if EM objectives for interference and crosstalk tests are not compromised.
c. possible use of simulator for the EM verification purposes is TBD.
d. Use of EQM if sufficiently representative electrically and functionally.

The EM units will remain at the Prime Contractor's premises following delivery until after payload in-flight commissioning.

3. The FM units shall have full flight standard verified by formal functional and environmental acceptance tests.

4. The FS units shall have full flight standard verified by formal acceptance tests.

In order to save cost the FS units

a. may be derived from refurbished qualification units if full flight worthiness can be demonstrated
b. may be reduced to repair kits for repair at manufacturer's site if pre-determined turnaround time is ensured. This approach has to be agreed with the project office on a case by case basis.

6.8.1.3 Spares Philosophy

The PI shall propose a spares philosophy at component and unit levels that is consistent with the above model philosophy as well as the specifics of the instrument, to be agreed with the ESA Project Office.

6.8.2 SYSTEM INTEGRATION AND TEST FLOW

TBW

6.8.3 GROUND SUPPORT EQUIPMENT

TBW

7 PRODUCT ASSURANCE REQUIREMENTS

7.1 General

All space products procured in the frame of a programme of the European Space Agency are required to conform to the Agency's Product Assurance (PA) requirements as laid down in the ESA-ECSS series of documents.

The ECSS-Q standards define the Product Assurance (PA) policy, objectives, principles and rules for the establishment and implementation of PA programmes for projects covering the mission definition, design, development, production and operations of space products including disposal. They shall be considered **informative documents to be tailored to the specific needs of the mission.**

Safety requirements imposed on ESA by the respective Launcher Authorities are mandatory on all flight hardware and software and hence the relevant requirements are applicable for all instruments.

Note: *Despite the fact that ESA and the selected Prime will act as formal interface to the Launcher Authorities this does not release the PI from the commitment to provide adequate inputs to ESA in order to comply with the applicable Launcher Authority Safety Requirements.*

The prime objectives of the PA requirements are:

- to establish confidence in the design;
- to enhance the overall mission integrity;
- to assure the safety of the system and its operations;
- to assure that failures in one element do not have detrimental effects on other elements

While the first two topics intend to assure a successful functioning and performance of an instrument, the latter two aim to assure the safety and integrity of the interface of the instruments with the spacecraft, other instruments and the launcher.

Following these two different objectives the PA requirements defined in this document as derived from applicable ECSS set of documents will be:

- fully applicable to the interface of the instrument with the spacecraft (and the launcher)
- partially applicable to the overall instrument to assure the success of the programme

The interface between an instrument and the spacecraft must be understood in a wider sense than simply mechanical, electrical or thermal, e.g.:

Additional to the “physical” interfaces:

- mechanical/dynamic:
  elements that contribute to the mounting, fixation, position of an instrument, a subassembly

a device or part of them and which can by its failure or faulty operation damage or render the capabilities of other elements of the platform

- electrical
  elements (harness or electronic equipment) which can be a source of any over voltage, under voltage, over current, under current (versus nominal design) or any unpredicted variation of an electrical signal of the interface circuit, capable to create any degradation to the electrical circuit characteristics or to the operational performance of the platform or any other instrument
- thermal
  elements which can cause any unexpected change in temperature or heat flux capable to generate major disturbance in the thermal balance of the platform or other instruments
- radiative electromagnetic
  elements which can cause any disturbance of the platform or other instruments by electromagnetic effect
- optical
  elements which can cause disturbance of the platform or other instruments by generation of reflexion, absorption, biasing or modification (stray light) of the optical flux to a sensor, detector or from a source
- contamination
  outgassing of materials that can contaminate other instruments or lead to degradation of surface coatings that can influence the thermal control of the instrument and the spacecraft

The following needs to be considered as interface relevant:

- control of materials and processes that can affect the structural integrity of the instrument and hence the spacecraft, and even the launch vehicle;
- outgassing of materials that can contaminate other instruments or lead to degradation of surface coatings that can influence the thermal control of the instrument and the spacecraft;
- qualification and acceptance testing of the instrument alone and after integration in the system;
- control of non-conformances to avoid effects on other instruments and schedule delays during integration;
- configuration control on documentation, hardware and software to make possible to build and to operate a complex integrated system.

Specific PA rules defined herein are aimed at controlling the phenomena that may propagate beyond the interface of the instrument to other parts of the spacecraft. With the help of a Failure Modes, Effects and Criticality Analysis prepared by the PI early in the design phase, the critical interfacing elements of an instrument will be determined and agreed between the Principal Investigator (PI) and the ESA selected Prime.

The PA Requirements and Guidelines defined here have been established to prevent potential problems, and past experience has shown that they are cost-effective and provide long term benefits to all parties participating in the programme.

Taking the above into account, the PA requirements and guidelines defined hereafter are derived from the entire set of applicable documentation, and have been tailored and limited to be relevant to the items contributed by the PI. In General:

1. All products procured in the frame of the Solar Orbiter programme of the European Space Agency, independently whether they will be procured under direct ESA industrial contract, direct PI responsibility or under responsibility of partner agencies, shall be conform to the "tailored" ECSS standards, as described in the following sections.

2. The PI shall further apply launch site and launch vehicle safety requirements and regulations as defined in these programme requirements.

7.2 Product Assurance Management

7.2.1 GENERAL

1. The Principal Investigator (PI) shall establish and implement an effective PA programme in accordance with the ECSS-Q-00A [NR9] to support the PA activities at programme level. The programme shall provide for the assessment and control of risks, and that acceptability of the residual ones is evaluated. This shall by:
   - the prevention or early detection of actual and potential deficiencies
   - the identification of system incompatibilities
   - the identification of aspects, which could affect project requirements having major impacts on safety, mission success and the related cost and schedule consequences

The basic implementation principles are to:

- ensure the allocation and availability of adequate resources, personnel and facilities to carry out the required PA tasks, (see 6.2.2)
- define, in a Product Assurance Plan all PA activities consistent with the Project objectives, requirements, criticalities and constraints, (see 6.2.3)
- ensure that lower level contractors / suppliers perform proper PA monitoring and control,
- ensure proper progress monitoring, reporting and visibility of all PA matters, in particular those related to alerts, critical items, non-conformances, changes, deviations, waivers, actions and/or recommendations resulting from reviews, inspection and audits, qualification, verification and acceptance.

2. The PI shall report on a regular basis as specified in section 7 of this EID-A the status of the product assurance programme implementation as part of the regular progress reporting.

3. Reporting of PA progress shall be part of the overall progress reporting and shall address:
   - risk management
   - specific PA activities and actions
   - critical items status

- non-conformance and waiver status

7.2.2 ORGANIZATION

1. The PI shall establish an effective PA organization tailored to the size and complexity of the programme.

Note: This requirement applies also to eventual supplier / contractors

2. The PI shall assign an instrument PA manager from the PA line organization (if existing in the PI organization)
   - reporting functionally to the instrument manager and
   - having unimpeded access to higher management, to the PA structures within the Prime Contractor Organization and ESA, who
   - will manage the PA activities within the instrument collaboration and
   - will coordinate these activities with the ESA designated Prime

3. The appointed instrument PA manager shall have sufficient organizational authority and independence to propose and maintain a product assurance programme in accordance to the Solar Orbiter product assurance requirements

4. The PI shall identify PA resource requirements and shall provide adequate resources to perform the required PA tasks. Trained personnel shall be assigned to the various PA activities.

5. In case the PI has no suitable facilities or experienced personnel, adequate measures shall be undertaken (including applying for additional funds) to grant the use of external facilities and / or ensure the training of personnel.

The use of National Agency resources, consultants and contractors should be considered for specific tasks for which in-house expertise is not available and where the investment may not be possible.

7.2.3 PRODUCT ASSURANCE PLAN

1. The PI (or his PA manager) shall prepare and implement a project product assurance plan as part of the management documentation.

2. The PI shall maintain the PA plan throughout the project life cycle. The PA plan may refer to clauses of the PI Organization Quality Manual and to in-house procedures.

Note: *The update rate of this document is related to the major review points as listed in Section 7.*

3. The PA plan shall describe the PA responsibilities within the instrument collaboration and eventually be extended to outside facilities and external personnel used during the project lifetime.

4. The PA plan shall describe in a structured manner the implementation phase addressing explicitly critical areas pertinent to the instrument development, such as magnetic, optical cleanliness, deployable items, safety items etc.

7.2.4 ESA/PRIME CONTRACTOR RIGHT OF ACCESS

For the purpose of product assurance and technical coordination ESA and / or its selected Prime have the right to perform or participate to, together with the Investigator, audits, surveys, source inspections, test reviews, mandatory inspections, etc., at the facilities of the PI and his contractors and suppliers.

ESA or its Prime contractor's participation shall not in any way replace or relieve the PI of his responsibility; it will be rather aimed at contributing to the identification of problem areas and assessing satisfactory progress.

1. Arrangements shall be made to permit designated ESA / Prime contractor personnel free access to all technical and programmatic documentation, areas and operations within the facilities of the PI and his contractors and suppliers in which work related to the Solar Orbiter Programme is being performed.

Proprietary rights of the PI and third parties will be fully respected.

7.2.5 CONTRACTOR AND SUPPLIER SURVEILLANCE

1. In case the PI procures equipment or services from contractors or suppliers, he shall impose on them a set of product assurance requirements derived from the requirements listed herein, and tailored to the criticality of the products or services being provided.

2. The delegation of product assurance tasks by the PI to another lower tier supplier shall be done in a documented and controlled way. The PI retains the responsibility towards ESA and its selected Prime.

7.2.6 IDENTIFICATION AND CONTROL OF CRITICAL ITEMS

1. The QA function shall contribute to the overall risk management activities by:
   - Supporting the identification and risk evaluation of critical items for which major difficulties or uncertainties are expected in:
     - demonstration of design performances
     - development and qualification of new product, processes and technologies

    - procurement, manufacturing, assembly, inspection, test, handling,
    - storage and transportation, which may lead to major degradation in the scientific performance of the instrument
- Monitoring and documenting the achievement of the specified risk reduction implementation and the corresponding verification measures throughout all project phases.
- Identifying single-point failures with a failure consequence severity classified as catastrophic, critical or major (Reliability Critical Items)
- Identifying items or procedures that do not comply with the applicable safety requirements, or which cannot be verified as complying with those requirements (Safety Critical Items).
- Identifying products that cannot be checked and tested after integration, limited-life products, products that do not meet - or can-not be verified as meeting – applicable maintainability requirements (Maintainability Critical Items)
- Identifying items who's structural failure may cause catastrophic or critical consequences (Fracture Critical Items)

2. A Critical Items List (CIL) shall be prepared as a summary of data from the different disciplines and with identical information, in accordance with the template ECSS-Q-20-04A [NR21].

3. An item shall be classified as critical if it meets the criteria defined in ECSS-Q-20-04A [NR21].

4. The complete CIL shall be updated to the main reviews. The Critical Items of the category MAJOR shall be maintained permanently and changes shall be reported as part of the progress report and during the progress meetings.

7.2.7 PA DATABASE

1. All PA-related data (such as NCR's, RFW's, EEE components list, DML, DPL, MIPs/KIPs) shall be stored in an electronic database. This shall allow to import and export data, electronically by email, from and to contractor, suppliers and the Agency.

2. The databases format and content shall be agreed with the Agency and imposed on all suppliers.

3. For the NCR database the Internet capable NCTS version of ESA shall be used for major NCRs which address violations against performance or interface requirements.

7.2.8 QUALITY RECORDS

1. The Contractor/suppliers shall maintain quality records to provide objective evidence of complete and effective performance of Quality Assurance activities and to demonstrate the achievement of the required quality.

2. Quality records shall be stored in a safe way to prevent alteration, loss or deterioration, for at least three years after the end of mission.

3. The Agency shall have the right to access quality records upon request.

7.3 Quality Assurance Management

7.3.1 GENERAL REQUIREMENTS

The PI shall establish a detailed **Quality Assurance (QA) Programme Plan** as part of the PA plan following the generic guidelines given in ECSS-Q-20B [NR10]. These guidelines shall be considered a reference document for the Solar Orbiter instruments. . Activities that have an impact on quality, dependability and safety shall be covered by written procedures. These shall be available to the Agency for review upon request.

1. If the PI institute/organization does not already provide a proven self-standing PA/QA organization, the PI shall establish (in collaboration with those responsible for PA/QA) a QA system covering the following tasks:

    - Documentation and Data Control including Quality Records and Stamp Control;
    - Traceability and Logbook (see 6.3.2);
    - Metrology and Calibration (see 6.3.3);
    - Non-Conformance Control System (see 6.3.4);
    - Alert System (see 6.3.5);
    - Handling, Storage and Preservation (see 6.3.6)
    - Statistical Quality Control and Analysis

Note: *Herewith related guidelines can be found in [NR10]*

2. The QA requirements shall be made applicable to:

    - flight models and spares;
    - manufacturing, assembly and integration facilities and tools / equipment interfacing directly with flight hardware (partially applicable)
    - hardware subjected to or participating in design verification / qualification testing with respect to the properties relevant for those tests;
    - portions of the GSE which interface directly with flight hardware or which can have an impact on safety (e.g. lifting devices).

3. The PI shall provide evidence that QA personnel and other personnel, whose performance affects the instrument quality, have followed adequate training programmes according to national or international standards. Especially those personnel performing critical processes or controlling critical processes shall be trained and certified according to the ESA accepted standards.

4. The QA management is an integral part of the configuration management. As such the QA function shall ensure that:
   - the as designed status is defined prior to manufacturing
   - the as-built documentation is properly defined, identified, traced to the as-designed baseline and maintained in order to reflect approved modifications;
   - items to be delivered comply with the as-built documentation.

