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Title

PAYLOAD INTERFACE DOCUMENT

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SUMMARY

The overall set of PID (PID-A and PID-B) documents forms the sole technical requirements link between MMS (as spacecraft contractor) and the payload Principal Investigators (or PI's as payload contractors) for the MARS EXPRESS programme.

The objective of the PID is then to ensure that :

- the payloads are designed, built and verified within the constraints imposed by the overall scientific package, spacecraft and vehicle
- the spacecraft contractor is able to design, build and verify the satellite in such manner that the payloads can be properly integrated in the system
- the spacecraft system can be successfully launched and operated to achieve the general ESA mission and science objectives
- the overall programme management follows clear, efficient and approved rules.

To these respects, the PID-A intends to define the MMS spacecraft requirements - in nature, value and format whenever applicable - while the PID-B will represent the PI reflected (and maintained) payload status with respect to these requirements.

The document is organized as follows :

- chapter 2 / General Informations and Requirements : this part is devoted to the ESA science requirements and to the general and physical description of the spacecraft, together with its life cycle (including the relevant environments) and operations, in-flight and on-ground, including those related to ESA ground segment and spacecraft AIT
- chapters 3 to 7 / Payloads Design and Interfaces Requirements : these chapters are detailing the mechanical, dynamic, thermal, optical, electrical, EMC, operations and software interface main requirements, in nature and value (definition of mutual constraints) and form (in which format express these constraints)
- chapter 8 / Payloads Verification Requirements : this part describes the development and verification approach settled at spacecraft level, and the detailed requirements applying on payloads
- chapter 9 / Payloads Product Assurance Requirements
- chapter 10 / Payloads Management Requirements.

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1. INTRODUCTION

1.1. Scope and Organization of the Document

The overall set of PID (PID-A and PID-B) documents forms the sole requirements link between ESA / MMS (as customer / spacecraft contractor) and the payload Principal Investigators (as payload contractors) for the MARS EXPRESS programme.

The objective of the PID (co-written by ESA and MMS) is then to ensure that :

- the payloads are designed, built and verified within the scope defined by the ESA mission and science objectives, and constrained by the spacecraft, vehicle and ESA ground segment capabilities
- the spacecraft contractor is able to design, build and verify the satellite in such manner that the payloads can be properly integrated in the system
- the spacecraft can be successfully launched and operated to achieve the general ESA mission and science objectives
- the overall programme management follows clear, efficient and approved rules.

To these respects, it is understood that :

- the PID-A intends to define the ESA scientific performance specifications and the MMS spacecraft requirements in nature, value and format whenever applicable
- while the PID-B will represent the PI reflected (and maintained) payload status with respect to these requirements.

The following payload subjects - irrespective to the « new developed » either « partially re-developed » or « re-flown » design / hardware status - are to be exhaustively covered :

- engineering and verifications (focussed on performance and interface aspects)
- ground and flight operations, including spacecraft AIT and mission operations
- product assurance and applicable standards
- schedule control and management.

The last subjects are settled in such a way that payloads and spacecraft are successfully developed within the very stringent constraints of the MARS EXPRESS programme.

The document is organized as follows :

- chapter 2 / **General Informations and Requirements** : this partis devoted to the **ESA science requirements**, to the general / physical **description of the mission and the spacecraft**, together with its **life cycle** (including environments) and **in-flight / on-ground operations** (at spacecraft AIT and ESA ground segment levels)

  This chapter is co-written by ESA and MMS, and the applicability of its various paragraphs is given in the following table :

Chapter 2	Written by	Applicability
Para 2.1 / Mars Express Science Requirements	ESA	Requirements
Para 2.2 / Spacecraft Physical Characteristics	MMS	For Information
Para 2.3 / Mission Timeline	MMS	For Information
Para 2.4 / Ground Operations Before Flight	MMS	For Information and Requirements
Para 2.5 / Launch Environment	MMS	Requirements
Para 2.6 / Flight Operations Concept	MMS	For Information
Para 2.7 / Attitude and Orbit Control System	MMS	For Information
Para 2.8 / Flight Environment	MMS	Requirements
Para 2.9 / Payload System Engineering	MMS	Requirements
Para 2.10 / Interfaces with the Ground Segment	ESA	Requirements

- chapters 3 to 7 / **Payloads Design and Interfaces Requirements** : these chapters are detailing the mechanical, dynamic, thermal, optical, electrical, EMC, operations and software interface main requirements, in nature and value (definition of mutual constraints) and form (in which format express these constraints)
- chapter 8 / **Payloads Verification Requirements** : this part describes the development and verification approach settled at spacecraft level, and the detailed requirements applying on payloads
- chapter 9 / **Payloads Product Assurance Requirements**
- chapter 10 / **Payloads Management Requirements**

1.2. The MARS EXPRESS Programme

The MARS EXPRESS mission is considered to be the first scientific mission which will pioneer a « cheaper » (in the sense of « capability driven ») approach - known as flexible (F) mission in the context of the Horizons 2000 general plan - to the procurement of ESA science programmes.

The spacecraft is planned to be launched in the very beginning of june 2003 (a two weeks narrow window), and to be implemented within very stringent financial conditons by ESA. These development streamlines lead to reconsider the spacecraft contractor involvment in the payload package development course itself, more deeply than for the more classical science programmes (like SOHO).

The spacecraft will consist of :

- a basic orbiting spacecraft (the « orbiter ») carrying a complete instrumentation set directed towards « surface » and « atmosphere » remote sensing
- with anyway the possibility of carrying one landing vehicle (the « lander ») - depending the overall spacecraft configuration - if its mass is lower than 60 kg.

While some orbiter instruments will be specifically developed for the MARS EXPRESS mission, it is to be noticed that it could be intended to re-fly a part (or the entirety) of some of the MARS 96 orbiting instrumentation. But, in any case the embarked hardware shall be qualified to the MARS EXPRESS specific mission environment.

1.3. PID Issue Schedule

Most of the formal PID issues are edited in conjunction with the main Payload Reviews (see the chapter 10 related to management for details).

Because of the nature of this rapidly evoluting (technical) interfaces domain, let us highlight here that, beside the formal issue and approval run of the changes, the various evolutions (likely to be proposed in the course of the programme) shall be extensively discussed in the frame of the payload-dedicated progress meetings, under the form of « red marked, comprehensive and exhaustive » amendments (the so-called Engineering Change Requests, or ECR's) of the relevant pages directly extracted from the current (and approved) PID-B issue.

The final dispositioning of these ECR's (after the various discussions and approvals) should as well take the form of a new « red marked » amendment of the PID-B relevant page to take place - as quickly as possible - in the PID to permit the maintaining of an updated and self-contained document.

1.4. Acronyms and Abbreviations

An non exhaustive list of acronyms and abbreviations is given hereafter.

- A -

AC	Alternating Current
AIT	Assembly, Integration and Test
AIV	Assembly, Integration and Verification
AOCS	Attitude and Orbit Control System
ASPERA	Analyzer of Plasma and EneRgetic Atoms

- B -

BOL	Beginning Of Life

- C -

CCB	Change Control Board
CDR	Critical Design Review
CIDL	Configuration Items Data List

- D -

DDVP	Design Development and Verification Plan

- E -

ECR	Engineering Change Request
EM	Experiment Manager
EOL	End Of Life
ESA	European Space Agency
ESOC	European Space Operations Center

- F -

FDIR	Failure Detection, Isolation and Recovery
FMECA	Failure Mode Effects Critical Analysis
FPT	Full Performance Test

- G -

GSE	Ground Support Equipment

- H -

HRSC	High Resolution Stereo-Camera

- I -

ICDR	Instrument Critical Design Review
IPDR	Instrument Preliminary Design Review
IQAR	Instrument Qualification and Acceptance Review
IRR	Instrument Requirements Review
ITT	Invitation To Tender

- L -

LEOP	Low Earth Orbit Phase
LPT	Limited Performance Test
LSB	Least significant Bit

- M -

MLI	Multi Layer Insulation
MOI	Mars Orbit Insertion
MSB	Most Significant Bit

- N -

NCR	Non Conformance Report

- O -

OMEGA	Observatoire pour la Minéralogie, l'Eau, les Glaces et l'Activité

- P -

PDR	Preliminary Design Review
PFS	Planetary Fourier Spectrometer
PI	Principal Investigator
PID	Payload Interface Document
PVP	Performance Verification Plan
P/L	Payload

- Q -

QR	Qualification Review

- R -

RF	RadioFrequency
RFW	Request For Waiver
RSE	Radio Sounding Experiment

- S -

SMM	Structural Mathematical Model
SPICAM	Spectroscopic Investigation of the Characteristics of the Atmosphere of Mars
SSRA	Sub Surface Radar / Altimeter
STMM	Simplified Thermal Mathematical Model
S/C	Spacecraft

- T -

TBC	To Be Confirmed
TBD	To Be Defined
TC	TeleCommand
TM	TeleMetry
TMM	Thermal Mathematical Model
TP	Tweested Pair
TRP	Temperature Reference Point
TSP	Tweested Shielded Pair

- U -

UORF	Unit Optical Reference Frame
URF	Unit Reference Frame
USO	Ultra Stable Oscillator

- V -

VCD	Verification Control Document
VSWR	Voltage Stationary Wave Ratio

- W -

WOA	Working Orbit Acquisition

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2. GENERAL INFORMATIONS AND REQUIREMENTS

2.1. Mars Express Science Requirements

TBD ESA.

2.2. Spacecraft Physical Characteristics

2.2.1. Spacecraft Configuration

The spacecraft preliminary design is based on a cubic like shape sizing about 1.5 m side length, with top and bottom platforms and four side panels.

A modular mechanical architecture provides accommodation surfaces and volumes fully dedicated to the science payload. Two adjacent faces are dedicated to the science payload. One of them will be nadir pointed during the observation sessions and lander communication relay phases about pericenter. The other face is kept away from the Sun during both science and Earth communication phases and represents a favourable surface for accommodating cold payload radiators.

The Onboard Data Handling is based on packet telemetry and telecommand. The Power generation is performed by solar arrays, the power storage by batteries. A standard 28 V main bus is offered to the payload instruments. Connections to the potential lander whilst attached to the orbiter will be provided through umbilicals. The Telemetry, Tracking and Command (TT&C) will transmits X band telemetry 8 hours per day via a High Gain Antenna depending of the Mars to Earth Distance. A telecommand rate of 500 bps (overall) is foreseen during up to 8 hour per day.

Note : a more complete spacecraft description will be available after S/C selection.

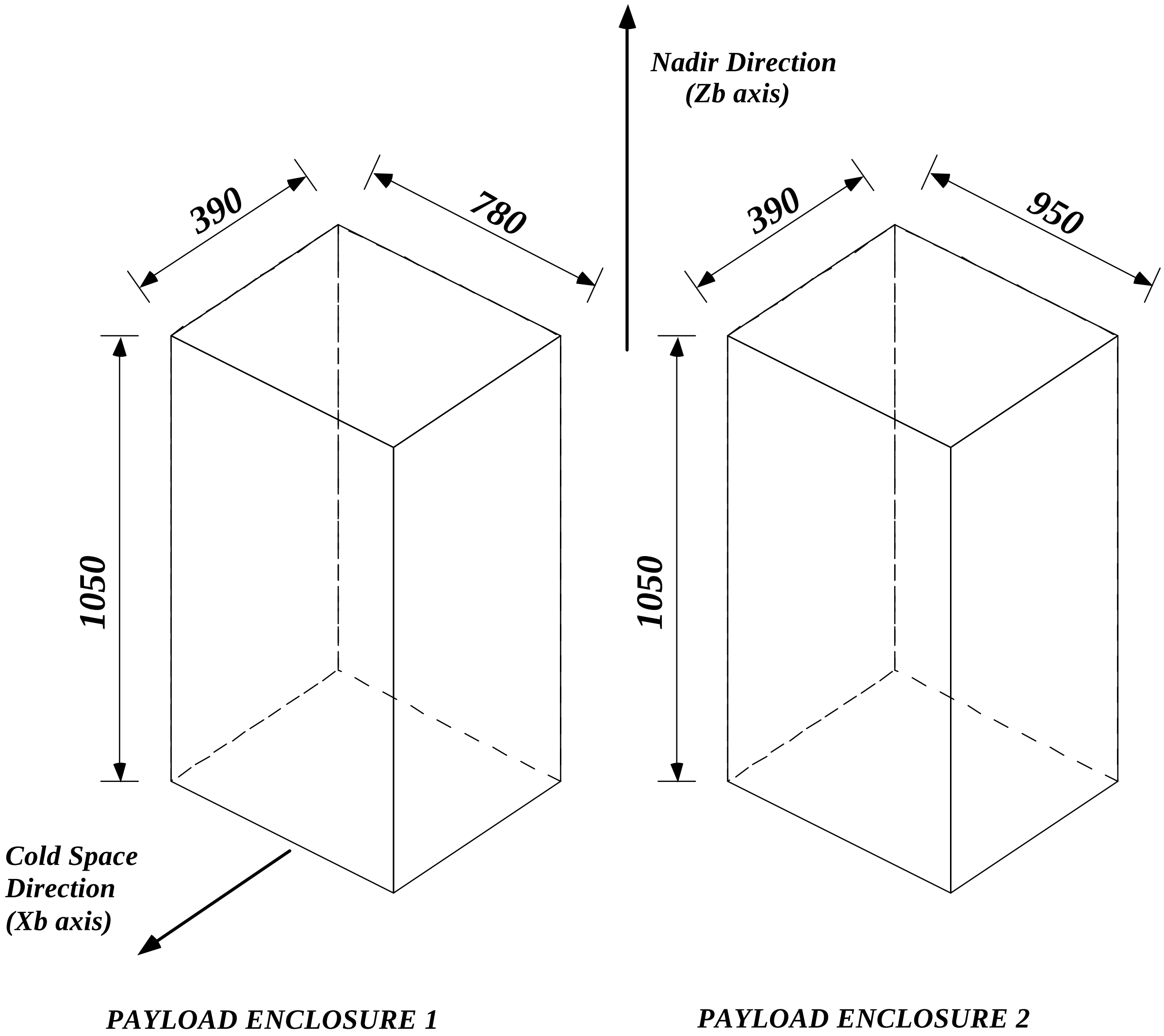
2.2.2. Spacecraft Reference Coordinate Axes

Spacecraft mechanical build axes [Xb, Yb, Zb]

The spacecraft mechanical build axes are fixed relative to the spacecraft geometry, forming a right handed orthogonal co-ordinate system with the origin located at the centre of the spacecraft lower flange/adapter interface plane. The spacecraft mechanical build axes will be used during spacecraft integration for positioning of every unit.

During the science observation phases around pericentre, the Zb axis is nominally nadir pointed and positively oriented toward nadir. Xb is normal to the orbital plane and positively oriented toward the antisun side of the orbital plane. As a consequence, Yb remains in the orbital plane and moves every about 180 days from an orientation close to the velocity direction to an orientation opposite to the velocity direction.

During the communication phases around Mars, The Zb axis is kept away from the Sun by more than 45 degrees. The Xb axis is kept in the half space opposite to the Sun.



Note : Some Payload elements (antennas, scanners, heads) may protude externally from the Payload

Figure 2.2/1: Science payload accommodation volume and spacecraft build axes

Spacecraft Optical Reference Axis [Xo, Yo, Zo]

For alignment purposes during integration, a set of spacecraft reference optical axes will be defined by a master optical cube located on the spacecraft. In the nominal case, these axes shall correspond to the spacecraft Mechanical Build Axes. After integration, the position of those equipment units requiring measurement or adjustment and equipped with an optical cube will be determined with regard to the Spacecraft Optical Reference Axes.

2.3. Mission Description

2.3.1. Mission Timeline

Five mission phases have been defined (Figure 2.3.1/1). They are the LEOP, Cruise, Mars Orbit Insertion, Working Orbit Acquisition and Science phases. An optional phase, called the Aerobraking phase, is also considered. Its purpose is to reach a working orbit with a smaller apocentre, not reachable with the chemical propulsion system under the current mass constraints.

- The Launch and Early Orbit Phase (LEOP) begins at the start of the launch countdown and ends after successful completion of Earth acquisition.

- The Cruise phase corresponds to the ballistic orbit starting immediately after Earth acquisition and ending at Mars arrival, before Mars capture manoeuvres and allowing for some instrument calibration activities.

- *The potential lander ejection phase (details depending on its definition)*

- The Mars Orbit Insertion (MOI) phase starts from the initiation of the orbit insertion sequence and ends after successful completion of the spacecraft braking. The main activity consists in a trajectory correction with a large ΔV.

- The Working Orbit Acquisition (WOA) consists in a series of manoeuvres to acquire the final orbital plane and reduce the apocentre to reach the final orbit. This phase starts after the braking manoeuvre at first pericentre (MOI) and ends when the spacecraft has reached the Working Orbit and is then declared ready for science data acquisition and transmission.

- The Science phase starts once the S/C has successfully reached the final orbit and lasts nominally two Martian years (1374 Earth days). It consists in science data acquisition from the different payloads, storage and transmission of these data to the Earth.

- The Aerobraking phase is an alternative phase to WOA for apocentre reduction. It can also be used at the end of the nominal mission to decrease the apocentre altitude for communication with US landers. *The Aerobraking phase is presently optional.*

Timeline	Phase	Activities
Day 1	LEOP	Launch window : May/June 2003. Mars Express S/C is mounted atop the Fregat stage of the Soyuz launcher. Rate reduction, Sun acquisition and then Earth acquisition is performed short after separation from launcher. The S/C communicates with the Earth with the LGA.
Day 2 to month 7	CRUISE	After injection into the escape orbit and S/C initialisation, preparation for the injection error correction starts (the launcher upper stage shall not be put on a Mars impact trajectory). This first manoeuvre (TCM-1) is performed 2 days after launch by thrusters and possibly the main engine, under real time Earth monitoring through LGA. A second orbit correction manoeuvre (TCM-2) occurring about 2 months after launch serves as a correction to TCM-1. Switch from the LGA to the HGA is performed about 5 days after launch (S/C to Earth distance is about 0.01 AU). Then the S/C maintains communications via the HGA by autonomous Earth tracking (Earth direction known from onboard ephemeris and S/C position known by ranging and tracking), excepted during mid-course manoeuvres performed without HGA coverage. One planetary conjunction takes place around day 85 after launch, interrupting the command access to the S/C during 6 days.
	LANDER EJECTION	*Details will depend on the lander definition*
Day 207 after launch	MARS ORBIT INSERTION	The insertion on the operational orbit is performed thanks to a large braking manoeuvre of 767 m/sec at first pericentre of the arrival hyperbola. This manoeuvre decelerates the S/C below the Mars escape velocity and places it on a highly elliptical orbit. The pericentre of the arrival hyperbola is selected at about 280 km altitude, i.e. low enough to minimise the required ΔV and high enough for safety reasons (trajectory errors and atmosphere avoidance). The targeted apocentre altitude is 130 000 km.
Day 207 to day 215	WORKING ORBIT ACQUISITION	The final working orbit is acquired by means of four successive manoeuvres : • An apocentre manoeuvre (187 m/sec) is implemented to acquire the adequate orbital inclination and tune the pericentre altitude. • Three pericentre manoeuvres (624 m/sec total) are performed to decrease the apocentre to the final altitude.
From day 215 to day 1590	SCIENCE	The orbiter is operating for 2 Martian years (1374 days) on its remote science and data relay orbit. The nominal orbit (so-called Z) has a period of 4.88 hours with a 280 km pericentre altitude, 6780 km apocentre altitude and a 90 deg inclination. Visibility with Perth ground station is 8 hours per day (including sharing with other satellites). Two Earth opposition periods occurring around day 245 and day 1016 after Mars Orbit Insertion will interrupt the communications with Earth for about 30 days each, during which the science operations will have to be reduced. An Earth conjunction around day 666 after Mars Orbit Insertion will affect the command link for about 8 days, preventing for confirmation of successful reception or up-link of new science plan during this period. End of nominal mission is planned 2 Martian years after Mars Orbit Insertion.
	AEROBRAKING (option)	*This phase concerns aerobraking and is an alternative to the Working Orbit Acquisition phase to reach the working orbit, or decrease the apocentre at the end of the mission for communication with US landers.*

Figure 2.3.1/1 : Overall mission timeline overview

2.3.2. Mission Phases and Spacecraft Modes

The Mars Express mission is achieved by three main operational modes (Figure 2.3.2/1), dedicated to Mars observation, RF communications and trajectory correction (with or without a main engine). These modes are completed by a Sun Acquisition Mode, a Earth Acquisition Mode and a Survival Mode, as well as a dedicated mode used during optional aerobraking activities.

The **Sun Acquisition Mode** and **Earth Acquisition Mode** are used for the initial attitude acquisition sequence or in case of failure. First, a robust Sun Acquisition is performed just after launch and solar array deployment. The Earth is then acquired thanks to the autonomous star pattern recognition. After completion of Earth acquisition, a switch to the Communication Mode can be performed. During Mars Orbit Insertion, the immediate switch to an attitude re-acquisition is prohibited (both the Sun Acquisition and Earth Acquisition modes cannot be entered during the MOI phase).

The **Communication Mode** is entered after Sun/Earth acquisition has been completed. This mode is used during almost the whole Cruise Phase, except when orbit corrections are performed (Orbit Control Mode. The Communication Mode is optimised for the transmission of all the scientific data to the Earth and for the TC reception (configuration, mission plan, etc.). An inertial attitude is selected to point the HGA towards the Earth and the Solar Array towards the Sun. In LEOP and at the beginning of the Cruise phase, the LGA is used instead of the HGA.

Both the **Fine Orbit Control Mode** and **Main Engine Boost Mode** perform the required trajectory corrections during the overall mission to acquire and maintain the final orbit. The Fine Orbit Control Mode is based on 10 N thrusters only, while the Main Engine Boost Mode also considers the 400 N main engine. The main engine is used for large ΔV during the Mars Orbit Insertion phase, the Working Orbit Acquisition phase and possibly during the Cruise phase providing the required ΔV is not too small (using the main engine in the Cruise phase is useful for main engine tests before MOI and for calibrating both the main engine and the accelerometers). The hardware configuration and FDIR principles depend on the criticality of the phase. An orbit control phase starts by a slew manoeuvre to orient the thrust in the adequate direction and ends with the reverse manoeuvre to point the S/C towards the Earth (Communication mode), so as to restore the link with the Earth.

The Science Phase on the final Mars orbit is spent either in the **Mars Observation Mode** near the pericentre or in the **Communication Mode** in the other part of the orbit for data down-link to the Earth. This phase can also includes orbit maintenance if required. The Mars Observation Mode is the operational mode used for the payload instruments operation. The S/C is pointed to the Mars nadir direction but could also follow a specific profile defined from ground. This mode starts and ends with slew manoeuvres for transition to and from Communication mode (inertial attitude).

A **Slew Manoeuvre Mode** is implemented to perform all the S/C attitude manoeuvres with the reaction wheels, prior to and after any orbit correction and twice per Mars orbit during the Science phase, to switch the S/C attitude from the Observation mode to Communication Mode and vice-versa.

The **Survival Mode** is triggered in emergency situations after several failures or unforeseen events. It is entered from any mode either autonomously upon failure detection or upon TC reception. During Mars Orbit Insertion, the immediate switch to the Survival Mode is prohibited.

The **Braking Mode** is used in the optional Aerobraking phase. It consists in pointing the anti-payload face along the S/C velocity direction. Aerobraking consists in using the Mars atmospheric drag to reduce the S/C velocity near the pericentre, thus leading to an orbit apocentre reduction. This technique can be used to reach a working orbit with a smaller apocentre (that cannot be reached with chemical propulsion under the current mass constraints). This mode can be envisaged in case of failure during Mars Orbit Insertion to retrieve a backup working orbit, and also if a satellite extra mass, such as an attached lander, is to be considered.

2.3.3. Mission Phases Description

2.3.3.1 Launch and Early Orbit Phase (LEOP)

The Launch and Early Orbit Phase (LEOP) begins at the start of the launch countdown and ends after successful completion of Earth acquisition.

The launch window extends from 21/05/2003 to 24/06/2003 which corresponds to arrival date (Mars Orbit Insertion) from 15/12/2004 to 15/01/2004.

The nominal launch date assumed in the Definition Study is 01/06/2003 at 19:10 UT (Greenwich mean time), leading to an arrival date on 25/12/2003 at 21 : 30 UT.

Injection of the Mars Express S/C onto the Mars transfer orbit is performed by the Soyuz launcher using the Fregat upper stage. The launcher directly places the S/C onto an escape hyperbolic orbit at injection. Upon detection of separation from the launcher, a fully automated sequence is initialised.

	S/C modes								
Mission phases (in sequence)	Sun Acquisition	Earth Acquisition	Com munication	Fine Orbit control	Main Engine Boost	Mars Observation	Slew Manoeuvre	Survival	*Ejection / Braking (options)*
LEOP	*	*						* (1)	
Cruise	* (2)	* (2)	* (3)	* (3)	* (5)		*	* (1)	
Mars Orbit Insertion	**prohibited**	**prohibited**			*		*	**prohibited**	
Working Orbit Acquisition	* (2)	* (2)	*		*		*	* (1)	
Science	* (2)	* (2)	* (4)	* (4)		*	*	* (1)	
Lander Ejection / Aerobraking (options)	** (2)*	** (2)*	** (4)*	*			*	** (1)*	*

(1) Upon multiple failure or unforeseen events.
(2) In case of anomaly requiring attitude re-acquisition.
(3) Except during conjunction (TC is not available).
(4) Except during conjunction (TC is not available) and opposition (TC and TM are not available).
(5) TBC : can be useful for main engine test and calibration for Mars capture.

Figure 2.3.2/1 : Mission phases vs. S/C modes

2.3.3.2 Cruise Phase

The Cruise phase corresponds to the ballistic orbit starting immediately after successful completion of Earth acquisition and ending at Mars arrival, just before Mars capture. The Mars Express cruise orbit with the main events is illustrated on Figure 2.3.3/1.

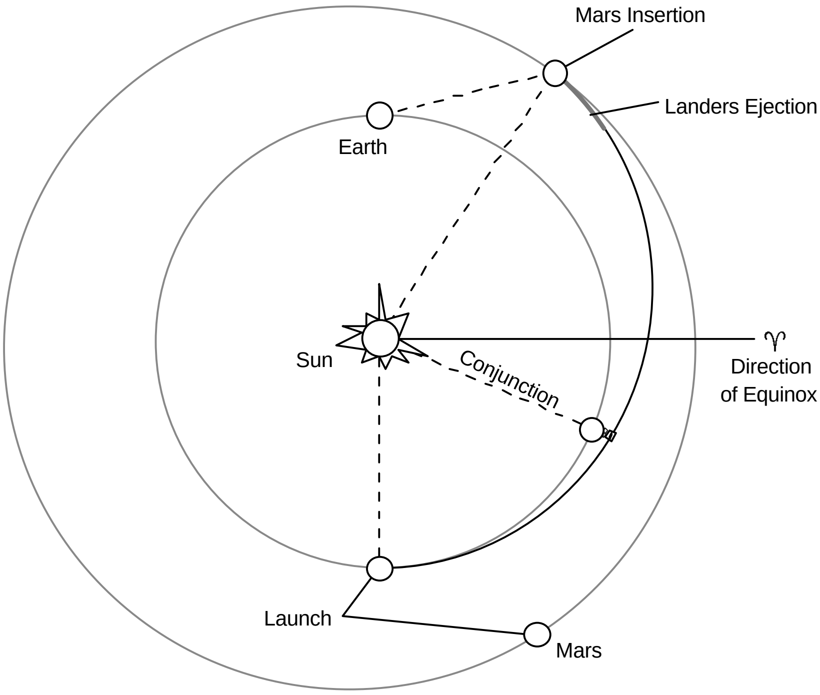


Figure 2.3.3/1 : Mars Express Cruise orbit

One planetary conjunction takes place around day 85 after launch interrupting the command access to the S/C for about 6 days.

The Cruise phase is dedicated to commissioning activities, and mainly consists in a complete subsystems and payload checkout and calibration.

No cruise science is foreseen. Nevertheless, if required, instruments can be operated according to needs to be defined (e.g. calibration, performance verification).

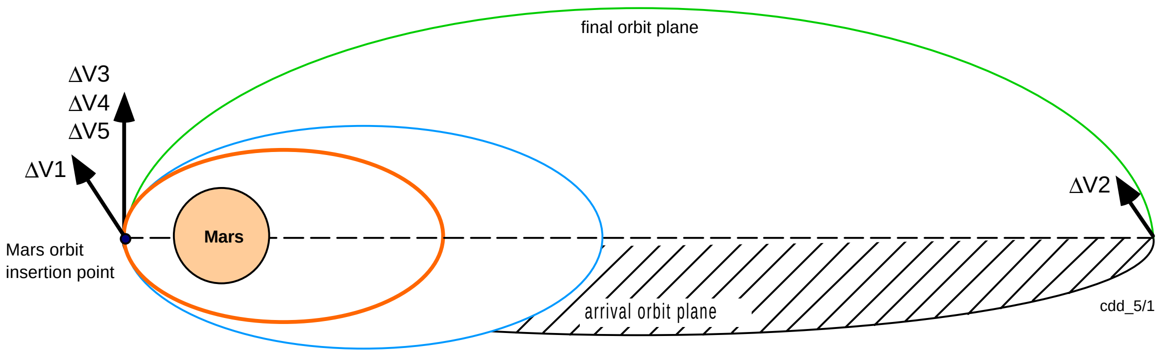
2.3.3.3 Mars Orbit Insertion and Working Orbit Acquisition Phases

The Mars Orbit Insertion (MOI) phase and the Working Orbit Acquisition (WOA) phases are the period of transition from the interplanetary trajectory (Cruise phase) to the final working orbit around Mars, where the Science phase starts. The MOI phase starts when arriving at Mars (end of the Cruise phase) and is followed by the Working Orbit Acquisition phase which ends when the satellite is declared ready for science data acquisition and transmission.

The insertion on the operational orbit is realised with a series of four manoeuvres (Figure 2.2.3/2) performed with the main engine :

- A first braking manoeuvre of about 767 m/sec (TBC) around the pericentre of the arrival hyperbola decelerates the S/C below the escape velocity and places it on a highly elliptical orbit around Mars. The pericentre of the arrival hyperbola is selected at 280 km altitude, i.e. low enough to minimise the required ΔV and high enough for safety reasons (trajectory errors and atmosphere avoidance). The target apocentre altitude is 130 000 km.
- An apocentre manoeuvre is then implemented to acquire the adequate orbital inclination and to tune the pericentre altitude. This manoeuvre requires about 187 m/sec.
- Three pericentre manoeuvres are then performed to decrease the apocentre to the final height. These manoeuvres require a total of about 624 m/sec.

Due to the high criticality of the MOI phase for the rest of the mission, all the payloads are OFF during this phase.



ΔV1 : first braking manoeuvre

ΔV2 : orbital plane turn manoeuvre

ΔV3 : first pericentre manoeuvre to reduce apocentre

ΔV4 : second pericentre manoeuvre to reduce apocentre

ΔV5 :third pericentre manoeuvre to reduce apocentre

Figure 2.3.3/2 : Mars Orbit Insertion Manoeuvres and Working Orbit Acquisition

2.3.3.4 Science Phase

The reference Mars orbit for the definition study is the so-called Z orbit. It has been computed by ESOC and results from a compromise between science mission requirements, launcher performances and S/C design constraints. The Mars orbit is an eccentric polar orbit with a period of 4.88 hours, a mean pericentre of 280 km and an apocentre of 6780 km.. Its inclination is about 90 deg, implying that the orbital plane remains almost inertial.

The Z orbit offers an interesting property for Mars observation : the rotation of the orbital plane and the precession of the line of nodes are phased with the rotation of Mars relative to the Sun. The relationship relating the first time derivative of the right ascension of the ascending node Ω and the argument of pericentre ω is $\dot{\Omega} - \dot{\omega} = 2.\omega_M$ where ω_M is the Mars orbital frequency around the Sun ($\omega_M = 10^{-7}$ sec^{-1}).

The Science phase starts once the S/C has reached the Working orbit and is declared ready for science operation, and lasts two Martian years (1374 Earth days). It consists in science data acquisition from the different payloads, storage and transmission of these data to the Earth.