5. Waivers shall be controlled via the Configuration Management System.

6. Waiver processing shall follow all of the related rules for CM. Suppliers as well as the PI, shall submit a Request for Waiver (RFW) to the Agency for approval when the disposition of a major NCR is “use as is” or “repair”, and there is a discrepancy with an Agency requirement, or when the Contractor or a Supplier wishes to deviate from a requirement for whatever reason. A distinction between “waiver” and “deviation” need not to be performed.

7. The waiver shall identify the baseline and specification affected, provide an estimate of the impact on cost, schedule, and logistics, and provide a justification for its request.

8. All waivers shall be listed and described in the RFW section of the PA electronic database, including minors and supplier RFW’s.

9. The waiver processing procedure and the waiver format shall be delivered to the Agency for review and approval.

7.3.2 TRACEABILITY AND LOGBOOK

1. Each part, material or product shall be identified by a unique and permanent part or type number.

Note: *To assure a full traceability the following controls shall be established:*
- *Identification numbers are assigned in a systematic and consecutive manner.*
- *Identification numbers of scrapped or destroyed items are not used again.*
- *Identification numbers, once allocated, are not changed, unless the change is authorized by the ESA or the selected Prime.*
- *The logbooks shall contain historical and quality data and information which is significant for operation of the item, including non-conformances, deviations and open tasks.*

2. The PI shall prepare and maintain system, subsystem and equipment logbooks (in accordance with annex B of [NR10]) for all operations and tests performed on the item during the period to be covered by the logbook.

3. Equipment logbooks shall start with the first qualification or acceptance test after assembly.

4. The log books shall accompany the hardware whenever it is placed under the custody of another organization. The log books will form part of the End Item Data Packages which are to be delivered for every item at the time of acceptance.

7.3.3 NON-CONFORMANCE CONTROL

1. A Non-Conformance Report (NCR) tracking system is required when a discrepancy is observed between a characteristic of deliverable hardware or software and the relevant specification, including drawings and test procedures. For Solar Orbiter, this term includes defects and failures, and requests to waive requirements.

2. The system shall provide for a disciplined approach to the identification and segregation of nonconforming items, the recording and reporting (Non-Conformance Report, NCR), review, disposition and analysis of non-conformances, and the definition and implementation of corrective actions.

The NCR template is provided in ECSS-Q-20-09.

3. Non-conformances shall be reviewed and dispositioned by a formal Non-Conformance Review Board (NRB). The originator's PA shall ensure that:

    - responsibilities and authorities for the disposition of non-conformances are properly defined
    - the NRB includes at least representatives from the PA and Engineering organizations
    - the Board to review non-conformances is chaired by the Product Assurance Management function;
    - all relevant Product Assurance experts are involved in the review, investigation and disposition of non-conformances;
    - all knowledge acquired from non-conformances results in preventive actions in all relevant engineering, manufacturing and Product Assurance fields.

7.3.3.1 Non-Conformance Classification

1. Non-conformances shall be classified as MAJOR which may have an impact on the next higher level requirements in the following areas:
    - safety of people or equipment;

- operational, functional or contractual requirements;
- reliability, maintainability, availability;
- lifetime;
- functional or dimensional interchangeability;
- interfaces with hardware and/or software of different contractual responsibility.

2. Additionally, any non-conformances shall be classified as major in the cases of:
    - changes to or deviations from approved qualification or acceptance test procedures;
    - project specific items which are proposed to be scrapped;
    - for EEE components, in case of:
    - lot/batch rejection during manufacturing, screening or testing at the manufacturer's facilities, if the purchaser proposes:
    - to use as-is the rejected lot/batch, or
    - to continue processing, rework or testing, although the lot/batch does not comply with the specified requirements
    - non-conformances detected after delivery from the manufacturer

3. **Minor** non-conformances are those which by definition cannot be classified as major. The following EEE discrepancies after delivery from the manufacturer may be classified as Minor:
    - random failures, where no risk for a lot-related reliability or quality problem exists;
    - if the form, fit or function are not affected;
    - minor inconsistencies in the accompanying documentation.

4. **In case of doubt, non-conformances shall be classified as major.**

5. The consequences of several different minor non-conformances on the same item shall be evaluated for proper classification.

7.3.3.2 Non-Conformance Reporting

1. The second party and the next higher contractual level (regarding ESA and/or its selected Prime and the PI as contractually linked) are to be informed of MAJOR NCRs within 48 hours of their discrepancy, notified of the date of a proposed NRB, progress towards closure and final closure.

2. The approval of the closure is required of the higher level. If the NCR is also MAJOR at that level other contractual levels must be included in the reporting, progress and closure process. Where NCRs affect ESA or the ESA selected Prime other users may be involved but the PI has the responsibility to inform these parties.

Any form may be used to document the NCR and it may be transmitted by Fax or E-mail providing the following information is recorded:

a. a unique reference number including the identifier of the originator's organization
b. date (and time if relevant) of the observation of the non-conformance
c. identification of the affected hardware/software
d. descriptions of the requirement including the paragraph and specification reference
e. description of the discrepancy
f. details if relevant of previous events and environment
g. name of the person describing the non-conformance
h. presumed or identified cause
i. proposed actions including date of the NRB
j. reference to any related NCRs
k. identification of any other hardware that may be affected

3. Subsequent reports should add pages numbered sequentially to previous reports, each page reflecting the original identification number. The final report should confirm that all actions are completed and that closure has been agreed by the relevant parties.

4. Reports between PIs, ESA, the Prime Contractor, Parts Agency and Spacecraft AIV and Test Houses shall be in the English language.

5. Minor NCRs must be reported to the next higher contractual level at least by means of a monthly report and shall be reviewed at the time of acceptance of hardware/software.

6. MAJOR NCRs will be treated at levels other than the originator, but, to avoid confusion, the originator's number must be preserved, perhaps with a suffix to identify the other level.

7.3.4 ALERT SYSTEM

ESA operates an Alert system to inform all affected ESA projects of technical problems of general nature concerning safety, parts, materials and processes (e.g. a serious deficiency discovered with the sealing of IC-packages by a specific manufacturer).

The notification of problems from any source will be screened by the Project Office for a first assessment of potential applicability to Solar Orbiter. If it is suspected or if it cannot be excluded that an instrument may be affected, the alert will be forwarded to the Principal Investigators with a request to evaluate the alert, to assess the relevance to the instrument and to take corrective actions as necessary to assure that the reported problem is avoided or eliminated on the instrument.

1. The PI shall assess incoming Alerts for applicability to the instrument and a response shall be provided to the ESA Project Office within 15 days after receipt of a formal Alert, either indicating its non-applicability or the appropriate actions (to be) taken.

7.3.5 HANDLING, STORAGE AND PRESERVATION

1. The PI shall prevent handling damage during all phases of manufacturing, assembly, integration, testing, storage, transportation and operation, by adequate:
    - protection of items during handling (e.g. red tags);
    - handling devices;
    - procedures and instructions (e.g. purging procedures).

7.3.6 QA REQUIREMENTS FOR PROCUREMENT

7.3.6.1 Selection of Procurement Sources

1. For the procurement of equipment, components, parts, materials and services the PI shall evaluate and select manufacturers, suppliers or contractors who have a demonstrated capability of supplying the items with the required properties and the necessary quality levels.

2. The demonstration of capabilities shall be based on the successful supply of items or services similar to those to be procured.

7.3.6.2 Incoming Inspections

1. Incoming Inspections shall be performed on procured items to check their compliance with applicable requirements.

2. The visual inspection for completeness and freedom from obvious damage or deficiencies which might result from transportation shall always be performed.

7.3.6.3 Procurement Requirements for EEE Parts

See section 6.6.5 on EEE parts.

7.3.7 QA REQUIREMENTS FOR MANUFACTURING & INTEGRATION

7.3.7.1 Manufacturing and Inspection Flow Chart

1. Before the beginning of the actual manufacturing the PI shall review the manufacturing readiness in front of the following aspects:
    - Status of product definition and requirements
    - Status of manufacturing, assembly, inspection and test documentation

- Validation status of manufacturing processes, with particular emphasis on critical processes.
- Availability of required production, measuring and inspection equipment, and calibration status, when relevant.
- Cleanliness of facilities, with respect to the required cleanliness levels

2. The manufacturing and assembly process shall be analyzed and the sequence of the various steps thoroughly planned.

3. Surveillance of manufacturing and assembly activities shall be performed by the designated quality assurance personnel, by means of inspections for:
   - critical parameters of the process;
   - satisfactory workmanship;
   - completion of individual manufacturing and assembly steps.

4. The planning of inspections shall take into account the complexity of the operations and their potential effect on the properties and integrity of the end product.

7.3.7.2 Key and Mandatory Inspection Points (KIP/MIP)

1. Among the inspections and test as part of the production sequence, some selected inspections shall be performed with participation of representatives from ESA.

2. A MIP shall require invitation at least one week before the event, and participation of ESA or its written agreement to proceed without ESA participation.

3. A KIP shall require the same invitation, but the notified activity may be performed as scheduled if there is no reaction from ESA.

4. The PI shall identify Key and Mandatory Inspection Points (KIP/MIP) in accordance with the following criteria:
   - when critical processes are performed
   - formal qualification and acceptance tests
   - when the manufacturing sequence is irreversible
   - when the manufacturing sequence makes the item difficult and costly to disassemble for inspection
   - when the manufacturing sequence or renders the location inaccessible for inspection

5. The PI shall propose a list of KIPs and MIPs to ESA together with the manufacturing and inspection flow chart at the ICDR and IPDR. The MIPs where is participation is required will be agreed with the PI.

7.3.8 INTEGRATION AND TESTING

7.3.8.1 Test Planning

1. An AIT planning shall be prepared as part of the DDV Plan, to cover all test requirements for development, qualification and acceptance test phases for the different models. Details shall be given of:
   - hardware configuration
   - test objectives
   - test parameters
   - test sequences (incl. initial and final test conditions)
   - acceptance/rejection criteria
   - test equipment (incl. test software) and accuracy required
   - test facilities involved
   - hazards
   - cleanliness of integration/test facilities

7.3.8.2 Test Procedures

1. Test procedures are required for all tests on deliverable hardware.

2. Test procedures shall be derived from the project requirements of the project AIT plan and shall completely and precisely define the methods and steps by which the tests specified by the relevant test requirements shall be carried out.

3. The test procedures shall include:
   - scope of the test, including the identification of the requirement being verified;
   - identification of the test object;
   - applicable documents, with their revision status;
   - test flow;
   - test organization
   - test conditions;
   - test equipment and set-up;
   - step-by-step procedure, including definition of specific steps to be witnessed by QA personnel
   - recording of data;
   - pass/fail criteria and test data evaluation requirements;
   - guidelines / criteria for deviation from test procedure and for retest (procedure deviation sheets).

Note: *Pass/Fail criteria shall be set allowing for test equipment accuracy and* measurement *uncertainty so that measured/indicated values can immediately be related to the required specification.*

4. All instrument level test procedures shall be submitted for review and approval by ESA for compliance with all related requirements 4 weeks prior to the Test Readiness Review and performance of the test concerned.

7.3.8.3 Test Facilities/Equipment

1. Test facilities required to conduct the test programme shall be specified in the AIT (test) plan and shall comply with the requirements of ECSS-Q-20-07. Their suitability confirmed well in advance of testing.

2. All test equipment including commercial test equipment shall be calibrated as required prior to use and shall remain within calibration during use.

3. Prior to unpacking and test of the equipment, the test facility shall have been set up in accordance with the applicable test procedure, and the facility cleared of all obstructions. The facility shall be inspected by QA who shall give approval for the commencement of tests.

4. During testing all measurements and tests shall be made in conditions in accordance with the cleanliness and contamination control requirements. Actual ambient test conditions shall be recorded periodically during the test period.

5. During tests, only persons associated with the test shall be permitted into the facility.

7.3.8.4 Test Witnessing

1. Critical development tests and formal qualification and acceptance tests shall be monitored or witnessed by QA personnel to ensure that applicable procedures are followed without errors, and that adequate records of the activities and test results are taken.

Note: *Test witnessing shall be considered when manual intervention is performed, at the setting-up, start and end of continuous fully auto-mated test sequences, or when no automatic recording of test parameters/results is available.*

2. The QA personnel shall document any variations to test procedures, deficiencies and non-conformances during the test, and monitoring the implementation of dispositions and corrective actions.

7.3.8.5 Test Reviews

1. A **Test Readiness Review** (TRR) shall be held prior to any formal instrument qualification and acceptance tests, to determine the following:

   - that the as-built configuration status of the test specimen conforms to the released design baseline or differences are acceptable and documented;

- status of existing non-conformances/failures, Requests for Waivers/, open work and assessment that open actions do not affect the test;
- availability and approval status of test procedures;
- verification that hazards and hazardous operations have been clearly identified within the test procedure and appropriate actions are implemented;
- readiness of test facility, personnel and associated equipment (cleanliness of test facility, calibration status and validity of all test equipment, including any software programme);
- identify recovery actions for the more probable contingencies in test (e.g. loss of pumping, cooling etc.)assignment of responsibilities during the test;
- conclusion whether to release for testing.

2. After major portions of qualification and acceptance tests (e.g. at end of EMC tests and at end of vibration tests), a **Post-Test Review** shall be held to determine that:

   - all portions and steps of the applicable procedure have been properly executed, and the test specimen and test equipment have been brought into a safe condition;
   - all deviations from or modifications to the initial test procedure which had to be made during the test were properly authorized;
   - all required data records are complete and at least a first assessment has been made to determine whether the parameters were within required limits, or whether there is a need for additional testing and/or further analysis of the results before a conclusion can be reached;
   - non-conformances/failures have been recorded and at least initial dispositions affecting continuation/completion of the test have been made by the appropriate Material;
   - conclusion, whether the test article can be released to the next step or the test set-up can be dismantled.

3. Test Review Boards shall include the following representatives of the PI: project management, AIT and product assurance.

4. ESA and its selected Prime shall be invited to attend instrument level Test-Readiness Reviews and Post Test Reviews, with a notification at least one week before the event.

7.3.8.6 Test Reports

1. A test report shall be provided for each test, including as a minimum:

   - a summary of test results;
   - an evaluation and verification of test results;
   - a list of Non-Conformance Reports raised during the test;
   - the as-run filled-in test procedure;
   - all test data including environmental test facility records (i.e. vibration plots, vacuum values, temperature and humidity figures, during tests);

- clean room environmental control data i.e. temperature, pressure and humidity, during qualification and acceptance tests.

7.3.9 QA REQUIREMENTS FOR ACCEPTANCE AND DELIVERY

1. The PI shall establish a formal acceptance process for all items delivered by collaborating institutes/organizations as well as from contracted industries.

2. The PI shall establish a formal acceptance process for all items delivered by his collaboration to ESA and its selected Prime.

3. The PI is responsible to organize a formal Delivery Review Board for instrument models to be delivered to ESA or its selected Prime.

The sole basis of this review is the End-Item Data Package (EIDP). The content of this EIDP is described in ***ECSS-Q-20B, annex C***.

4. The PI shall prepare and deliver the EIDP at least 10 days before the DRB takes place.

The DRB consisting of the PI representatives, ESA and selected Prime representatives will review the data package and agree on the consent to ship of the H/W.

It shall be remarked that a "consent to ship" is not automatically considered a formal acceptance. The formal acceptance of the instrument might be subject of closure of open actions, retests etc., in which case a delta DRB might be held.

In this respect the DRB is responsible for authorizing the shipment of the items under acceptance, and certifying by writing that:

a. The items conform to the contractual requirements and to an approved design configuration.
b. The items are free from material and workmanship deficiencies.
c. All non-conformances are closed-out, or corresponding plans, compatible with the delivery, are accepted.
d. The relevant EIDP is complete and accurate.

7.3.10 QA REQUIREMENTS FOR SUPPORT EQUIPMENT

Ground Support Equipment (GSE) is clarified as:
“Optical, mechanical, fluidic, electrical and software support equipment or systems used for calibration, measurements, testing, simulation, transportation, handling... of space segment or of space segment elements.”

1. For all above defined GSE items which will be mechanically or electrically connected to FM units the same acceptance requirements applies as for FM units.

7.4 Safety Assurance

7.4.1 GENERAL

1. The PI shall implement a safety assurance programme comprising:
   - the identification and control of all safety related risks with respect to the design, development and operations of the instrument
   - the assessment of the risks based on qualitative and quantitative analysis as appropriate
   - the application of a hazard reduction precedence and of control measures of the residual risks.

The hazard reduction process consists of the following sequence of activities, performed in sequence:

a) Hazard Elimination - Select design technology, architecture and operational characteristics to eliminate hazards and hazardous conditions from the design and operational concepts.
b) Hazard Minimization
   i) Select the least hazardous design architecture, technologies, and operational characteristics to minimize the severity of the associated hazardous events and consequences.
   ii) Reduce the probability of occurrence of the hazardous condition.
c) Hazard control
   i) Safety devices – Control hazards through the use of automatic safety devices as part of the system, subsystem or equipment. Safety inhibits shall be independent and verifiable.
   ii) Hazard control - Warning devices - Use devices for the timely detection of the condition and the generation of an appropriate warning signal. This shall be coupled with emergency controls of corrective action for operators to safe or shut down the affected subsystem.
   iii) Hazard control - Special procedures – Only when it is not possible to reduce the magnitude of a hazard through the design, the use of safety devices or the use of warning devices, special procedures shall be developed to control the hazardous conditions for the enhancement of safety. Special procedures may include emergency and contingency procedures, procedural constraints, or the application of a controlled maintenance programme.

Note: *The objective of safety requirements is to establish methods to be followed during the design, development, fabrication, assembly, integration, testing, transportation, ground operations, launch and orbital operations. These methods will ensure that the risk of hazardous consequences to personnel, flight hardware and facilities are minimized.*

2. The PI shall identify the responsibility in his team and a contact person for safety related aspects. Description and planning of safety related activities shall be included in the Product Assurance Plan.

7.4.2 REQUIREMENTS

The requirements for safety assurance are governed by the requirements imposed on ESA by the launcher authority, complemented by requirements imposed by ESA itself and those of the applicable national safety standards and regulations in the country of origin.

1. Therefore the design of the instrument and its associated GSE and their operation shall conform to TBD.

2. The consequences of identified hazardous events (according to FMECA) shall be categorized as follows:
   I. CATASTROPHIC
      - loss of life,
      - life threatening or permanently disabling injury or occupational illness;
      - loss of an element of an interfacing manned flight system;
      - long term detrimental environmental effects;

   II. CRITICAL
      - temporary disabling, but not life-threatening injury, or temporary occupational illness;
      - loss of major damage to flight systems, major flight system elements, or ground facilities;
      - loss of, or major damage to, public or private property; or
      - short term detrimental environmental effects.

3. All kind of hazardous events shall not propagate across the interfaces.

The ESA selected Prime will act on behalf of ESA as “Payload Authority” for the launcher Interface. It will assure that safety data resulting from the design and operation of an instrument will be integrated into the safety considerations for the system and vice versa the Prime will identify and control the detailed safety requirements to be met by the payload.

7.5 Dependability Assurance

7.5.1 GENERAL

This section is based on ECSS-Q-30B [NR12] and ECSS-Q-30-02A [NR11], which are tailored here to the instruments and their interfaces with other elements of the spacecraft.

Prime objectives of the reliability assurance activities are:

- to establish and list in a systematic way all possible modes of failure, in order to identify weak elements of the design for improvements, and to support the safety analyses by pinpointing potential hazards (FMECA, HSIA and SPF sections);
- to assist in the optimization of system reliability and redundancy concepts with comparative reliability assessments for alternative design options and trade off studies (see section on Numerical Reliability Assessment, 6.5.5);
- to prevent the propagation of failures to other instruments or to the spacecraft (see section on Worst Case Analysis, 6.5.6).

1. The PA Plan shall describe how compliance with the programme dependability requirements will be met and reliability assurance activities will be interrelated and coordinated with parallel engineering and safety activities.

2. The various steps for the initiation, update and finalization of the reliability analyses shall be identified in the PA plan.

3. Hardware or software failures shall not propagate to cause additional failures or the hazardous operation of interfacing hardware. The Contractor shall prove the capability of the design to sustain:
   - a single failure or operator error without critical consequences, and
   - any combination of two failures/ operator errors without catastrophic consequences.

4. Failure tolerance need not to be applied to: primary structures, load-carrying structures, structural fasteners, load-carrying elements of mechanisms, and pressure vessels. In these cases, the requirements of design for minimum risk shall be applied.

7.5.2 DEPENDABILITY ANALYSIS: FAILURE MODES, EFFECTS AND CRITICALITY ANALYSIS (FMECA)

1. A comprehensive Failure Modes, Effects and Criticality Analysis (FMECA) shall be performed on the functional and physical design (functional FMECA and design FMECA) of the entire instrument and any GSE interfacing physically or functionally with the instrument. In all cases the FMECA shall identify how each failure mode is detected.

2. The purpose of the FMECA shall be to identify all failure modes of the system and rank them in accordance with the severity of the effects of their occurrence. Furthermore, it shall be to:

   - determine the effects of each failure on the performance of the function under analysis;

- identify all single point failures, classify them according to the severity of their effects, and propose actions to eliminate them from the design;
- establish how the detection, diagnosis, correction, and verification of each failure can be unambiguously implemented.

3. FMECA shall be carried out in accordance with [NR11] clause 4.1 and 4.6, for all operational modes of the instrument during orbital phases, launch phases and also for ground testing, if not covered by analyses of the other phases.

Note: *The FMECA shall be performed on the basis of the lowest level of design definition which is available at the successive steps in the design and development process, e.g. initially starting with assumed failure modes of basic functions, later at assembly level and finally at instrument level as necessary to cover potentially critical effects. Later, for mechanisms from part level upwards; else from functional blocks without redundancy upwards. The logical sequence of the FMECA shall include the following steps:*

- *to identify the item under consideration and its function;*
- *to identify the assumed failure modes for that item or function;*
- *to analyze and describe the effect of the assumed failure mode on the function of the assembly under consideration and the effects on related and higher level assemblies and functions;*
- *to identify observable symptoms for the assumed failure mode or its effects (e.g. automatic function monitoring or house-keeping data and telemetry; in orbit or during test).*
- *to establish what provisions are inherent in the design:*
- *to compensate the effect of the malfunction (e.g. switching to redundant unit, automatically or by telecommand),*
- *to isolate the fault, or*
- *to switch to contingency operational modes;*
- *to identify the criticality category of the failure effect according to the definition given below and, specifically, whether the item is a Single Point Failure (SPF).*
- *provide remarks and recommendations if applicable or necessary or desirable modifications for the design or operations (e.g. elimination of SPFs).*

4. The following Failure Effect Severity Categories shall be used in the FMECA:

Catastrophic:

- Propagation of failure to other subsystems / assemblies / equipment
- Loss of functionality

Critical:

- Loss of functionality
- the failure effect is not confined to the instrument.

Major:

- Degradation of functionality

Negligible:

- Any other effect.

5. The following attributes shall be added to the criticality category as appropriate:
   -the suffix “S” shall be used to indicate safety impacts.
   -the suffix “R” shall be used to indicate redundancy

6. The PI shall submit an updated FMECA at each instrument design review.

7.5.3 HARDWARE/SOFTWARE INTERACTION ANALYSIS (HSIA)

1. A hardware / Software Interaction Analysis (HSIA) shall be performed in conjunction with the FMECA.

2. The HSIA shall systematically address the hardware / software interface of a design to ensure that hardware failure modes are being taken into account in the software requirements and design. Detailed requirements are provided in [NR11].

3. The HSIA shall be performed for flight H/W controlled by on-board S/W.

4. The HSIA shall be performed for safety critical / elements of the GSE as identified in the FMECA and Hazard Analysis controlled by S/W.

5. The HSIA shall be attached to the FMECA.

7.5.4 SINGLE POINT FAILURES

1. On the basis of the FMECA, the PI shall identify Single Point Failures (SPF) and take the necessary actions to eliminate or reduce them. All residual SPFs shall be identified in a SPF List in accordance with template in [NR11], to be a section of the FMECA, with a rational for retention.

2. This rationale shall include an engineering assessment of the likelihood of occurrence, a discussion of the measures, if any, that might be taken to eliminate the SPF, and special provisions to reduce the probability of occurrence or the potential failures effects.

3. The PI shall take the necessary action to eliminate Single Point Failures (SPF) related to interface critical elements.

4. Any remaining SPF shall be approved by ESA through the Request for Waiver procedure.

7.5.5 RELIABILITY PREDICTION

1. Reliability prediction techniques shall be used to optimize the reliability of a design against competing constraints such as cost and mass.
2. The failure rates and methods used in reliability predictions shall be as specified by the Prime.

7.5.6 WORST CASE ANALYSIS

1. The PI shall perform a Worst Case Analysis (WCA) in accordance with [NR11], in parallel to electronic design and development activities.
2. The WCA shall cover assemblies interfacing with other spacecraft elements to demonstrate that interface requirements (e.g. leakage current) are not violated, taking into account parameter variations of components resulting from initial tolerances, environmental effects (e.g. temperature), ageing, radiation doses, wear-out etc. over the operating life.
3. For electronic components the parameter variations defined in PSS-01-301 [NR13] shall be taken into account. Other values have to be substantiated with support from test data (e.g. end of long-term life test limits from qualification tests).
4. The replacement of sensitive parts or circuit redesign shall be considered if the WCA indicates a potential problem due to violation of de-rating requirements or marginal end-of life performance due to aging.
5. The adequacy of margins in the design of electronic circuits, thermal and electromechanical systems shall be demonstrated by analysis or test.
6. The analysis work shall start during the early design phase and reflect the current design status, and updated as necessary at least for the design reviews.

7.6 EEE Parts Selection and Control

7.6.1 GENERAL

Parts quality play an essential role for the overall chance of success of the spacecraft mission, and therefore their selection and control shall be paid high attention.

In the following, ECSS-Q-60 A [NR14] has been tailored for the definition of the component requirements to be applied for the instruments.

These requirements apply to flight standard hardware and to components coming into direct contact with flight standard hardware such as the interfacing connectors from GSE cables. For Engineering Models, components shall be used which are equivalent in form, fit, function and materials (e.g. if thermal vacuum tests would be done on EM) but particular quality assurance provisions are not needed.

1. The PI shall prepare a Component Control Plan as part of the Product Assurance Plan. This plan shall describe how the component programme will be carried out with identification of the tasks which will be carried out by the PI, or by procurement agents, test houses or consultants as applicable.

The terms "Parts" and "Component" are used here as synonymous.

7.6.2 COMPONENT PROGRAMME MANAGEMENT

Since a deficient identification of the needed components, the usually long delivery times, and the evaluation and tests can have serious impact on the overall schedule, the activities of the component procurement programme need to be planned thoroughly and progress must be closely monitored.

1. The PI shall define the responsibility for the component engineering and procurement activities within this team and he shall nominate a contact person for coordination with ESA.

2. The PI shall provide as part of the project management plan a EEE parts procurement plan, identifying possible long lead items and eventual re-qualification of parts requiring additional time and effort.