The timeline for the science phase around Mars has for primary objectives the determination of the best sharing of orbiter resources between payloads. Such sharing will be detailed once the payload sciences objectives will be accurately defined and the payload instruments selected.

As illustrated on Figure 2.3.3/3, each orbit is split into a Mars nadir-pointed observation phase, during which instruments science data are collected, and an Earth-pointed communication phase, during which data are down-linked to Earth at high rate. These two phases require two S/C attitude manoeuvres per orbit.

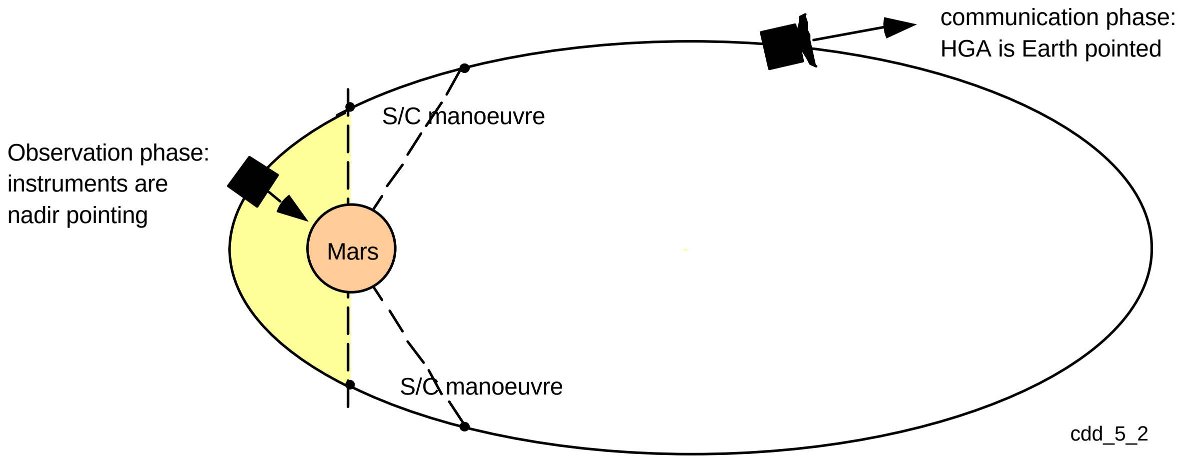
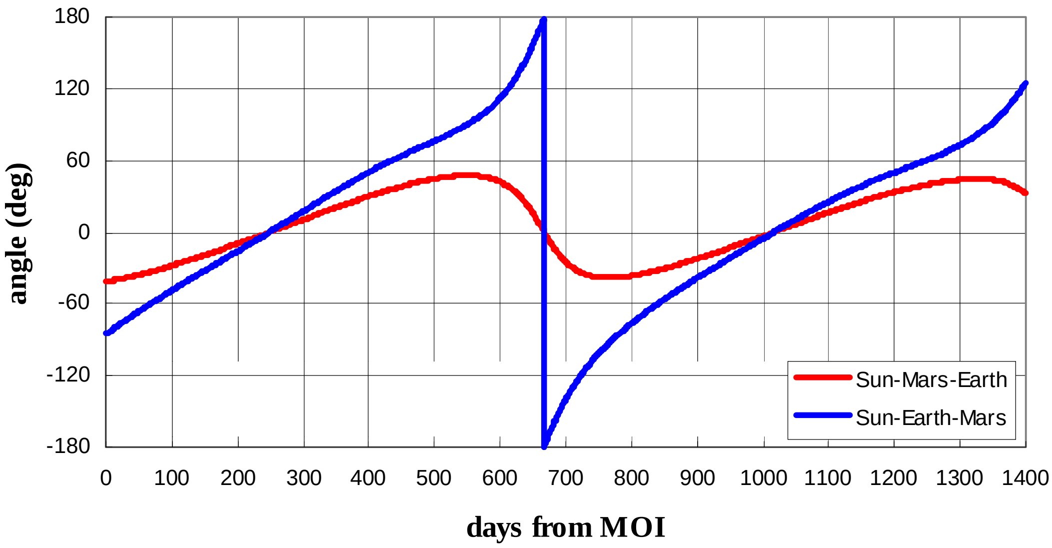


Figure 2.3.3/3 : Single Mars orbit time-line

As shown in Figure 2.3.3/4, one solar opposition and one solar conjunction occur during the first Martian year, leading to RF communication blackout periods of several days. A solar opposition is characterised by a small Sun - Mars - Earth and Sun - Earth - Mars angles (typically 3 deg) and a solar conjunction by a small Sun - Mars - Earth angle and by Sun - Earth - Mars crossing 180 deg.

- An Earth opposition occurs around day 248 after MOI leading to the interruption of Earth communication for 30 days. During this period, both TM and TC links are subject to blackout. Therefore, it is proposed that the main mission be suspended, since no data can be down-linked to ground, but still to acquire landers data, which will be down-loaded to Earth at the end of the blackout period. In both cases, the autonomy management will guarantee that the S/C enters the Survival Mode in case of failure, with no need for ground intervention.
- An Earth conjunction occurring around day 666 after MOI and lasting about 8 days will affect the command link, preventing for confirmation of successful reception or up-link of new science plan during this period. During this period, no TC can be sent to the S/C due to Sun illumination of the HGA, but TM remain available. A master time-line defined for a duration of TBD days will allow nominal science operations during this period.
- A second solar opposition occurs around day 1016 after MOI, from day 1001 to day 1032, leading to the interruption of Earth communication for 30 days.



Solar opposition occurs around day 248 (first Martian year) and day 1016 (second Martian year).

Solar conjunction occurs around day 666 (first Martian year).

Figure 2.3.3/4 : Sun-Earth-Mars conjunction and opposition during the 2 Martian year Science phase

As shown in Figure 2.3.3/5, visibility with the Perth ground station varies from about 5 hours to 12 hours per day, with a mean value around 9 hours. Nevertheless, due to ground sharing with other satellites (e.g. ROSETTA), the visibility period to be taken into account during the Science phase is 8 hours per day.

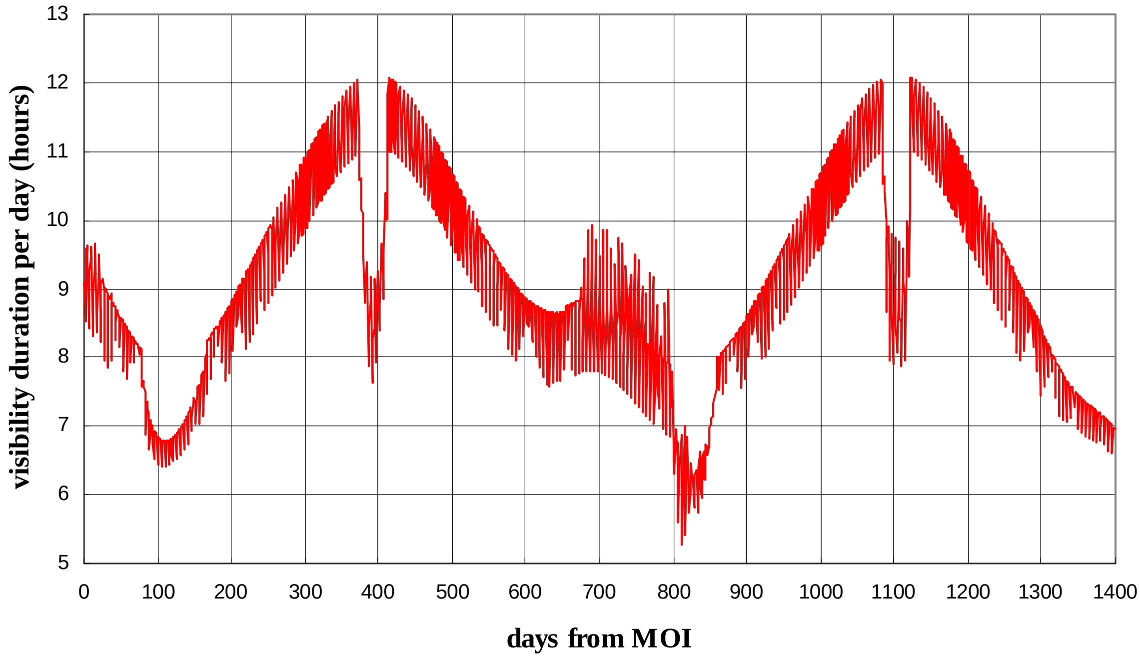
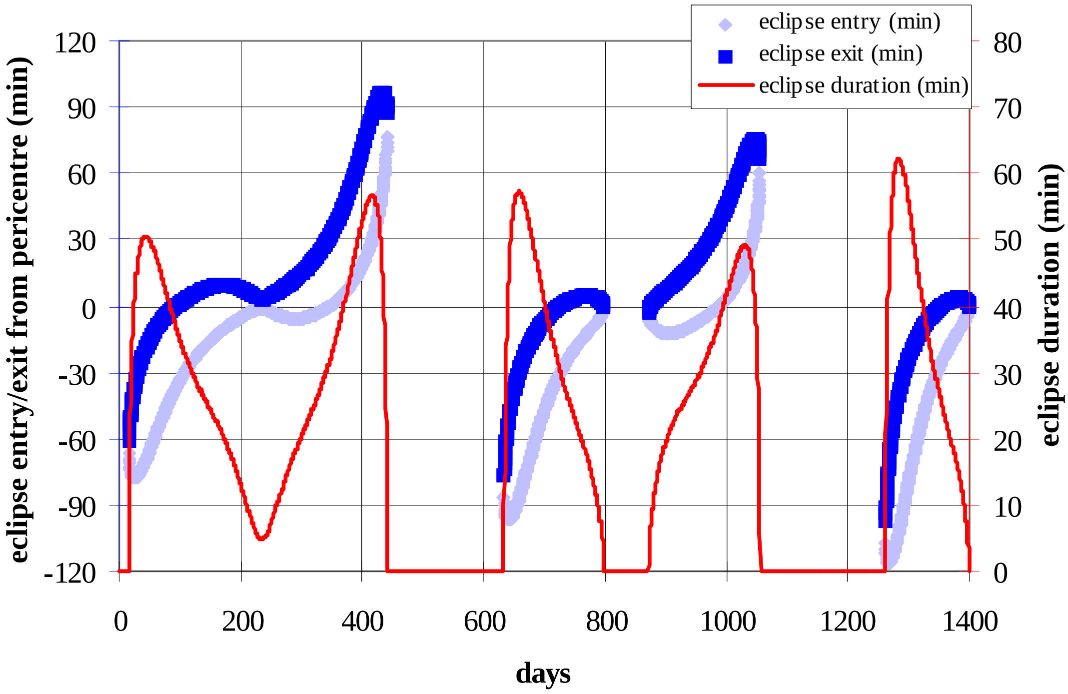


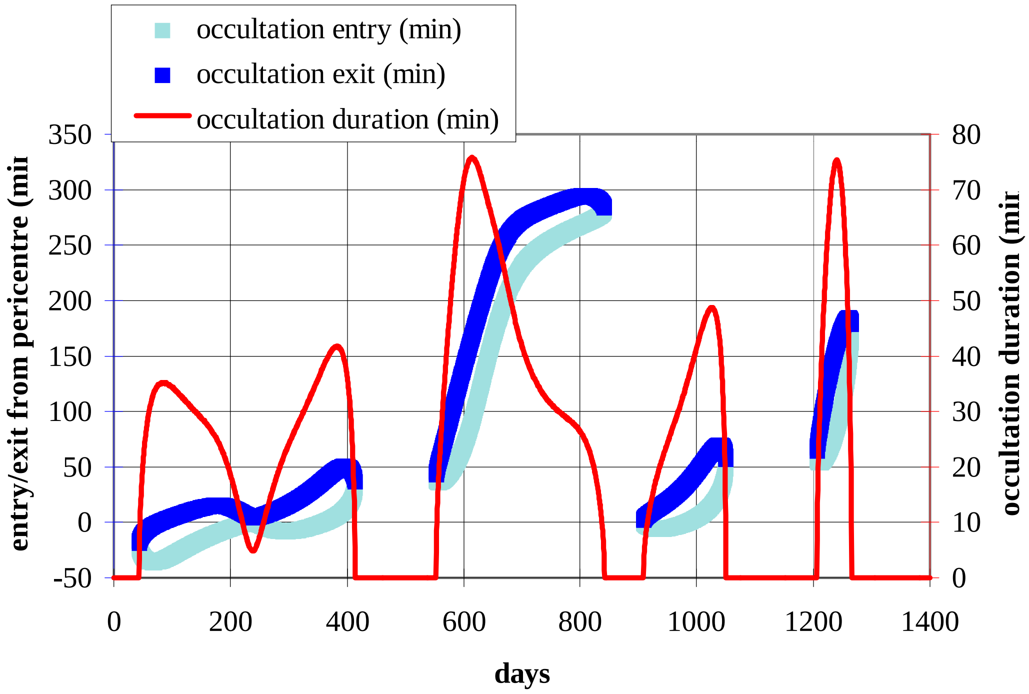
Figure 2.3.3/5 : Perth station visibility during the Science phase

Earth and Sun occultation periodically occur during the Science phase as shown in Figure 2.3.3/6 and Figure 2.3.3/7. These periods induce constraints on payload operations, RF communication and power delivery.



Eclipse Period	Starting day	Ending day	Duration (days)	Localisation
1	15	441	427	around pericentre
2	634	798	165	near pericentre
3	871	1056	186	near pericentre
4	1262	1399	138	all along one apocentre to pericentre path

Figure 2.3.3/6 : Eclipse period entry/exit and duration for the Mars Working orbit



Occultation Period	Starting day	Ending day	Duration (days)	Localisation
1	43	413	371	about pericentre
2	552	842	291	all along the orbit
3	909	1050	142	close to pericentre
4	1206	1265	60	close to apocentre

Figure 2.3.3/7 : Earth occultation period entry/exit and duration for the Mars Working orbit

2.3.3.5 Aerobraking Phase (option)

The Aerobraking phase is an optional phase that takes place after Mars Orbit Insertion as an alternative to chemical propulsion to reach the final orbit, and/or at the end of the nominal mission, to decrease the apocentre altitude for communication with US landers.

Aerobraking is a technique for reducing the apocentre by using the Mars atmosphere drag to reduce the S/C velocity near the pericentre.

2.3.3.6 Lander Ejection Phase (option)

TBD

2.4. Ground Operations Before Flight

2.4.1. General

The on-ground lifetime includes the integration and test programme at S/C level, any potential storage period, and the pre-launch operations.

Specific activities (e.g. refurbishment) after on-ground storage can be required to guarantee the in-orbit compliance to the functional and performance requirements of the instruments. These have to be defined in the P/L User Manual (for use in ground operations).

2.4.2. Spacecraft Level Integration and Validation

2.4.2.1. General Sequence

After a specific incoming inspection, each instrument will be subjected to various tests throughout the integration and test campaigns on the spacecraft.

As a general rule, these instrument integration and test activities shall satisfy different achievements :

- Mechanical and electrical assembly on the spacecraft
- Alignment (and related measurements)
- Interface thorough verification (mainly electrical, but thermal as well for instruments relying on spacecraft level thermal control)

Instrument proper performance / health verification on the spacecraft

All the integration and test operations involving the instruments shall be discussed between the PI's and the Spacecraft Contractor, together with the appropriate conditions and requested facilities.

To this respect, it is baselined :

. to perform S/C AIT under class 100,000 (Fed-Std-209E) in cleanliness controlled environment (contamination samples will be periodically measured) or in using specific protections around it if this cleanliness class is not guaranteed

. to run ambient pressure integration and test activities in various conditions (e.g. wrt gravity) including those flight ones for which the instrument dissipative equipments are confined in the nominal flight volume (no ventilation system is anticipated at spacecraft level)

. to use high class nitrogen for chamber refilling after thermal tests (or insure specific refilling capabilities for specific instruments if judged more practical)

. to run environment tests without specific measurement transducers, i.e. with no others than those nominally requested through spacecraft level analyses.

The resulting contamination level at instrument level will be then kept lower than TBD.

2.4.2.2. General Requirements

The instruments shall be provided with the specific GSE and manpower support necessary to integrate, operate and test them.

All the specific conditions related to the post-delivery activities (e.g. any orientation constraint wrt gravity for mechanism release, any precaution related to human body ESD...) - including hazards - shall be detailed in the P/L User Manual.

The instruments, under their ground operation conditions, shall be able to withstand transport loads when integrated on spacecraft.

2.4.3. Requirements on Payload Ground Support Equipments

2.4.3.1. Spacecraft Simulator

A Spacecraft Simulator can be provided (on request) to each PI whose instrument interfaces comply with the standard electrical interfaces.

Details of this simulator are TBD.

2.4.3.2. Specific Payload Ground Support Equipments

Payload Ground Support Equipments shall satisfy classical ESA programme rules.

Their interfaces shall comply with those of the S/C Ground Support Equipments.

Details of these interfaces are TBD.

2.4.4. Pre-Flight Simulation

During the pre-flight phase, ESOC performs its final simulation programme including the validation of both the Flight Operation Plan and the mission control system.

2.4.5. Launch Operations

The launch campaign ground conditions are TBD.

The SOYOUZ rocket is potentially transported horizontally up to the launch pad.

All instruments will be off during launch : payload electronics are not powered.

2.5. Launch Environment

2.5.1. Launch Mechanical Environment

The payload will see mechanical environment as defined in following sections (based on ESA PSS and SOYOUZ User Manual, TBC).

2.5.1.1. Sine Vibration

Axis	Frequency (Hz)	Level
All	5/22.75	± 12 mm
	22.75/80	25 g
	80/100	10 g
	100/150	2.5 g

Figure 2.5.1/1 : Sine Vibration Environment (TBC)

2.5.1.2. Random Vibration

For units with mass ≥ 50 kg

Axis	Frequency (Hz)	Level
All	20 - 100	+ 3 dB/oct
	100 - 400	0.07 g^2/Hz
	400 - 2000	-3 dB/oct

For units with mass ≤ 50 kg

Axis	Frequency (Hz)	Level
All	20 - 100	+ 3 dB/oct
	100 - 400	$0.05 * \frac{M+20}{M+1}\ g^2/Hz$
	400 - 2000	-3 dB/oct

Figure 2.5.1/2 : Random Vibration Environment (TBC)

2.5.1.3. Acoustic Vibration

Octave band (Hz)	Flight level

(Center frequency)	(dB)
31.5	129
63	134
125	140
250	138
500	137
1000	130
2000	120
4000	118
Overall level (Ref. : 0 dB = 2 x 10^{-5} Pa)	144

Figure 2.5.1/3 : Acoustic Vibration Environment (TBC)

2.5.1.4. Shock

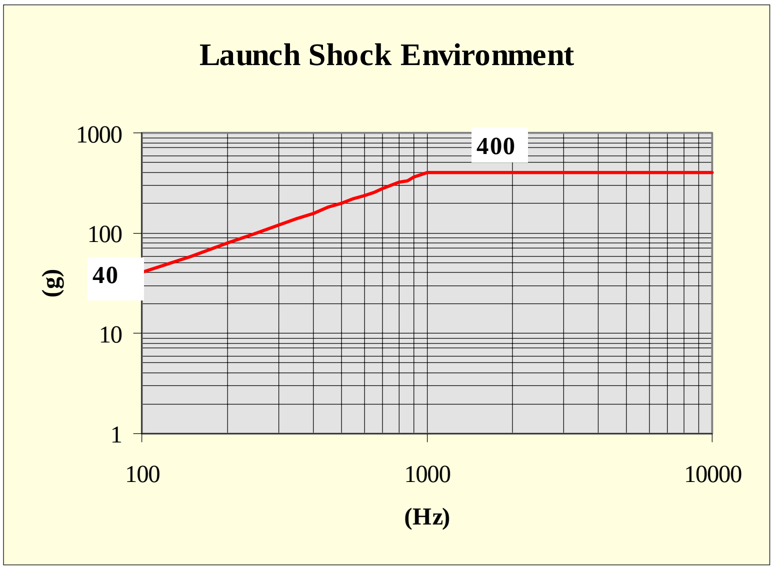


Figure 2.5.1/4 : Launch Shock Environment (TBC)

2.5.1.5. Quasistatic Loads

Quasistatic (Limit) Loads are defined as those loads that the experiment may be subjected to during its life time with a certain probability level.

Limit Loads as defined shall be considered :

- Applied at the experiment C.o.M.,
- Acting simultaneously along the 3 axes of the experiment.

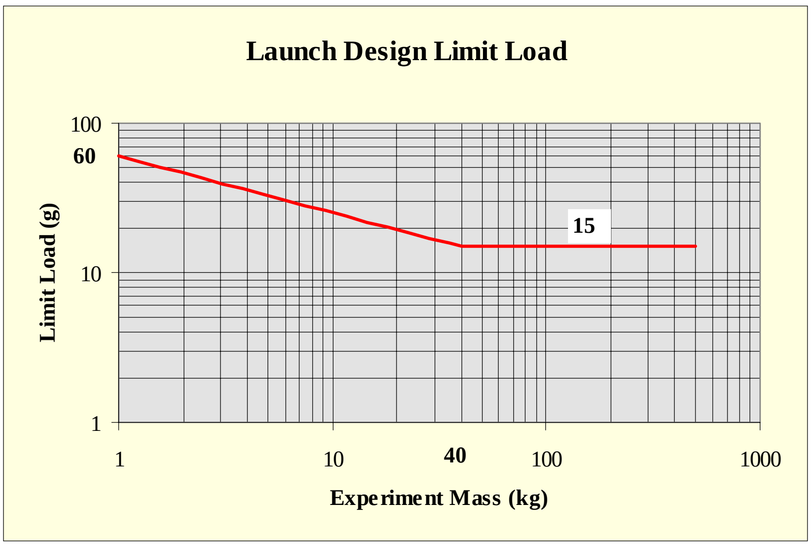


Figure 2.5.1/5 : Design Limit Load (TBC)

2.5.2. Launch Pressure / Thermal Environment

2.5.2.1. Depressurisation

The typical variation of the static pressure within the Launcher Payload Compartment volume is shown in figure 2.5.2/1 (from Soyuz User Manual, TBC).

The maximal slope does not exceed 2000 Pa/sec.

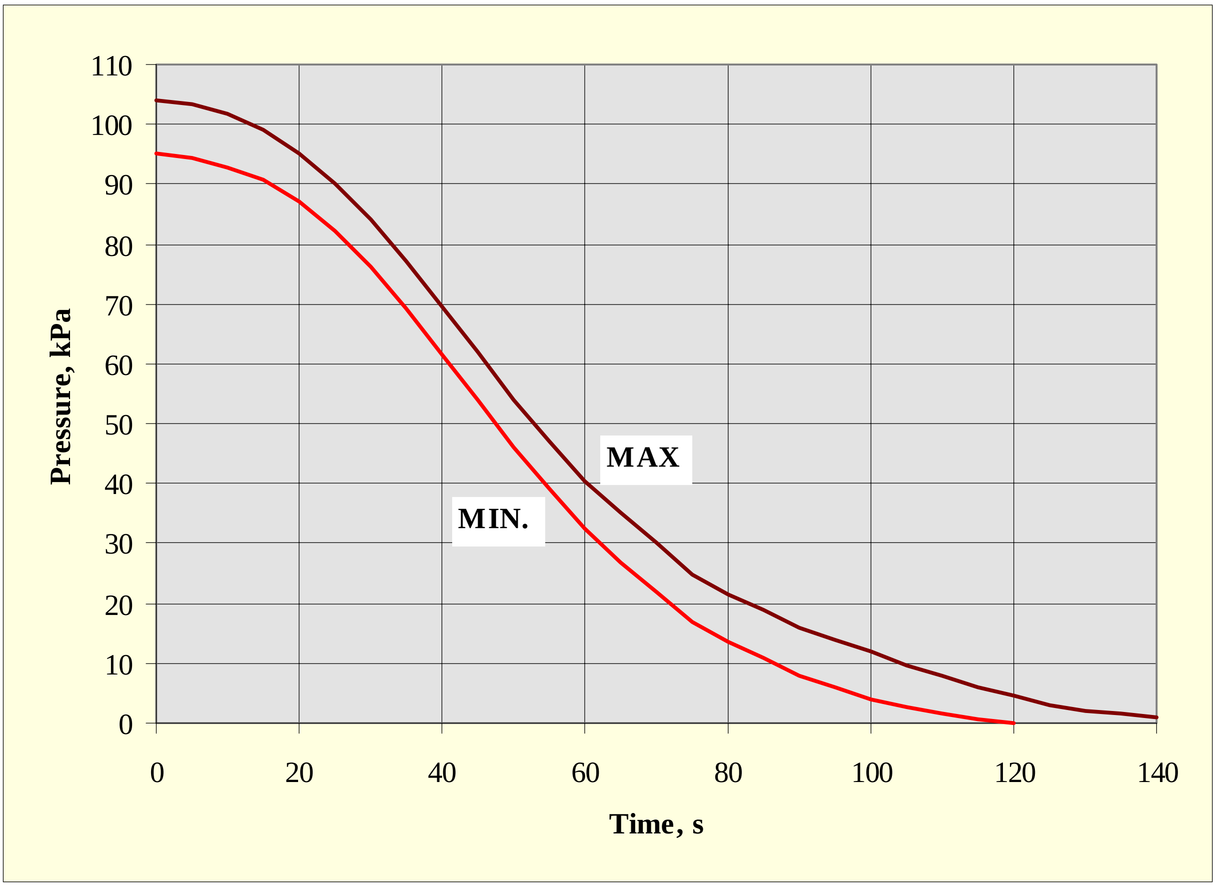


Figure 2.5.2/1 : Static Pressure Variation (TBC)

2.5.2.2. Thermal Flux

The thermal flux will be (from Soyuz User Manual, TBC) :

- Under fairing < 800 W/m2 isotrope, during 3 min (TBC).
- After fairing jettison < 1135 W/m² on a plane perpendicular to Zb axis, during 7 min (TBC).

  + Earth albedo and thermal radiation.

2.6. Flight operations concept

The Mars Express mission features the specific autonomy and operations requirements of deep space planetary missions with time critical operations. The mission profile imposes specific spacecraft functions and performances in terms of operations, operability and commandability mechanisms, and failure management.

The Mars Express mission is characterised by a variable communication round trip delay with Earth, several periods without communications (no TC during planetary conjunction, no TM and no TC during planetary opposition with the Sun and Sun plasma in between) and one critical phase, namely the Mars Orbit Insertion phase or the lander ejection (if confirmed), requiring a high level of autonomy. The operations during Mars Express mission will have to take into account general requirements linked to ESOC operations and specific requirements due to the deep space environment:

The mission operations (including LEOP) of the Mars Express mission will be conducted by ESA/ESOC facilities and staff. ESOC facilities will support all required elements of training and mission preparation, mission control, instrument planning and operations, and generation of specific measurement data products. The spacecraft will be compliant with the operational procedure of ESOC.

Command and control of the Mars Express satellite during nominal operations and science data acquisition will be performed by the Perth 32 m ground station (or Sardinia if available). Additional stations (TBD) will be used in the early phase of the mission and during critical phases.

The S/C to Earth distance varies from approximately 0.5 to 2.5 AU. Therefore, the resulting communication round trip travel time with Earth excludes closed loop control operations for time critical phases neither nominal such as orbit control nor after an on-board failure. The large distance to Earth also drives the transmission rate, and leads to relatively accurate antenna pointing requirements, combined with data storage capabilities and possibly data compression facilities.

The Mars Express mission generates various modes for the spacecraft, payload operation oriented, communication oriented or trajectory control oriented. The spacecraft will have a sufficient autonomous capability to manage these modes :

- The operations of the orbiter during the Science phase will be possible using either direct communication with the Earth or via on-board intelligence and suitable on-board data storage followed by periodic data transfer to Earth.
- During the Cruise phase, several orbit correction manoeuvres will be performed, and some operations may be done on the payloads (status checks, calibrations, etc.). The criticality of the insertion manoeuvre is high since an interruption of the manoeuvre or a bad execution of the ΔV, resulting from a failure, would lead to a loss of the mission. No cruise science is foreseen.

In order to fulfil the mission objectives and support the overall autonomy and operation concept, the spacecraft features the capability to perform autonomously a complete sequence of operations once initiated and the capability to stay in a safe state during periods of several days or several weeks.

In case of failures, the autonomy requirements include an extensive capability of failure passivation and a capability to ensure the same functions after the failure or to continue the same operations without interruption.

As illustrated in Figure 2.6/1, the long duration of the link with the Earth, and the existence of black out periods require on on-board capability to perform complete operational sequences autonomously or to stay in a safe state during several weeks.

Subject or Event	Constraint	Autonomy requirement
S/C to Earth distance Link duration	0.5 to 2.5 AU up to 40 min (round trip)	Ability to perform complete autonomous operational sequences
Attitude manoeuvre with a fixed HGA	Link interruption	
Opposition phase Earth - Sun - Mars	No TC and no TM	Ability to continue the mission or stay in a safe state
Conjunction phase Sun - Earth - Mars	No TC	Ability to continue the mission or stay in a safe state

Figure 2.6/1 : Link characteristics and autonomy need

2.7. Attitude and Orbit Control System

AOCS basic concepts

The AOCS for Mars Express has to ensure a high pointing stability performance to the payloads for the Mars observation like an Earth observation satellite, but also to provide a good accuracy on the ΔVs for the Mars insertion manoeuvre and the final orbit acquisition. Some phases like the Mars insertion manoeuvre require also a high level of autonomy.

The basic concepts selected for Mars Express are the following :

- The selection of a three axes stabilised control during the whole mission increases the operational flexibility : attitude manoeuvres dedicated to the payloads are easier,
- The use of Star Trackers enables to cover the nominal operations of almost all the mission phases : this solution offers a great flexibility for the science mission which can be optimised and defined very late without changing the basic AOCS concepts,
- The use of wheels, Star Trackers and gyros during the observation phase allows to ensure to the payloads the best pointing performances,
- The wheels sizing is defined to enable a great number of AOCS operations with the wheels such as the attitude manoeuvres between the observation and the communication phase.
- The use of the propulsion is mainly limited to the main trajectory manoeuvres performed before the final orbit acquisition. On the final orbit around Mars, the propulsion is only used for the wheel off-loading, for the orbit maintenance from time to time if necessary, and in case of major failure leading to an attitude reacquisition or the Safe Mode.

AOCS typical performances

The following table provides the typical performances of the AOCS during operational phase on the final orbit. On board data concerning the attitude or the orbit are available for the payloads.

Attitude knowledge w.r.t a stellar direction	0.05°
Pointing accuracy w.r.t a stellar direction	0.06°
Attitude knowledge w.r.t the Nadir direction	0.12°
Pointing accuracy w.r.t the Nadir direction	0.15°
Rate stability	0.003°/s
Pointing stability over 10s	0.005°
Pointing stability over 60s	0.009°
On board orbit knowledge	4 km

AOCS mode logic

The AOCS modes logic results from the different attitudes to be ensured by the AOCS during the different phases :

- The selection of a fixed High Gain Antenna leads to define a communication mode during which the spacecraft attitude is such that the antenna is pointed towards the Earth, and the same mode is used at AOCS level.
- On the final orbit around Mars, two modes will be therefore used :

  the first one dedicated to the planet observation near the pericentre, during which the Spacecraft is basically Nadir pointed,

  and the second one dedicated to the communication near the apocentre as explained above.
- Other pointing directions can be envisaged on the final Martian orbit :

  - During the observation phase, a total manoeuver duration of 20 mn per orbit at more than 40deg from the nadir direction is currently considered as a maximum.

  - During the phase dedicated to the communications specific pointing directions could be ensured provided that they do not reduce significantly the capability of data transmission.