7.6.3 COMPONENT ENGINEERING

7.6.3.1 Prohibited Materials and Components

1. Components with the following characteristics shall be prohibited except where specifically agreed on case-by-case by ESA:
    - Limited life
    - Known instability
    - May cause a safety hazard
    - May create a reliability risk leading to loss of function.

2. Use of components with known instability shall be avoided unless specifically approved. Examples of unstable components are:

    - Plastic encapsulated semiconductors
    - Components containing the following materials:
    - Beryllium oxide
    - Cadmium
    - Lithium
    - Magnesium
    - Mercury
    - Radioactive material
    - Pure tin (electroplated or fused)
    - Hollow core resistors
    - Potentiometers
    - Non-metallurgically bonded diodes
    - Non-solid tantalum capacitors with silver case
    - Dice with no glassivation
    - Unpassivated power transistors
    - Wet slug tantalum capacitors (except for CLR79 construction using double seals and a tantalum case)
    - Any component whose internal structure uses metallurgic bonding with a melting temperature not compatible with the end-application mounting conditions
    - Wire link fuses

It must be noted that the requirements of this paragraph apply to the entire instrument, not only to critical interface circuits.

3. The supplier shall ensure that non-hermetically sealed materials of components meet the requirements of ECSS-Q-70 A [NR15] regarding outgassing, flammability, toxicity and/or other criteria required for the intended use.

7.6.3.2 Radiative Sensitive Components

1. The PI shall perform a radiation analysis to evaluate single-event and total-dose effects on instrument operation for all components used in flight hardware that are exposed to radiation environment, including those components used in COTS units.

The radiation environment to be considered for mission is described in the [IR2].

As a baseline, the component shall be radiation resistant to 100 Krad (TBC). However, the use of components which can withstand radiations lower than 100 Krad (TBC) but not less than 20 Krad (TBC) may be considered after a sector analyses, provided that sufficient shielding can be foreseen. ECSS-10-04 A [NR16] provides guidelines for good radiation design practices.

ESA Radiation Design Handbook [NR16] is a valuable source of radiation data. On components for which available data indicate sensitivity to the expected radiation environment, additional shielding and/or lot acceptance testing may have to include radiation testing to demonstrate that the batch of components (or wafers) intended for flight-application is acceptable.

If no radiation data are available on specific components, radiation testing will have to be performed for evaluation (characterization test of commercial equivalents and LAT – see 7.6.5.3 - for flight standard components).

ESA is prepared to provide advice as far as possible on the selection of radiation hard component types, or potential precautions or testing as may be necessary.

7.6.3.3 Component De-rating

In order to enhance the reliability during operation, the components shall not be stressed to the maximum rated values established by the manufacturer, but only to the de-rated values specified in [NR13].

Drift and degradation of performance parameters (e.g. increase of leakage current of diodes) as specified in [NR13] shall be taken into account in the design of electronic circuitry. If insufficient data are specified there, the end-of-life limits of qualification tests may be used.

The verification activities for these requirements are specified in section 7.5.6 (Worst Case Analysis).

7.6.4 COMPONENT SELECTION AND APPROVAL

7.6.4.1 Preferred Components

1. The European Preferred Parts List (EPPL) [NR17] and the ESA/SCC Qualified Parts List shall be used as the primary basis for component selection.

2. All components used in flight hardware shall comply with the following standards as a minimum (TBC):

Microcircuits	MIL-PRF-38535 class Q, or ESA SCC level C	
Transistors/ diodes/ optocouplers	MIL-PRF-19500 JANTXV, or ESA SCC level C	
Hybrids	ESA-PSS-01-608 level C or MIL-PRF-38534 class H	
Passives	ER-MIL failure rate P (exponential law) failure rate B (Weibull law) or ESA SCC level C	or CECC generic spec level B

Switches	MIL-STD-1132, or ESA SCC level B	
Crystals	ESA SCC 3501 level B	
Relays	MIL class S, or ESA SCC level B	

3. In non redundant units components meeting the following standards shall be used:

Microcircuits	MIL-PRF-38535 class V, or ESA SCC level B	
Transistors/ diodes/ optocouplers	MIL-PRF-19500 JANS, or ESA SCC level B	
Hybrids	ESA-PSS-01-608 level B or MIL-PRF-38534 class K	
Passives	ER-MIL failure rate R or S (exponential law) failure rate B (Weibull law) or ESA SCC level B	or CECC generic spec level B
Switches	MIL-STD-1132, or ESA SCC level B	
Crystals	ESA SCC 3501 level B	
Relays	MIL-R-39016, or ESA SCC level B	

7.6.4.2 Non PPL Listed Components

1. The selection of components which are not in the PPL shall be based on the knowledge of technical performance, qualification status or qualifiability, and history of previous usage in similar applications.

2. Preference shall be given to components from sources which would necessitate the least evaluation/qualification effort.

7.6.4.3 Component Approval

Components used in flight standard hardware of an instrument are subject to ESA approval.

Component types will be approved by ESA if at least one of the following criteria applies:

- they have been qualified according to the requirements of the applicable SCC specification or to equivalent requirements;
- they have successfully passed the component evaluation and approval programme as outlined in paragraph 7.6.4.4 below;
- they have received circuit type approval as outlined in ESAPSS-01-608 [NR18] (for hybrid integrated circuits).

Type approval will be given if equivalence to ESA/SCC qualification requirements can be demonstrated via existing data or by similarity to qualified components. This information shall be provided on or attached to the Part Approval Document. The actual qualification status of the selected manufacturer shall be checked prior to procurement.

Component approval includes approval of the manufacturer, the procurement specification (and amendments) with definition of all technical requirements, applicable screening and lot acceptance tests and the evaluation/qualification programme, if applicable. Copies of procurement specifications which are not readily available at ESA, shall be provided with the Part Approval Document.

Approval by ESA is given by the signature on the Part Approval Document (PAD) (see 7.6.4.5). An approval reference shall be entered on the DCL to maintain traceability of ongoing work.

7.6.4.4 Component Evaluation and Qualification

1. In case a valid and acceptable qualification cannot be demonstrated, a component evaluation and qualification test programme shall be implemented.

2. This programme shall cover the following elements:
   - Design assessment for the parameters of the component which are essential for the intended application and which justify the use of a non-preferred part.
   - Constructional analysis of the selected part (minimum three components) to assess the standards of fabrication and assembly, potential failure modes, materials and processes which may lead to deterioration or malfunction.
   - Manufacturer assessment to assure that the organization, facilities, production control and inspection system are adequate.
   - Evaluation and qualification tests corresponding to those defined in the ESA/SCC specifications for similar technologies.

Further details for an evaluation/qualification programme are outlined in [NR14].
Experienced consultants or procurement agents may have to be used by the PI to perform these tasks.

3. If applicable, the evaluation/qualification programme and the test results for a specific component to be qualified for use on Solar Orbiter shall be provided with the Parts Approval Document (see 7.6.4.5 below).

7.6.4.5 Application for Part Approval

1. A Part Approval Document (PAD) shall be prepared and submitted for approval for all parts intended to be used for the instruments, after performing type reduction as described in 7.6.4.6.

2. The PAD shall be in accordance with [NR14 Annex]. A minimum of 20 (TBC) working days shall be included in the schedules to allow for the ESA review of the PAD.

7.6.4.6 *Declared Components List (DCL)*

1. All components to be used on flight standard hardware shall be listed in a Declared Component List, which shall be completed stepwise as the selection of components and the approval process progresses.

This list will be used for comments and advice by components experts from ESA for type reduction or substitution and for evaluation of potential for a coordinated procurement for various instruments.

2. The DCL shall identify the instrument / instrument unit and the design status to which it is applicable.

3. The parts shall be grouped according to the families identified in the ESA PPL and the list shall be in accordance with [NR14].

4. The Investigator shall prepare and submit at the latest at the Instrument Baseline Design Review a first issue of the DCL, to be regarded as the first choice of components which is subject to further efforts on standardization and coordination.

5. The final version shall be available at the time of the Instrument Critical Design Review.

7.6.5 PROCUREMENT REQUIREMENTS

7.6.5.1 *Procurement Specifications*

1. Each type of component used by the PI shall be controlled by a procurement specification, or series of specifications, which must be approved by ESA.

2. The PI shall make maximum use of approved specifications issued under existing European component specification systems, either CECC or ESA/SCC as appropriate.

7.6.5.2 *Component Screening and Burn-In*

1. All components to be incorporated into flight-standard hardware shall be subjected to screening test.

2. The screening test requirements shall be so designed that the accumulated stresses will not jeopardize the component reliability.

3. The following ESA/SCC test levels for the screening of components for the full instrument shall be applied:

- Level B: for active components and critical passive components like crystals, filters, cermet-fuses, relays and switches;
- Level C: for other passive components not listed above.

SSC Testing levels:

- testing level 1: applicable for critical flight-standard hardware
- testing level 2: applicable for maintainable, non-critical flight hardware or single instruments

For components procured outside Europe or not in accordance with ESA SCC, screening levels (also called testing or quality levels) equivalent to those specified above may have to be considered.

Alternative acceptable levels are:

- JAN S, for active components;
- MIL failure rate R or S for passive components.

In any case lot traceability shall be assured by the component manufacturer, starting from the wafer to the final product.

All screening test shall be performed at the component manufacturer's premises or at an approved source.

7.6.5.3 Lot Acceptance Test (LAT)

1. It shall be ensured that all components shall be subjected to Lot Acceptance Testing (LAT) as defined in the ESA/SCC specifications or QCI (Quality Conformance Inspection) as defined in the United States Military specifications. The levels shall be as defined below:

   - Level LAT1 or QCI compatible: the component is neither ESA/SCC nor United States Military qualified at the time of the procurement and level LAT2 is not applicable.
   - Level LAT2 or QCI compatible: the component is not space qualified but has successfully supported other long life and/or high reliability space programmes and the reliability/evaluation data are still valid for the current design.
   - Level LAT3 or QCI compatible: all cases not included in level LAT1 or LAT2. Level LAT3 tests may be replaced by incoming inspection. Level LAT3 tests may be omitted for qualified ranges of components (e.g. 54HC, ...).

7.6.5.4 Hybrid Circuits

1. Hermetic hybrid circuits, if used, shall be procured in accordance with ESA PSS-01-608 [NR18], to be complemented by a detail specification, from sources which are "capability approved" for all relevant technologies, as per ESA PSS-01-606 [NR23] for thick film, and per ESA PSS-01-605 [NR24] for thin film.

2. In case hybrid circuits are required from a source which is not yet approved, an evaluation and acceptance testing programme shall be performed as defined in [NR24] or [NR23].

3. All add-on components shall be selected as defined herein and shall meet the requirements of [NR18].

4. For each hybrid circuit a PAD shall be established, including all add-on components, and submitted to ESA for approval.

7.6.6 COMPONENT QUALITY ASSURANCE

TBW

7.6.7 OFF-THE-SHELF EQUIPMENT

1. Any Off-The-Shelf (OTS) equipments that the PI is expecting to use shall be pre-agreed with the ESA Project office.

2. The PI shall review the components used in OTS equipment to verify compliance with the requirements of this document. The review shall consider the used parts' list, radiation hardness, the derating rules, Worst Case Analysis and the equipment design. COTS components shall be treated as non standard parts.

Special requirements may be imposed on OTS equipment (TBC)

7.7 Materials and Process Selection and Control

7.7.1 GENERAL

In the following, [NR15] has been tailored and summarized here for the definition of the materials, mechanical parts and processes requirements to be applied for selected payload instruments.

7.7.2 MATERIALS AND PROCESS SELECTION AND APPROVAL

1. The PI shall be responsible for the selection of materials and processes, and for demonstrating their suitability for the intended application.

2. To this end, the PI shall plan and enforce an effective material control and standardization programme. Materials and processes shall be selected in accordance with the criteria summarized in TBW; full details are given in ECSS-Q-70 A [NR15].

3. Materials, mechanical parts and processes shall be approved by ESA before they can be used for the production of flight standard hardware as outlined below; detailed instructions are provided in [NR15].

4. The PI shall submit to ESA for approval a Declared Material List (DML) (see [NR15]), a Declared Mechanical Parts List (DMPL) (see [NR15]) and a Declared Process List (DPL) (see [NR15]). Materials and process used by Co-Investigators and/or contractors shall be consolidated in the lists produced by the PI.

5. For materials or mechanical parts with limited or no test data available, the PI shall submit a Material or Mechanical Part Request for Approval in accordance with [NR15], proposing an evaluation programme. ESA will provide advice on and approve the evaluation programme and its results. ESA may request material samples for additional evaluation and comparative testing. These samples shall be provided with a Material Identification Card in accordance with [NR15].

6. Critical processes shall be identified in the DPL and PIs shall submit Process Requests for Approval (see [NR15]) together with the DPL. Critical processes are those which can have an effect on the structural integrity of the instrument are novel or the quality of which cannot be assessed solely by examining the end product (e.g., bonding, potting, painting and soldering).

7. The PI shall identify in these documents all materials involved (also cleaning agents), all processing steps, in-process and final inspections and tests, and the proposed test programme for process evaluation. At the end of the evaluation programme, a report shall be submitted to ESA upon which the approval of the process will be based.

8. A first issue of the DML, DMPL and DPL shall be submitted in the conceptual design phase for ESA comments and guidance for replacement of unacceptable materials and processes with suitable ones.

9. DML, DMPL and DPL shall be updated to reflect the degree of definition of the design in the following phases of the programme and revisions shall be provided for each of the project design reviews.

10. The review/approval activities and all necessary evaluation / qualification programmes for materials and processes shall be scheduled such that they will be finalized at the Instrument Baseline Design Review (IBDR) (start of manufacturing of qualification/flight hardware).

7.7.3 MATERIALS CONTROL

1. Each type of material to be used shall be covered by a procurement specification or standard. The contractor is encouraged to use existing international and national standards

at the maximum extent, in order to expedite the approval process of the DML. When developed by the PI, procurement specifications shall be made available upon request to ESA for review; proprietary rights will be respected.

2. Receiving inspections, lot/batch testing and in-process inspections/test shall be carried out to the degree necessary to ensure that variable and significant characteristics of the materials are within required limits.

3. Lot/batch acceptance test reports shall be kept at the investigator's or contractor's plant for at least 10 years together with other historical manufacturing/production records for the assemblies.

4. Non-conformances on materials and processes shall be recorded and treated as specified in section 7.3.4 of this document.

5. The PI shall be responsible for the performance of all inspections and tests necessary for evaluation, qualification and production surveillance.

ESA reserves the right to require samples of raw or processed materials for evaluation and testing in its own or other laboratories.

6. Mechanical parts (for bolts/nuts at least for size M4 and larger) shall be covered by procurement specifications including all technical requirements and adequate quality assurance provisions.

7. Materials, semi-finished products, and parts shall be procured from sources which can demonstrate previous deliveries of products with the required characteristics and quality or which have been formally qualified.

8. The name of the source/manufacturer shall be entered in the DML together with the name of a back-up source for critical procurement.

Printed Circuit Boards should preferably be procured from ESA qualified sources.

9. Material design allowables used in mechanical stress analyses shall correspond to "A values" as defined in MIL-HDBK-5 [NR25] or equivalent documents. Strength values for mechanical parts shall not be assumed to be higher than the values specified for the relevant qualification and acceptance tests.

7.7.4 PROCESS CONTROL

1. Each process used by the PI and listed in the DPL shall be covered by a process specification or standard.

The PI is encouraged to make maximum use of existing ESA specifications or ESA approved specifications/standards produced by international organizations and national agencies, because they reflect a consolidated experience and in order to expedite the approval process of the DPL. The complete list of approved documents and standards is contained in [NR26] and [NR27].

2. When developed by the PI, process specifications / procedures shall include sufficient in-process and final inspections and controls to ensure that characteristics of the product are within the required limits. Process procedures shall be made available or accessible to ESA upon request for review.

7.8 Software Product Assurance

7.8.1 GENERAL

1. An effective Software Product Assurance (SPA) programme shall be implemented. It shall ensure that:
   - software design requirements are properly specified;
   - formal definition documents are issued;
   - standards, practices and conventions are applied (e.g. logic structure, coding, commentary);
   - design and development activities are subjected to formal reviews;
   - all testing carried out to formal test procedures;
   - configuration management control procedures are applied.

2. The SPA requirements shall be applicable to:
   - flight S/W (application and operating S/W);
   - GSE S/W.

7.8.2 SOFTWARE PRODUCT ASSURANCE ACTIVITIES

1. The following fundamental tasks of SPA activities shall be performed:
   - establishment of standards and quality assurance procedures.
     Examples of ESA software engineering standards are listed in ECSS-Q-80 [NR28]; in-house software standards shall be approved by ESA.
   - participation in writing coherent development, analysis, production and test plans for PA related issues;
   - participation in reviews, audits and meetings;
   - ensuring adherence to standards and procedures;
   - liaison with configuration management;
   - involvement in problem reporting and resolution;
   - control of supplies/contractors;

- validation and acceptance test follow-up including non-conformance control

Note*: The SPA can be part of the overall Product Assurance Plan. As such the verification of this requirement can be assessed in combination of that Plan.*

7.8.3 SOFTWARE PRODUCT REVIEWS AND INSPECTIONS

1. The Software Development shall be done in accordance with ECSS-E-40 A [NR20]. During the software development process the following key reviews and inspections shall be performed:
   - System Requirements Review (SRR);
   - Preliminary Design Review (PDR);
   - Critical Design Review (CDR);
   - Qualification Review (QR);
   - Acceptance Review (AR);
   - Operations Readiness Review (ORR)
   - Software Inspection on Source Listing;
   - Review of Test procedures and test plans;
   - Witnessing of tests;

2. The traceability shall be ensured during all development and test phases from requirements via intermediate steps, down to code.

Formal acceptance release is mandatory for each step.

7.8.4 HARDWARE/SOFTWARE INTERACTION ANALYSIS (HSIA)

This subject is covered by section 7.5.3.

7.8.5 SOFTWARE CONFIGURATION MANAGEMENT

1. Software and software related documents shall be placed under configuration control not later than the start of integration of the individual software modules.

2. Configuration management and change control activities shall be performed in accordance to the configuration management requirements (see section 8.6).

7.8.6 SOFTWARE PROBLEM REPORTING

1. Software non-conformance shall be treated as defined in the Non-Conformance Control section, but the terminology used may be different.

7.9 Cleanliness and Contamination Control

1. The PI shall define in the PA plan the criteria and tasks for the contamination control, taking into account the guidelines provided in ECSS-Q-70-01A [NR29].

2. After establishing the cleanliness requirements for his instrument, the PI shall identify the provisions, activities and verification methods necessary to achieve the cleanliness levels through all stages of fabrication, handling, transportation and testing. Also the precautions and provisions to be taken during the integration, transportation and launch preparations of the spacecraft shall be defined, and the ESA selected Prime shall be notified accordingly so that the necessary arrangements can be made in due time.

3. The following potential contamination sources shall be considered:
   - choice of materials;
   - lack of degreasing of raw materials;
   - residues from cleaning agents, fluxes or machine lubricant;
   - insufficient curing and bake-out of materials;
   - handling of flight hardware with bare hands or dirty tools;
   - inadequate clean room clothing or discipline of personnel in clean rooms;
   - condensation of moisture or contaminants on cold surfaces during tests or transportation;
   - suitability and cleanliness of packing and packaging materials.

4. Appropriate provisions for their control shall be defined for facilities and procedures, and their implementation shall be verified.

5. During the design of the instrument it must be kept in mind that the environment encountered during the integration phase and launch preparations of the spacecraft (usually class 100 000) is not of the same high cleanliness standard which can be achieved in a laboratory where sensitive equipment is assembled. Therefore, protection devices shall be incorporated in the design, and also provisions for cleaning sensitive areas at later integration phases shall be identified, if necessary.

6. Bake-out in vacuum at elevated temperatures of contamination sensitive items before integration into the instrument shall be considered as an effective method to reduce the molecular contamination accumulated.

8 MANAGEMENT REQUIREMENTS

8.1 Introduction

The organization and management structure of the scientific team is an important element in assuring a timely success in the development and flight of a scientific instrument. Clear roles must be defined and respected in order to ensure proper information flow between the many parties involved in the instrument development, the spacecraft contractors, the scientific community and the ESA project office. The requirements of this section will be the unique reference on the responsibilities and methods of resolving problems and disputes between the PI group and any of the other stake holders in the instrument, spacecraft and mission development.

The implementation of the Solar Orbiter programme has to meet the various and multidisciplinary scientific objectives within the given financial envelope. The managerial complexity and the timely availability of payloads will significantly contribute to the overall programmatic risk. It is therefore essential that the PI is conscious of the risks and contributes to their minimization by adhering to the programme requirements established in this section.

The following sections will address:

- in section 8.2 the organization and responsibilities of each key participant in the programme
- in section 8.3 the communication rules within the project
- in section 8.4 the project phases and the herewith related progress monitoring
- in section 8.5 the project reviews and meetings
- in section 8.6 the configuration management
- in section 8.7 the deliverable items
- in section 8.8 the overall schedule

8.2 Organization and Responsibilities

8.2.1 ESA SOLAR ORBITER PROJECT OFFICE

The management of the Solar Orbiter mission will be under the responsibility of the ESA Project Manager located at ESTEC, Noordwijk, The Netherlands. The ESA Project Manager has full responsibility for all aspects of the development, launch and initial operations of the mission.
If, in the interest of the overall programme, significant technical and/or programmatic changes to an experiment are necessary, then ESA shall be responsible for the definition of the required change to be implemented by the PI.

The ESA Project Manager will be directly supported in the execution of the programme by the staff of the ESA Project Office located at ESTEC, organized into five main responsibility areas (TBC) namely, System, Spacecraft, Payload, AIV and Product Assurance.

The Solar Orbiter Instruments are managed by Payload engineers under the overall responsibility of the Solar Orbiter Project Manager. The Payload engineers will deal with the day to day Instrument activities and follow up on a regular basis the progress to ensure that they meet the Solar Orbiter programme objectives.

The Payload engineers will in particular:

- Coordinate with the Principal Investigator's team the day to day Instrument related matters;
- Control the technical interfaces defined in the EID, including the assessment, finalization and approval of change requests;
- Oversee acceptance tests of the Instrument deliverable items as part of the delivery procedure to the Industrial consortium;
- Supervise and coordinate with the Principal Investigator the support and inputs required for the spacecraft system test activities, the launch campaign and the operations in flight.
- Coordinate with the PI and the industrial Prime Contractor all deliverables needed by either the PI or the Prime Contractor in relation to the accommodation of the instruments in the spacecraft.

The ESA Project Office will fulfill its function until the completion of the spacecraft in-flight commissioning phase. An ESA Mission Operations Manager will be appointed, who will be responsible for the conduct of the mission operations from the end of the commissioning phase until the end of the mission.

The project manager is also supported by administrative and project control functions.

8.2.2 PROJECT SCIENTIST

The ESA Project Scientist is responsible for ensuring the scientific objectives of the mission are achieved, through the verification of instrument performance and science operation planning. As such she/he is the formal interface for all scientific matters.

The ESA Project Scientist will organize regular Science Working Team (SWT) meetings in support of the above objectives.

The ESA Project Scientist will monitor the state of the implementation and readiness of the instrument operations and scientific data processing infrastructure.

After the in-orbit commissioning phase, the Project Scientist is specifically responsible for:

- Coordinating the scientific operations according to the policy and guidelines established by the SWT;

- Supporting the short term planning of payload operations in conjunction with relevant PIs including resolution of conflicting payload operations requests and contingency support;
- The Project Scientist will coordinate the creation of the scientific products, their archiving and distribution to the scientific community;
- Coordinating the scientific operations with the Mission Operations Manager under terms and conditions defined by the Solar Orbiter Project Manager.

8.2.3 PI RESPONSIBILITIES

It is overall responsibility of the Principal Investigator to ensure that the complete Instrument is financed, developed and implemented within the mission and schedule constraints of the approved Solar Orbiter Programme. The full financial coverage for this activity shall be committed by the Funding Agency of the Lead Nation. Within this frame she/he is in particular responsible for the overall instrument management aspects described herewith below.

The PI shall take full responsibility for the instrument programme and retain at all times full authority within the Principal Investigator Team over all aspects related to procurement and execution of the programme. In this context, the PI shall be able to make commitments and make decisions on behalf of all other participants in the instrument programme. He shall organize all efforts, assign tasks and guide other members of the instrument consortium

8.2.3.1 Instrument Management

The organization of the Principal Investigator is the responsibility of the Principal Investigator. Specific requirements of the instrument organization are:

1. A single Principal Investigator (PI) shall be defined for each instrument.

2. The PI shall be responsible for:
    a. Sole managerial and decision making authority interfacing with the ESA Solar Orbiter Project Office.
    b. Appointing an instrument development manager to manage the day to day activities of the instrument development team.
    c. Provision of financial control in order to assure necessary resources to achieve the agreed delivery dates of all deliverables including technical data and instrument models.
    d. Providing instrument support to system level anomaly investigations, tests, reviews, operations and scientific activities arranged by ESA.
    e. Creating and maintaining a EID Part B which details the instrument design and interfaces answering requirements in the EID part A.

f. Ensuring compliance with all ITAR regulations in a timely manner. Surveillance requirements arising from ITAR regulations shall be reported to ESA and any costs associated with such requirements shall be borne by the PI.
g. Support and attendance to Science Working Team meetings as called by the ESA Project Scientist. As far as scientific requirements are concerned the PIs are committed to the Science Working Team to whom the Science Performance Report is submitted on regular basis (at every project review).

3. The PI shall produce a Management Plan covering the proposed investigation for the entire duration of the mission and shall include the following:
   a. The contribution of each institution must be clearly indicated and the responsibilities of each participant described in detail including deliverables and dates.
   b. Organigrammes containing the names of all partners: PI, CO-I's, Instrument Development Manager, and all key personnel. The PI will show, in particular, how he/she will participate in the overall activities.
   c. For all personnel the qualifications and experience of the team must be clearly indicated along with the fraction of time to be spent on the project.

4. The PI shall comply with the scientific data policy of the Agency as defined in the Science Management Plan.
   a. Provision of inputs for the definition and implementation of the science operations planning, and data handling and archiving concepts;
   b. The level and nature of the support of the definition and implementation of the Solar Orbiter ESA scientific data archive, as part of the pre-launch tasks.

5. The PI shall ensure the timely delivery of all deliverable items according to scheduled dates defined in section 8.7.

6. The technical interface of the experiment to the Industrial Prime contractor shall be supported.

7. The PI shall participate in technical working groups and control boards as requested by the ESA Project Office (e.g. environmental control board).

8. The PI shall support ESA management requirements (e.g. investigation progress reviews, programme reviews, change procedures, product assurance, etc.), as outlined in the EID-A.

8.2.3.2 Science Management

1. The PI shall monitor the compliance of the instrument design to the scientific requirements outlined in the Solar Orbiter Science Requirements Document [IR5].

2. The PI shall attend meetings of the Science Working Team and Groups, as appropriate; report on instrument development, and take a full and active part in their work.

3. The PI shall ensure adequate calibration of all parts of the instrument, both on the ground and in space. This includes the provision of all required calibration data to the Science Operations Centre along with a full instrument technical and science user manual for use by the general science user community.