2.8. Flight environment

2.8.1. Flight Mechanical Environment

The payload will see mechanical environment defined in shock and dynamic loads as follows :

2.8.1.1. Spacecraft Appendage Deployment Shock

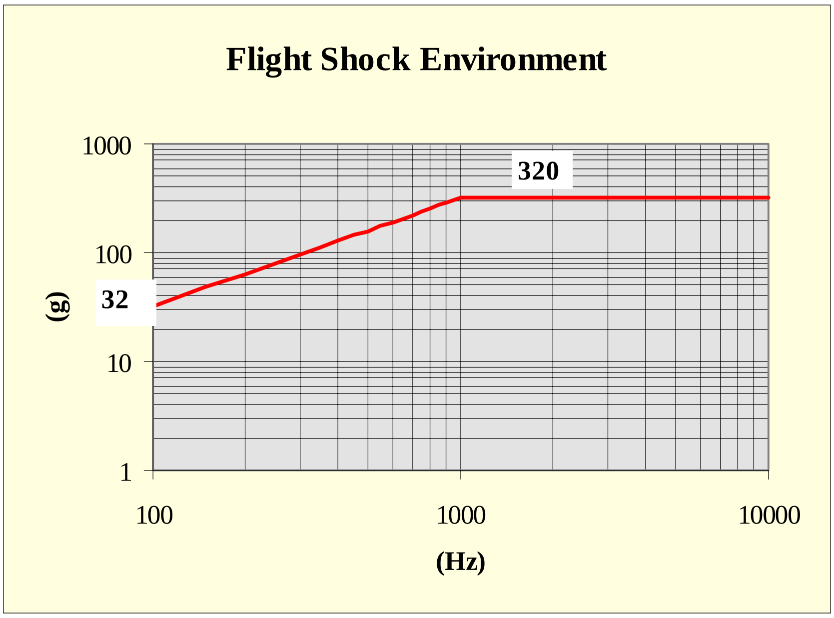


Figure 2.8.1/1 : Flight Shock Environment (TBC)

2.8.1.2. Low Frequency Disturbances Generated By The AOCS

The Payload design shall be compatible with the following dynamic environment generated by the AOCS at low frequency (< 10 Hz) when using the thrusters.

Perturbation Levels at S/C Centre Of Mass (*)	**Trajectory correction before final orbit acquisition**	**Initial attitude acquisition or reacquisistion**	**On final orbit**
Linear acceleration	0.9 m/s²	0.05 m/s²	0.05 m/s²
Angular acceleration	1 °/s²	1 °/s²	1 °/s²
Angular rate	0.15 °/s	5 °/s	0.12 °/s

Figure 2.8.1/2 : Flight Dynamic disturbances (low frequency, all numbers TBC)

(*) linear level at experiment location are increased by TBD participation of angular acceleration, depending of applicable experiment offset from S/C CoG.

2.8.1.3. High frequency disturbances (> 10 Hz) : micro-vibrations

The Payloads shall be compatible with the micro-vibrations generated by the Spacecraft, mainly due to the reaction wheels, and possibly other instruments. These disturbances are TBD today.

2.8.2. Flight Thermal Environment

2.8.2.1. Main Flight Phases

2.7.2.1.1. Cruise Phase and Lander Ejection

The cruise phase is defined by the end of the launch and the beginning of the Mars Orbit Insertion. During this period the Spacecraft will be in deep space; the payload will be off except during health check.

The thermal environment seen by the Spacecraft and the Payload is TBD, depending on the configuration, orientation in space and power state ; Any surface of the spacecraft may be shaded or directly exposed to Sun illumination for long durations.

Solar distances range between 1 AU (TBC) and 1.65 AU (TBC).

For thermal design purpose the following characteristics (solar aspect angles are presentlyTBD) shall be used :

	Solar Irradiance (W/m^2)	**Planet Albedo**	**Planet Emission (W/m^2)**
Earth	1290 to 1420	0.34	250
Mars	490 to 710	0.16	145

Figure 2.8.2/1 : Planet Thermal Environment

2.8.2.1.2. Mars Orbiting

The thermal environment seen by the Spacecraft and the Payload is TBD depending on the configuration, orientation in space and power state.

Any surface of the spacecraft may be potentially shaded or directly Exposed to Sun illumination for long durations.

2.8.2.2. Spacecraft Thermal Control

The Control of the Spacecraft internal thermal environment throughout all mission phases is achieved by means of a combination of passive and active thermal control techniques. See Figure 2.8.2/2.

The basic idea for the Thermal Control System (TCS) is to insulate the Spacecraft from the external environment by means of classical Multi Layer Insulation (MLI).

Radiator areas are, however, envisaged on the +Z floor and the lateral panels with the aim to reject the heat generated by high dissipative and/or high power density spacecraft units.

The radiator areas are covered (whenever required) with white paint, Secondary Surface Mirror or Optical Solar Reflectors (OSR) in order to reduce the potential impact from appendages reflected Sun heat flux.

All internal units shall be black painted to increase mutual radiative exchanges, hence decreasing thermal gradients. MLI will be applied to propellant and pressurant tanks of the Reaction Control System (RCS).

Electrical heaters are foreseen to maintain Spacecraft interior and/or specific units within the required limits.

Finally, the TCS is supported by Spacecraft avionics to perform acquisition of the thermal telemetry data, control and surveillance of the electrical heaters.

2.8.2.3. Spacecraft Thermal Characteristics

Spacecraft MLI blanket outer layer IR emittance shall be 0.45 to 0.75 (TBC).

Spacecraft MLI blanket outer layer solar absorptance shall be TBD to TBD (preliminary 0.45 may be assumed).

Spacecraft MLI blanket effective emittance shall be 0.01 to 0.03 (TBC).

To be included after S/C contractor selection

Figure 2.8.2/2 : Spacecraft Thermal Preliminary Concept (TBD)

2.8.2.4. Definitions

Temperature Reference Point (TRP)

The Temperature Reference Point is defined at the mounting interface between the experiment unit and the Spacecraft on the experiment unit side.

This is the conductive interface temperature experienced by the units. The Experimenter shall select the particular attachment point to which the Temperature Reference Point shall apply and identify it on the Thermal Interface Drawing.

Environmental Temperature

This is the predicted radiative internal and/or external temperature of the Spacecraft.

This is the radiative interface temperature experienced by the units.

Operating Temperature

This temperature shall be maintained at the TRP whilst the experiment is operating. This temperature shall be independent of experiment operating modes.

Operating Rate of Change

This temperature stability shall be maintained at the TRP over the operating temperature range.

Non-Operating Temperature

This temperature shall be maintained at the TRP when the experiment is not operating.

Unit Switch-on Temperature

The Experimenter shall specify if the non-operating temperature range at the TRP can prevail at the time of experiment switch-on or if an alternative temperature range must apply at switch-on.

Predicted Temperature Range

This is the nominal temperature range the unit may experience, taking into account the worst case combination of modes, environment and parameter degradation, excluding failure cases.

Design or Extreme Worse Case Temperature Range

This is the extreme temperature range the unit may experience, taking into account in addition uncertainties in parameters (like view factor, surface properties, contamination, radiation environment, conductance, dissipation).

Qualification Temperature Range

This is the extreme worst case temperature range (defined for the operating and non-operating mode of the unit) for which a unit is guaranteed to function nominally, fulfilling all required performances with the required reliability.

Acceptance Temperature Range

The acceptance temperature range, defined for the operating and non-operating mode of the unit, is obtained from the qualification temperature range after substraction of suitable qualification margin. This is the extreme temperature range that a unit may be allowed to reach, but not exceed, during all envisaged mission phases (based on worst case assumptions).

Interface Temperature Budget

The thermal interface is defined using two type of identified temperatures :

- Temperature Reference Point (TRP). The temperature at the TRP is the only one controlled and guaranteed by the S/C Thermal Control Subsystem.
- Environmental temperature. The environmental temperature range shall be used by the unit provider to perform its thermal design.

The temperature ranges of tables 2.8.2/1 and 2 correspond to an extreme worst case prediction.

The thermal interface is defined by the range of temperature in operating and non-operating mode and by the temperature rate of change in the operating mode.

	Ops	Non-ops
S/C Internal	-10 to 40°C	-20 to +50°C
S/C External	TBD to TBD °C	TBD to TBD °C

Table 2.8.2/1 : TRP Temperature Range Budget (TBC)

	Ops	Non-ops
S/C Internal	-10 to 40°C	-20 to +50°C
S/C External	TBD to TBD °C	TBD to TBD °C

Table 2.8.2/2 : Environment Temperature Range (TBC)

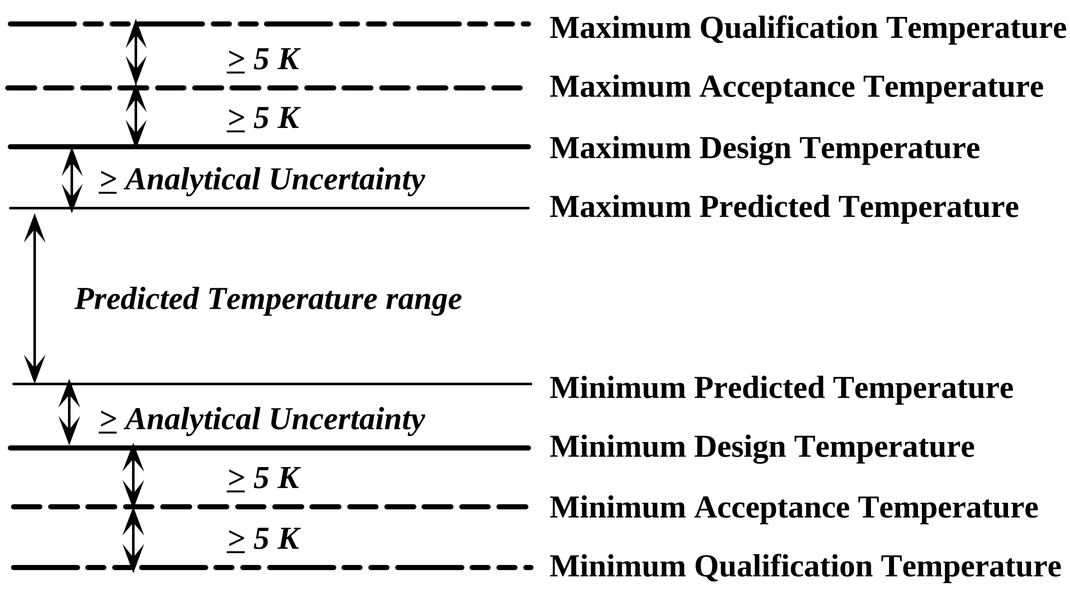


Figure 2.8.2/3 : Temperature Margin Policy

2.8.3. Mars Express Orbital Environment

During its 5-year mission, the Mars Express S/C will be subjected to ionised particles and meteoroids.

2.8.3.1. Ionising radiation environment

The payload shall be designed to withstand the effects of the varying flux of high energy particles encountered during the mission. Radiation effects can be summarised as radiation deposit dose and Single Event Upset (SEU) / Latch-up effects. The integrated dose can degrade the performance of the payload and possibly be the cause of failures. Radiation impinging on electronics circuits/detectors can produce an increase of the background noise. SEU and latch-up phenomenon are described in Section 6.7.

First, a transfer to Mars suppose crossing the Van Allen radiation belts. The Soyuz-Fregat launcher will place the S/C on the escape orbit to Mars so that the S/C will cross the radiation belts only once. Worst case radiation dose is received when crossing the South Atlantic Anomaly (SAA). The standard AE8MAX-AP8MAC trapped radiation environment models can be used to assess the radiation dose level received by the S/C during this mission phase.

During the cruise phase (outside the Earth magnetosphere), the whole S/C will be subjected to solar flares and Galactic Cosmic Rays (GCR).

- Solar flares are composed of high energy protons and heavy ions. A flare can last a few hours to a few days. The computation of solar flares energy spectrum for the 4.5-year Mars mission is performed with the Feynman model.

- Galactic Cosmic Rays are a continuous flux of high energy particles (heavy ions). The GCR spectrum can be computed for the Mars Express mission using standard NASA model (COSMIN, COSMAX models). The GCR energy spectrum is not expected to be at its maximum during the Mars Express lifetime as a maximum solar activity corresponds to a minimum GCR activity.

The radiation environment in the vicinity of Mars mission is not severe. The charged particle radiation environment at Mars is not expected to exceed the weak charged particle environment encountered during the interplanetary transfer.

The applicable dose-depth curve for the Mars Express mission is given in Figure 2.8.3/1. This curve is applicable for all instruments, whatever its location within the S/C. It shall be assumed at instrument level, that no protection is brought by the satellite. This approach is justified by the fact that the radiation environment for the Mars Express mission is not severe compared to that of a typical Earth mission.

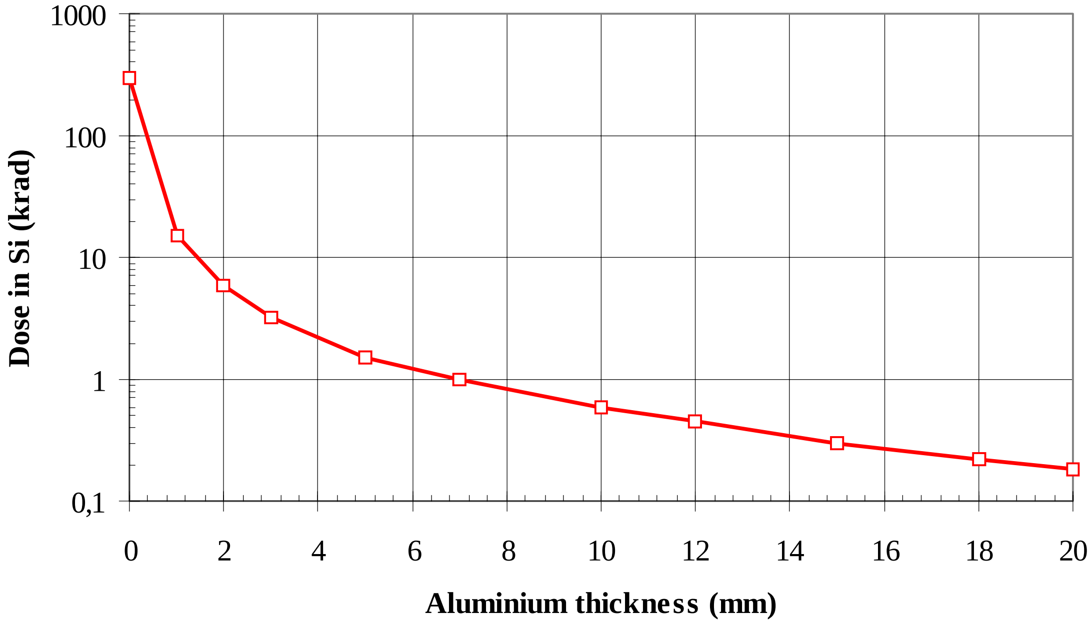


Figure 2.8.3/1 : Applicable dose-depth curve for the Mars Express electronics and payloads

2.8.3.2. Meteoroids

During its journey to Mars, the Mars Express S/C will encounter two different sources of dust and meteoroids : the particles coming from the asteroids belt and the particles from the comets. The former type distinguishes isotropic population of meteoroids and localised particles streams along well defined interplanetary orbits.

The meteoroids fluence spectrum applicable for all instruments is given in Figure 2.8.3/2 : the mission fluence is varying from about 4. 10^{+3} particles per square meter weighting more than 10^{-12} g down to 3.10^{-7} particles per square meter weighting more than 1 g. This environment is applicable for any instrument, whatever its location within the S/C. It shall be assumed at instrument level, that no protection is brought by the satellite.

The mean impact velocity is about 17 km/s, and the mean particle mass density is about 0.5 g/cm^3 for comet particles and about 3.5 g/cm^3 for asteroid ones.

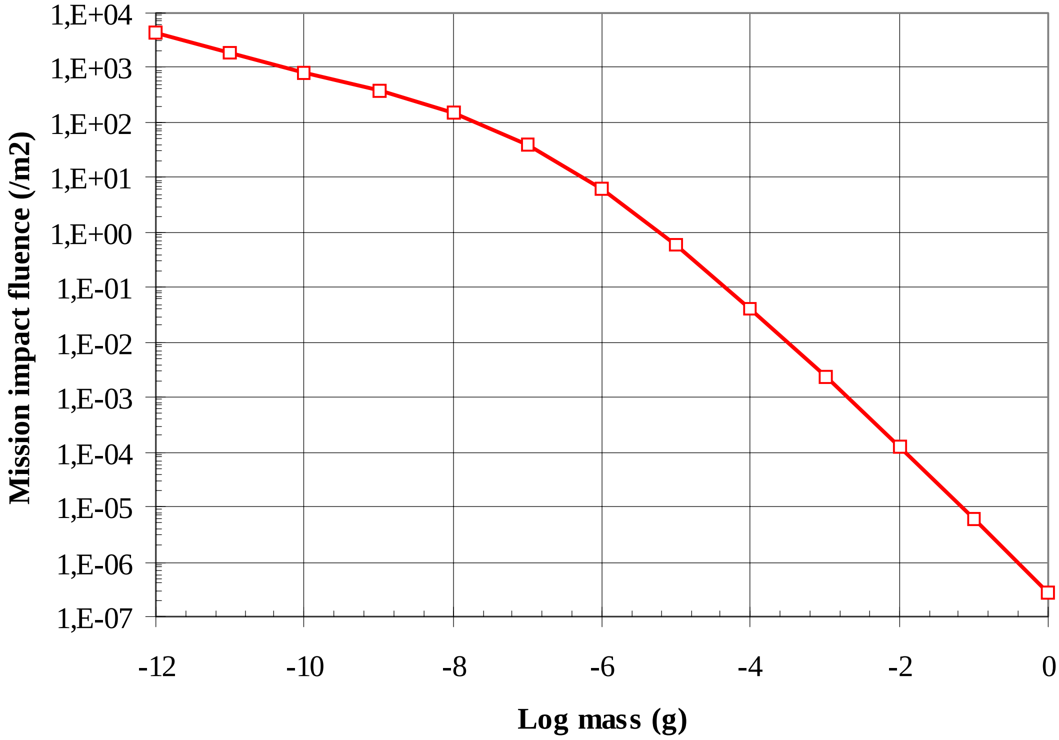


Figure 2.8.3/2 : Applicable meteoroids fluence for the Mars Express payloads

2.8.3.3. Surface Material Erosion and Resulting Contamination

Specific attention shall be paid to instrument material surface erosion and resulting damaging effects as well as resulting contamination on optical or detector surfaces (especially, for instruments with cold operational assemblies and/or cold optics working in IR wavelengths).

The reference Martian atmosphere environment model for the Mars Express mission is the NASA/MSFC Mars Global Reference Atmospheric Model (MARS-GRAM 96).

2.8.4. Contamination Coming from the Spacecraft

The contamination coming from the spacecraft will be kept lower than TBD.

2.9. Payload System Engineering

The PI shall perform all studies, analyses, simulations and testing as necessary to demonstrate the instrument conformance with the specified requirements over the mission lifetime and environments.

The PI shall demonstrate :

(i) that the instrument design conforms to the requirements (design qualification verification), and with adequate margins (in particular as interfaces and resources are concerned)

(ii) and that the flight hardware and software conform to a qualified design, is free of materials and/or workmanship deficiencies, and will function within the specified operational conditions (acceptance verification).

The overall verification program (see part 8 for detailed requirements) shall allow to verify that any individual requirement specified is fulfilled either by test, or by analysis, or by similarity, or by a combination of any of these methods.

The engineering requirements details are presently TBD.

2.10. Interfaces with the Ground Segment

TBD ESA.

Table of Contents

3. MECHANICAL/OPTICAL DESIGN AND INTERFACES

3.1. General Configuration

3.1.1. Interface Boundaries

Instruments shall be designed to be geometrically and mechanically as independent as possible from the satellite in order to simplify and permit flexibility in the payload module configuration and integration activities. The instrument shall be designed as either a single unit (containing all electronics, antennae, optical devices...) which is mounted onto or inside the satellite, and so finally integrated, or a set of different elements (electronics, sensor, antennae, harness, wave guides..) which are integrated at payload module level.

If, in order to meet its performance, the Instrument has to be mechanically and/or thermally decoupled from the Spacecraft, this related insulation is provided by the Instrument and any effect brought by this additional material shall be considered in the Instrument performances/budgets.

This concerns for example (but not restricted to) :

- instrument mass and power budget increases brought by mechanical/thermal insulation,
- instrument alignment degradation potentially brought by thermal washers or mechanical dampers,
- instrument field of view requirements increased by potential instrument on-ground/in-orbit realignment devices.

3.1.2. Physical Envelope

The external unit dimensions in both launch and in-orbit modes, including mounting lug and connector envelopes, shall be recorded in the PID-B. Unit overall dimensions shall be considered as a dynamic envelope (unit fixed on an infinitely rigid basis), and given to a tolerance of 1 mm or 10 % of the real size, whatever is the lowest.

3.1.3. Field Of View Requirements

Where relevant, the instrument shall define in its specific PID-B its field of view requirements referenced to the instrument coordinate system (X_u, Y_u, Z_u) F_u.

The definition shall cover the following points :

- scanning mode presentation and scanning plane definition,
- instrument vertex as a three dimensional position in the instrument coordinate system, and aperture shape,
- instrument boresight direction, as the centreline or nadir line of the instrument field of view (if possible),
- instrument instantaneous field of view (IFOV),

- instrument field of view : its definition shall consider different types of limitations for different zones of the field of view, clearly justifying the technical bases for the different limitations. For instance, in a first zone, the requirement could be the absence of other bodies, in a second zone, the absence of highly reflective bodies, and in a third zone, the absence of specular reflecting surfaces. These fields of view shall be defined in angles related to the boresight direction.

In addition, the instruments shall define any requirement on solar exclusion field of view.

If any instrument unit is provided with an alignment cube, its viewing directions shall be described in the field of view requirements.

3.1.4. Physical Properties

3.1.4.1. Mass Properties

The mass allocated to the instrument shall include the total instrument hardware that is intended for flight, i.e. instrument, mounting attachments and specific interface hardware (e.g. thermal washers) if required.

Mass properties shall be established and recorded in the PID-B. The record shall account for all mass states and mass dynamics attributable to deployable, consumable, moving or jettisonable materials or assemblies.

Mass shall be determined to the greater accuracy of 0.05 kg or 1% of instrument mass.

3.1.4.2. Centre Of Gravity

The Centre of Mass shall be determined to an accuracy better than 5 mm spherical error, in launch / deployed configurations, referenced to the instrument coordinate axes.

3.1.4.3. Moments Of Inertia

Moments of Inertia at the centre of mass location, about each major axis are to be determined for all instrument configurations to an accuracy of 1% of the total instrument moment of inertia for that axis, referenced to the instrument coordinate axes.

The moments of inertia are defined as follows :

$$\begin{bmatrix} I_{xx} & -I_{xy} & -I_{xz} \\ -I_{xy} & I_{yy} & -I_{yz} \\ -I_{xz} & -I_{yz} & I_{zz} \end{bmatrix} \quad \text{with :} \quad \begin{aligned} I_{xx} &= \int (y^2+z^2)dm \\ I_{yy} &= \int (x^2+z^2)dm \\ I_{zz} &= \int (x^2+y^2)dm \end{aligned} \quad \text{and} \quad \begin{aligned} I_{xy} &= \int xydm \\ I_{xz} &= \int xzdm \\ I_{yz} &= \int yzdm \end{aligned}$$

All these data (mass and inertias) shall be given with their respective uncertainties.

3.1.5. Mechanical / Optical Interface Control Drawing

The instrument mechanical and optical configuration and its interface requirements and dimensions, shall be fully detailed in one (or more) Interface Control Drawing(s), that will be fully referenced by the Instrument PID-B.

This drawing shall detail all co-ordinate systems utilised and their relationship to each other, together with the principal instrument interfaces.

The instrument shall have a right handed orthogonal co-ordinate reference system (X_u, Y_u, Z_u) F_u as defined in TBD. These axes shall be referred to on all drawings and any finite element description.

The instrument physical characteristics shall be detailed together with indication of volumes required for moving and deployable parts. Where access to clear fields of view is required this shall also be shown.

Provision of instrument CAD models will be asked with the following media and file formats :

- operating system : HP-UX
- media type : 4 mm DAT tape
- media format : TAR
- file format (by order of preference) :
    - 3D IGES (assembly as one entity)
    - 3D IGES (assembly with each part in separate IGES file)
    - 3D IGES
    - DXF
    - 2D IGES

For the unit equipped with an optical reference cube, the normal to the faces of that cube will define the Unit Optical Reference Frame. This frame shall be nominally parallel to F_u.

The location of the mirror shall be clearly identified in the PID-B drawing(s).

3.2. Structural Design

3.2.1. General Requirements

The experiment shall be designed to withstand the environments it will encounter during its lifetime without degradation to its performances and without detrimental influence on the spacecraft or other experiments' performance. The mechanical loads produced by these environments shall include :

- fabrication and assembly loads (e.g. welding, interference fitting) handling and transportation loads,
- test loads (including thermal stresses),
- Soyuz-Fregat (TBC) launch loads (vibration, thermal and depressurisation) operational loads (including thermal, attitude and orbit control induced loads).

The design of the experiment handling and transportation devices shall be such as to produce loads far lower than the predicted flight loads.

Manufacturing and assembly induced loads shall also be minimised or properly relieved.

Note : Test induced stresses are usually the most important factor limiting the life of structural items, particularly fracture critical items.

Therefore if, for instance, a structure used for qualification is proposed to be reused as a flight spare, care should be exercised in determining well in advance that this is possible from a structural point of view.

The design should therefore also consider the case of replacement of these critical parts.

3.2.2. Strength Requirements

Limit Load

Limit Load - factored by design factor - shall be used for design of bolts, feet and adjacent structure.

It shall be used for static sizing of the structure but the actual unit dynamic behaviour is not taken into account.

It shall be applied at the experiment centre of gravity.

Limit load acts simultaneously along the spacecraft longitudinal axis (Zb) and on an axis perpendicular to Zb (circular envelope).

Dynamic Load

Sine, random, acoustic vibration and shock design load (e.g. qualification) - factored by safety factor - shall be used for analysing the dynamic stress induced in the structure.

Design Factor

- Ultimate factor (below which no rupture, buckling, deformation leading to loss of functionality shall occur) : See Table 3.2.2/1.
- Yield factor (below which no permanent deformations shall occur) : See Table 3.2.2/1.
- Proof factor (the minimum to which an item shall be tested) : See Table 3.2.2/1.

Component	YIELD	ULTIMATE	PROOF
Unpressurised Items	1.5	2.0	
Pressurised Items (at Maximum Expected Operating Pressure (MEOP))		4.0	2.0

(TBC)

Table 3.2.2/1 : Yield, Ultimate and Proof Safety Factor

- Additional factor: See Table 3.2.2/2.

Material	KADD
Metallic materials, screw, rivet welding, face wrinkling of honeycomb	1.0
Carbon fibre, honeycomb (shear compression)	1.0
Bonding, structural inserts (axial loading)	1.1
Buckling strength	1.3
Honeycomb in tension, equipment (axial loading) if specific tests are done inserts in honeycomb to verify the margin if not 3.6	1.7
Metallic material (ultimate/yield < 1.2) if specific tests are done to verify the margin if not	1.0 1.7

(TBC)
Table 3.2.2/2 : Additional Safety Factor

Success Criteria

These loads (or the stresses they produce) shall be compared with minimum allowable loads (stresses) of materials and/or structural items to verify that:

$$\frac{\text{allowable load / stress (yield and ultimate)}}{\text{limit load / stress x design factor x additional factor}} > 1$$

With a slightly different terminology, the verification shall therefore show that the margin of safety, defined as:

$$\frac{\text{allowable load / stress (yield and ultimate)}}{\text{limit load / stress x design factor x additional factor}} - 1 > 0$$

Note: Pressure vessels and pressurised components are not covered by these definitions. The safety factors required vary with materials, type of tests performed, size etc. and will be defined on a case by case basis.

3.2.3. Stiffness Requirements

All experiment units shall show a first global resonance frequency higher than 100 Hz when bolted at their flight interfaces to a rigid fixture. Resonances of internal items (PCB's, discrete large components) should also be higher than 150 Hz to avoid coupling with inputs from the launcher/spacecraft, which as a consequence of high amplification factors might compromise their functional capabilities. A design margin of 20 Hz shall be considered to take into account the material characteristics variation and the modelling idealisation.

Attention is drawn to the peak frequency ranges of random acoustic inputs where inputs to internal components might reach high values.

3.3. Mechanisms Design

3.3.1. General Requirements

Mechanisms shall be functionally analysed to determine loads deriving from their activation, both in-orbit or on ground, as applicable.

Mechanisms shall be designed to the same criteria as all other structural items.

They shall be also compliant with the EMC and cleanliness requirements.

They shall therefore:

- withstand without degradation all the environments they will be subjected to during their life,
- be designed with the same loads and safety factors as other structural items,
- fulfil the minimum frequency requirement of 100 Hz in their stowed condition (same margin applicable),
- be subjected to fracture mechanics procedures as required by safety analysis.

Elements to be deployed in orbit (or otherwise showing important changes in their configuration between the launch and the orbital phase) shall also be verified under orbital environmental loads.

3.3.2. Mechanism Perturbation

Deleted

3.3.3. Lifetime Requirement

All mechanisms shall be designed for a lifetime covering the number of operations during their predicted service life multiplied by the factors of Table 3.3/1.

Actuation type	Actuation Quantity	Factor
Number of ground testing	minimum = 10	x 4
In-orbit predicted cycle	1 to 10	x 10
	11 to 1000	x 4
	1001 to 100000	x 2
	> 100000	x 1.25

Figure 3.3/1 : Mechanism Lifetime Factors (TBC)

Service Life shall include both in orbit and ground operations necessary for functional tests, system tests etc.

As actuations, a full cycle or full revolution of the mechanism is defined.

3.3.4. Torque Margin Requirements

The mechanisms shall be functionally designed to provide a minimum torque and force 3 times (up to 10 in specific cases) greater than the worst case predicted operational resistive torque.

The resistive torque/force will be the combination of the following resistive contributions, as applicable, each factored by the relevant design factors (TBC) :

Mass:	1.1
Inertia:	1.1
Friction:	3.0
Hysteresis:	3.0
Harness:	3.0
Spring forces:	1.2

(TBC)

3.3.5. Pyrotechnic Devices

All pyrotechnic and other one shot devices shall be redundant. Redundancy shall be provided for each function by duplication up to, and including, the level of initiators and the mechanical actuator, where required.

All pyrotechnic initiators shall be compliant with the PSS-03-301.

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

Any proposed pyrotechnic actuator must have been qualified according to Design Verification Requirements (section 8), and all materials and processes used shall meet the Product Assurance Requirements (section 9).

3.3.6. Shock Load Induced by the Lander Ejection Mechanism

The Lander separation mechanisms shall induce a load below the level defined in Figure 3.3/2.

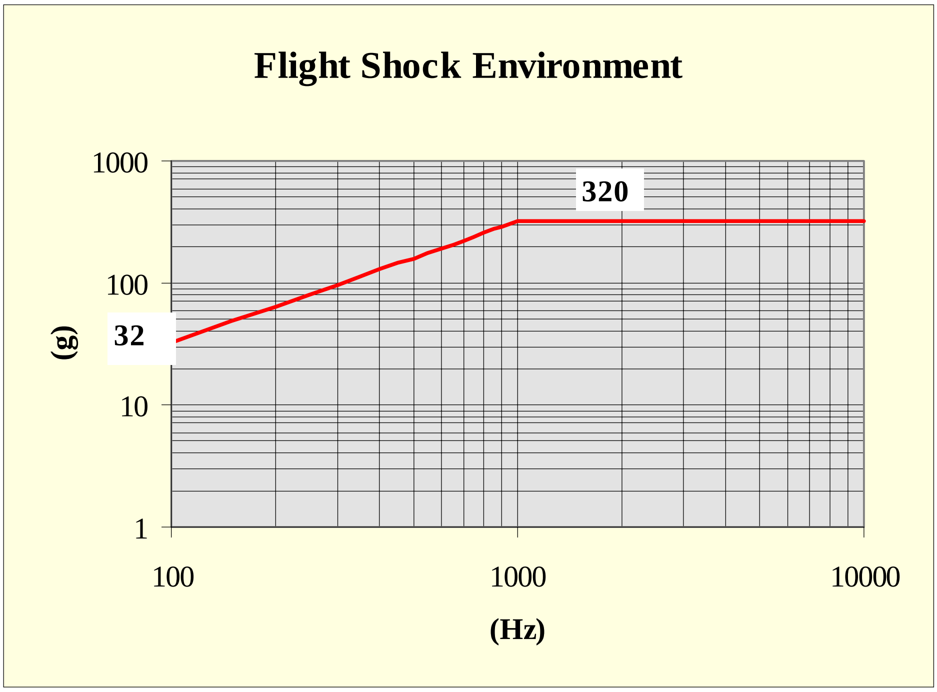


Figure 3.3/2 : Allowable Shock Level at Lander Orbiter Interface

3.4. Mounting Attachment and Handling

3.4.1. Attachment Requirements

The following requirements must be fulfilled by the unit attachments :

- The attachment points shall be designed to guarantee the connection of the units or subassembly to the satellite structure throughout the ground and orbital life of the spacecraft in a fail-safe manner.
- The boxes are required to be hard mounted. The use of dampers is not recommended and shall be agreed with the Spacecraft Project office on a case by case basis.
- Rectangular shaped boxes shall be provided with at least four attachment points near the corners of the unit
- The unit configuration itself shall not hinder the installation and removal of the attachment bolts.
- For experiment sensors with complex shapes, the location and number of attachment points shall be agreed with the Spacecraft Project Office.
- The contact area and free or spot faced area shall be unpainted and provide a good electrically conductive surface.
- All attachment points shall be in a common plane within + 0.1 mm.

- The mechanical mounting interface shall be consistent with the thermal and EMC design requirements.
- The unit shall use potted inserts in the core of the S/C panel.
- For each of the Lander Ejection Modules, 3 (TBC) hard points will be provided at TBD locations of the S/C +Z face. Hard point configuration : TBD together with Lander Ejection Module interface definition.

Bolts	
tvpe	: M4 or M5
Material	: Titanium (TBC)
Supplier	: Prime Contractor provided for spacecraft integration
Dimensions	: length = 11 mm
Other sizes of bolts or material shall only to be used after agreement with Spacecraft Project Office, and procurement shall be agreed with the prime contractor and the experiment.	
surface finish	: TBD
torque	M4:2.5Nm +/-10%
	M5 :6.0 Nm +/- 10 %

Washer	
Material	: Titanium or Stainless Steel (TBC)
Supplier	: Prime Contractor provided for spacecraft integration
Dimensions	M4: Diameter 9 mm, thickness 0.8 mm M5: Diameter 10 mm, thickness 1.0 mm

Table 3.4/1 : Bolt and Washer definition

3.4.2. Lug design Requirements

General requirement
Number of lugs per unit (*) : > 4
Ground clearance of the unit base: Minimum 1 mm with respect to the mounting area of the attachment lugs

(*)The number of lugs shall meet fail-safe design requirements.

Foot pattern design requirements	
Distance between holes	:
	M4 : > 100 mm
	M5 : > 120 mm
Tolerance of attachment hole	: distance +/- 0.1 mm
location from reference hole	: pitch circle dia. +/- 0.1 mm
	: angles +/- 1 arcmin (if applicable)

Hole design requirements	
Diameter	M4 : 4.3 - 0 / + 0.1 mm
	M5 : 5.3 - 0/ + 0.1 mm
Angle of attachment hole	: 90° +/- 0.5° for untapped holes to attachment surface
Distance between attachment	M4 : ≥ 8 mm
holes and unit side walls:	M5 : ≥ 9 mm

Lug design requirements	
Attachment lug thickness	: 4 mm -0 / 0.1 mm
Free width between webs (if applicable)	M4 : min 16 mm M5 : min 18 mm
Edge radius	: ≥ 0.5 mm

Lug surface requirements	
Spot face of Free area, parallel to mounting plane for bolt head and washer	M4: 11 mm dia. -0 / + 0.5 mm M5: 13 mm dia. -0 / + 0.5 mm
Spot face area parallelism w.r.t. mounting plane	: ≤ 0.05 mm
Counterbore depth	: 0.2 + 0.1 mm

Lug contact requirements	
Surface flatness of attachment lugs	: ≤ 0.05 mm
Mounting surface per lug	M4 : min 16x20 mm M5 : min 18x22 mm
Roughness	: ≤ 1.6 microns R.A

Table 3.4/2 : Lug Design Requirements

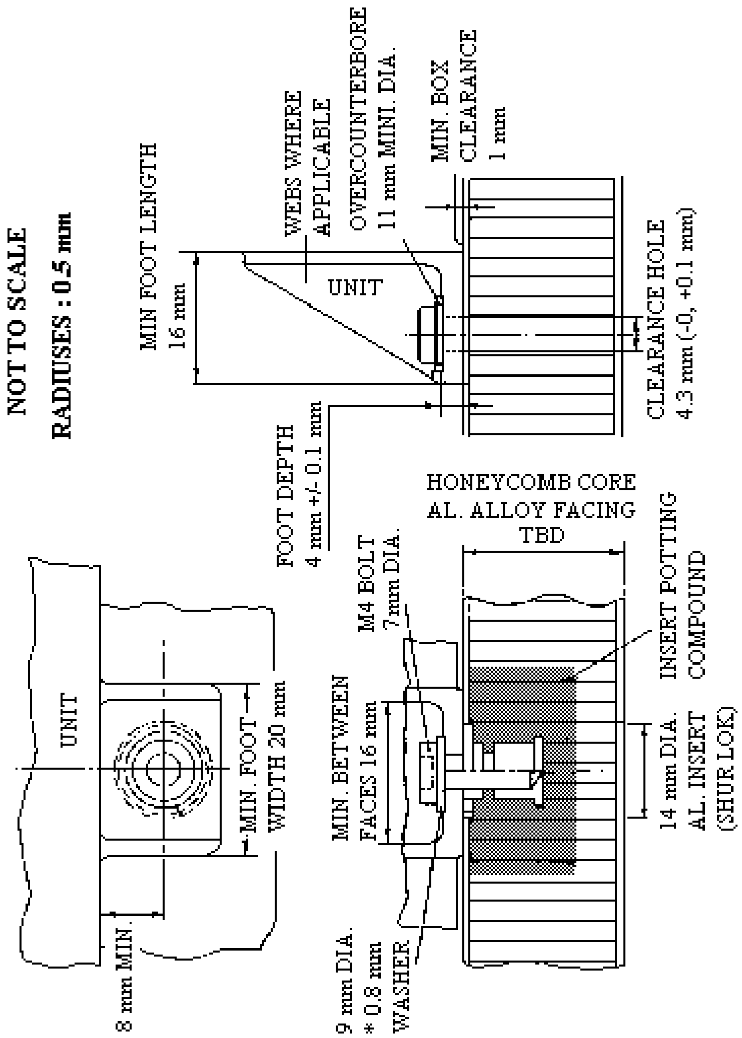


Figure 3.4/2 : Typical Mounting lug (M4 Bolt)

3.4.3. Loads Requirements

The S/C will provide 2 type of fixations M4 and M5 and 2 categories of inserts standard or reinforced (face to face).

The nominal ultimate allowable insert loads is given in table 3.4/3.

Insert Type	Allowable Load
Standard	TBD
Reinforced	TBD

Table 3.4/3 : Nominal Ultimate Strength Allowables for inserts

Actual insert allowable load of the inserts pattern configuration:

Due to their proximity, groups of insert induced a mutual reduction of the strength capability;

$S^*_{1u} = S_{1U}\ \mu_1$

S_{1U} = nominal Ultimate strength capability of the insert 1

S^*_{1U} = reduced Ultimate strength capability due to neighbouring inserts

μ_1 = reduction factor due to the inserts 2 to n

$$= \frac{1}{1 + \sum_{j=2}^{j=n} \frac{1-\mu_{1j}}{\mu_{1j}}}$$

μ_{1J} = reduction factor due to the insert j

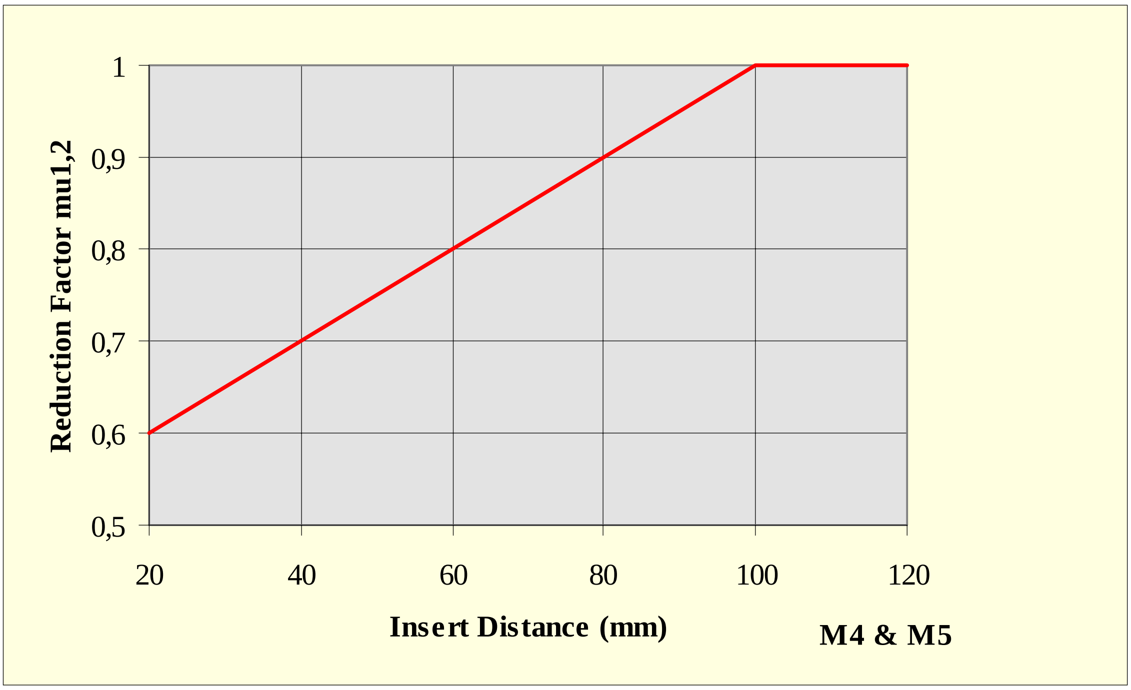


Figure 3.4/3 : Strength reduction factor of 2 adjacent inserts

Units mutual influence

The actual load capability of each insert is subject to the proximity of the inserts of others units, so the individual actual load capability shall be assessed by the Prime Contractor, and the classification of each fixation shall be defined.

Load Combination

The experiment unit and its foot pattern shall be designed with respect to the insert Ultimate allowable load (reduced by the unit foot pattern mutual influence).

The force applied at each standard attachment point shall comply with the condition below, when the unit is subjected to unit Ultimate Limit Load (Nominal * ultimate safety factor) applied at CoM simultaneously along the three axes:

$$\left(\frac{N}{N_u}\right)^2 + \left(\frac{S}{S_u}\right)^2 < 1$$

N = normal applied load

S = shear applied load

N_u = allowable normal load

S_u = allowable shear load

3.4.4. Reference Hole

One of the attachment holes on a unit shall be specified as the reference hole and must carry the identification letter R. This shall be clearly indicated on the mechanical interface drawings.

The dimensioning of the attachment hole pattern shall be specified with respect to the URF.

3.4.5. Accessibility Requirements

The design of the experiment and attachments shall provide sufficient accessibility to enable the mounting and removal of the experiment or unit with normal standard tools.

Where this requirement cannot be applied, the Experimenter shall provide a kit of tools as a part of the experiment MGSE such that the mounting bolts can be tightened from an accessible position.

3.4.6. Handling Provisions

Units weighing more than 10 kg shall be equipped with handling points (e.g. threaded bushes) which will enable the connection of special handles, provided by the Experimenter, for use during the (de)integration of the unit.

As any other piece of hardware used only during ground operations, such handles shall be clearly identified as non-flight item (red anodized and a red flag carrying the notation "NOT FOR FLIGHT" attached to them).

Such items shall be clearly identified on the relevant Interface Control Drawing.

3.5. Aperture Covers

3.5.1. Requirements

Experiment units that incorporate a sensor aperture:

- shall be delivered with a dust/protective cover over the aperture.

These covers can either be removable before flight or be part of the experiment and be deployed in orbit.

3.5.2. Removable Covers (Non-Flight Items)

All removable covers shall be :

- each cover shall be identified,
- be easily identifiable as non-flight hardware; red anodising is a usual finish and a red flag carrying the notation: "NOT FOR FLIGHT", attached to the device makes them doubly protected,
- clearly marked on the drawings as non-flight,
- normally be removed during system tests where flight configuration is mandatory i.e. thermal vacuum testing or vibration testing,
- be accompanied by a detailed procedure for their removal by Spacecraft Project Office personnel or their Contractors.

3.5.3. Deployable Covers (Flight Items)

Any unit that absolutely requires an aperture cover at all stages of spacecraft integration and launch shall incorporate a deployable cover activated by telecommand.

- This deployable cover shall be classified and designed as a mechanism.
- The Experimenter shall provide a description of the cover deployment and retention system.
- The cover could be a one shot item or a reversible one.
- This system shall therefore be agreed with the Spacecraft Project Office. Solutions which minimise variations of required experiment volume will be preferred.

3.6. Electrical Connection

3.6.1. Electrical Connectors

Connector Location

- In order to avoid problems with cable routing and cable harness support fixation, all electrical connectors shall be located close to the unit mounting plane.
- Connectors shall be spaced sufficiently to allow easy mounting and installation of the harness. The spacing, orientation and location of connectors shall follow the requirements indicated on Figure 3.6/1.
- The exact location shall be agreed with the Spacecraft Project Office.

All connectors shall be located within ± 0.75 mm of the nominal position.

The screw lock assembly of the connector shall be tightened as follows:

- female part (i.e. screw mounting the connector to the box) torque = 0.45 N*m,
- male part (i.e. screw connecting harness connector to the box connector) torque = 0.34 N*m.

3.6.2. Ground Point Requirements

Each unit shall provide a grounding point for test purposes; M4 x 6 bolt, near but not less than 20 mm above the mounting plane and easily accessible (side or rear faces). A torque of 3 N*m will be applied during AIV activities.

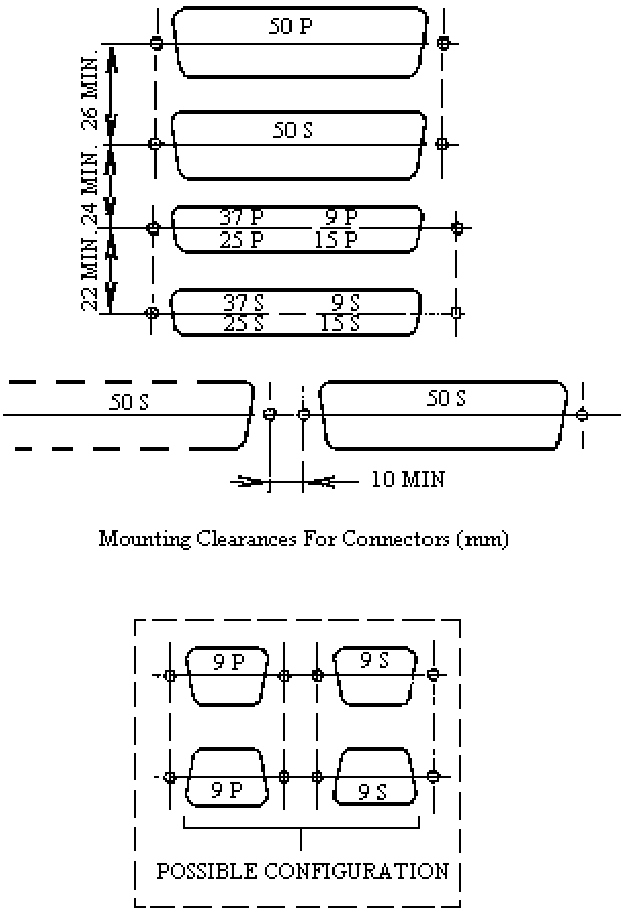
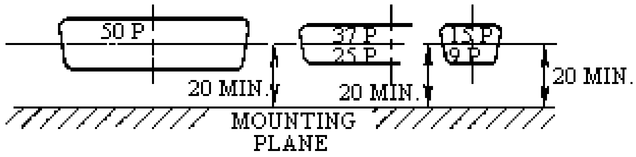
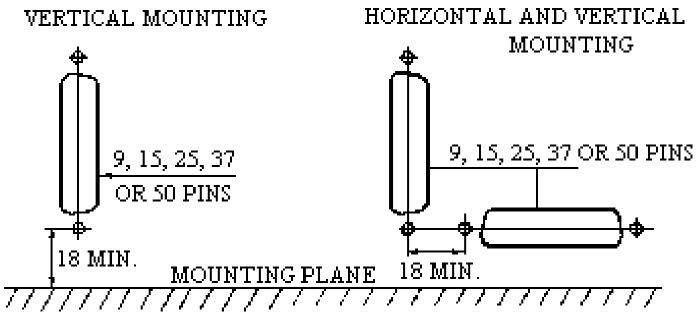


Figure 3.6/1 : Connectors Accommodation



Minimum Distances Between Mounting Holes Of Any CANNON "D" Type
Connector Required For Heat Shrink Boots and Screw Lock Assemblies Clearance



Spacing Between Adjacent Connectors As For Adjacent Mounted Connectors

Figure 3.6/2 : Connectors Accommodation

3.7. Positioning and Alignment

3.7.1. Definitions

Unit Reference Axes (URF) [Xu, Yu, Zu]

Defined in section TBD.

Unit Optical Reference Axes (UORF) [Xuo, Yuo, Zuo]

For any unit equipped with an optical reference cube, the normal to the faces of that cube will define the Unit Optical Reference Axis.

During the alignment phase this reference system will be aligned with regard to the spacecraft optical axes. Thus attention is drawn to the criticality of this reference and therefore to the reliability of its fastening system and the possibility of providing a redundant reference.

This cube shall be mounted on a fixed part of the experiment, the position shall be agreed with the Spacecraft Project Office and it shall have the following characteristics:

- the minimum size of the faces shall be 15 x 15 mm,
- the quality of its surface compatible with autocollimation techniques (flatness better than lambda/4).

The cube shall have one mirror surface perpendicular to the experiment boresight axis.

Payload/Spacecraft Interface Plane Axes [Xp, Yp, Zp]

The Payload/Spacecraft Interface Plane Axes are used to establish a local reference system for each payload unit at the mounting interface on the spacecraft side. The orientation of these axes will be measured before payload integration.

In the ideal case these axes will be coincident with the nominal payload unit reference axes.

3.7.2. Experiment Field of View and Boresight Requirements

The Experimenter shall define in the URF and (if relevant) in the UORF system :

- the boresight direction,
- the field of view,
- the vertex.

Boresight direction

Direction of boresight shall be defined with respect to the different mission phases - spacecraft scanning requirements.

This is the centerline of the Field of View.

This direction shall be defined in terms of Azimuth and Elevation such that :

- Azimuth is the angle between the boresight 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 boresight direction and its projection in the Xu-Yu plane. This angle is counted positive from the Xu-Yu plane to +Zu axis and varies in the range +/- 90 deg.

Field of view

This shall be defined in term of :

- SFOV (Scientific Field of View) which is the field of view (fixed or scanning) used for scientific observation.
- UFOV (Unobstructed Field of View) which is the field of view inside which no unit surfaces (either of the spacecraft or other units) are allowed.

Those two Fields of View shall be defined as:

- either a half angle of a cone centred on the boresight direction,
- or by two half angles SOE and SOA from the boresight direction such that :
    - SOE is the half angle in the plane containing the X-axis and the boresight axis.
    - SOA is the half angle in the plane containing the boresight axis and perpendicular to the one containing SOE.

In order to optimize the accommodation of the experiments, the Experimenter shall also describe the tolerable obstructions in his field of view in terms of direction, opening angle and separation distance to obstructions.

Vertex

The experiment has to define the position of the vertex of the Field of View angles in the URF system.

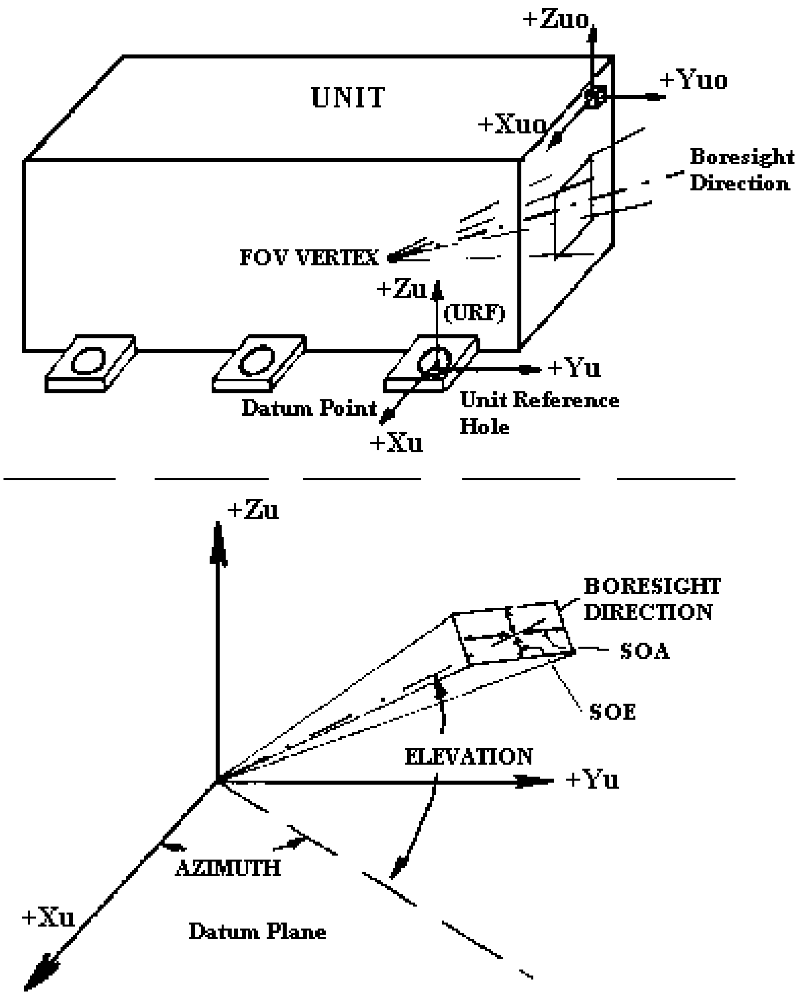


Figure 3.7.2 : Boresight direction, Field of View and Vertex Definition + Stray Light Requirements

3.7.3. Alignment Budget

Payload alignment requirements shall be defined between the Spacecraft Mechanical Build Axes [Xb, Yb, Zb] and the Payload/Spacecraft Interface Plane Axes [Xp, Yp, Zp] or the Payload unit reference axes (URF) [Xu, Yu, Zu].

In case of optical alignment, the measurement and adjustment will be done between Spacecraft Optical Reference Axes [Xo, Yo, Zo] and the Unit Optical Reference Axes (UORF) [Xuo, Yuo, Zuo].

The alignment budget will be split into azimuth and elevation contribution, because of the nature of the alignment errors.

The alignment will be defined in term of accuracy, knowledge and stability as shown on Figure 3.7.3/1 and Table 3.7.3/2.

The accuracy is defined as the total circle of error (worst case build-up tolerances).

Knowledge and stability will be defined at 95% confidence level.

The stability is essentially composed of thermal cycling induced structural deformation.

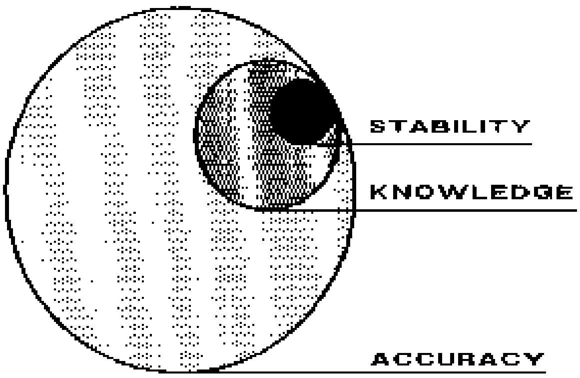


Figure 3.7.3/1 : Alignment Terminology (circle of error)

CAT	Accuracy	Knowledge
1	interfaces axes tolerances integration stresses launch 0 g + centrifugal force outgassing stability	integration stresses launch 0 g + centrifugal force outgassing stability
2	interfaces axes measurement measurement accuracy integration stresses launch 0 g + centrifugal force outgassing stability	measurement accuracy integ rati on stresses launch 0 g + centrifugal force outgassing stability
3, 4 **5, 6**	URF(O) measurement (adjustment) measurement accuracy launch 0 g + centrifugal force outgassing stability	measurement accuracy launch 0 g + centrifugal force outgassing stability

Table 3.7.3/2 : Accuracy and Knowledge sources

Alignment procedures are classified into 5 categories:

CAT 1:	no adjustment alignment relying on mechanical tolerances no measurement
CAT 2:	no adjustment alignment relying on mechanical tolerances no measurement except the mounting I/F plane
CAT 3:	no adjustment alignment relying on mechanical tolerances mechanical measurement of the URF
CAT 4:	adjustment (standard method) mechanical measurement of the URF
CAT 5:	no adjustment alignment relying on mechanical tolerances and optical measurement
CAT 6:	adjustment optical measurement

Table 3.7.3/3 : Alignment Categories

Typical mounting tolerance achievable during nominal integration activity : see table 3.7.3/4

TBD

Table 3.7.3/4 : Alignment Budget

3.7.4. Experiment Requirements

To allow the accommodation of each unit on the spacecraft, the experimenter shall define:

- the angle between the required boresight direction and the Payload panel normal axis. This angle is counted from the +Xb axis within the range 0 to 180 deg,
- the acceptable accuracy, knowledge and stability of the above angle in terms of azimuth and elevation. These values, in correlation with the experiment internal (including mounting error) alignment accuracy*, knowledge and stability will be used to compute the mechanical mounting accuracy knowledge requirement of the unit.

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

Based on the final accommodation lay-out, each unit will be mechanically positioned with respect to the spacecraft mechanical build axes.

The **alignment baseline** is Category 1 or Category 5 (CAT 1 or CAT 5).

Every unit having a mounting accuracy requirement less stringent than the position accuracy to be achieved will not be adjusted. After mounting on the spacecraft the position of those units will be determined analytically.

The other units which have a mounting accuracy requirement more stringent than the achievable position accuracy, shall be adjusted by utilising shimming and slack at the mounting interface of the unit. The adjustment activity will use mechanical or optical measurement.

The adjustment activity will be under the Prime Contractor's responsibility. However the Experimenter shall :

- provide :
  - As part of his mechanical interface the means to adjust his unit (i.e. shims, screws, eccentrics...).
  - Special mounting design to be considered to allow sufficient adjustment capability of the sensor position in all three axes.
  - The boresight measured direction in the URF system and w.r.t. the UORF and the accuracy of those measurements.
  - For approval by the Spacecraft Project Office, 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.
- demonstrate :
  - That the adjustment activities will not introduce stresses in the experiment and in the spacecraft structure (or that the stresses are quantified and stay below an acceptable level).

➢ produce a design of his unit which :

- Minimises the adjustment angle during the alignment phase (i.e. the boresight position w.r.t. the mounting plane), in order to stay in a conventional alignment method.

Coalignment refers to the relative alignment between the payload spacecraft interface plane axes of experiment unit 1 and those of experiment unit 2.

The definitions of accuracy, knowledge and stability are applicable for co-alignment of sensors.

If two or more experiment sensors require to be co-adjusted, their design shall include the means of adjustment as necessary.

3.7.5. Pointing Budget

Deleted

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4. POINTING AND DYNAMICS

4.1 Attitude manoeuvres and profiles

The instruments shall define the attitude manoeuvres and profiles necessary for the mission with respect to the basic attitudes of the Spacecraft on the orbit around Mars :

- during the observation phase near the pericentre, the Spacecraft is basically Nadir pointed (one face of the Spacecraft oriented towards the planet centre).
- during the communication phase, the Spacecraft is basically Earth pointed (the High Gain Antenna is pointed towards the Earth).

The description of the attitude manoeuvres shall include the angular amplitude of the manoeuvre with respect to the basic attitude, and the total duration spent in this attitude.

The attitude or rate profiles necessary for the planet observation shall be also described (angles and rates in function of time), together with their related inaccuracies.

The acceptability of these manoeuvres or profiles will of course depend on the compatibility between the different P/L needs at the same time. In addition to this obvious criterion, the manoeuvres to be performed during the observation phase will have to be compatible with the wheel kinetic momentum capacity and torque (see section 2.6), and the manoeuvres to be performed during the communication phase will have to be compatible with data transmission constraints and will be preferably placed for instance during the periods without ground links.

4.2 Sensitivity to sun illumination

In case of failure, requiring a Spacecraft mode change, the pointing may be unknown during several thousands of seconds, before the end of the attitude acquisition. During this period, it is not possible to ensure to the payload a minimum angle with respect to the sun, and there could be a direct sun illumination to sensitive elements. The instruments could be however switched OFF by the Spacecraft during the failure management mode following the failure detection.

The instruments shall be therefore designed to be robust to such events and shall include a protection of sensitive elements from the sun if necessary.

4.3 Pointing performances

The instruments shall express their need concerning the performances of the Spacecraft AOCS :

- The onboard attitude knowledge as the performance of the attitude estimator. These data can be used in real time, for instrument internal purposes.

- The a posteriori attitude knowledge as the result of a TBD post processing performed on Earth. This need shall be expressed for information only of the S/C contractor, in order to clarify what is under its responsibility.
- The attitude pointing accuracy as the pointing performance of the attitude control of the system in real time onboard.
- The attitude pointing stability over a given period as the maximum angular evolution of the Spacecraft attitude during a given duration of interest for the instrument.

The performances required for the attitude knowledge or the attitude pointing may be referred to an absolute Reference Frame (linked to stars for instance) or to a Mars Reference Frame (Nadir pointing for instance).

4.4 In Flight alignments

The instrument shall define the need for in flight alignment calibration, if necessary, with respect to the AOCS reference sensors or with respect to other instruments. This calibration, could be more accurate than on ground since it uses a direct measurement on celestial objects and provide an information on the alignment between two optical axes without intermediate references, such as mirrors or cubes. Its efficiency will be however limited by the in flight evolutions of this alignment (thermoelastic effects for instance).

4.5 Payload flexible behaviour

The instrument flexible behaviour in the range [0 - 10 Hz] shall be described through a deliverable model when their eigen frequency is in this range. This model, for the operational configuration, shall provide the cantilever frequency of the mode, the modal participation factors with their signs, and the damping factor. The value of the damping factor shall be justified.

This description shall be given with the uncertainties on the parameters.

The flexible modes of the deployed instruments shall have to be compatible with the thruster control of the AOCS used during the attitude acquisition modes and the Safe Mode, that will be entered by the Spacecraft after some failures. This could lead to some constraints on the flexible mode parameters, such as a frequency greater than 0.07-0.1 Hz (TBC).

4.6 Compatibility with the Spacecraft dynamic environment

Low frequency disturbances generated by the AOCS

The Payload design shall be compatible with the following dynamic environment generated by the AOCS at low frequency (< 10 Hz) when using the thrusters.

	Trajectory corrections before final orbit acquisition	Initial attitude acquisition	On final orbit
Linear acceleration	0.9 m/s^2	0.05 m/s^2	0.05 m/s^2
Angular acceleration	1 $°/s^2$	1 $°/s^2$	1 $°/s^2$
Angular rate	0.15 °/s	5 °/s	0.12 °/s

All the figures correspond to the Spacecraft centre of mass and are TBC. Figures for each instrument location to be computed

Environment during deployments

The instruments, in a non operating mode, shall be compatible with the dynamic loads generated by the deployment of the Solar Array or other Payloads. These loads are TBD.

High frequency disturbances (> 10 Hz) : micro-vibrations

The Payloads shall be compatible with the micro-vibrations generated by the Spacecraft, mainly due to the reaction wheels, and possibly the Solar Array Drive Mechanism and other instruments. These disturbances are TBD today.