4. The PI shall participate in the definition of the science operations and data handling, shall participate in the definition of the payload flight operations timeline and shall support the Science Operation Centre.

5. The PI shall exploit the scientific results of the mission and assure their diffusion as widely as possible.

6. The PI shall provide the scientific data (raw data, calibrated data, and higher level data), including relevant calibration products, to the Solar Orbiter archive in a format that will be agreed with the ESA SOC for application by the general science community.

8.2.3.3 Hardware Procurement

1. The PI shall define the functional requirements of the instrument and auxiliary equipment (e.g. MGSE, EGSE, CGSE, etc.) at instrument and spacecraft system level.

2. The PI shall ensure the development, construction, testing and delivery of the instrument. This shall be performed in accordance with the standards, technical and programmatic requirements defined in the Experiment Interface Document (EID).

3. The PI shall ensure that the instrument is appropriate to the objectives and lifetime of the mission, and to the environmental and interface constraints under which it must operate.

4. The PI shall ensure that all procured hardware is compliant with ESA requirements as defined in the EID.

5. The PI shall deliver a Flight Model and Flight Spares in accordance with the technical requirements defined in the EID-A, together with the relevant Ground Support Equipment.

6. The PI shall provide the necessary equipment to process their data as agreed with ESA and specified in the EID-A.

7. The PI shall ensure that all procured hardware is compliant with ESA requirements, through participation in technical working groups and control (e.g. cleanliness) boards, as requested, and that the hardware allows system level performance compatibility to be maintained.

8.2.3.4 Software Development

1. The PI shall ensure the development, testing and documenting of all software necessary for the control, monitoring, testing, operation and data reduction/analysis of the instrument, in accordance with the rules and guidelines established in the EID-A.

2. The PI shall ensure the delivery of such instrument specific software and its documentation to ESA, or elsewhere, in accordance with approved ESA guidelines, procedures and schedules. This includes the provision of software required in the SOC as agreed in section 7.6., including the support during software integration (where applicable) and operations support at the SOC.

3. The PI shall ensure the development, testing, documentation and delivery of the software required during instrument system level tests in real time or off-line mode including auxiliary software (instrument EGSE and interfaces) as defined in the EID.

4. The PI shall maintain and update all derived software and its documentation for the duration of the mission including a post operations (archiving) phase.

8.2.3.5 Verification

1. The PI shall establish and conduct a full instrument level AIV program and provide support to the system level AIV program in accordance with the requirements in section 6, Verification Requirements.

2. The PI shall support the system level integration and test activities related to and involving the instrument. This includes the System Validation Tests, involving the spacecraft and the ground segment.

3. The PI shall deliver adequate verification models of the instrument to the prime contractor, as required to verify system interfaces. The envelope of this delivery is ruled by the EID-A, in accordance with technical programme needs.

8.2.3.6 Product Assurance Coverage

1. The PI shall provide product assurance functions in compliance with the requirements of the EID-A section 7. This includes the safety requirements as defined by the relevant launch authorities.

8.2.3.7 Operations

1. The PI shall support all operational phases by providing the necessary manpower and/or expertise (training) to the ESA Project Team. The level of support shall be defined and agreed with the ESA Project Office.

2. The PI shall provide support for preparation and implementation of the mission and science operations up to the end of the mission, in accordance to the EID-A requirements, including:
   - Delivery of a full instrument technical and science user manual for use by the MOC and by the general science user community
   - Inputs to the data-base
   - Inputs to the Flight Operations Plan
   - Support the instrument flight operations, e.g. switch-on, commissioning, diagnostics, etc.

3. The PI shall support operations through team expertise including resolution of anomalies and malfunctions of the instrument.

4. The PI shall provide expertise support during critical mission phases at the MOC/SOC.

8.2.3.8 Data Processing and Dissemination

1. The PI shall support the implementation of data processing, analysis and reporting according to plans established in collaboration with SOC.

2. The PI shall provide the data analysis facilities (hardware and software) and manpower that is needed for achievement of the coordinated research within the Solar Orbiter programme until end of mission.

3. The PI shall deliver preliminary calibrated data of the experiment in due time for payload operations and planning as agreed by the SWT.

4. The PI shall participate in and provide required calibrated data for scientific workshops in order to facilitate collaborative studies required to meet the scientific objectives of Solar Orbiter.

5. The PI shall provide due acknowledgement to ESA in all published material

6. The PI shall collect the preliminary mission data products from SOC and deliver back final data products for further dissemination and archiving.

8.2.3.9 Financial Responsibilities

The financial status of will have to be guaranteed by the relevant national Funding Agency of the Lead Nation. The relevant Funding Agency will be considered responsible vis-à-vis ESA for all what concerns financial matters related to the selected investigations. The Funding Agency of the Lead Nation shall be responsible for the funding arrangements of the complete instrument. It shall ensure that adequate funding is available at the required time for all aspects of the instrument and its support. A funding margin should be provided, not only to cater for experiment evolution, but also to finance changes deemed necessary by the ESA Project Office.

Co-I teams are required via their national funding agencies to seek agreement with the Lead Funding Agency, which retains full responsibility for the IFE development and is the sole contact with ESA with respect to the Letter of Commitment

8.2.3.10 Communications and Public Relations

1. The PI shall adequately support ESA science communications and public relations activities.

2. The PI shall provide data and scientific results to ESA in a timely manner and in a form suitable for public relations purposes.

8.2.4 SCIENCE WORKING TEAM

The Solar Orbiter Science Working Team SWT will advise the ESA Project Office on all the scientific aspects related to the Solar Orbiter mission. The SWT will establish guidelines for the science operation and determine the long and short term planning. The tasks and responsibilities of the SWT are defined in more detail in the Solar Orbiter Science Management Plan.

The SWT shall be supported as a minimum by the participation of the PIs and the instrument Technical managers. Participation of the experiment Co-Investigators, team members, inter-discipline scientists and Industry are left to the decision of the PI and the ESA Project Office.

8.2.5 PRIME CONTRACTOR'S RESPONSIBILITIES

During the Formulation Phase and Definition Phase, when there will be two competitive contractors engaged in studying instrument accommodation and technology developments all contacts will be made via the ESA project office concerning questions of interface. This method of communication is to ensure that where confidentiality is needed that it will be maintained.

Once the Prime Contractor is chosen, after an ITT, they will be responsible for the management and maintenance of the EID-B during the implementation phase. As such, they will be in direct contact with the PI for the instrument interface management (TBC).

In the Implementation phase ESA will retain the overall responsibility and supervision.

8.3 Communications within the Programme

An effective information exchange within the Principal Investigator as well as with the stake holders in the mission is necessary to ensure all parties are working to the same baseline assumptions.

8.3.1 FORMAL COMMUNICATIONS WITH PROJECT OFFICE

All **formal** communication concerning technical and programmatic aspects shall be made between the Principal Investigator and the ESA Project Manager. No other party shall have formal authority, without written delegation.

Note: *Formal communication is defined to be a communication with a registration number in the configuration control system, independently of the medium used to transfer it (mail, fax, e-mail).*

Any formal communication interchanged between PI and PS or other ESA entity shall be copied to the ESA Project office.

1. The Principal Investigator shall provide an interface to allow electronic transfer of data (documentation, progress reports including schedule information, changes, technical data, etc.) between the Principal Investigator, the ESA selected Prime and the ESA Project Office compatible with the Agency's scientific project infrastructure.

Note: *Details to be elaborated at later stage.*

8.4 Project Phasing and Planning

8.4.1 OVERALL PROGRAMME PLANNING

- Instrument AO — October 2007
- Programme submission to ESA's Cosmic Vision Down Select — Fall 2009
- Announcement of Cosmic Vision Selection / Start of 20-month Definition Phase — January 2010
- Confirmation of Cosmic Vision M1 Missions — October 2011
- RFQ for Phase B2/C/D — November 2011
- Initiation of Phase B2/C/D — January 2012
- Launch — January 2017

Further programme milestones will be established in the course of Phase B1.

8.4.2 BASELINE INSTRUMENT MASTER SCHEDULE

1. The PI shall establish and submit to the ESA Project Office and its selected Prime a **Baseline Master Schedule** in line with the Solar Orbiter Project Schedule (see section 8.8) covering all the instrument programme activities identified in the Work Breakdown Structure.

2. Directly from the Baseline Master Schedule, a set of bar charts shall be created, covering:
   - Overall instrument programme
   - Individual instrument models
   - Instrument model integration and testing
   - Detailed bar chart of critical activities

3. Changes to the Baseline Master Schedule shall only be made with the approval of the ESA Project Office.

4. The resources and fraction of time available for all personnel shall be given throughout the instrument development cycle and the following mission phases:

   - Instrument Development Phase
   - Science Operations Phase
   - Archival phase

8.4.3 PROGRESS CONTROL AND REPORTING

8.4.3.1 General

The **technical and programmatic** aspects of each instrument programme will be assessed between the ESA Project Office and each Principal Investigator through:

- regular progress reporting,
- instrument progress meetings,
- a cycle of formal Instrument Reviews.

The **overall scientific performance** will be monitored by the ESA Project Office during the review cycle and through the regular progress reporting supplied by the PI. **Detailed scientific aspects** will be reviewed within the context of the Solar Orbiter Science Working Team, as defined in the Solar Orbiter Science Management Plan.

8.4.3.2 Reporting

1. The Principal Investigator shall submit 5 days after the end of the month (TBC), a **Monthly Progress Report** in which the current status of each activity is described and problem areas

or potential problem areas are highlighted together with identification of proposed remedial action.

2. The Monthly Progress Report shall include the following topics:
    - Overall summary, covering scientific and technical performance, status of design changes and open ECR's, overall progress status,
    - Design Development and Verification status, covering status of design definition and verification of interfaces, test and calibration, GSE, operations,
    - PA status, including NCR and RFW status,
    - Programmatic status, including schedule and milestone reports,
    - Science Performance status,
    - Problem areas and corrective actions.

3. The Monthly Progress Report shall be concise and submitted in the format of TBD (to be provided later).

8.4.3.3 Schedule and Monitoring Reporting

1. For each milestone, the PI shall maintain a record of the baseline achievement date, the forecast achievement date and the actual date achieved.

2. In order to track the progress, the PI shall provide to the ESA Project Office a monthly schedule report as part of the reporting procedure as described above in section 8.4.3.2.

During the manufacture and test phases the frequency of schedule reports may be increased should the Agency judge progress to be critical.

8.4.4 PROJECT BREAKDOWN STRUCTURES

In order to clearly identify the instrument, the scope of the work and the responsibilities involved, the following structures will be created and maintained by the Principal Investigator:
- the Product Tree (PT) to break down the instrument into its components, both hardware and software,
- the Work Breakdown Structure (WBS) to define the scope of the work and the responsibilities involved.

8.4.4.1 Product Tree

1. A Product Tree shall be developed by the PI, depicting a product oriented breakdown of the instrument into successive levels of detail.

2. The Product Tree shall be submitted to the ESA Project Office and shall be maintained under configuration control.

8.4.4.2 Work Breakdown Structure

1. A Work Breakdown Structure shall be developed by the PI, based on its agreed Product Tree and extending the applicable elements to include appropriate development models and support functions necessary to produce all the deliverables.

2. For each Work Package, the PI shall complete a Work Package Description (WPD).

3. The PI shall ensure all the responsibilities assigned to manage or to perform in all the Work

4. Packages are identified in the Principal Investigator organization chart (see section 8.2.3).

5. The WBS shall be submitted to the ESA Project Office and maintained up to date throughout the project.

8.5 Meetings and Reviews

Meetings and Reviews of instruments development as well as reviews at system level are a normal part of the procurement process for space equipment.

1. The PI shall organize regular progress meetings with the Solar Orbiter Project including instrument members at least quarterly or as required.

2. Ad-hoc meetings shall be supported when requested by the ESA Solar Orbiter Project Office to address critical subjects at the time.

3. The PI shall provide the resources to prepare review data packages as defined in TBD and support fully the review processes at instrument, ground segment and mission level as defined hereafter:

    a. Instrument Level
        i. Instrument Science Requirements Review (TBC)
        ii. Instrument Preliminary Design Review
        iii. Instrument Qualification Review
        iv. Instrument Critical Design Review
        v. Instrument Delivery Review Board
        vi. Other TBD Reviews as required

b. Ground Segment Level
    i. Ground Segment Requirements Review
    ii. Ground Segment Design Review
    iii. Ground Segment Implementation Review
    iv. Ground Segment Readiness Review

c. Mission Level
    i. System Requirements Review
    ii. Preliminary Design Review
    iii. Critical Design Review
    iv. Qualification Review
    v. Flight Acceptance Review
    vi. Flight Readiness Review
    vii. Mission Commissioning Results Review
    viii. TBD

4. The PI shall participate and support the Science Working Team meetings called by the Solar Orbiter Project Scientist.

8.5.1 INSTRUMENT PROGRESS MEETINGS

These meetings will be conducted between the ESA Project Office / the selected Prime and each Principal Investigator with the objective of ensuring that the interface technical design integrity of the experiment, its compatibility with the spacecraft system, and instrument programmatics are proceeding in a manner which will not jeopardize the overall programme. As a minimum, the Principal Investigator shall be represented by their PI, Co-I's, as required, the Instrument Manager as well as local project managers.

1. Regular Instrument Progress Meetings (Quarterly TBC) shall be held on the premises of the PI during the design, development and verification programme of the instrument. The frequency may be changed on request of the ESA Project Office depending on the severity of problems that may accumulate.

2. Detailed technical problems occurring on either side of the interface shall be flagged during these meetings and corrective actions, including their schedule impact, agreed and implemented.