If necessary, the instrument supplier shall describe the sensitivity of the instrument to such micro-vibration disturbances indicating :

- the operating mode concerned and the effect of the disturbance (degradation of optical performance, mechanical effect...),
- the frequency concerned,
- the amplitude (microradians, mm/s^2) generating this effect.

4.7 Payload generated disturbances

Identification of the disturbing behaviour

The behaviour of the instrument that generate a dynamic disturbance to the Spacecraft and other instruments shall be identified and described, separating :

- the non recurring transient events (deployments, unlocking device, etc...),

- the recurring transient events,
- the continuous behaviour.

Technical description of the disturbing mechanism

For each disturbance identified, a complete analytical model shall be provided including the following items :

- geometrical description of the rotation/translation axes,
- description of mass and/or inertia in motion,
- description of actuators disturbances or defects,
- description of the control laws for the mechanism,
- description of possible static unbalances (when the centre of mass of the rotating assembly is not aligned with the rotation axis).
- description of possible dynamic unbalances (when the products of inertia of the rotating assembly with respect to the rotation axis are not zero).

This description shall be given with the uncertainties on the parameters and the identified limitations of the proposed model.

Frequency of the disturbances and micro-vibrations

The disturbance analysis shall cover the low frequency range [0 - 10 Hz] that can be partially compensated by the Attitude Control of the Spacecraft, and the high frequency range [10 - 150 Hz] that concern micro-vibrations phenomena. For the micro-vibrations, the description of the disturbance (amplitude, frequencies) could replace a model.

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5. THERMAL DESIGN AND INTERFACES

5.1. Payload Thermal Control Principle

The Experiment shall be designed to withstand the extreme temperatures which may be experienced during transportation, testing, launch and flight.

The Spacecraft Thermal Control Subsystem shall control the interface of all Payload with the:

- heat exchange requirements,
- interface temperature specification,
- interface thermal model.

From the thermal control point of view, the units fall in two categories:

- **Individually controlled units**, for units conductively and radiatively decoupled from the Spacecraft and performing their own thermal control.
- **Collectively Controlled units**, for units conductively and radiatively coupled from the Spacecraft and thermally controlled by the S/C.

From the thermal accommodation point of view, three categories can be defined, as a baseline **Internal Units** belong to the collectively controlled category and **External Units** belong to the individually controlled category. The case of partially **protruding Units**, shall be reviewed case by case, agreed between the Experimenter and the Spacecraft Project Office and defined in the relevant PID-B.

In principle, some S/C radiator area could be also allocated, upon request and in case by case basis, to Payload units. Those radiators cannot provide cryogenic or near-cryogenic temperatures.

5.1.1. Individually Controlled Units

The thermal design of these units shall be under the responsibility of the Experimenter, controlled by defined interfaces between the unit and the spacecraft.

Within Experiment boundaries

To comply with the heat exchange requirements,

- The conductive insulation needed between the unit and the Spacecraft shall be sized, procured and implemented by the Experimenter.
- The radiative insulation needed by the unit shall be defined, procured and implemented by the Experimenter.
- The Interface Thermal Mathematical Model shall be provided by the experimenter for verification.

The internal thermal design, including radiator sizing -where applicable - shall be the responsibility of the Experimenter. In case of individually controlled unit heat dissipation through spacecraft radiators, the unit dissipation characteristics (thermal power and temperature level) shall be defined by the Experimenter in each relevant unit mode.

The units may be equipped with Spacecraft controlled heaters to be used when the unit is switched off. The sizing, procurement and installation of these heaters are under the Experimenter responsibility. Those heaters shall be controlled using installed Spacecraft powered thermistors or thermostats.

To satisfy the unit individual thermal control requirement, the experimenter may decide to use internal operational heaters. This heater power required shall be budgeted within the experiment power resource allocated.

When the unit heat transfer is to be performed conductively from unit to spacecraft radiator via transfer device (straps of fluid loop), this transfer device is designed and procured under Experimenter responsibility, and is then connected by spacecraft contractor to the relevant spacecraft radiator with agreed interfaces and procedures.

The design shall be verified by analysis and test. Analysis shall verify that all parts are within the specified temperature range in extreme worst case.

Within Spacecraft boundaries

The heat exchange requirement and the interface temperature specification shall be defined by the Spacecraft Thermal Control.

The Spacecraft shall control the non operational heaters power.

The Spacecraft shall verify through various analytical cases that the thermal design of the unit :

- satisfy the unit operational requirements in terms of temperature ranges and heater/radiator sizing,
- has a conductive heat exchange between the unit and the spacecraft compatible with the spacecraft thermal control performances. In case of difficulties, the spacecraft may impose more stringent requirements than the one of section 5.2.1 below.

The Spacecraft shall provide the following data :

- as a result of each thermal analysis campaign :
  - I/F conductive temperature,
  - incident (but not adsorbed) solar flux per external node,
  - incident (but not adsorbed) IR flux per external node (Tsink eventually),
  - nodal temperatures,
  - conductive heat flow budget and eventually revised conductive heat flux requirement,
  - results assessment (e.g. unit heater and radiator sizing, impact on unit operations...)
- flight procedures applicable to the thermal control of the experiments during their non-operational phases.

The Spacecraft shall, after unit integration on the S/C, ensure that :

- the unit temperature and its ambient lies within specified range during storage,
- the unit temperature are met for the non-operational cases by adjusting the S/C thermal subsystem and/or the S/C powered heaters according to the agreed flight procedures,
- the unit conductive interface temperature requirements are met for the operational cases by adjusting the S/C thermal subsystem.

5.1.2. Collectively Controlled Units

Those units will be controlled by the S/C Thermal Control Subsystem, but their internal thermal design shall be under the responsibility of the Experimenter.

With Experimenter boundaries

The internal thermal design is the responsibility of the Experimenter, who shall provide adequate information about unit internal design and the internal requested temperature.

Within Spacecraft boundaries

The interface thermal design is the responsibility of the Spacecraft Thermal Subcontractor who will adjust the conductive and radiative unit interface to suit specific unit designs, and shall provide the corresponding hardware.

The S/C thermal control will monitor a temperature measurement point belonging to the unit housing and will ensure that it stays within an agreed temperature range.

The design will be verified by analysis and test. Analysis shall verify that all parts are within the specified temperature range in extreme worst case.

5.2. Thermal Interfaces Requirements

Those interfaces shall be defined in the Thermal Interface Control Drawing and summarised in the Interface Control Thermal Mathematical Model.

5.2.1. Conductive Interface

The mounting interface shall comply with the mechanical and EMC requirements.

For individually Controlled Units, the Experimenter may use conductive isolation, such as glass fibre washers or stand-offs.

To meet the Heat Exchange Requirements of Section 5.3 or to achieve the desired internal unit temperature, any conductive isolation necessary shall be on the experiment side of the interface and hence shall be Experimenter provided.

Individually Controlled Units shall be designed such that their heat flux to the S/C, by a conductive link, per foot is not greater than 0.5 W/cm² (collectively controlled units) and 0.1 W/cm² (individually controlled units).

For Collectively Controlled Units, experiment unit mounting areas shall not be painted or anodised, in order to obtain a good conductive thermal contact with the Spacecraft.

5.2.2. Radiative Interface

The Heat Exchange requirements of Section 5.3 and the desired internal unit temperature shall be achieved by the selection of finishes.

- **Protruding units**:

  The emissivity of protruding units shall be agreed between the Experimenter and the S/C Project Office.

  The emissivity class shall be defined in relevant EID.B

- **Individually controlled units** :

  The recommended coatings for the external surfaces are (in this order):

  - multi layer insulation (MLI),
  - vacuum deposited aluminium (VDA),
  - gold-plated, silver-plated or alodine.

- **Collectively controlled units** :

  They shall be designed with an emissivity > 0.8 (black anodise recommended, paint not recommended), the final trimming shall be performed by the Spacecraft Contractor.

5.2.3. Spacecraft Powered Heater Interface

For Individually controlled experiment heaters, the heater power will be allocated according to Section 5.3. shall be controlled by the spacecraft via a latching current limiter. Their sizing, design and installation is an experimenter responsibility and shall be agreed with the Spacecraft Project Office on a case by case basis.

For collectively controlled units, it is envisaged to bond some S/C heaters directly onto one unit face.

5.2.4. Thermistor Interface

Spacecraft Powered Thermistor

Spacecraft Powered Thermistors shall be used to monitor the experiment health and safety (with respect to the non operating temperature range and the switch on temperature).

These thermistors shall be used to control Spacecraft powered heaters.

The experimenter shall select the S/C powered thermistor from the standard types defined in the OBDH requirements; utilisation of different types shall by agreed with the Spacecraft Project Office.

Experiment Powered Thermistor

In addition, temperature monitoring of selected points within an experiment shall be provided by the Experimenter and embedded in the experiment housekeeping data to cover the following cases:

- Experiment operational health and safety monitoring.
- Experiment operational temperature and performance monitoring.

Conditioning/acquisition of these sensors shall be the Experimenter's responsibility.

5.2.5. Thermal Hardware

All external surfaces shall be electrically conductive, according to the EMC requirements of Section 6.5.

Coatings and Finishes

These shall be selected from the advised list of Coatings and Finishes.

Specularly reflecting external surfaces shall be avoided where possible.

Surface finishes shall be cleanable after ground handling or shall be protected.

Surface finishes shall be compliant with the chemical cleanliness requirements defined in Sections 2.7 and 9 and in particular they shall not chip, dust or peel when exposed to the launch and flight environment.

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

MLI : All thermal blankets shall be protected during ground activities.

5.3. Design Requirements

The Experiments shall be designed to comply with the Spacecraft interfaces and their own units requirements in term of temperature and heat exchange.

5.3.1. Temperature Requirements and Budget

The Experimenters shall define their internal temperature requirements in the PID.B. The temperature reference point (TRP) temperature range shall be agreed with the S/C project office.

The design shall take into account the Interface temperature Budget.

The design shall be supported by analysis which will predict nominal worse case and extreme worse case temperatures.

Uncertainty margin policy

The individual contributions of parameter variations are considered as independent and statistical and hence shall be added in the root-sum-square manner.

The modelling or idealisation error is assumed to be of a systematic nature and therefore shall be added linearly, this error shall be agreed with the Spacecraft Project Office.

In the absence of any uncertainty analysis an initial uncertainty margin of 10°C shall be retained in general and 20°C for highly sensitive items.

The temperature margin policy is defined in section 2.8.2.4.

5.3.2. Heat Exchange Requirement and Budget

The **heat exchange budget** defines the amount of heat that can be transferred (conductively and radiatively) between the experiment unit and the Spacecraft.

Collectively Controlled Units

There is actual budget for this category because the heat exchange is equal to the power dissipated.

The experimenter shall base the heat rejection mechanism in the internal unit thermal design as follows :

- Internally mounted units : from unit internal through unit baseplate.
- Externally mounted units: from unit internal through unit external structure.

The design of the Spacecraft Thermal Control will take this mechanism into account when defining the ratio of conductive versus radiative external heat rejection.

Individually Controlled Units

The heat exchange of these units shall be restricted by unit design and the mounting interface.

The conductive heat exchange applies at the interface between the unit and the Spacecraft and includes the conductive heat through the harness.

5.4. Thermal Interface Control Documents

The general characteristics of the instrument to spacecraft thermal interface and the requirements for its thermal control shall be as given below. For each instrument, a detailed description of its interface and thermal requirements shall be included in the instrument PID-B.

5.4.1. Thermal Interface Control Drawing

The Instrument Supplier shall prepare and supply Thermal Interface Drawings which shall define the total thermal interfaces. These drawings and their issue shall be included in the instrument PID-B. The interface requirements given below may be defined either in the PID-B or in this Thermal Interface Drawing. It shall, at least, contain the following data :

- overall layout,
- dimensions - overall size including thickness and their attachment,
- Temperature Reference Point (TRP),
- radiator areas,

- external surface optical properties,
- apertures (position and size),
- blankets,
- blanket performance,
- optical properties of box in/outside and protruding parts, apertures... (BOL, EOL if applicable),
- non operational heater location,
- spacecraft powered thermistor location (if applicable),
- grounding of MLI.

5.4.2. Conductive Interfaces

The PID-B shall contain a definition of the conductive interface. If this is the same as the mechanical interface then specific reference shall be made to the relevant section or drawing in the PID-B.

For each Collectively Controlled Unit the baseplate contact area with the PLM shall be defined.

5.4.3. Radiative Interfaces

The external finishes of the instrument (MLI, coatings, finishes etc.) shall be defined along with their optical properties at BOL and EOL.

For each instrument radiator, its area, field of view, required radiator temperatures and rejected heat shall be defined in the instrument PID-B. The instrument thermal design shall be capable of some adjustment to the radiator size and finishes to obtain a satisfactory thermal design with the fields of view obtained with the spacecraft configuration.

5.4.4. Thermal Heat Capacity

The Thermal Heat Capacity of each unit or module comprising the instrument shall be defined in the PID-B.

5.4.5. Instrument Temperature Measurement

The location, type and electrical interface of all devices used for instrument temperature measurement shall be defined in the PID-B.

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6. ELECTRICAL AND SOFTWARE

6.1. GENERAL

All parameters in Section 6 — Electrical and Software are specified under Worst-Case End-Of-Life conditions, unless otherwise explicitly notified.

Beginning-Of-Life criteria shall be derived by the Instrument contractor from the specified parameters for testing and acceptance of all on-board equipment.

6.2. POWER SUPPLY

6.2.1. Spacecraft Power Supply Characteristics

The Instruments will be provided with switchable protected and regulated power lines.

The power lines regulated voltage is V_{BUS} = 28 V ± 1%, as measured at the Spacecraft instrument distribution star point.

The source impedance for each power line will be as follows :

TBD

The power lines will be controlled by the Spacecraft by means of Solid State Power Switches (SSPC), so as to ensure the following functions :

- Instrument connection and disconnection,
- current limitation during a fixed duration in case of instrument Over-Current Detection (OCD),
- automatic disconnection of the instrument in case of instrument Over-Current Detection (OCD) exceeding the current limitation duration.

The SSPCs characteristics will be as follows :

- I_{NOM} = 2.5 A, 5 A or 10 A (depending on Instrument power consumption)
- $120\% \times I_{NOM} \le I_{LIMIT} \le 140\% \times I_{NOM}$
- $t_{TRIP\text{-}OFF}$ = 10 ms ± TBD ms

6.2.2. Power Allocation

The Mars Express Payload power will be as defined here after :

	OFF	Launch	Cruise	Stand-By	Operation
Mean Power	0 W	0 W	20 W (TBC)	55 W	140 W
Maximum Instantaneous Current	0 A	0 A	4 A (TBC)	4 A (TBC)	10 A (TBC)

6.2.3. Power Interface Design Requirements

All Instruments shall operate with nominal performances for steady state voltages comprised between 26 V and 29 V.

All Instruments shall be compatible with the following characteristics :

- During the satellite AIT phase, the power will be switched ON and OFF every day.
- The voltage rise and fall rates on the regulated power lines will be less than 50 V/ms.

All Instruments shall withstand, without damage or permanent performance degradation, the following conditions, irrespective of their configuration (modes, ON, OFF, etc.) :

- Permanent voltage in the range 0 V to 30 V (TBC),
- ± 28 V voltage between the primary 0 V and the mechanical ground,
- Short circuits of any duration on power buses.

6.2.4. High-Voltage Requirements / High-Voltage Equipment

The presence of high voltage parts shall be taken into account and proper discharge tests shall be performed so as to prove the hardening of the Instrument against high voltage breakdown.

Any high power microwave Instrument shall be designed and tested such that a 6 dB margin between the maximum expected power levels and the onset of multipaction is established.

All Instruments shall be designed so as to withstand without any degradation operation at any pressure between ambient and flight conditions.

In particular, any Instrument which has components operating at a voltage in excess of 100 V shall be designed so as to ensure that corona discharge and arcing do not occur for pressure between ambient and flight conditions.

When applicable, this requirement shall be demonstrated during the Instrument thermal vacuum test.

High voltage Instrument shall be designed to withstand the effects of arcs. The susceptibility of high voltage Instrument to arcing shall be characterised.

In case of intrinsic non-compliance with the here above requirements, all precautions and utilisation constraints aiming at avoiding instrument degradation due to high voltage effects shall be clearly listed in the Instrument User Manual.

6.3. DEPLOYMENT DEVICES

6.3.1. Pyro Devices

6.3.1.1. Functional Requirements

Pyro Devices actuation is under the responsibility of the spacecraft, and shall not be initiated by Instruments circuits.

Pyro lines are redundant.

Nominal and redundant Pyro circuits may be fired in any desired sequence. The delay between the beginning of a pulse and the beginning of the following one will be at least 100 ms.

The maximal delay is 1 mn (TBC) in nominal case.

6.3.1.2. Electrical Characteristics

The Pyro initiator device shall be DASSAULT 1 TAP WH 40 (AMD/BA) or OEA PN 4205100 type (TBC).

The resistance of the initiator filament will be : R = 1.05 Ω ± 0.10 Ω.

Pyro device interconnection harness resistance will be less than 100 mΩ.

In this case, the pyrotechnic command delivered to the initiator — when taking the harness into account — will have the following characteristics :

- current : $5\ A \leq I \leq 12.5\ A$
- pulse duration : 45 ms ± 10 ms
-

6.3.1.3. Pyro Lines Allocation

Up to 8 (TBC) redunded Pyro lines may be provided by the Mars Express spacecraft to the instruments for release of deployable items.

6.3.2. Other Deployment Devices

TBD

6.4. DATA HANDLING

6.4.1. General Description

Payload related Data Handling activities are as follows :

- payload Command & Control, including Instrument configuration and monitoring,
- payload Master Schedule management,
- payload Failure Detection Isolation and Recovery (FDIR) management,
- Science Data acquisition and storage,
- On-Board Time and Synchronisation Pulses distribution.

In addition, payload dedicated processing resources can be offered by the spacecraft processor.

High interface standardisation, based on circuits definition inherited from other ESA programmes allow to achieve optimum flexibility and to secure interface validation tasks.

As a baseline, Science Data are provided by the Instruments in the form of ESA PSS-04-106 standard Source Packets. The Mars Express policy is that low rate scientific data are collected through ESA TTC-01-B standard 16 bit Serial Acquisition links (baseline), or through a spacecraft mastered MIL-STD-1553 B data bus, and then fed into the Central Mass Memory. High rate scientific data (above 100 kbps) are directly routed to the Memory input through standard high speed digital links.

Configuration and mission commands towards the Orbiter Instruments are uplinked from Earth to the Spacecraft processor, which dispatches them to the Instruments. Landers telecommand packets are stored in the Spacecraft processor (or optionally on the lander elements on-board the spacecraft), and are dumped when in view of the landers.

A hierarchical FDIR policy will be applied to the payload. As a baseline, instruments featuring a built-in Command & Control processor will perform instrument level failure detection and passivation (no autonomous reconfiguration required). Each instrument processor will then be monitored by the spacecraft processing resources, which will perform corrective actions in case of failure. Non-intelligent instruments will be directly monitored by the spacecraft processing resources.

6.4.2. Interface Redundancy

The Data Handling will deliver main and redundant lines to the spacecraft items for each required function.

The redundant interface shall be used only in case of failure of the spacecraft main Data Handling section or other major spacecraft or payload malfunctions.

It is the user's responsibility to interconnect these lines according to the reliability scheme adopted to avoid failure propagation.

6.4.2.1. Spacecraft to Instruments Interface Redundancy

Data Handling signals coming from the Spacecraft will be available on two functional lines (one main and one redundant).

In case of a non-redunded Instrument, the Instrument shall connect to both (main and redundant) lines as per Figure 6.4.1. This ORing of the main and redundant signal shall be done in such a way that a permanent high or low level on one receiver does not invalidate the signal on the other receiver.

For a redunded Instrument, the nominal line shall be connected to the nominal branch and the redundant line to the redundant branch as per Figure 6.4.2. This shall be done in such a way that a single component failure has no impact on the performance of the other line coming from the Spacecraft.

6.4.2.2. Instruments to Spacecraft Interface Redundancy

Signal coming from the Instrument (telemetry data) will be acquired by the Spacecraft through two functional lines (one main and one redundant).

In case of a non-redunded Instrument, the Instrument shall feed not only the nominal input of the Spacecraft interface, but also the redundant one, each on dedicated lines as shown in Figure 6.4.3.

For a redunded Instrument, the nominal line of the Instrument shall be connected to the nominal Spacecraft interface input and the redundant line to the redundant Spacecraft interface input as per Figure 6.4.4. This shall be done in such a way that a single component failure has no impact on the performance of the other line acquired by the Spacecraft.

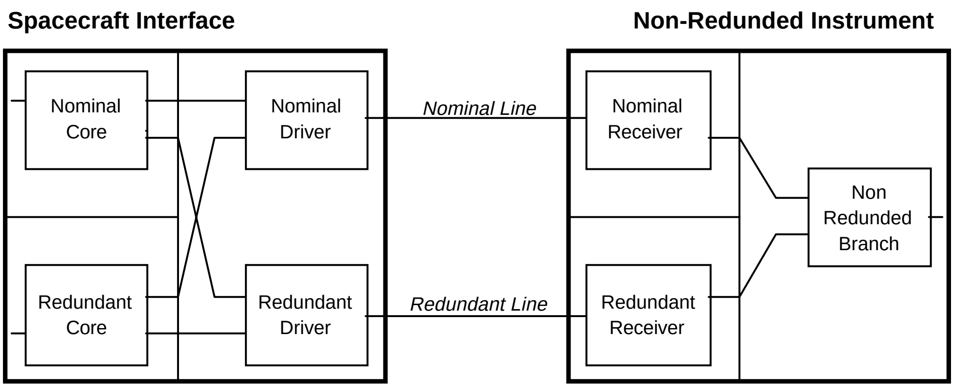


Figure 6.4.1 : Command Scheme for Non-Redunded Instrument

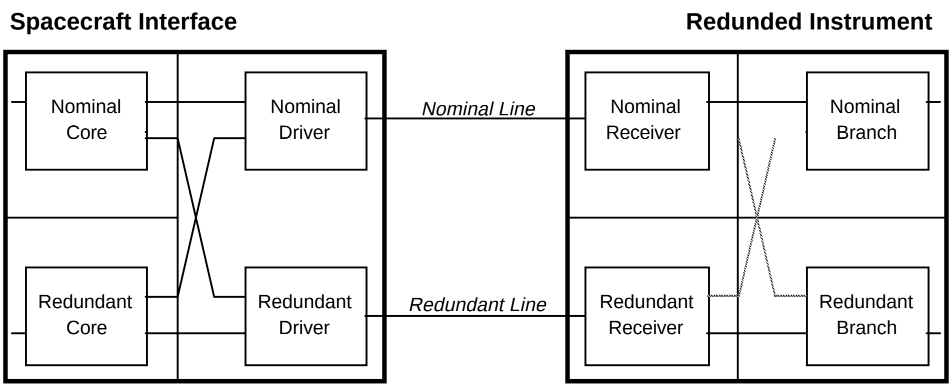


Figure 6.4.2 : Command Scheme for Redunded Instrument

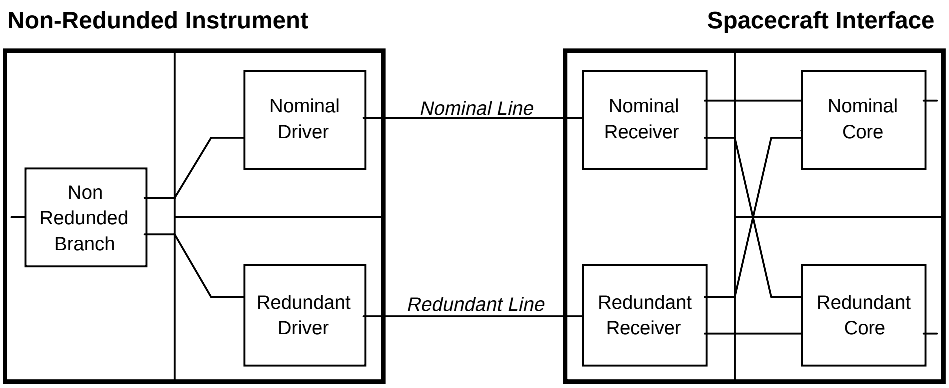


Figure 6.4.3 : Acquisition Scheme for Non-Redunded Instrument

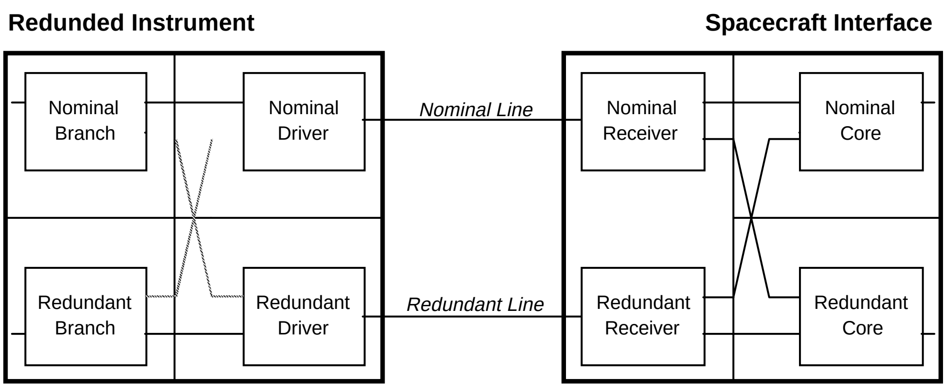


Figure 6.4.4 : Acquisition Scheme for Redunded Instrument

6.4.3. Science Data

6.4.3.1. Functional Description

As a baseline, Science Data collection will be performed on standard links.

Science Data with an instantaneous data rate lower than 80 kbps (TBC) will be collected as follows :

➢ through ESA TTC-01-B standard 16 bit Serial Acquisition lines, in order to be transmitted towards the spacecraft processor on the spacecraft data bus,

or

➢ through a MIL-STD-1553 B standard data bus data bus, the master of which is the spacecraft processor.

Science Data with an instantaneous data rate higher than 80 kbps (TBC) will be collected on IEEE-1355 like High Rate Data Links.

6.4.3.2. Source Packet Structure

Science Data shall be packetised at the instruments level, according to the Source Packet definition of ESA PSS-04-106 Packet Telemetry standard (see Figure 6.4.5).

Source Packets shall not be filled in with dummy data.

It is recommended not to use large source packets, due to the risk of packet loss during ground transmission.

Packet Header (48 bits)							**Packet Data Field (variable)**		
Packet ID 16 bits				Packet Sequence Control 16 bits		Packet Length 16 bits	Packet Field Header 72 bits	Source Data Variable	Packet Error Control (optional) 16 bits
Version Number 3 bits	Type 1 bit	Data Field Header Flag 1 bit	Appli-cation Process ID 11 bits	Segmen-tation Flags 2 bits	Source Sequence Count 14 bits				

Figure 6.4.5 : Telemetry Source Packet Structure

6.4.3.2.1. Packet Header Definition

The Source Packet Header fields definition is given here after.

PACKET ID

Version Number (Bit 0 - 2)

The version number bits shall be set to a value of 1.

Type (Bit 3)

The type bit shall be set to zero for all telemetry source packets.

Data Field Header Flag (Bit 4)

Indicates the presence of the Data Field Header and shall be set to 1.

Application Process ID (Bit 5 - 15)

The Application Process ID field shall uniquely identify the on board source of the packet. Guidelines for the choice of Application Process ID (APID) values across the spacecraft subsystems and experiments are given in TBD.

Note : Each Application Process ID is logically associated with the Source Sequence Count sub-field of the Packet Sequence Control field.

PACKET SEQUENCE CONTROL

Segmentation Flags (Bits 0, 1)

In the Source packet, the Segmentation Flags shall always be set to '11'B indicating that the source packet is not segmented by the data source.

Source Sequence Count (Bits 2 - 15)

The Source Sequence Count field shall represent the actual sequence count. A separate source sequence count shall be maintained for each APID and shall be incremented by 1 whenever the source (APID) releases a packet. Ideally, this counter shall never re-initialise, however under no circumstances shall it *short-cycle* (i.e. have a discontinuity other than to a value zero). The counter shall wrap around from 2^{14} -1 to zero, and shall start at zero at power on of the unit.

Note : The counter corresponds to the order of release of packets by the source and enables the ground to detect missing packets.

PACKET LENGTH

The Packet Length field shall specify the number of octets (minus one) contained within the Packet Data Field as binary value expressing (Number of octets in Data Field - 1).

For Mars Express, the maximum length of a Telemetry Source Packet Data Field is 4105 octets, including 4096 source data and 9 data field header octets.

6.4.3.2.2. Packet Data Field Definition

The Packet Data Field shall contain the information specific to the generating source on-board the Mars Express spacecraft.

DATA FIELD HEADER

All Data Field Headers shall have this same basic structure as shown on Figure 6.4.6 :

Time	PUS-Version	Checksum Flag	Spare	Packet Type	Packet Subtype
48 bits	3 bits	1 bit	4 bits	8 bits	8 bits
Mandatory	Mandatory	Mandatory	Mandatory	Mandatory	Mandatory

Figure 6.4.6 : Source Packet Data Field Header

Time

Defines the time that the acquisition of the data in the packet was initiated.

The time code format is provided in TBD.

PUS-Version

PUS-Version 1= '000'$_{BIN}$.

Checksum Flag

Indicates if there is a Packet Error Control Field at the end of the Packet Data Field.

Not used = '0'$_{BIN}$.

Spare

Set to all zeros, i.e. '0000'$_{BIN}$.

Packet Type

Indicates the type to which the telemetry source packet relates. The telemetry source packet types applicable to Mars Express are defined in TBD.

Packet Subtype

Together with the Type, the Subtype shall uniquely identify the nature of the telemetry contained within the telemetry source packet. The same Packet Type and Subtype definitions shall apply to all applications. The relationship between Telecommand and Telemetry Packet Types and Subtypes and the users to which they apply is given in TBD.

SOURCE DATA

The Source Data field shall constitute the data element of the TM reports to ground.

PACKET ERROR CONTROL

The Packet Error Control field shall not be used.

6.4.3.3. High Rate Data Acquisition Protocol

Protocol Layers

The High Rate Data (DS-Link) transmission protocol includes the following layered structure, derived from the IEEE 1355 Standard, as shown in Figure 6.4.7.

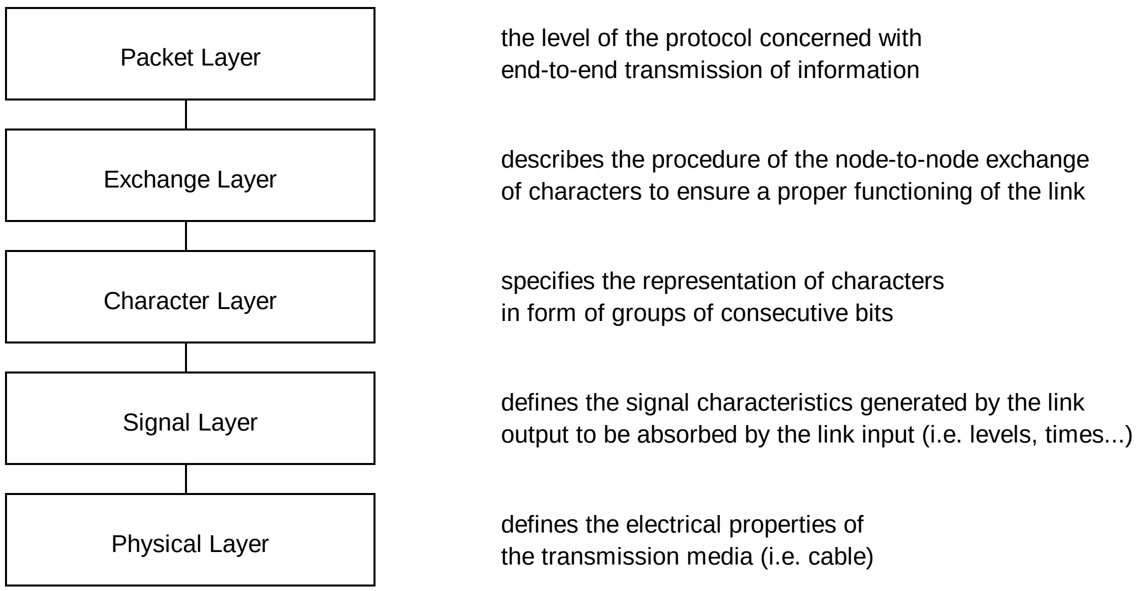


Figure 6.4.7 : 1355 Protocol Layers

The Physical Layer shall be as defined in Section • .

The Signal Layer shall be as defined in Section • .

The Character Layer shall be according to the IEEE 1355 Standard for DS-Link Character Level as described hereafter.

The Exchange-Layer shall be according to the IEEE 1355 Standard for DS-Link Exchange Level as described hereafter.

Character Layer

Bits shall be transmitted in groups called *characters*.

Note : The character level provides the service to the higher levels of the transmission of a continuous sequence of characters on a link.

Characters shall be either Data Characters or Control Characters. Each character shall consist of

- a parity bit,
- a control bit to distinguish between data and control characters,
- 8 bits of data (data character) or 2 indication bits (control character).

The data and control character structure is defined in Figure 6.4.8.

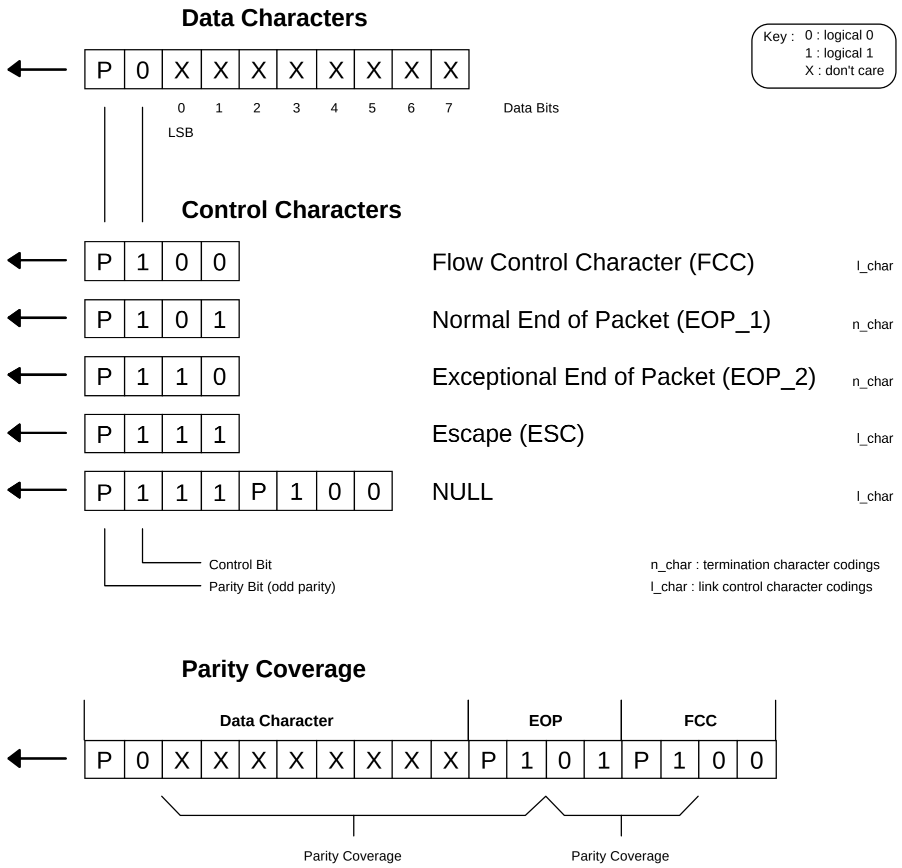


Figure 6.4.8 : High Rate Data Link Character Encoding and Parity Coverage

The parity bit in any character shall cover the information content of the previous character plus the control bit of the current character.

Odd parity checking shall be used.

Exchange Layer

Initialisation

After activation or a reset, the link shall maintain both signals at their reset level until started. Thereafter it shall only send NULL characters unless and until TBD. After that, the link shall commence normal operation.

Flow Control

Once a link output has been started it shall send characters continuously until instructed to cease operation or an error is detected. In the absence of other characters, the DS Link transmits NULL characters. The link shall only be active during each entire transmission sequence

In normal operation, characters provided for transmission are sent when there is flow control credit available. Flow control credit is issued by TBD.

6.4.4. Command & Control

As a baseline, instruments Command & Control will be performed by the spacecraft in the following way :

- through a MIL-STD-1553 B standard data bus data bus, the master of which is the spacecraft,

  or

- through ESA TTC-01-B standard Memory Load and 16 bit Serial Acquisition links, interfacing with the spacecraft internal data bus.

Telecommand Protocol

TBD

Telemetry Protocol

TBD

6.4.5. Standard Data Links

6.4.5.1. MIL-STD-1553 B Data Bus Physical Layer

The basic structure of MIL-STD-1553 B data bus words is given in Figure 6.4.9.

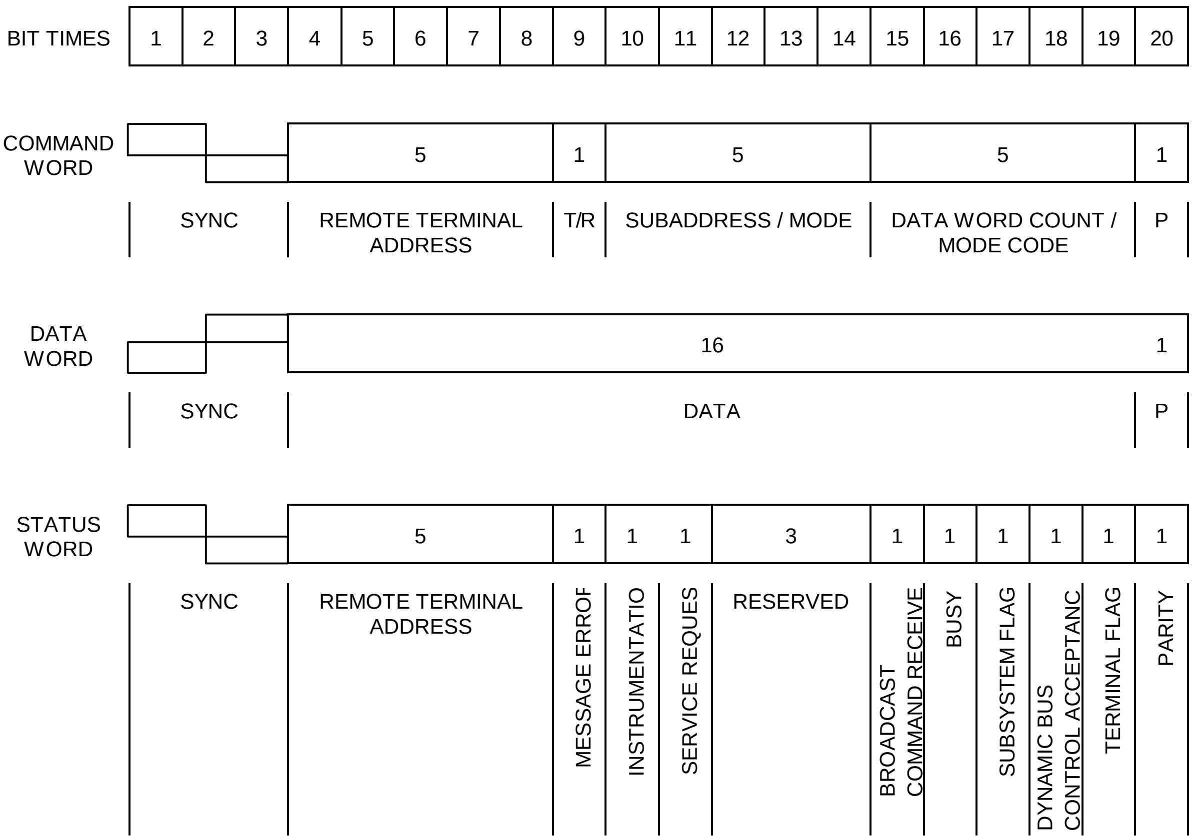


Figure 6.4.9 : MIL-STD-1553 B Data Bus Words Format

6.4.5.2. TTC-01-B Interfaces

6.4.5.2.1. Memory Load Commands

The purpose of the memory load command (or 16 Bit serial load command) interface is to transfer a 16 bit word, in serial form, from the Spacecraft to the Instruments.

The Memory Load Command interface consists of three lines for command transmission, issued from the Spacecraft :

- one data line (serial 16 Bit NRZ-L)
- one clock line (serial data transfer clock)
- sampling line(s) (address line(s))

Each Instrument shall be provided with only one redundant clock (Serial Data Transfer Clock) for Memory Load Command and Serial Digital acquisition, i.e. the clock is shared.

Only one redundant line for memory load data shall be provided by the Spacecraft even if more than one channel is distributed to the Instrument. Addressing of the channel(s) shall be performed by the specific Memory Load Sampling line(s).

Each Memory Load Command line signal shall be transmitted via a Standard Balanced Digital Link, with the cable shield connected at both sides of the interface to chassis ground.

The memory load data provided to the Instrument will be a serial NRZ-L PCM signal :

- A bit '1' corresponds to a positive differential voltage (high level voltage of true line / low level of comp line, with respect to signal ground),
- A bit '0' corresponds to a negative differential voltage (low level voltage of true line / high level of comp line, with respect to signal ground).

Outside of the memory load transfer period, the quiescent state of the 'true' lines shall be :

- the high level for the clock and the sample lines,
- the low level for the data lines.

Clock and data lines may be active outside the memory load transfer period.

The memory load command channel interface circuitry, including driver and receiver, as well as the interconnecting harness shall be as shown in Figure 6.4.10 and Figure 6.4.11.

Memory Load Command Timing Diagrams

The phase relation between the three signals for the 16 bit Memory Load Command will be as depicted in Figure 6.4.12.

The nominal value for the transfer clock frequency ($1/t_5$) will be 262.144 kHz, variable with the initial setting of the central On-Board Clock.

The various signal durations shown in Figure 6.4.12 will be as follows :

- t_1 = 61.0 µs
- t_2 = 3.8 µs + 1.5 /- 1.0 µs
- t_3 = 118.3 µs + 1.0 /- 1.5 µs
- t_4 = 26.7 µs + 1.0 /- 1.5 µs
- t_5 = 3.8 µs

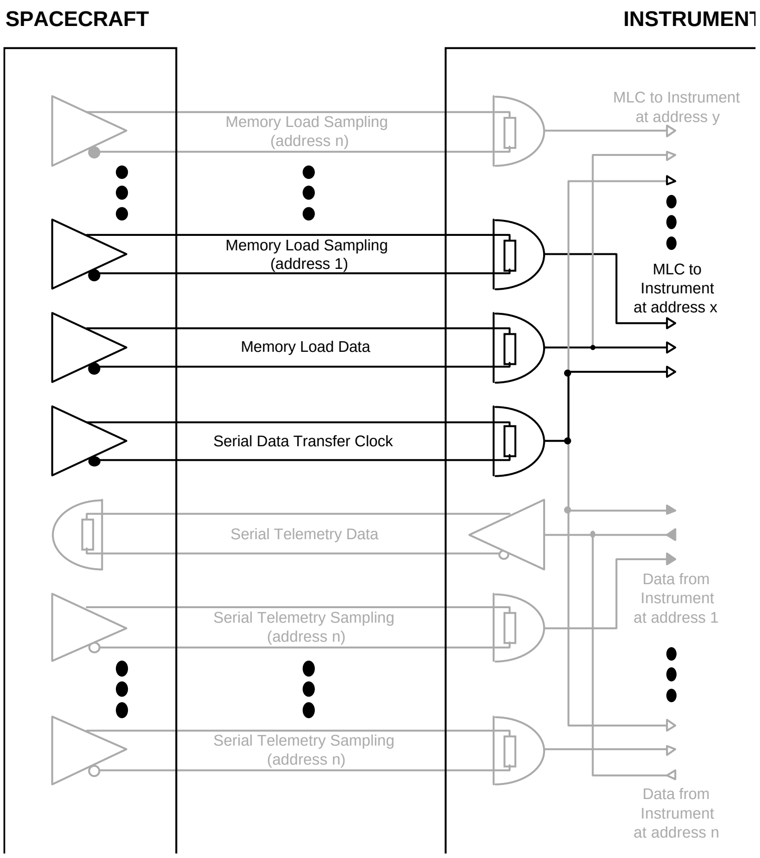


Figure 6.4.10 : Memory Load Command Interface

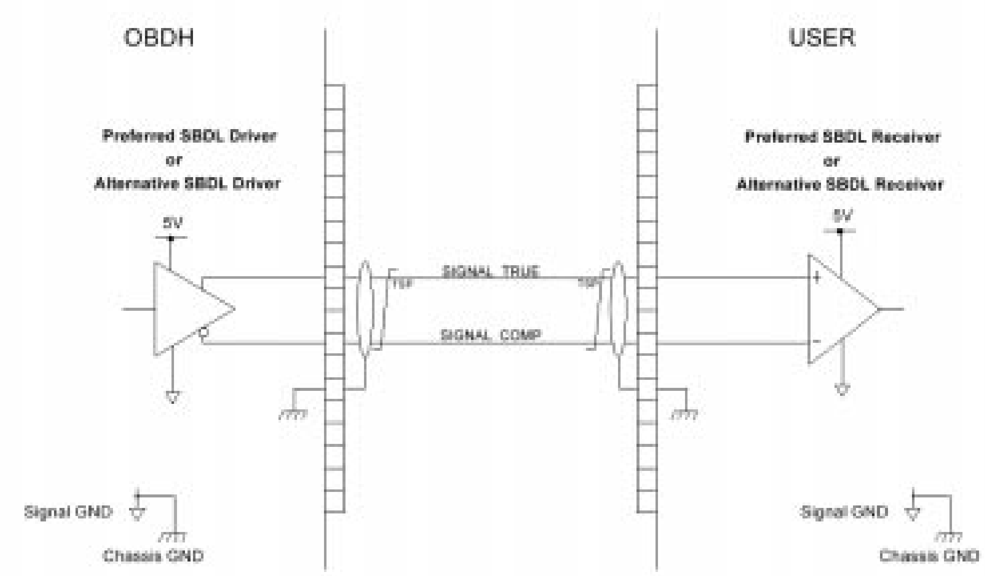


Figure 6.4.11 : Memory Load Command Typical Circuit Implementation

- t_6 = 1.9 µs + 0.5 µs
- t_7 < 3.8 µs
- t_8 < 1.4 µs + 0.5 /- 0.6 µs
- t_9 > 1.4 µs

The transmission rate will be 16 bit within 122 µs (131.072 kbps) with nominally no gaps in between the individual 16 bit Memory Load commands.

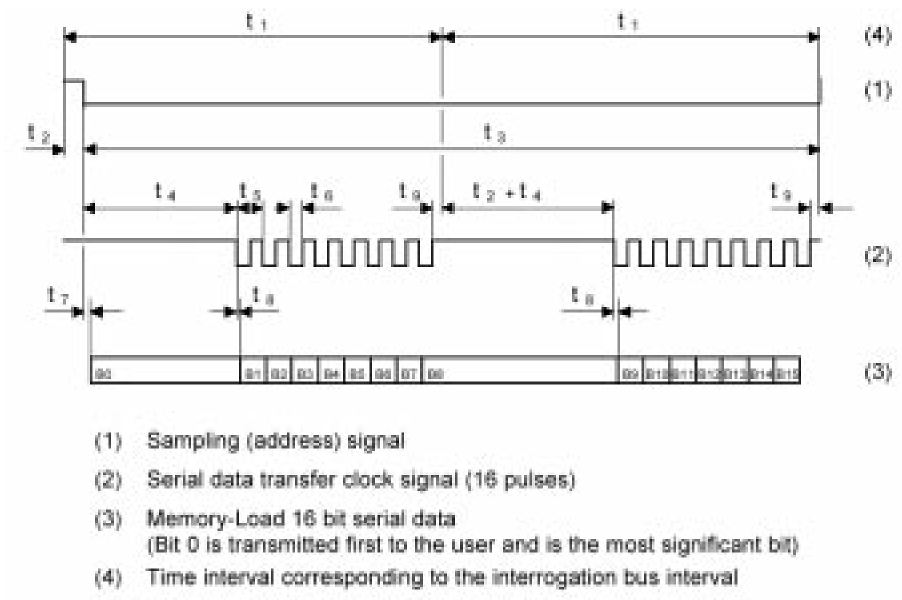


Figure 6.4.12 : Memory Load Command Timing Diagram

6.4.5.2.2. Serial 16 Bit Digital Channels

The Spacecraft will provide serial digital telemetry input channels for acquisition of both housekeeping data and Science Data. The purpose of the interface is to transfer, in serial form, a 16-bit data word from the user to the Spacecraft.

The function of the serial data input channel comprises three lines for telemetry transmission: the data line, issued from the Instrument, as well as the clock and sampling line, issued from the Spacecraft. The specific sampling / address line(s) defines the channel, as well as the time during which the transfer of the serial data will take place from the Instrument to the Spacecraft.

Each Instrument will be provided with only one redundant clock (Serial Data Transfer Clock) for Memory Load Command and Serial Digital acquisition, i.e. the clock is shared.

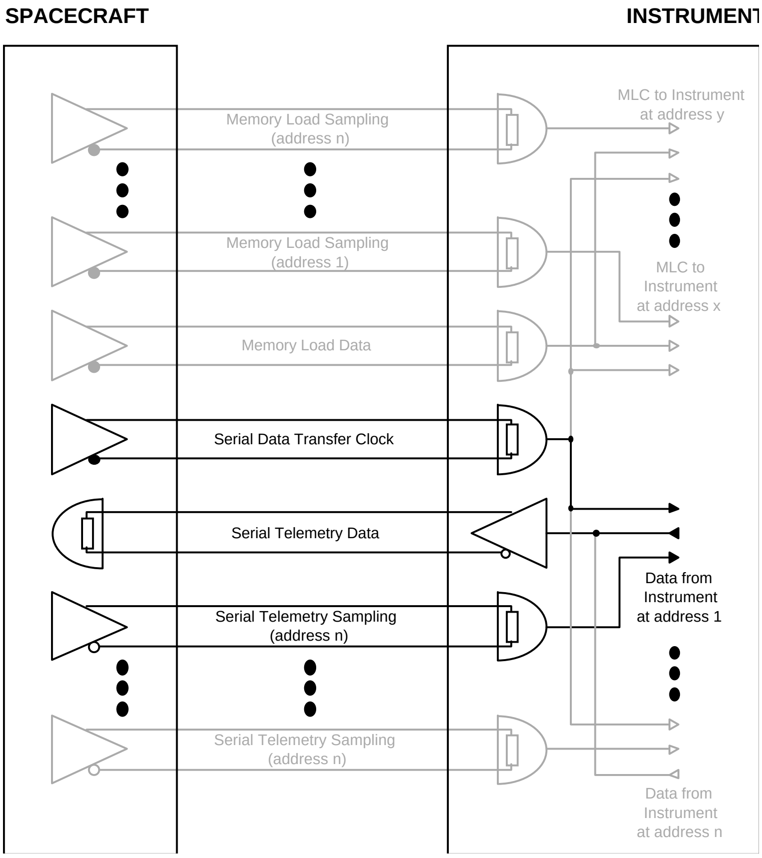


Figure 6.4.13 : Serial 16 Bit Digital Telemetry Interface

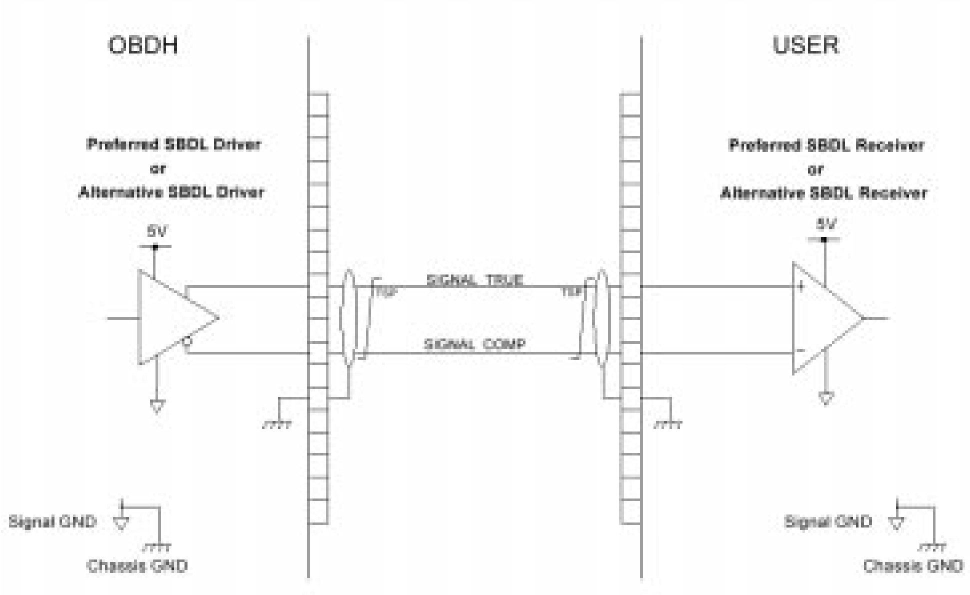


Figure 6.4.14 : Serial 16 Bit Digital Sampling and Transfer Clock Typical Circuit Implementation

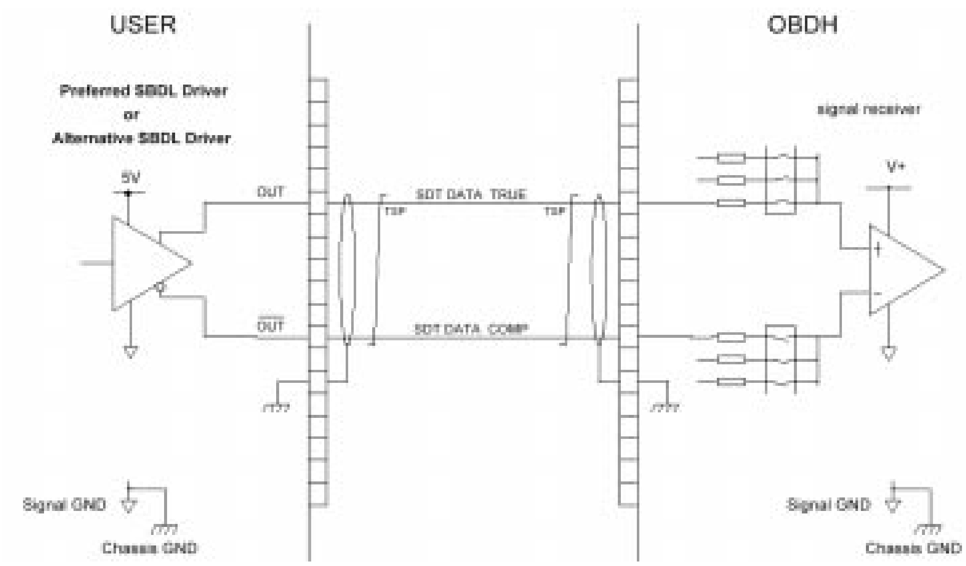


Figure 6.4.15 : Serial 16 Bit Digital Data Line Typical Circuit Implementation

Only one redundant line for serial digital data shall be provided by the Instrument even if more than one channel is acquired by the Spacecraft. The channel(s) shall be addressed by the SDT Sampling line(s).

Each Serial Digital Telemetry line signal shall be transmitted via a Standard Balanced Digital Link, with the cable shield connected at both sides of the interface to chassis ground.

The serial digital data provided by the Instrument shall be a serial NRZ-L PCM signal. A bit '1' corresponds to the high level and a bit '0' corresponds to the low level on the 'true' line, with respect to signal ground. The Most Significant Bit (MSB) of the serial data shall be transmitted first (from the Instrument to the Spacecraft) within the sampling interval. The acquisition of the following bits shall then be controlled by the trailing edge of the transfer clock.

Outside of the serial data transfer period, the quiescent state of the 'true' lines shall be the high level.

The clock line may be active outside the serial data transfer period.

The serial 16 bit digital channel interface circuitry, including driver and receiver, as well as the interconnecting harness shall be as shown in Figure 6.4.13, Figure 6.4.14 and Figure 6.4.15.

Serial 16 Bit Digital Telemetry Timing Diagrams

The phase relation between the three signals for the 16 bit serial channel shall be as shown in Figure 6.4.16.

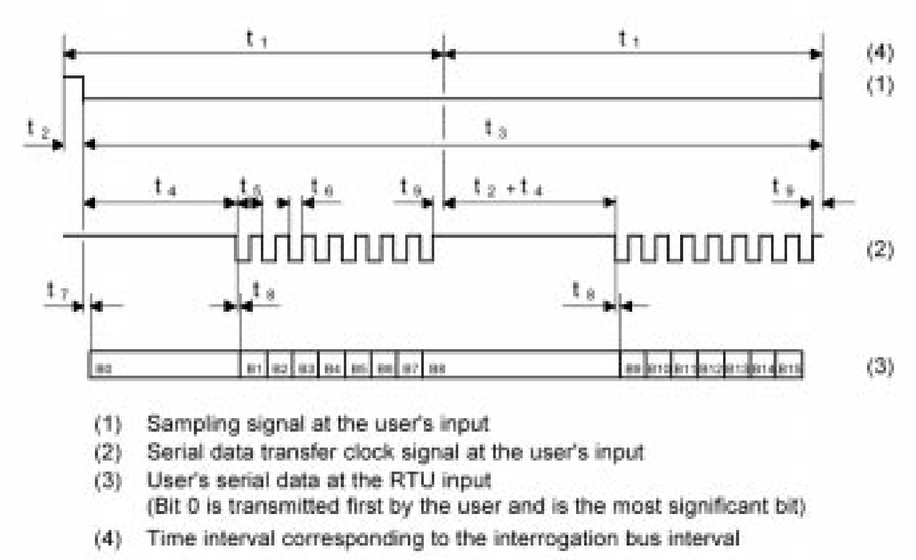


Figure 6.4.16 : Serial 16 Bit Digital Timing Diagram

The various timing values shown in Figure 6.4.16 shall be as follows:

- t_1 = 61.0 µs
- t_2 = 3.8 µs + 1 µs
- t_3 = 118.3 µs + 1 µs
- t_4 = 26.7 µs + 1 µs
- t_5 = 3.8 µs
- t_6 = 1.9 µs + 0.5 µs
- t_7 < 15.3 µs
- t_8 < 1.1 µs
- t_9 > 1.4 µs

The Acquisition rate will be 131.072 kbps.

The time interval (start to start) between two consecutive telemetry 16 bit read-outs shall be :

122 µs + 1 µs.

The Instrument is responsible to reload the channel register if consecutive read-out on the same telemetry channel is required.

6.4.5.3. IEEE-1355 High Rate Data Links

The High Rate Data interface is used to acquire high speed data (i.e. with an instantaneous rate higher than 80 kbps TBC) from the Payload Instruments.

The High Rate Data interface shall be implemented close to the DS-link defined in the IEEE STD 1355-1995.

On Mars Express, the baseline is that High Rate Data flow only from the Instruments to the Central Mass Memory input.

A DS link shall consist of one (TBC) unidirectional, point-to-point link from the Instrument to the Central Mass Memory input. The unidirectional interface shall consist of two lines, issued from the active data source (see Figure 6.4.17) :

- DS_Data (serial bit stream),
- DS_Strobe (decoded strobe).

The data-strobe system shall carry an 'encoded' clock as shown in Figure 6.4.18. The receiving device will synchronise to the incoming data (asynchronous).

On power on, the DS link outputs shall hold both the data and strobe signals at logic '0' (i.e. the reset level).Each connection will be a differentially balanced pair link via impedance controlled transmission line with the cable shield connected at both sides of the interface to chassis ground.

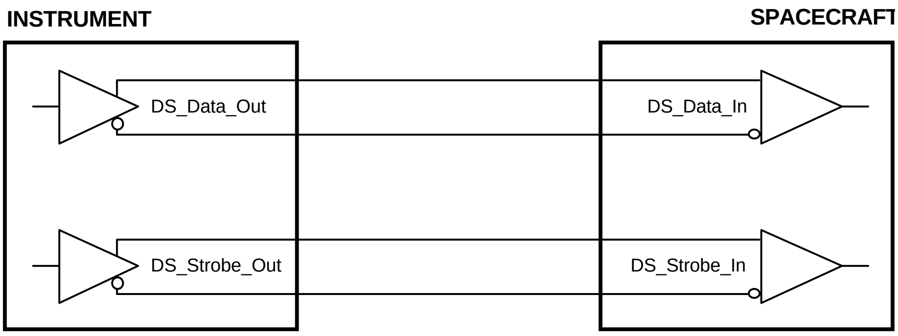
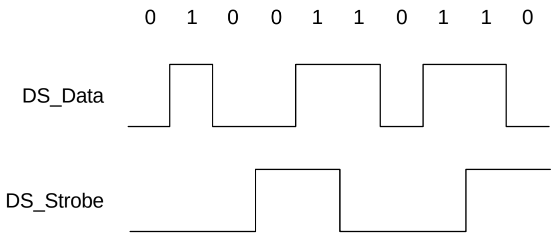


Figure 6.4.17 : High Rate Data Link



Note : The strobe line shall change state each time when the next bit on the accompanying data line has the same value than its previous one. (Clock recovery by XORing)

Figure 6.4.18 : DS Link Signal Encoding

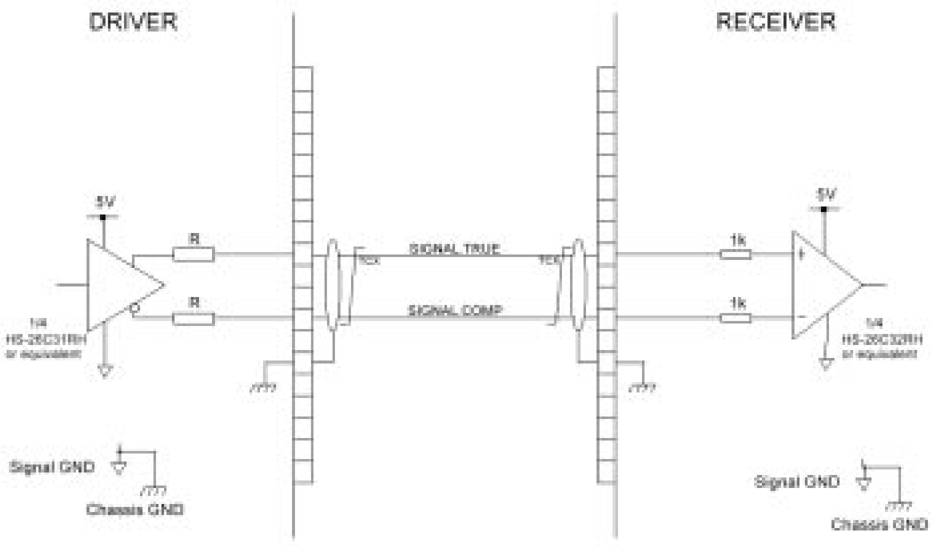


Figure 6.4.19 : High Rate Data Link Typical Circuit Implementation

The serial termination resistors at the driver output shall be optimised (each $R_0 = Z_0/2$) to match exactly the transmission line impedance (Z_0) to avoid overdamping or over-/undershooting effects at the receiver input.

The interface circuitry, including driver and receiver, as well as the interconnecting harness shall be as shown in Figure 6.4.17 and Figure 6.4.19.

Source Circuit

Circuit type	Balanced CMOS driver
Bit rate	$\leq$ 10 Mbps
Low level output voltage	$0\text{ V} \leq V_{OL} \leq 0.55\text{ V}$
High level output voltage	$2.45\text{ V} \leq V_{OH} \leq 5.5\text{ V}$
Differential output voltage	Logical 1 : $+1.9\text{ V} \leq V_{OD} \leq +5.5\text{ V}$
	Logical 0 : $-5.5\text{ V} \leq V_{OD} \leq -1.9\text{ V}$
Rise and fall times	$\tau_R \leq 20$ ns, $\tau_F \leq 20$ ns
Current limitation	$I \leq 100$ mA
Maximum fault voltage	Tolerance : $-3\text{ V} \leq V_{OD} \leq +12\text{ V}$
	Emission : $-0.5\text{ V} \leq V_{OD} \leq +7\text{ V}$

Receiver Circuit

Circuit type	Differential receiver, CMOS compatible
Low level input voltage	$V_{ID} \leq -1$ V
High level input voltage	$V_{ID} \geq +1$ V

Switching shall only occur for voltage steps greater than 1 V.