3. The PI shall maintain and publish minutes of meetings to all participants and stake holders in the instrument.

8.5.2 MISSION REVIEWS

8.5.2.1 Objectives

In the course of the Solar Orbiter programme a series of reviews have been planned in accordance with the development schedule of the different elements forming the Solar Orbiter mission.

The objective of the Mission Reviews is to ascertain the satisfactory status of advancement and to verify the compliance of the technical and programmatic progresses with the overall programme requirements. Mission Reviews shall consist in the global assessment of

- Spacecraft (System Review)
- Instruments
- Ground Segment (Operational and Science)
- Launcher

The Principal Investigator will be invited to attend Mission Reviews.

8.5.2.2 Review Process

The review process will be ruled by **dedicated procedure** addressing all elements the programme consists of. The reports of the underlying reviews will be analyzed and the adequate closure of action items will be noted.

The role of the Board, consisting of senior ESA Executive and Scientific Staff is to elaborate **synthetic and harmonized conclusions to** the various Review issues and **formulating actions / or actions plan** to solve the identified discrepancies. The Board is chaired by the ESA Director of Science.

8.5.2.3 Review Sequence

The Mission Reviews will follow after the completion of the element reviews.

The sequence of Mission Reviews is as follows:
TBD

8.5.3 INSTRUMENT REVIEWS

8.5.3.1 Objectives

Instrument Reviews will be conducted by the ESA Project Office for each instrument selected for the Solar Orbiter Programme. The objectives will be to ensure that the instrument design will achieve the anticipated **science objectives** and that it **complies with the technical** interface requirements of the EID. **Programmatic aspects** like scheduled delivery dates and their compatibility with system level requirements will also be screened.

The Instrument Review will typically anticipate the System Level Reviews, where the results of the instrument review is taken into consideration as formal input and where possible inconsistencies have to be brought to a final solution.

The Review Board composition will be made up of ESA Project Office, Prime contractor personnel and invited specialists. It will be chaired by the designated Solar Orbiter Payload Responsible together with the Project Scientist. The PI and his scientific and technical team shall support the review and its panel and board sessions with appropriate manpower and expertise.

Documentation to be reviewed will consist of Review Data Packages, as detailed in TBD and provided to the review authority in due time as laid down in TBD and in accordance to dedicated Review Procedures to be issued for the occasion.

The output of the review may provide recommendations for consideration by the ESA Project Manager or the Principal Investigators in technical or programmatic areas. Where requested, either party shall provide a formal response to such recommendations according to the above mentioned Review Procedure.

8.5.3.2 Review Process

The review process will be ruled by **dedicated procedure** according to the following guidelines:

- Typically one month before Review due date, the Principal Investigator will submit the data package according to the to be specified review procedure
- ESA and its selected Prime will jointly issue RID's and transmit them to the PI organizations
- The PI shall include the position of the Principal Investigator, clarifying misunderstanding or propose corrective measures
- The PI answers will be submitted to the review panels for endorsement and agreement on corrective measures to the issues raised
- Where answers are not satisfactory or where major programmatic interests are impacted the panel will address the issue to the Board.

The role of the Board is to elaborate **synthetic and harmonized conclusions to** the various Review issues and **formulating actions / or actions plan** to solve the identified discrepancies.

8.5.3.3 Review Sequence

The review sequence will insure a consistent approach and interrelationship between the instrument reviews and the system or satellite reviews, which are considered higher level reviews. The following sections will define the detailed objectives and their precise interrelationship.

It is realized that, aside from the formal ESA reviews as defined in this document, instruments might want to conduct further instrument reviews, e.g. for internal monitoring of progress, request from funding agencies. In order to avoid duplication of effort combination of instrument internal and formal ESA reviews can be envisaged, as long as the objectives of both reviews match.

The sequence of Reviews is TBD.

8.6 Configuration Management

8.6.1 GENERAL

An effective configuration management scheme shall be established within the Principal Investigator in order to ensure all hardware, software and documentation is fully traceable with history and exact definition of the data or hardware at all times.

1. The Solar Orbiter Project documentation coding system (Ref: TBD) shall be employed to provide references for all configurable items.

8.6.1.1 Objectives

The objectives of Configuration Management are to establish:

- a **configuration identification** baseline system which defines through approved specifications, interface documents and associated data the requirements for the instrument,
- a **configuration control** system which controls all the changes to the identified configuration of the instrument,
- a **configuration accounting** system which documents all changes to the baseline configurations, maintains an accurate record of configuration change incorporation, and ensures conformity between the end item As Built Configuration (ABCL) and its appropriate design and qualification identification (CIDL including waivers).

8.6.1.2 Responsibilities

1. The PI shall be responsible for managing the configuration of his instrument and the lower level products of which it consists. For this purpose, he shall set up the necessary organization and means for satisfying the objectives and requirements of configuration management.

2. The PI shall also impose configuration management requirements on contractors and suppliers as appropriate for the items being provided to the instrument. For this purpose, the PI shall ensure compatibility between their own configuration management and the one implemented by all other participants to their instrument programme.

3. The PI shall be responsible for the implementation and operation of a Configuration Control Board (CCB) at his level.

8.6.2 CONFIGURATION REQUIREMENTS

8.6.2.1 Configuration Identification

1. Configuration baselines shall be established with respect to requirements, design and verification.

2. The Requirements Baseline shall include:
   - Instrument System Specification
   - Instrument System Support Specification
   - Interface Control Documents

3. The Design Baseline shall include:
   1. Design Specification
   2. Drawings
   3. Manufacturing Procedures

   4. The Verification Baseline shall include:
      - Control and Inspection Procedures
      - Operating and Handling Procedures
      - Test Procedures

   5. Configuration baselines shall be established and reviewed at each Instrument Review. Baselines may also be established and reviewed as required at selected intermediate stages.

   6. The "as designed" baseline shall be established at the Instrument Hardware Design Review.

   7. Verification documents including design analyses and test reports shall make reference to the configuration status of the design or the hardware or software being evaluated.

8.6.2.2 Configuration Control

1. As an integral part of his management structure, the PI shall set up a configuration control procedure for his instrument in such a manner that the status of all aspects of his experiment such as the design and manufacturing of hardware and development of software can be unambiguously defined at any time.

2. The control procedure shall allow the ESA Project Office to conduct a configuration audit at any point in the programme in order to obtain the up-to-date status of the instrument. The approval right for changes initiated by any party is exclusive right of the ESA Project Manager.

8.6.2.3 EID Configuration Control

The requirements defined in the EID Part A and the agreements documented in the EID-B and annexed documents (Engineering, Management and PA Plan) will be subject to configuration control and must reflect the up to date agreed configuration baseline between the ESA Project Office and its selected Prime and the PI.

1. Once the EID-B is signed by ESA and its selected prime on one hand and the PI on the other side changes to these documents shall be handled using the Engineering Change Request (ECR) form (TBD).

2. Deviations (not fulfillments) from the requirements defined in EID A, B will be handled using the Request for Waiver (RFW) form.

The **Engineering Change Request (ECR)** may be initiated at any time by either the PI or ESA / Prime in writing. It shall be completed by the raising party with all relevant entries. All ECR's shall be circulated / addressed to the other two parties involved (ESA, Prime and PI).

Note: *The ECRs shall be addressed to the PI, ESA Project Manager and Prime Project Manager, with copy to the relevant Payload or instrument managers.*

Principal Investigators shall ensure that adequate resources and funding are available to them for execution of a proposed change prior to submittal to the Agency.

The **Request For Waivers (RFW)** may be initiated at any time by either the PI or ESA / Prime in writing. It shall be completed by the raising party with all relevant entries. All RFW's shall be circulated / addressed to the other two parties involved (ESA, Prime and PI).

Processing of a Request for a Change or Waiver

3. Following its receipt, the ECR / RFW is submitted to the Change Control Board (CCB) of the receiving part who shall process the request and take a decision on the change (ECR / RFW) disposition within 4 weeks.

The Configuration Control Board (CCB) of the receiving party shall process the request and if agreeable, approve the implementation of the change or approve the waiver.

In case of disagreement each party has the right to call for a joint Change Review Board which shall be chaired by the ESA Project Manager. If no consensus can be reached the ESA Project Manager shall finally disposition the request in the interest of the overall programme.

No activities on the proposed change shall be started prior to the written approval of the ECR.

Numbering of Changes and Waivers

4. Each ECR or RFW shall be identified by an individual number. This number shall be used on all subsequent correspondence. Following ECR or RFW dispositioning, this number shall not be used again. Each PI shall register and control his own numbering sequence.

The number comprises:
- Abbreviation of Engineering Change Request or Request For Waiver (ECR or RFW)
- Instrument identification (3 letter abbreviation)
- Instrument sequence number (four digits)

Note: *Example ECR/XYZ/0100*

8.6.2.4 Configuration Status Accounting

1. The current status of all configured documents shall be sent to the ESA Project Office as part of the reporting procedure required in section 8.4.3.

2. Configuration Item Data Lists (CIDL) listing all the documents and their applicable issues and revisions which define the configuration baseline shall be prepared and submitted for each Instrument Review.

3. The PI shall establish and maintain As Built Configuration Lists (ABCL) listing all the documents and their issues and revisions defining the as built configuration.

4. Differences between the as designed baseline and the as built configuration list shall be identified for all qualification and flight hardware and software. The validity of all design verifications, including analyses and tests, shall be assessed for all the differences and modifications from the as designed baseline.

8.7 Deliverable Items

8.7.1 DELIVERABLES TO THE SPACECRAFT

8.7.1.1 Interface Documentation

1. The PI shall deliver and maintain all relevant interface documentation throughout the project lifetime.

8.7.1.2 Mathematical Models

1. The PI shall deliver a Structural Mathematical Model (SMM) of his instrument, as defined in section 6.3.1.

2. The PI shall deliver a Thermal Mathematical Model (TMM) of his instrument, as defined in section 6.3.2.

These instrument mathematical models shall be updated as the design progresses. They will serve as input to the spacecraft mathematical models and may require revision at various points in the Solar Orbiter development programme.

8.7.1.3 Instrument Models

1. The PI shall deliver the following instrument models as defined in section 6.8.1:
   - Structural and Thermal Model (STM),
   - Electrical Model (EM),
   - Flight Model (FM),
   - Flight Spares (FS).

2. Each delivery shall include, as appropriate, instrument hardware, on-board software and ground support equipment.

3. Each item delivered shall be accompanied by an End Item Data Package (content TBD).

4. Prior to delivery, each item shall undergo formal acceptance on the basis of mutually agreed acceptance programme.

5. Shipment of the instrument models and any other equipment required by either the Agency or the PI shall be the financial responsibility of the PI. This responsibility shall extend to return for repair and return of all equipment following launch.

6. The points of delivery of all items will be determined later in the programme and be included in this document.

7. Any insurance deemed necessary by the Principal Investigator for his equipment during shipment or whilst on the premises of the Agency, it's Contractors or on the launch site, shall be the financial responsibility of the Principal Investigator.

8. All ITAR papers necessary for shipment shall be obtained by the PI prior to the required shipment date and shall include all the delivery destinations for launch to orbit.

9. The build standard of each model shall be defined in EID-B and agreed with the ESA Project Office.

10. The FM and the FS shall be fully calibrated before delivery.

11. The PI shall support the system level integration and test activities as well as the launch preparation by supplying the appropriate manpower and expertise.

8.7.1.4 On-Board Software

1. The instrument on-board software shall be delivered together with the corresponding instrument model.

2. The on-board software shall either reside in the instrument in a non volatile memory or be delivered in a format such that it can be loaded through the spacecraft telecommand uplink.

3. In addition to the flight software, special test software for instrument diagnostics and failure investigation may be required.

4. The on-board software to be delivered shall comply with the ESA software standard ECSS TBD.

5. The PI remains responsible for the maintenance of the instrument software after delivery up to the end of mission.

6. The PI shall support the verification of updated instrument software at system level.

8.7.1.5 Ground Support Equipment

1. Together with each instrument model the PI shall deliver the Mechanical Ground Support Equipment (MGSE) necessary to transport, handle and integrate the instrument hardware, accompanied with appropriate documentation and proof load and calibration certificates.

2. Together with each instrument model the PI shall deliver the Electrical Ground Support Equipment (EGSE) necessary to stimulate the instrument and to perform quick look analysis of instrument TM during system tests.

3. It shall be designed such that it can be integrated into the system EGSE set-up.

4. The instrument EGSE software to be delivered with the EGSE equipment shall comply with the ESA software standard ECSS-E-40 [NR20].

5. The Instrument Station in charge of performing quick look analysis of instrument scientific TM shall communicate with the Central Checkout Equipment (CCE) via a LAN with TCP/IP protocol.

6. TM data exchange over the LAN will be at Source Packet level following the EGSE protocol to be defined by the contractor.

7. The Instrument Station software shall be designed such that it can be reused in the Science Ground Segment to the maximum extent.

8. The CCE will be in charge of the instrument's housekeeping by sending the necessary commands and by monitoring the housekeeping TM for health and status monitoring.

9. The instrument ground support equipment shall remain at the spacecraft integration site until launch.

10. The PI shall remain responsible for the maintenance of this equipment.

11. The PI shall also provide the necessary manpower and expertise support to integrate the instrument EGSE into the system EGSE.

8.7.2 DELIVERABLES TO THE GROUND SEGMENT

8.7.2.1 Deliverables to the Operational Ground Segment
TBW

8.7.2.2 Deliverables to the Science Ground Segment
TBW

8.7.2.3 Review Deliverables
1. The PI shall deliver Review Data Packages for each scheduled Instrument Review detailed in section 8.8.2.

2. The Review Data Package shall include information regarding:
    - configuration
    - scientific performance,
    - design and verification,
    - operations
    - product assurance.