Differential input impedance	$Z_I \geq 10\text{ k}\Omega$
Maximum fault voltage	Tolerance : $-3\text{ V} \leq V_{OD} \leq +12\text{ V}$
	Emission : $-0.5\text{ V} \leq V_{OD} \leq +7\text{ V}$

Harness will be a twisted shielded pair with a 125 Ω characteristic impedance.

6.4.6. Discrete Telemetry

As a baseline, the spacecraft will acquire discrete telemetry lines from the instruments for housekeeping and monitoring purposes. The following telemetry types can be acquired by the PIU :

- status telemetry,

- analogue voltage acquisitions,
- thermistor acquisitions.

6.4.6.1. Status Telemetry

Status telemetry will be used to acquire the status of relays switched by High Level Commands, in the form of a dry contact (TBC).

A closed contact, corresponding to a switched on or enabled function, will be encoded as a 0 (TBC) in the Mars Express telemetry flow.

An open contact, corresponding to a switched off or disabled function, will be encoded as a 1 (TBC) in the Mars Express telemetry flow.

Electrical Characteristics

Circuit Type	Relay contact (= dry contact status)
Contact Resistance	Closed contact : $R \leq 50\ \Omega$
	Open contact : $R \geq 1\ M\Omega$
Current	$I \geq 10$ mA
Voltage Tolerance	$-16\ V \leq V_{FAULT} \leq +16\ V$, as seen through a 1.5 kΩ source impedance

6.4.6.2. Analogue Voltage Acquisitions

Analogue Voltage Acquisitions will be used to monitor instrument level health criteria, such as DC/DC Converter secondary voltages or primary currents.

Analogue Voltage encoding will be performed on 8 bits at Mars Express spacecraft level, with #00 (TBC) corresponding to 0 V and #FF (TBC) corresponding to 5.12 V (full scale).

Emitter characteristics

Circuit Type	TBD
Output Voltage Range	$0\ V \leq V_{ANA} \leq 5.12\ V$
Output Current	$I \leq 10$ mA
Output Impedance	$Z_O \leq 5\ k\Omega$
Maximum Fault Voltage	Tolerance : $-3\ V \leq V_{OD} \leq +12\ V$
	Emission : $-0.5\ V \leq V_{OD} \leq +7\ V$

Receiver characteristics

Circuit Type	Differential receiver
Input Impedance	$Z_I \geq 6\ M\Omega$ (except when Spacecraft interface is off : $R_I \geq 1\ k\Omega$)

Input Capacitance $C_I \leq 180$ pF

6.4.6.3. Thermistor Acquisitions

Thermistor Acquisitions will be used to monitor instrument level temperature health criteria, such as DC/DC Converter temperature or focal plane temperature.

The thermistor type shall be TBD.

Both thermistor bundles shall be floated with respect to the instrument chassis, the primary reference points and the Unit Secondary Reference Point (USRP).

Acquisition voltage $V \leq 28$ V (TBC)

6.4.6.4. Discrete Telemetry Allocation

The spacecraft will acquire the following Discrete Telemetries from the whole set of Payload Instruments :

- up to 16 (TBC) Status Telemetry,
- up to 16 (TBC) Analogue Voltage Acquisitions,
- up to 16 (TBC) Thermistor Acquisitions.

6.4.7. Configuration Commands

High Level Commands (HLC) will be sent to the instruments from the spacecraft for instrument configuration purposes.

Electrical Characteristics

Command Type	Single Ended Driver
Passive Level	$0 \text{ V} \leq V_{OL} \leq 0.5$ V
Active Level	$12 \text{ V} \leq V_{OH} \leq 16$ V
Command Duration	11 ms to 30 ms
Rise and Fall Times	$50 \text{ ns} \leq \tau_R \leq 500$ ns (when loaded by 2 nF)
	$50 \text{ ns} \leq \tau_F \leq 500$ ns (when loaded by 2 nF)
Current Capability	$I \geq 180$ mA
Current Limitation	$I \leq 300$ mA
Receiver Input Capacitance	$C \leq 300$ pF
Receiver Fault Voltage	Tolerance : $-2 \text{ V} \leq V_{OD} \leq +18$ V
	Emission : $-2 \text{ V} \leq V_{OD} \leq +25$ V

High Level Commands shall directly interface with the power relays associated with the different instrument functions or optocouplers, without it being necessary to use any other means.

No switching whatsoever shall be performed by the relays when submitted to pulses with the following characteristics :

- $0\ V \leq V \leq 16\ V$
- $d \leq 100\ \mu s$

No switching whatsoever shall be performed by the relays when submitted to pulses with the following characteristics :

- $0\ V \leq V \leq 2.5\ V$
- $d \geq 11$ ms

6.4.7.1. Configuration Commands Allocation

The spacecraft will generate up to 32 (TBC)Configuration Commands towards the whole set of Payload Instruments.

6.4.8. Emergency Signals

The Mars Express spacecraft will generate Emergency Signals towards the instruments for emergency switch off in case of instrument failure which might impact the spacecraft integrity.

The Emergency Signal shall be used by the Instruments for which it is not possible to perform emergency switch-off by external disconnection of the input power line.

Upon Emergency Signal reception, the instrument shall immediately switch off.

The Emergency Signal processing by the instrument shall not involve the instrument software.

As far as possible, the Emergency Signal shall directly trigger power relays or optocouplers, except in case instrument operational constraints prevents it.

Any time constraint between the Emergency Signal reception and the actual instrument switch off (due to e.g. need to close instrument protection cover or need for an internally sequenced switch off procedure) shall be reported to the Mars Express Prime Contractor for agreement.

Electrical Characteristics

Command Type	Single Ended Driver
Passive Level	$0\ V \leq V_{OL} \leq 0.5\ V$
Active Level	$12\ V \leq V_{OH} \leq 16\ V$
Command Duration	11 ms to 30 ms

Rise and Fall Times	50 ns $\leq \tau_R \leq$ 500 ns (when loaded by 2 nF)
	50 ns $\leq \tau_F \leq$ 500 ns (when loaded by 2 nF)
Current Capability	I $\geq$ 180 mA
Current Limitation	I $\leq$ 300 mA
Receiver Input Capacitance	C $\leq$ 300 pF
Receiver Fault Voltage	Tolerance : - 2 V $\leq V_{OD} \leq$ + 18 V
	Emission : - 2 V $\leq V_{OD} \leq$ + 25 V

No emergency switch-off whatsoever shall be performed by the input interface circuit when submitted to pulses with the following characteristics :

- 0 V $\leq$ V $\leq$ 16 V
- d $\leq$ 100 µs

6.5. CLOCKS AND SYNCHRONISATION

6.5.1. Clock Line

The Mars Express spacecraft will deliver one Clock Line to all the Instruments in order to allow internal time stamping of Instrument generated Source Packet.

Electrical Characteristics

Differential Output Voltage	Logical '1' : 3.5 V < V_{OD} < 5.5 V Logical '0' : -5.5 V < V_{OD} < -3.5 V
Emitter Rise and Fall Times	(τ_R, τ_F) < 700 ns (EOL) when loaded with 1.2 nF (differential)
Receiver Circuit Type	Differential Receiver CMOS compatible
Differential Input Voltage	Low : V_{ID} < - 1 V High : V_{ID} > + 1 V
Receiver Hysteresis	$\Delta V > 1$ V
Differential Input Impedance	Z_{ID} > 10 kΩ (DC: > 10 kΩ ; AC: 120 Ω in series with 100pF)
Maximum Fault Voltage	Tolerance : -3 V to +14 V Emission : -2 V to +14 V (3)
Clock Frequency	f = 131,072 Hz (= 2^{17} Hz)
Duty Cycle	47.5% ≤ δ ≤ 52.5%
Clock Stability	Initial Setting : $\Delta f \leq 3$ ppm (TBC) @ 25 °C Long Term Stability : $\Delta f \leq 30$ ppm (TBC) with time and temperature

The Clock Line reception circuit shall be implemented as per Figure 6.5.1.

6.5.2. Synchronisation Pulse

The Mars Express spacecraft will deliver a Synchronisation Pulse to the instruments in order to allow internal time stamping of Instrument generated Source Packet.

The Synchronisation Pulse will be delivered continuously to the Instruments.

Electrical Characteristics

Differential Output Voltage	Logical '1' : 3.5 V < V_{OD} < 5.5 V Logical '0' : -5.5 V < V_{OD} < -3.5 V

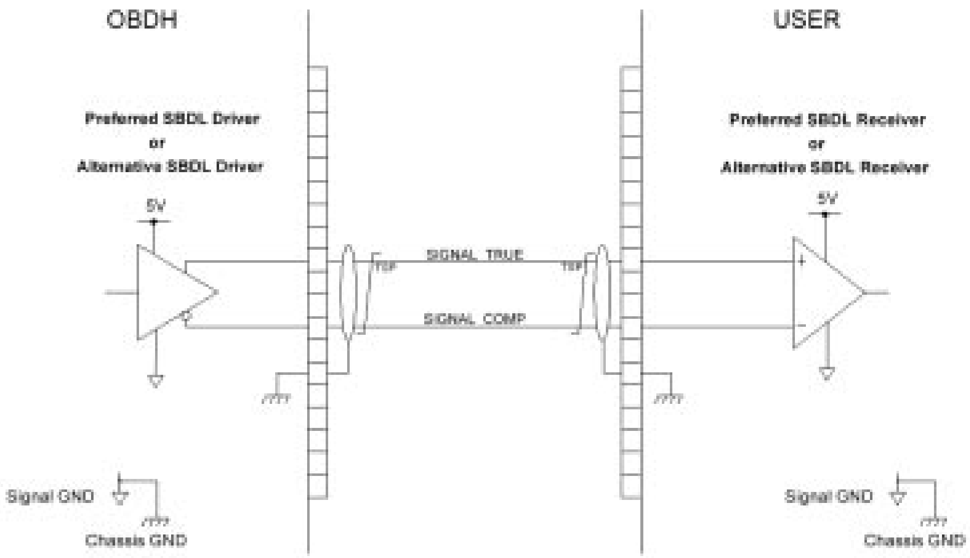


Figure 6.5.1 : Clock Line and Synchronisation Pulse Distribution Circuit Typical Implementation

Emitter Rise and Fall Times	(τ_R, τ_F) < 700 ns (EOL) when loaded with 1.2 nF (differential)
Receiver Circuit Type	Differential Receiver CMOS compatible
Differential Input Voltage	Low : V_{ID} < - 1 V High : V_{ID} > + 1 V
Receiver Hysteresis	$\Delta V > 1$ V
Differential Input Impedance	Z_{ID} > 10 kΩ (DC: > 10 kΩ ; AC: 120 Ω in series with 100pF)
Maximum Fault Voltage	Tolerance : -3 V to +14 V Emission : -2 V to +14 V (3)
Pulse Width	$3.8\ \mu s \leq d \leq 62\ \mu s$
Jitter	less than 2 µs

The Synchronisation Pulse reception circuit shall be implemented as per Figure 6.5.1.

6.5.3. On-Board Time Management

At Instrument switch-on and after request (typically once per day TBC), a Time Packet will be distributed to the respective Instruments.

The rising (first) edge of the next Synchronisation Pulse shall be used to set the Instrument internal timer with the On-Board Time transmitted within the Time Packet.

The instrument shall maintain an image of the Mars Express spacecraft On-Board Time in the form of a register with the following characteristics :

- On-Board Time coded on TBD bits
- LSB = TBD
- Behaviour upon overflow : TBD
- Synchronisation accuracy with respect to Mars Express spacecraft On-Board Time : TBD

6.6. PAYLOAD SOFTWARE AND AUTONOMOUS FUNCTIONS

6.6.1. Satellite/Payload Software Interface

The command/control of all instruments is performed by the Central Processor on-board the spacecraft. The Central Processor manages the flow of housekeeping and scientific data for the different instruments and offers packetisation service to scientific data and has the capability to write and read into/from the Central Mass Memory.

A payload application software runs in the Central Processor and interfaces with the payload software by means of TM/TC.

The payload software shall be autonomous with respect to the spacecraft software. The only interface between the Central Processor and the payload software consists in TM/TC protocols.

The payload interface with the spacecraft shall be done through a dedicated « Payload Processing Unit » (PPU), well identified in the payload hardware. Lower payload software levels relative to the PPU shall be transparent to the Central Processor.

For new payloads, the interface between the Payload Processing Unit and the spacecraft Central Processor shall be done through a MIL-STD-1553 bus. For existing payload, the interface shall be clearly described (serial lines, MIL-STD-1553, OBDH, etc.).

The Central Processor is able to deliver auxiliary data to the payload. Therefore, payloads are requested to describe the auxiliary data required from the spacecraft for correct functioning.

Payload monitoring will be performed at payload PPU level, by means of high level information such as temperature, secondary voltage, secondary current, units status, FDIR information, etc. Those data shall be transmitted to the spacecraft.

6.6.2. General Rules for Payload Software

All payload software shall comply with the ESA software standard PSS-05-0.

In order to ease in-flight maintenance, it is recommended to respect the following rules :

- Functionally distinct areas of memory shall be assigned to code, fixed constants and variable parameters.
- Payload software shall be structured and spare gaps put between modules, so that modifications can be made without affecting other module positions in the memory.
- It shall be possible to modify any individual software parameter or constant by command from the ground.

- Information to indicate all action of operational significance taken by on-board software in a complete, unambiguous and timely manner shall be available in the TM housekeeping or event packets.
- At experiment power switch on, the instrument shall first enter a safe mode with software running in PROM, allowing for command execution, housekeeping TM generation and RAM dump (this requirement allows recovery from a failed patch in the RAM software).
- If check of RAM code fails, then the instrument shall enter the Safe Mode.
- The Safe Mode shall allow dump and patch of RAM software as well as a jump to RAM code.
- Maintain a watch-dog to detect lock-ups and endless loops (this should trigger a recovery to a safe mode).
- Keep record of detected failures and indicate them to the DMS in an event packet. This record shall be cleared by TC only.
-

6.6.3. On-Board Memory Loading

It shall be possible to load any memory area from ground.

The areas of memory to be loaded shall be structured so that the TC packets needed for this are compatible with the maximum packet size limitation of one packet.

Every TC packet needed to update any area of memory shall be self consistent in such a way that :

- the successful load shall not depend on previous packets,
- if the packet is rejected or if the load fails for this packet, it may be up-linked alone at a later time,
- the rejection or failure of a packet shall be clearly and unambiguously indicated by a TC verification packet giving the TC packet sequence number and reason for failure in a standard way (see packet services).

The packets required to up link the full change shall be accepted by the experiment at the full up-link command rate without the need for any pauses between packets.

Patches should remain even after the experiment has been switched off.

6.6.4. On-Board Memory Dumping

Any memory area shall be accessible for dumping to ground, upon request by the ground. The dump request shall by a standard service that specifies the name of the memory to be dumped, the start address and the length of the dump.

Only one single TC shall be required to dump multiple areas of memory, even if several packets are required to convey the dump data to the ground.

6.6.5. Autonomy Concept and Operability

The Mars Express baseline autonomy concept is such that :

- Low level monitoring and actions are performed by the instrument and signalled to the ground and DMS by event and housekeeping packets.
- High level monitoring and recovery actions are performed by the ground or DMS using predefined procedures and operating on event and housekeeping packets only.

The instrument shall inform the DMS about all autonomous actions it has performed together with all data required for monitoring this action, in a single event packet.

The instrument shall perform a full boot-up to an operational mode at power switch on.

Instrument mode switching shall be performed by a single high level command.

The instrument shall perform a self check at power switch on and inform the DMS in an event packet. It is recommended to use a failure flag for critical cases.

The instrument shall perform self monitoring at regular intervals. If an error is detected, then the instrument shall enter the Safe Mode autonomously, inform the DMS in an event packet and await DMS to perform a predefined procedure to recover experiment.

The instrument shall implement any required time-lining, sensor sweeps/stepping using a single high level command.

6.7. CONNECTORS AND HARNESS

6.7.1. Connectors Design Requirements

6.7.1.1. Connector Types and General Characteristics

Connectors shall be space qualified. They shall be Hi-Rel golden connectors (TBC) on the PFM and FM models.

EM may potentially be used as FM or PFM integration spares, and thus subject to thermal vacuum testing.

EM units shall be equipped with golden connectors housings featuring Hi-Rel pins and/or sockets (TBC).

Please note that golden connector housings are only required for Sub-D type connectors.

The use of MDM microconnectors is prohibited on the Mars Express Satellite.

Instrument mounted connectors which provide electrical power shall feature female contacts (sockets).Equipment-mounted connectors shall have male contacts (pins), except for those which deliver power (regulators, converters).

One dedicated connector shall be used for each power supplying function.

Connector savers shall be used during ground testing and shall be supplied by the instrument contractor.

Connector pins carrying redundant functions shall be physically separated and isolated from each other

Functional connectors shall not include test points.

Male test connectors accessible by the Mars Express Prime Contractor at the integrated module level shall be fitted with protective metallic shell that may be used in flight. These shells shall be supplied by the instrument contractor.

Coaxial links shall use dedicated coaxial connectors. Test points may, however, be found on coaxial/non-coaxial hybrid connectors.

All connector shells shall be made of metallic material.

Equipment-mounted connectors shall be arranged in such a way that the harness connectors can be mated and demated without the need for special tools and without touching any neighbouring connectors. In particular, the minimum free space between neighbouring connectors shall be 5 mm.

6.7.1.2. Connector Contacts

At the time of the preliminary definition review (PDR), each connector shall offer a 10% margin of unused contacts.

As far as possible, signal type separation shall be respected at the level of the connectors : only one signal category shall be used on the same connector.

The Unit Secondary Reference Point (USRP) shall be made accessible to the Mars Express Prime Contractor via a connector.

Connector pins with the same name shall be interconnected inside the unit in question.

The equipment shall be responsible for grounding all housing contacts, by means of a wire less than 6 cm in length with a resistance of less than 15 mΩ.

6.7.2. Harness Design Requirements

6.7.2.1. Main Characteristics

Depending on the signals they carry, cables are classified into four categories :

- Category 1 : Power supply and relay power commands,
- Category 2 : Low-frequency electrical signals (frequency < 100 kHz or pulses with rise and fall times exceeding 5 µs),
- Category 3 : Pyrotechnic orders,
- Category 4 : High-frequency electrical signals (frequency > 100 kHz or pulses with rise and fall times of less than 5 µs).

The design requirements are as follows :

- All cables shall be space qualified.
- All conductors connected in parallel shall be so connected within the relevant unit or on a Terminal Junction Module designed to that effect.
- The harness will be defined and/or modified in accordance with the EMC requirements, the results of design studies, and any problems encountered during the integration and testing of the EM modules. The overall physical design of the cabling system shall thus ensure maximum flexibility.

6.7.2.2. Separation of Cables and Functions — Redundancy

Where different categories of cables are to be routed together with no metallic separation, every effort shall be made to maintain, wherever possible, a separation of at least 5 cm along the entire harness routing. This requirement is particularly important for cable of Categories 1 and 3. Any non-compliance with this rule shall be submitted to the S/C Contractor for approval.

Cables shall be provided so as to ensure redundancy as defined at the level of the connector pin function.

In cases where these requirements cannot be met due to physical constraints, cables may be grouped, in order of preference, as follows :

1) categories 2 and 4

2) categories 1, 2 and 4

Harness wires carrying redundant functions shall be physically separated and isolated from each other

6.7.2.3. Cable types and signals

Harness design rules are defined depending on the Signal Categories.

Category 1

Twisted, unshielded cables shall be used, unless otherwise recommended following EMC analysis or test results.

Category 2

Twisted, shielded pairs or triplets with shield grounded to mechanical ground at both ends shall be used. In some special cases, groups of more than three twisted and over-shielded cables may be used, subject to the approval of the Module Contractor.

Category 3

Each Pyro device shall be powered by twisted shielded pairs, physically separated from other categories cables

Category 4

The cables shall be routed close to the satellite structure. They shall be made out of twisted, shielded pairs grounded to mechanical ground at both ends, or coaxial cables.

RF links shall use coaxial cables or waveguides.

6.7.2.4. Cable shields

With the exception of coaxial cables, cable shields shall not be used as intentional return paths.

Coaxial cables shall not be used wherever the risk of interference can be reduced by any other means.

The length of unshielded sections of cable shall be kept to a minimum. As far as possible, such sections shall be entirely contained within connector shells.Shield continuity shall be ensured from source to user, including the passage through connectors, junction boxes and units.

The Category 3 cables shielding shall be grounded in several points.

The Category 3 cables shielding shall be continuous and with a total coverage over the bundle (2p coverage).

The Category 3 cables shielding may be interrupted at the level of interface connector brackets : in this case, the shielding interruption shall be as limited as possible, and a connector pin shall be allocated to the shielding.

This does not apply to the Pyro device connector itself, which is by definition limited to two (2) pins.

RF shields shall extend to the interior of connector shells, all the way to the connector contacts.

All shields shall be grounded to the instrument housing by the most direct means possible, each shield being grounded individually (no series-linked chains of ground connections).

Each shield shall be grounded to the metallic housing of the unit containing the circuits to be protected by one of the two following solutions (in order of preference).

Solution 1

- the shield shall be grounded to connector body, or to connector EMC shell (if present) by the harness contractor,
- a good continuity between the connector body and the equipment structure shall be ensured by the equipment.

Solution 2

- the shield shall be grounded directly to a point on the equipment structure by means of a wire less than 10 cm in length with a resistance of less than 20 mΩ.

In a general way, shields shall be grounded at both ends and at the level of each dismountability.

Where necessary, multiple shielding may be used. In such cases, the innermost shield shall generally be grounded at one end to the mechanical ground of the unit or to the return line, while the outermost shield shall be grounded to the structure at several points.

In any case, each shield shall be grounded separately.

The overall harness shield shall be grounded at both ends and also at intermediate points approximately every 25 cm along its length.

6.7.2.5. Labelling

All connectors shall be labelled at each end.

Labels shall remain both accessible and readable after cables have been fitted on the satellite structure.

6.7.2.6. Wiring Installation

Wiring installation requirements shall be derived from a selected MIL-STD or equivalent design standard.

6.7.2.7. Distribution of Twisted Wires

Where twisted wires are used, they shall, wherever mechanically possible, have a number of twists per metre greater than the values indicated below in Figure 6.7.1.

This last requirement is only applicable for cables which do not come as twisted pairs supplied by a manufacturer complying with ESA SCC requirements for twisted pairs.

	Number of twists per metre		
Wire Gauge	**2 wires**	**3 wires**	**4 wires**
8	15	12	9
12	21	15	12
16	30	21	15
20	48	36	24
22, 24, 26	54	43	30

Figure 6.7.1 : Twisted Wires Characteristics according to Wire Gauge

6.7.2.8. Insulation Resistance

As seen through the harness connectors and when not mounted on the equipment connectors, the insulation resistance shall be greater than 100 MΩ as measured under a 500 V voltage :

- between any contacts of the connector,
- between any contact of the connector and the connector shell.

However, for cables associated with thermistors, heaters, and connectors fitted with EMC filters, the voltage under which the insulation resistance is measured may be reduced to 50 V.

6.7.2.9. Power supply cable derating

Power supply cables shall be derated in accordance with the requirements stipulated in ESA PSS-01-301 (*Derating Requirements Applicable to EEE Components for ESA Space Systems*).

In case several cables are used for the same power supply, the derating shall be performed considering one failed cable (cut or disconnection).

6.8. GROUNDING REQUIREMENTS

The main goal for the rules concerning electrical potential reference is to ensure that EMC requirements are met starting from the satellite design phase. The proper application of these rules will allow the following :

- to prevent unwanted voltages from appearing on equipment housings,
- to prevent the accumulation of electrostatic charges on insulating surfaces,
- to reduce mutual interference effects between the different circuits via electromagnetic radiation and other coupling phenomena,
- to provide design teams with an equipotential reference surface, which is particularly important for RF links.

6.8.1. Secondary 0 V points

The return line of each secondary winding of a power converter will be considered as a reference point hereafter referred to as the power converter Secondary 0 V.

Initially, all converter Secondary 0 V are insulated from one another by design.

In the case where the circuits powered by a power converter or a set of power converters are housed in one single unit, the power converter(s) Secondary 0 V shall be interconnected to form the Unit Secondary Reference Point (USRP).

In the case where the circuits powered by a power converter or a set of power converters are housed in several units, forming a unit assembly, the corresponding power converter(s) Secondary 0 V may be either interconnected within the unit featuring the converter(s) so as to form the unit assembly Unit Secondary Reference Point (USRP) or remain insulated up to the inputs of the different user units, in which case they shall be interconnected within one of the units so as to form the unit assembly Unit Secondary Reference Point (USRP).

Interconnections between the Secondary 0 V of different PCBs shall be performed through a low impedance, so as to reduce the risk of common mode interference.

6.8.2. Equipment-Level Insulation of Primary Reference and Secondary 0 V Points

All insulation performances shall be met for any instrument operating and non-operating mode.

An insulation impedance of at least 1 MΩ in parallel with no more than 50 nF as measured under 28 V DC voltage shall be guaranteed at instrument level in the following cases :

- between Primary Reference Points,
- between Primary Reference Points and the instrument chassis,

- between the Primary Reference Points and the USRP,
- between the the Secondary 0 V Point and the instrument chassis, when a Secondary 0 V Point is grounded to the instrument chassis by an external bar,
- insulation between the Secondary 0 V Point and the equipment chassis, when a Secondary 0 V Point is grounded to the chassis of another equipment.

When used, the external bar shall be supplied by the instrument supplier.

If agreed by the Mars Express Prime Contractor, the chassis may be referenced to the satellite structure via a damped capacitive network.

6.8.3. TM/TC Signal Return Lines

Measurement Signals

Any instrument generating measurement signals shall provide at least one return line.

The use of a common return line for several measurement signals is however only authorised in the following cases :

- if the instrument supplier can demonstrate that the impedance of the return line is low enough to prevent any noise immunity degradation of the relevant circuits (cross-talk),
- if the different return currents are never present at the same time,
- if the concerned signal is a parallel digital link.

The reference point or points for these return lines shall be well defined so as to avoid any unwanted low impedance loop. In particular, no return line shall be bonded to the mechanical ground at both ends.

With the exception of RF signals, single ended receiver and transmitter shall not be used at both ends of a measurement signal line.

Single ended receivers are only allowed for thermistors and relay contacts telemetries. For all other measurement signal interfaces, differential or insulated receivers shall be used.

The double ended receivers (differential or insulated interface circuits) shall be designed so as to ensure a common mode insulation of at least 1 kΩ up to a frequency of at least 100 kHz.

This common mode insulation performance shall be verified between the receivers inputs (tied together) and the secondary power return.

Telecommands

In case of ORing relay commands from different units, the insulation between the return lines shall be respected (no less than 1 MΩ DC).

6.8.4. Structure Currents

In a general way, no electrical current shall be flown through the structure, in either a permanent or transient way.

This requirement is particularly important for return currents from secondary power supplies, including those for RF equipment.

This requirement is also important for return currents due to the execution of nominal switching commands.

6.9. ELECTROMAGNECTIC AND RADIOFREQUENCY COMPATIBILITY

6.9.1. Frequency Control

The frequency ranges given in Figure TBD may be used only by the units or instruments indicated. For all others instruments, intentional or unintentional emissions within these frequency bands is strictly forbidden.

6.9.2. Emission Requirements

Conducted Emissions

TBD

Radiated Emissions

If a Radiated Emission requirement is not met, investigations and additional tests shall be under taken to determine to what extent it is due to the cable harness and the test equipment.

When measured over the frequency range [150 kHz - 18 GHz] using method RE 02 of MIL-STD-462, the electric field measured at 1 meter from the unit under test shall not exceed the limits indicated in Figure TBD.

6.9.3. Susceptibility Requirements

Conducted Susceptibility

TBD

Radiated Susceptibility

If a Radiated Susceptibility requirement is not met, investigations and additional tests shall be under-taken to determine to what extent it is due to the cable harness and the test equipment.

The Unit Under Test shall not suffer of malfunction or present any performance degradation when exposed to an E field radiation with the levels specified in Figure TBD.

The test method to be used is the RS 03 method of MIL-STD-462 in the range [150 kHz - 18 GHz]

For RF receivers, the test is applicable only outside the receivers bandwidths.

The radiated E field shall be amplitude modulated by a sine wave at 1 kHz with a modulation depth of 50%.

Items of equipment not fully shielded (e.g. unit with unshielded openings other than those intended for connectors) shall be tested at all functional transmission frequencies.

6.9.4. ESD Requirements

6.9.4.1. Space Environment ESD

The Mars Express space environment may produce electrostatic discharge (ESD). Therefore, the Instruments shall not be affected by ESD, when in a configuration TBD.

If the Instrument is in OFF configuration, it shall be verified that the Instrument will not be stressed or damaged by the ESD.

If the Instrument is in ON configuration, it shall be verified that the Instrument will not be stressed or damaged by the ESD, nor will it good functioning be affected by the ESD.

The following ESD profiles shall be considered :

TBD

6.9.4.2. Human Body ESD

The human body ESD can occur during ground operation (e.g. AIT).

Therefore, the instruments shall meet the requirements as defined in the standard CEI-801-2.

In particular, they shall withstand without any permanent degradation of the specified parameters beyond tolerances given by the relevant specifications the following discharges :

- direct repetitive discharges of 8 kV under 150 pF ± 10% and 330 Ω ± 10% on the case of each connector,
- direct repetitive discharges of 2 kV under 150 pF ± 10% and 330 Ω ± 10% on the connector pins and or sockets connected to ESD sensitive circuits.

Test procedures and set-up are provided in the CEI-801-2 standard.

6.10. ELECTRICAL INTERFACE CONTROL DOCUMENT

6.10.1. Definition of the Electrical Interface Control Document

Interfaces will be formally controlled within the so-called Electrical Interface Control Documents.

The Electrical Interface Control Document will present all the electrical properties and any additional useful detail for each instrument.

The Electrical Interface Control Document data shall be provided by the instruments suppliers. They will be consolidated in the Satellite Electrical Interface Control Document, under the responsibility of the Mars Express Prime Contractor.

The instrument Electrical Interface Control Documents shall include at least the following data :

- the power consumed by the instrument, provided within the Power Data Sheets (PDS),
- the list of connectors for each instrument unit,
- the Interconnection Data Sheets, providing the pin function for all interface connectors between the instrument and the Mars Express platform,
- the Protections Data Sheet (PDS),
- the Interface Circuits Data Sheet (ICDS), to be provided for all interface circuit between the instrument and the Mars Express platform,
- the grounding diagram,
- the switching diagram of primary and secondary power distribution for each equipment.

6.10.2. Power Data Sheet (PDS)

6.10.2.1. Data Sheet Overall Description

The instrument power budgets shall be provided in the form of filled in Power Data Sheets, as defined in Figure 6.10.1.

The Power Data Sheets shall be filled in according to the guidelines given in Sections 6.10.2.2 to 6.10.2.6.

Blank data sheets are available from Matra Marconi Space as EXCEL™ 7.0 files for Windows™ 95.

MARS EXPRESS	POWER DATA SHEET	Ref. : MEX-IC- Issue : 0 Date : 26/02/98 Page :

INSTRUMENT :

		TYPICAL BEGINNING OF LIFE							WORST CASE END OF LIFE						
Equipment	Power	Margin	Mean	Min.	Max.	Peak	Peak	Peak Duty	Mean	Min.	Max.	Peak	Peak	Peak Duty	Failure
	POWER LINE # 1														
	POWER LINE # 2														
	POWER LINE # 3														
	POWER LINE # 1														
	POWER LINE # 2														
	POWER LINE # 3														
	POWER LINE # 1														
	POWER LINE # 2														
	POWER LINE # 3														
	POWER LINE # 1														
	POWER LINE # 2														
	POWER LINE # 3														
	POWER LINE # 1														
	POWER LINE # 2														
	POWER LINE # 3														
	POWER LINE # 1														
	POWER LINE # 2														
	POWER LINE # 3														

Figure 6.10.1 : Equipment Power Data Sheet

6.10.2.2. Header

Ref., Issue, Date

The reference, issue and date of the Power Budget document in which the Power Data Sheet is inserted.