The contents of the Review Data Package for each review will be outlined by the ESA Project Office prior to the review.

8.8 Schedules

A detailed schedule down to component and sub-assembly level is an invaluable tool to coordinate all the members of a Principal Investigator and to demonstrate the commitments for delivery to the spacecraft.

1. The PI shall create and maintain a detailed instrument development schedule in an agreed TBD format deliverable to the ESA Solar Orbiter Project Office quarterly or on demand in case of urgency.

2. The PI shall agree and maintain deliverable dates with the Solar Orbiter Prime Contractor and ESA Solar Orbiter Project Office.

3. All ITAR related approval aspects shall be clearly identified and included in the planning

8.8.1 OVERALL PROJECT SUMMARY SCHEDULE

1. All **milestones** specified by the ESA Project Office shall be included in the schedule and be agreed by the Principal Investigator.

2. The PI shall identify **additional milestones** as required and agree them with the ESA Project Office and its selected Prime.

3. All interfaces, such as procurement items, ITAR permissions, hardware deliveries, reviews, etc. shall be clearly identified.

4. The schedule shall reflect the result of detailed task analysis and critical review of all the activities associated with the instrument programme.

5. It shall contain all activity interdependencies durations and constraints.

6. Based on precedence type network, the schedule shall be so constructed that automatic analysis of time earliest and time latest for critical events can be performed and critical paths identified.

8.8.2 PROJECT REVIEW SCHEDULE

The instrument reviews will take place according to the following planning:

- January 2010 Instrument System Requirements Review (ISRR)
- January 2011 Instrument Preliminary Design Review (IPDR)
- January 2012 Instrument Critical Design Review (ICDR)

- February 2014 Instrument Qualification Review (IQR)
- December 2014 Instrument Flight Acceptance Review (IFAR)

The review data package shall be delivered 6 weeks (TBC) before the concluding Board meeting. The dates are as follows:

Review	Data pack Del.	Review Board
• ISRR	6 weeks before	7 yrs prior to launch
• IPDR	6 weeks before	6 yrs prior to launch
• ICDR	6 weeks before	5 yrs prior to launch
• IQR	6 weeks before	just after EM delivery
• IFAR	6 weeks before	just prior o FM delivery

8.8.3 BASELINE SCHEDULE OF DELIVERIES BY PI

1. Instrument Deliveries shall be in line with the overall project constraint:
    - **STM: 4 yrs prior to launch January 2013**
    - **EM: 3 yrs prior to launch January 2014**
    - **FM: 2 yrs prior to launch January 2015**

2. The instrument Thermal Mathematical Model shall be updated according to the following plan:
    - Issue 1 TBD
    - Issue 2 TBD
    - Issue 3 TBD
    - Issue 4 TBD
    - Issue 5 TBD
    - Issue 6 TBD

3. The instrument Structural Mathematical Model shall be updated according to the following plan:
    - Issue 1 TBD
    - Issue 2 TBD
    - Issue 3 TBD
    - Issue 4 TBD
    - Issue 5 TBD
    - Issue 6 TBD

4. Instrument deliveries to the Operational Ground Segment shall be according the following plan:
    - TBD

5. Instrument deliveries to the Science Ground Segment shall be according the following plan:
    - TBD

8.8.4 BASELINE SCHEDULE OF DELIVERIES BY ESA/PRIME CONTRACTOR

TBD

9 DOCUMENTS

Following the definitions of the ECSS (European Cooperation for Space Standardization), documents relevant to the Solar Orbiter Project are classified as Normative and Informative Documents. Normative documents are referenced in the text of the EID-A as specific requirements which call up the section in the specified document. Informative documents are listed for information to the PI but are not formally requirement documents.

9.1 Normative References

[NR1] Mission Requirements Document for Solar Orbiter, Sol-EST-RS-0049, Issue 3, June 2006 (all Sections)

[NR2] SpaceWire - Links, nodes, routers and networks, ECSS-E-50-12, Jan 2003

[NR3] ECSS-E-30 Part 1A

[NR4] ECSS-E-10-12

[NR5] ECSS-E-10-4

[NR6] ECSS-E-50-12

[NR7] ECSS-E-20A section 6.3.3.3d and par. 5.9

[NR8] ECSS-E-30-02

[NR9] ECSS-Q-00A

[NR10] ECSS-Q-20B
Quality Assurance

[NR11] ECSS-Q-30-02 A

[NR12] ECSS-Q-30 B

[NR13] PSS-01-301

[NR14] ECSS-Q-60 A
EEE Components Control

[NR15] ECSS-Q-70 A

Materials, Mechanical Parts and Processes

[NR16] ECSS-10-04 A
Space Environment

[NR17] ECSS-Q-60-01 A
European Preferred Parts List and its Management

[NR18] ESA PSS-01-608
Generic Specification for Hybrid Micro-circuits

[NR19] ECSS-E-30-01a

[NR20] ECSS-E-40
Software

[NR21] ECSS-Q-20-04A

[NR22] ECSS-Q-20-09

[NR23] ESA PSS-01-606
Capability Approval Programme for Hermetic Thick Film Hybrid Micro-Circuits

[NR24] ESA PSS-01-605
Capability Approval Programme for Hermetic Thin Film Hybrid Micro-Circuits

[NR25] MIL-HDBK-5

[NR26] ECSS-Q-70-08 A
The manual soldering of high-reliability electrical connections

[NR27] ECSS-Q-70-28 A
Repair and modification of printed circuit board assemblies for space use

[NR28] ECSS-Q-80
Software Product Assurance

[NR29] ECSS-Q-70-01A
Contamination and cleanliness control

9.2 Informative References

The following documents as of the current issue, as indicated or most recent in the event of updates, are possible sources of clarification for the content of the Solar Orbiter EID.

[IR1] Solar Orbiter Payload Definition Document – issue 6 – ref. SOL-EST-SP-00705 – October 2007

[IR2] Solar Orbiter Environmental Specification – issue 1, revision 3 – ref TEC-EES-03-034/JS – January 2006

[IR3] Soyuz User's Manual, ST-GTD-SUM-01, issue 3, revision 0, April 2001

[IR4] Study of the Interaction between the Rosetta Orbiter and the Cometary Plasma, Final Report of ESA Contract 12398, ESA Space Environments and Effects Section 1999

[IR5] Solar Orbiter Science Requirements Document – issue 1, revision 0, December 2003

[IR6] Solar Orbiter Mission Analysis – issue 1, revision 1 – ref. MAO-WP-483 – November 2005

[IR7] Atlas Launch System Mission Planner's Guide – ref. CLSB-0409-1109 - Revision 10a, January 2007

10 ACRONYMS

AAS	Alcatel Alenia Space
ABCL	As Built Configuration List
AC	Alternating Current
ACC	Attitude Control Computer
ADC	Analog to Digital Converter
AFT	Abbreviated Functional Test
AIT	Assembly, Integration and Test
AIV	Assembly, Integration and Verification
AME	Absolute Measurement Error
AOCS	Attitude and Orbit Control System
APE	Absolute Pointing Error
ASIC	Application Specific Integrated Circuit
BOL	Beginning of Life
CAN	Controller Area Network
CCB	Configuration Control Board
CCE	Central Checkout Equipment
CDMU	Command and Data Management Unit
CF	Cold Finger
CoG	Centre of Gravity
CIDL	Configuration Item Data List
CoM	Centre of Mass
COR	Coronagraph
CPU	Central Processing Unit
CRB	Contamination Review Board
CSA	Charge Sensitive Amplifier
DAC	Digital to Analog Converter
DC	Direct Current
DCL	Declared Components List
DDV	Design Development Validation
DHS	Data Handling System
DML	Declared Material List
DMPL	Declared Mechanical Parts List
DMS	Data Management System
DNEL	Disconnect Non Essential Loads
DPD	Dust Particle Detector
DPL	Declared Process List

DPU	Digital Processing Unit
DRB	Delivery Review Board
DS	Document Specification
DSM	Deep Space Maneuver
DSP	Digital Signal Processor
DTMM	Detailed Thermal Model
EAS	Electron Analyzer System
ECR	Engineering Change Request
ECSS	European Cooperation for Space Standardization
EGSE	Electrical Ground Support Equipment
EID	Experiment Interface Document
EIDP	End Item Data Package
EGSE	Electrical Ground Support Equipment
EMC	Electromagnetic Cleanliness/Compatibility
EMCB	Electromagnetic Cleanliness Board
EMI	Electromagnetic Interference
EOL	End of Life
EPD	Energetic Particle Detector
EPPL	European Preferred Parts List
EPS	Electrical Power Subsystem
EPT	Electron and Proton Telescope
EQM	Electrical Qualification Model
ESD	Electro-Static Discharge
EUI	Extreme Ultraviolet Imager
EUS	Extreme Ultraviolet Spectrometer
EUV	Extreme Ultra-Violet
FDT	Full Disc Telescope
FEE	Front End Electronics
FEM	Finite Element Model
FFT	Full Functional Test
FIFO	First In First Out
FM	Flight Model
FMECA	Failure Modes, Effects and Criticality (Analysis)
FOSU	Factor of Safety – Ultimate
FOSY	Factor of Safety - Yield
FOV	Field of View
FPGA	Field Programmable Array
FPU	Floating Point Unit
FS	Flight Spare
GAM	Gravity Assist Maneuver

GEO	Geostationary Earth Orbit
GSE	Ground Support Equipment
HA	Hazard Analysis
HE	Hot Element
HELEX	Heliophysical Explorers
HETn	High Energy Telescope with neutron detection
HFR	High Frequency Receiver
HGA	High Gain Antenna
HIS	Heavy Ion Sensor
HRT	High Resolution Telescope
HSIA	Hardware Software Interaction Analysis
HTHGA	High Temperature High Gain Antenna
H/W	Hardware
IBDR	Instrument Baseline Design Review
ICDR	Instrument Critical Design Review
ICU	Instrument Control Unit
IDP	Instrument Development Plan
IGMM	Interface Geometrical Mathematical Model
ILS	Instrument Line of Sight
I/O	Input/Output
IPDR	Instrument Preliminary Design Review
IR	Infra Red
ISS	Internal Stabilization System
ITMM	Interface Thermal Mathematical Model
ITO	Indium Tin Oxide
LAT	Lot Acceptance Test
LEMMS	Low Energy Magnetospheric Measurement Subsystem
LEOP	Launch and Early Orbit Phase
LET	Linear Energy Transfer
LCL	Latching Current Limiters
LCPM	liquid crystal polarization module
LET	Low Energy Telescope
LFR	Low Frequency Receiver
LGA	Low Gain Antenna
LISN	Line Impedance Stabilization Network
LOS	Line Of Sight
MAG	Magnetometer
MCP	Micro Channel Plate
MGA	Medium Gain Antenna

MGSE	Mechanical Ground Support Equipment
MICD	Mechanical Interface Control Document
MIP	Mission Implementation Plan
MIRD	Mission Implementation Requirements Document
MLI	Multi Layer Insulation
MOC	Mission Operations Centre
MRF	Mechanical Reference Frame
NCR	Non-Conformance Report
NGD	Neutron and Gamma-ray Detector
NIEL	Non-Ionizing Energy Loss
NIS	Normal Incidence Spectrometer
NRB	Non-Conformance Review Board
OBDH	On-Board Data Handling
OTS	Off-The-Shelf
PA	Product Assurance
PAD	Parts Approval Document
PAS	Proton and Alpha particle Sensor
PCU	Power Converter Unit
PDD	Payload Definition Document
PDE	Pointing Drift Error
PDMU	Payload Data Management Unit
PDU	Power Distribution Unit
PHA	Pulse Height Analysis
PI	Principal Investigator
PMT	Photo Multiplier Tube
PS	Project Scientist
QA	Quality Assurance
QCI	Quality Conformation Inspection
QM	Qualification Model
RAM	Random Access Memory
RES	Radiated Electric Field Susceptibility
RFW	Request for Waivers
RID	Review Item Discrepancy
RMS	Radiated Magnetic Susceptibility
RPE	Relative Pointing Error
RPW	Radio and Plasma Wave analyzer
RTC	Remote Terminal Controller
RTU	Remote Terminal Unit

S/C	Spacecraft
SciRD	Scientific Requirements Document
SDT	Solar Orbiter Science Definition Team
SEL	Single Event Latch-up
SEP	Solar Energetic Particle
SEU	Single Event Upset
SIS	Supra-thermal Ion Spectrograph
SMM	Structural Mathematical Model
SOC	Science Operations Centre
SPA	Software Product Assurance Program
SPE	Solar Particle Event
SPF	Single Point Failure
SpW	Space Wire
SRP	System Reference Point
SSAC	Space Science Advisory Committee
SSMM	Solid State Mass Memory
SSWG	Solar System Working Group
STE	Supra-Thermal Electron detector
STIX	Spectrometer Telescope Imaging X-rays
STM	Structural Thermal Model
STR	Star Tracker
SWA	Solar Wind Analyzer
SWG	Science Working Group
SWT	Science Working Team
TBC	To Be Confirmed
TBD	To Be Determined
TBS	To Be Supplied
TBW	To Be Written
TC/TM	Tele-command / Telemetry
TCS	Thermal Control System
TDA	Technology Development Activity
TDP	Technology Development Plan
TM	Telemetry
TMM	Thermal Mathematical Model
TNR	Thermal Noise Receiver
TOF	Time-Of-Flight
UFOV	Unit Field of View
UOAF	Unit Optical Alignment Frame
UORF	Unit Optical Reference Frame
URF	Unit Reference Frame

URP	Unit Reference Point
UV	Ultra-Violet
VIM	Visual-light Imaging Magnetograph
VLS	Variable Line Spacing
WBS	Work Breakdown Structure
WCA	Worst Case Analysis
WPD	Work Package Description