Page

The number of page out of the total number of pages for the instrument.

Instrument

The instrument which the data sheet is dedicated to.

6.10.2.3. Modes and Power Lines Definition

Instrument Mode

The instrument mode for which the power figures are provided.

Up to 6 functional modes can be provided in each Power Data Sheet.

In the case where the equipment has more than 6 functional modes, several Power Data Sheets can be used.

In the case where the power consumptions are identical for several functional modes, one single line may be used for all the modes.

Power Line

In case several switchable protected power lines are dedicated to an instrument, the power line for which the power figures are provided.

6.10.2.4. Typical Beginning Of Life Power Consumptions

The Typical Beginning Of Life power is the power which is expected to be measured during instrument acceptance tests. This raw figure corresponds to the basic one, to be taken into account for the satellite power budget computations. Therefore, it shall not include any design margins, since instrument contingencies will be consolidated at system level.

Margin Code

The code corresponding to the instrument contingency to be taken into account, related to design maturity and/or instrument milestones.

The definition of margin codes is given in Figure 6.10.2.

The margin code shall correspond to the first event to occur between the expected design maturity and the corresponding instrument milestone achievement.

Code	Instrument Design Maturity	Contingency
CE	Conceptual estimate — up to IRR	15%
PD	Preliminary Design — up to IPDR	10%
ED	Elaborate Design — up to ICDR	8%
MM	EM/QM Measurement — up to QRR	5%
AM	PFM Actual Measurement — up to FAR	3%

Figure 6.10.2 : Margin Codes Definition

Mean Power

The steady state power consumed by the instrument when the power bus is set at its 28 V mean voltage, and with a 25 °C temperature.

Steady state power implies the power consumed permanently by the instrument during the considered functional mode, excluding any power peak or transient.

Min. Power

The minimum steady state power consumed by the instrument as a function of power bus input voltage and equipment temperature.

Max. Power

The maximum steady state power consumed by the instrument as a function of power bus input voltage and equipment temperature.

Peak Power

The power consumed during a peak, i.e. corresponding to an event of finite duration during the considered functional mode.

The power peak can be periodic (motor actuation, periodic use of equipment function,…) or not (power ON inrush, relay switching,…).

The peak power shall be given at 28 V mean power bus voltage and with a 25 °C instrument temperature.

In case several peaks may be encountered in the same functional mode, then they shall all be listed with their characteristics. In this case, lines may be deleted from the data sheet, so that only one page is used per data sheet.

Peak duration

The duration of the peak.

Peak Duty Cycle

For periodic power peaks only, the peak repetition duty cycle.

The peak repetition duty cycle is defined as d = T1T2, according to Figure 6.10.3.

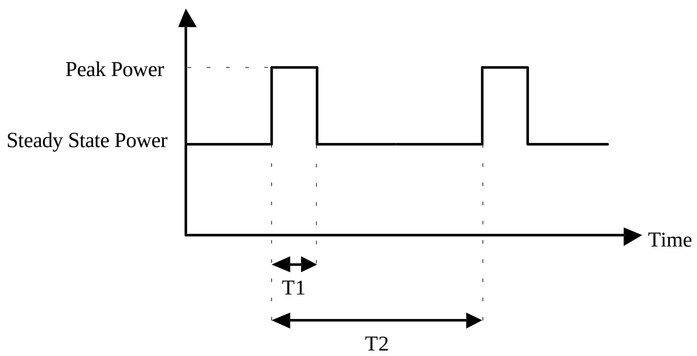


Figure 6.10.3 : Peak Repetition Duty Cycle

6.10.2.5. Worst Case End Of Life Power Consumptions

The Worst Case End Of Life power is the power which is computed when taking all the possible unfavourable factors into account : components initial dispersion, ageing, radiation effects,… By definition, the Worst Case End Of Life power shall be higher than the Typical Beginning Of Life One. Still, as it will be taken into account among others for harness, protections and monitoring thresholds sizing, it shall be lower than the maximum allocated power.

The Worst Case End Of Life power shall not include any design margins, since instrument contingencies will be consolidated at system level.

Definitions for Mean, Min. and Max. Powers, Peak Power, Duration and Duty Cycle are identical to the ones given in Section 6.10.2.4. — Typical Beginning Of Life Power Consumptions.

6.10.2.6. Failure Power Consumption

This is the absolute maximum permanent power which is foreseen to be consumed by the instrument in case of dissipative failure. This power shall be provided as a Worst Case End Of Life power, according to the definition given in Section 6.10.2.5 — Worst Case End Of Life Power Consumptions. It may be higher than the maximum allocated power.

6.10.3. List of Connectors

A list of connectors shall be provided for each instrument unit, as defined in Figure 6.10.4.

6.10.4. Interconnection Data Sheet

6.10.4.1. Data Sheet overall description

An Interconnection Data Sheet as defined in Figure 6.10.5 shall be provided for all interface circuit between the instrument and the Mars Express platform.

The Interconnection Data Sheets shall be filled in according to the guidelines in Sections 6.10.4.2 and 6.10.4.3.

Blank data sheets are available from Matra Marconi Space as EXCEL™ 7.0 files for Windows™ 95.

6.10.4.2. Header

Ref., Issue, Date

The reference, issue and date of the Interface Control Document in which the Interconnection Data Sheet is inserted.

Page

The number of page out of the total number of pages for each connector.

Instrument

The instrument which the data sheet is dedicated to.

Connector #

Number of the connector as allocated by the contractor according to TBD.

Connector Type

Name of connector manufacturer.

Connector Ref.

Complete reference of the connector.

MARS EXPRESS	LIST OF CONNECTORS	Ref : Issue : Date :

INSTRUMENT :

CONNECTOR #	CONNECTOR REFERENCE	FUNCTION	COMMENT

Figure 6.10.4 : List of Connectors Data Sheet

MARS EXPRESS	INTERCONNECTION DATA SHEET	Ref. : MEX-IC- Issue : 0 Date : 26/02/98 Page :

INSTRUMENT :

PIN	FUNCTION	SIGNAL TYPE	CURRENT		RESISTANCE		FREQ	VOLTAGE RANGE				CAPACITANCE			I/F TYPE	MODI
								SOURCE		LOAD						
			SOURCE MAX (A)	LOAD MAX (A)	SOURCE (Ω)	LOAD (Ω)	(Hz)	MIN (V)	MAX (V)	MIN (V)	MAX (V)	SOURCE (pF)	LOAD (pF)	TOTAL I/F (pF)		

Figure 6.10.5 : Interconnection Data Sheet

6.10.4.3. Data Sheet Columns Definition

Pin

The numbering of the pin of the connector (maximum 3 digits). All the pins of the connector shall be listed even if unused.

Function

Identification of the signal which goes through the pin (maximum 30 characters). Unused pins shall bear the mention SPARE in the Function column.

Signal type

Figure between 1 and 4 identifying the type of signal, according to the 4 categories as defined in Section 6.7.2.

Current

Maximum specified intensity which goes through the pin :

If the unit is a SOURCE, the maximum intensity is written in column SOURCE.

If the unit is a LOAD, the maximum intensity is written in column LOAD.

A maximum of 6 digits (including possible exponent) is allocated for the maximum current in each column.

Example : 525E-3 (for 525 mA)

Resistance

Resistance of the I/F circuit

If the unit is a source, the actual source resistor value is written in column SOURCE and the specified minimum load value in column LOAD.

If the unit is a load, the actual input resistance value is written in column LOAD and the specified source value in column SOURCE.

A maximum of 7 digits (including possible exponent) is allocated for the resistance in each column.

Example : 10E+3 (for 10 kΩ)

Voltage Range

Voltage which is seen by the pin.

If the unit is a source, the values are written in columns SOURCE MIN and SOURCE MAX.

If the unit is a load, the values are written in columns LOAD MIN and LOAD MAX.

A maximum of 6 digits (including possible exponent) is allocated for the voltage range in each column.

Example : 125E-3 (for 125 mV)

Capacitance

Capacitance of the interface circuit.

If the unit is a source, the aperture of the interface circuit (actual value) is written in column SOURCE, and the capacitance of the cable plus receiver, seen by the unit, is written in column TOTAL I/F.

If the unit is a load, the capacitance of the interface circuit (actual value) is written in column LOAD. Nothing is written in the 2 other columns (SOURCE and TOTAL I/F).

A maximum of 6 digits (including possible exponent) is allocated for the capacitance in each column.

Example : 600 (for 600 pF)

Bandwidth

Required bandwidth for the signal which goes through the pin. This column is only applicable of signals of type 2 or 4 (otherwise "NA" is written in this column).

A maximum of 6 digits (including possible exponent) is allocated for the bandwidth.

Example : 512E+3 (for 512 kHz)

I/F Type

This column is used to refer to the interface drawing of the interface circuit to be gathered in the Interface Circuit Data Sheets.

Allocation : for a source : S001, S002,....

for a load : E001, E002,....

Modi

To be marked with a star (*) if any of the signal parameters (including function or pin allocation) has been modified with respect to the last issue.

6.10.5. Protection Data Sheet (PDS)

The Protection Data Sheet as provided in Figure 6.10.6 shall be filled in for each protection/function.

In addition, a simplified interface circuit diagram shall be provided for each protection/function.

6.10.6. Interface Circuit Data Sheet (ICDS)

An Interface Circuit Data Sheet shall be provided for all interface circuit between the instrument and the Mars Express platform.

An Interface Circuit Data Sheet shall also be provided for power bus interfaces of each instrument unit, featuring mainly the input filter.

The following data shall at least be found in the Interface Circuit Data Sheets :

- circuit reference (E*** or S***), as identified in the relevant Interconnection Data Sheet,
- function name,
- connector and pin numbers,
- circuit diagram of the first interface, featuring the component values and references and the reference voltages.

6.10.7. Grounding and Isolation Diagram

The grounding and isolation diagram will be used to verify the correct implementation of the Distributed Single Point Grounding (DSPG).

The following data shall at least be found in the grounding and isolation diagram :

- primary power grounding,
- secondary power grounding,
- signal power grounding,
- interface circuit grounding,
- electronic box grounding,
- isolation between different grounds or references,
- principal interface circuit diagram,
- cable shields grounding,
- cable types used between units (TP, TSP, coaxial lines, triaxial lines,…),
- redundancy and cross-strappings.

The symbols used in the Grounding Diagram shall be as per Figure 6.10.7.

6.10.8. Power Distribution Switching Diagram

The Power Distribution Switching Diagram shall show all elements performing switching functions and all associated commands.

The mode symbols used in the Power Distribution Switching Diagram shall be as per Figure 6.10.8.

TBD

Figure 6.10.6 : Protection Data Sheet

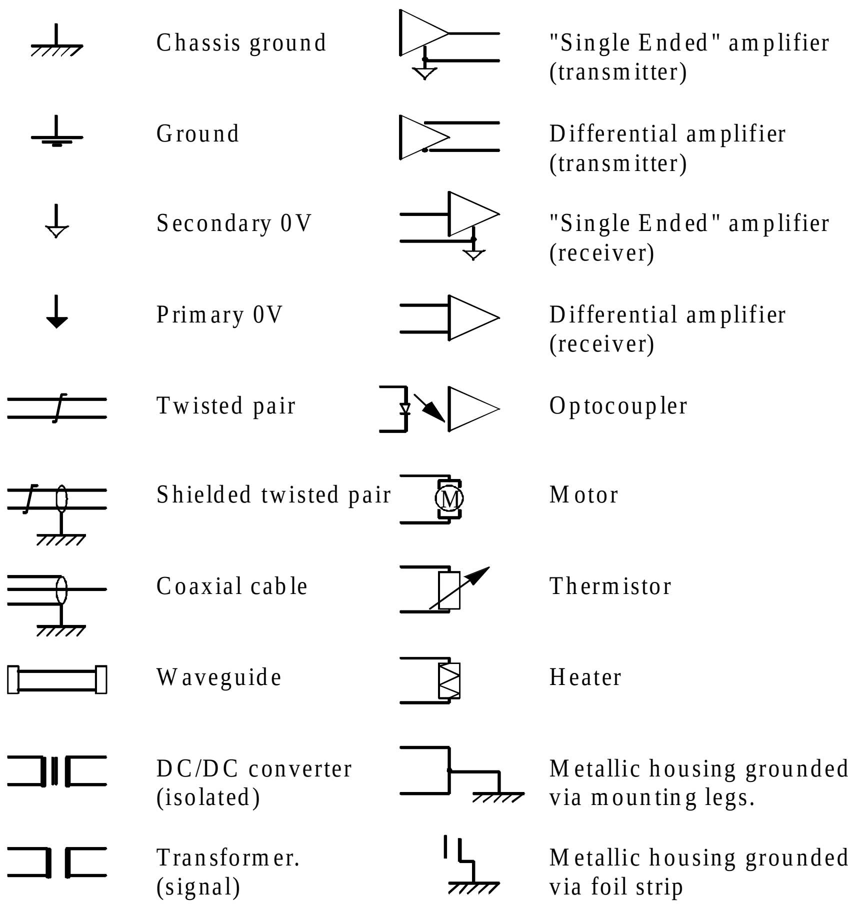
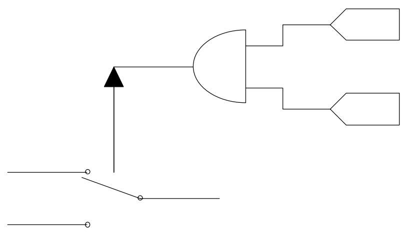


Figure 6.10.7 : Switching Diagram

Relays switched by simultaneous use of n commands (AND)



Relays switched by any command among n different ones (OR)

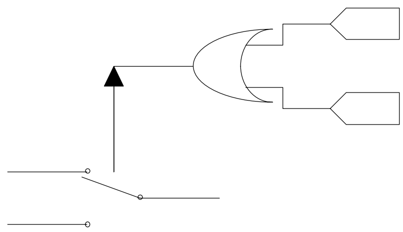


Figure 6.10.8 : Power Distribution Switching Diagram

6.11. SINGLE EVENT UPSET AND LATCH-UP

Single Event Upset (SEU) can occur in digital electronic components due to the passage of a single high energy ion through the device. The ion generally originates either from a heavy Galactic Cosmic Ray (GCR) nucleus or from high energy particles produced during a solar flare.

These upsets can result in transient software errors : logic, memory (single or multiple bits), loss of synchronisation, and permanent damages : latch-up, charge deposition in oxide traps, displacement damage in Silicon, insulator breakdown.

Demonstration shall be made that the normal functioning of the instrument will not be disrupted by SEU and/or latch-up phenomenon. Demonstration shall be made that the normal functioning of the satellite and the other payload will not be disrupted by SEU and/or latch-up phenomenon.

The methodology to ensure design immunity against cosmic particles shall be clearly identified : combination of selection of latch-up free components, use of fault-tolerant design rules, testing, analysis, computer simulations.

If it appears that the sensitivity of a device is above the SEU threshold, then adequate precautions shall be taken both at experiment level by e.g. watchdog monitoring and at functional level by use of error detection and correction.

On the detection of a latch-up state, the device shall be switched off, removing the latch-up condition, and then switched on again. The device is then re-usable ; however, systems using RAM will be in an arbitrary state and, thus, must be reset. Detection of latch-up can be performed either by sensing excessive current taken by the device or by detecting induced system malfunctions using watchdogs.

The applicable energetic ion LET spectrum for the design of the instrument is depicted in Figure 6.11-1. This spectrum is applicable for all instruments, whatever its location within the S/C. It shall be assumed at instrument level, that no protection is brought by the satellite.

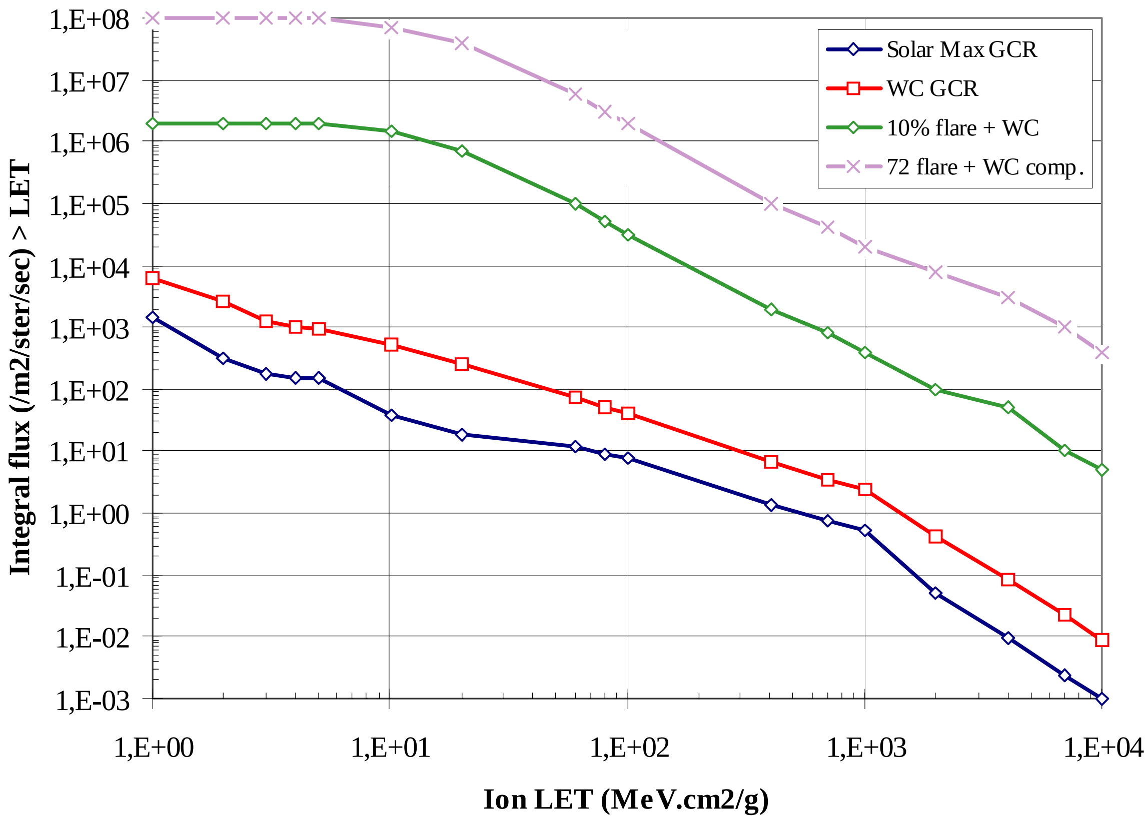


Figure 6.11-1 : Applicable Energetic Ion LET Spectrum for the Mars Express Electronics and Payloads

6.12. LIST OF TBDs AND TBCs

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7. FLIGHT OPERATIONS

7.1. Description of Instruments Flight Operations

This paragraph identifies the functional requirements for the Mars Express instruments which are considered as essential to ensure the operability of the Mars Express spacecraft.

The instrument will be operated from the ground (see Section 2.9 for Ground Segment interfaces, including science data extraction and distribution) through the satellite, in the different flight operating modes described in Section 2.2.

7.1.1. General concept

Operational aspects must be taken into account from the conceptual design phase of the instrument hardware and software. The instrument operations concept shall result in simple and safe flight operation procedures. In specific operating conditions only (design deficiency late discovered), additional ground segment tasks may be defined (through a TBD procedure) to implement well identified mission recovery actions.

The basis for instrument operations is defined in the instrument Flight Operation Manual (FOM). Its quality is the key point for efficient and safe operations in both routine and contingency cases. This document is required for early instrument reviews in order to support the iterative process between the instrument and the satellite in order to optimise the whole operational concept. The instrument FOM is finally integrated in the satellite FOM.

In critical mission phases (see description of the mission time-line), there shall be no requirement for the ESOC Control Centre to send TC in reaction to anomalies detected from the TM (response time consistent wrt S/C to Earth distance). Situations in which the Control Centre is expected to react within a short time shall be reduced to a minimum and be well defined. Situations in which the Control Centre is expected to react within a short time must be easily and unambiguously recognisable in the available TM, without the need for complex processing.

The instrument shall be able to survive autonomously (i.e. without ground intervention) during solar conjunction and solar opposition. These predicted periods can last up to 30 days.

The instrument shall be able to survive a un-predicted ground outage period of at least 3 days (TBC).

The spacecraft will enter the Survival Mode (except during the Mars Orbit Insertion phase) if any hazard exists which affects the mission objectives, mission life or spacecraft safety. Survival Mode will initiate a switch-off sequence for electrical loads not essential for spacecraft survival.

7.1.2. Instrument commandability

7.1.2.1. General

The instrument shall be functionally self-contained with minimal command and control interfaces from the satellite. However, it shall be fully controllable from the ground.

The instrument shall be designed so that no single malfunction of instrument software or single instance of erroneous commanding, can permanently damage or degrade the satellite and other instruments hardware and their use.

The execution of any single command shall not lead to any permanent instrument damage. Exception to this requirement shall be well identified on a case by case analysis and an exhaustive list of critical TC shall be provided.

The sequencing of payload critical operations shall be clearly identified and organised as safety routines (e.g. pyro operations, payload mechanism motion, etc.).

The instrument shall be able to operate safely during the whole Mars Express mission duration. Nevertheless, no cruise science is foreseen at satellite level. The instrument will be OFF during the Mars Orbit Insertion phase (most critical phase of the mission). The instrument will also be OFF during the optional Lander Ejection and Aerobraking phases.

The instrument shall provide the functionality of individually inhibit / enable from the ground any automated functions within the instruments.

It shall be possible to recover from any instrument configuration, by corrective actions using normal command procedures. It is recommended to use patchable software.

Instrument power loss may occur at any time and in any instrument mode or transition between instrument modes without leading to any instrument damage. Sound and safe recovery from this undefined instrument off configuration shall be possible when the instrument power is available again.

7.1.2.2. Command categories

Three categories of TC are identified depending on their criticality :

- Category 1 commands are potentially hazardous commands which, if executed at the wrong time, could cause loss of or significant degradation to the mission. **These commands shall not be used by instruments.**

- Category 2 commands are vital commands, which although not hazardous are essential to the success of the mission. Redundant TC shall be provided for all Category 2 commands, by means of an alternative route.

- Category 3 commands are configuration-dependant commands. These may only be sent if a particular configuration of a subsystem or payload prevails, or alternatively, must not be sent under certain defined conditions. It shall be possible to unambiguously determine from the "normally available" TM, the conditions under which Category 3 TC may be sent during in orbit operations.

Erroneous TC shall not be executed. In that case, TM shall be provided to indicate the reason why any command is not executed (incorrect form or structure, invalid subsystem mode etc.).

Critical TC and critical operational procedures shall be clearly identified and described in the Flight Operations Manual (FOM).

7.1.2.3. TC verification

TM shall be provided to allow complete and unambiguous verification of the reception and execution of all commands either sent directly from the ground or stored on-board and released at a later time or generated autonomously on-board.

The TM for command reception verification shall be generated by the « receiving » application, and shall indicate whether the TC was properly received and contained valid data.

After successful command reception, TM for command execution verification shall be generated indicating the verification status of each TC. This TM shall indicate the progress in the various stages of command execution as applicable.

All memory load (register load) commands shall be directly confirmed by echo parameters which reflect exactly the value contained in the register.

7.1.3. Instrument observability

7.1.3.1. General requirements

The instrument shall provide the ground with all the data required for the correct execution and monitoring of the nominal and contingency operations. The data shall be such that complete and unambiguous evaluation of instrument status and performance is possible without the need for reference to the TC history.

Instrument on-board events and parameters shall be observable by the ground through its TM. This reporting shall be such that continuity is ensured (no event lost) and accurate datation is provided. In particular, instrument modes shall be unambiguously observable via the TM data. The beginning and ending of modes shall be observable and dated.

The TM data shall report the following information as a minimum :

- successful / unsuccessul execution of commands (including pyro commands),
- minima and maxima of analogue/ digital housekeeping points,
- out-of-range parameter.

It shall be possible to determine at any time the current status of any switchable on-board unit. All changes in instrument and unit status shall be notified in the TM, without requiring the ground to request the status information.

All mission critical functions shall be monitored by at least two independent parameters, to be determined on a case by case analysis.

In case of detection by the ground segment system of an out-of-range monitored parameter associated with one or more instruments, the ground segment will put these instruments in a safe configuration, allowing further actions from the ground.

The instruments shall give the satellite the opportunity to detect and isolate the following failures at instrument level :

- electrical overload on the power supply lines, due to hard or partial short circuits, or otherwise non nominal power consumption in the instrument, including the pyro lines,
- corruption of protocol, formats, bit pattern, on data interfaces,
- non execution of the satellite emitted commands,
- signal outage on instrument generated data lines,
- hard or partial short circuit on data/command lines.

When redundancy is foreseen, the TM shall enable the ground to identify the failed item, and to identify and utilise the healthy remainders for reconstruction of a back-up mode. Any reconfiguration shall end with a fully defined configuration status of all involved units and software.

Suitable sampling sequences and frequencies shall be provided for all related parameters which require direct correlation or combination needed for ground data processing.

It is recommended to provide TM information from direct measurements rather than from secondary effects. It is recommended to provide current values or current status rather than changes since the last readout.

7.1.3.2. TM packets

All inputs used by on-board processes shall be accessible to the ground via housekeeping or diagnostic TM packets. The following auxiliary data shall be available within the TM :

- Data to establish the validity of the information within a packet.
- Data which may be required to process other data in the packet in order to derive the operational information.

Sufficient information shall be provided in the TM to permit the ground to unambiguously re-construct the sampling time of any data within the packet, nominally by providing time information in each packet.

It shall be possible to determine the absolute (on-board) sampling time of all housekeeping parameters within TM packets to an accuracy of TBD msec.

It shall be possible to determine the relative sampling time of any two housekeeping parameters to an accuracy of TBD msec. This shall be possible even if the parameters are telemetered in different packets.

If the same housekeeping TM parameter appears several times in the same TM packet, then the parameter shall be sampled regularly in time (at an interval which is a sub-multiple of the packet interval).

A TM parameter shall always have the same structure and interpretation even if it appears in several different TM packets.

It shall be possible to derive the location of a parameter within a TM packet.

In the case of the loss of a TM packet during transmission, it shall be possible to identify the lost packet unambiguously.

It shall be possible to invoke the automatic generation of housekeeping TM packets by an on-board event (e.g. after the occurrence of an on-board anomaly).

7.1.4. Instrument autonomy

The instrument shall not perform autonomous reconfiguration.

For the Science phase of the mission, ground station support to the satellite is not available for about 66 % of the time (with a single station). Any immediate interrogation of the satellite, should an on-board anomaly occurs, is therefore not guaranteed, and it is expected that the control centre may detect those on-board anomalies several hours after their occurrence.

The design of the instrument shall therefore provide sufficient autonomous control functions in combination with fault tolerance, in order to ensure, for a period of about 16 hours without ground contact, that instrument internal failure will not endanger other instruments and/or satellite subsystems.

The instrument design shall allow to supersede the on board autonomy at any time from ground.

7.2. Instrument Flight Operations Manual

As input to the Mars Express satellite Flight Operation Manual (FOM) and to the Flight Operations Plan, each instrument supplier shall deliver the instrument Flight Operation Manual.

This document shall contain all the information needed for correctly operating the instrument both in nominal and emergency conditions, in order to prevent the operational staff from making any utilisation error. It shall allow the user to easily find his way towards a particular information, such as an equipment configuration set-up method, the utilisation constraints, the definition of a given unit electrical interfaces or the recommended sequencing of a given operation.

The instrument FOM shall include the following information as a minimum :

- the precise functional description of the instrument down to equipment level and software elements, its scientific objectives, and the data collected during the mission,
- the description of the instrument configurations and modes during the different mission phases, including the list of operational constraints,
- the software design description,
- the instrument redundancy concept,
- the instrument command and control including the list of critical commands,
- the instrument functional block diagram and a switching diagram showing the location of the TM outputs and TC inputs,
- the nominal operational scenario,
- the requested ground segment support, including at least the TM validity conditions, range, and calibration curves,
- all the procedures required to operate the instrument both in nominal and emergency conditions,
- the instrument data base, containing a complete definition of all instrument housekeeping TM and TC.

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8. VERIFICATION REQUIREMENTS

8.1. General

The Principal Investigator shall, in a systematic manner, verify the instrument design and build status against each requirement specified in the PID/A and then reflected in the PID/B..

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).

This section establishes the verification requirements for the qualification and flight certification of the various instruments and units. It gives specific test levels and durations, and it describes qualification and acceptance tests, together with analytical methods for implementing the verification requirements. As a general requirement, all the typical situations (and not only the worst ones) as resulting from the mission analysis will have to be tested to first insure the proper functioning under nominal conditions.

8.2. Definitions

The following definitions are to be used :

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.

Acceptance Verification : 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.

Re-certification : Tests intended to demonstrate that initially flight accepted hardware which requires refurbishment is acceptable for flight. The re-certification is a limited repetition of the acceptance verification and serves also as a quality control to detect deficiencies and provides the basis for delivery of items under terms of contract or agreement.

Incoming inspection : Inspection and functional tests to declare that the item is ready for integration onto the spacecraft.

Functional Test : The operation of a unit in accordance with defined operational procedures to determine that performance is within the specified requirements.

Functional 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.

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 Vacuum Test : A test to demonstrate the validity of the design in meeting functional goals: it also demonstrates the capability of the test item to operate satisfactorily in vacuum at temperatures based on those expected for the mission. The test can also uncover latent defects in design, parts and workmanship.

Static Load Test : A test conducted to verify that the design can withstand the maximum combination (longitudinal and lateral) of static loads which occur during the various segments of the flight profile. It consists of steady state accelerations and quasi-static loads which are structure borne loads generated by the launch vehicle in the low frequency range

Sinusoidal Vibration Test : A test conducted to verify that the design can withstand the mechanical environment of the low frequency sinusoidal and transient vibrations.

Acoustics/Random Vibration Test : A test conducted to verify the capability of the design to survive 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.

Shock Test : A test conducted to verify the design under the environment induced by shocks produced by the launcher during events such as stage separation and by the spacecraft during events such as pyro firings.

Electromagnetic Compatibility (EMC) : The condition that prevails 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.

8.3. Payloads Design, Development and Verification

8.3.1. Design, Development and Verification Plan

8.3.1.1. General

A design development and verification plan (DDVP) shall be prepared that defines the tests and analyses that collectively demonstrate that hardware and software complies with the requirements called up in this volume.

The DDVP shall describe the overall approach to accomplish the instrument qualification and acceptance programme. It shall also show the characterisation and calibration activities which will be performed on the instrument.

When appropriate, the interaction of the test and analysis activity shall be described.

For each analysis activity, the plan shall include objectives and a description of the method used.

The DDVP shall be supported by the verification test procedures and completed by test reports.

8.3.1.2. Test matrix

As part of the DDVP, the PI shall provide a test matrix which is in essence a synthesis of the verification programme. The test matrix summarises all the tests that will be performed on each experiment unit.

The purpose of the test matrix is to provide a ready reference to the contents of the test programme in order to prevent the deletion of a portion thereof without an alternate means of accomplishing the objectives. It has the additional purpose of ensuring that all flight hardware has been exposed to various environment tests which are sufficient to demonstrate acceptable workmanship.

The test matrix shall provide traceability of the qualification heritage of hardware. All models shall be included in the test matrix which shall be prepared in conjunction with the initial DDVP and shall be updated as changes occur.

The test matrix shall also include the characterisation and calibration activities of the instrument.

8.3.2. PID Verification Control Document

The PID Verification Control Document (VCD) is the basic and exhaustive document recapitulating the means (analysis, test, other...) which are proposed to verify the PID items.

This document is usually edited under tabular form, and is tracing the overall verification documentation (reference of the various analysis definitions, study notes, test requirements, procedures and reports) used by the instrument team.

For the final qualification and acceptance reviews, it comprises as well - for each item - all the quantified results of the various analyses or tests, just to be compared to specified values.

8.3.3. Test procedures

For each verification test activity conducted at the instrument level, test procedures shall be prepared that describe the test objectives and success criteria, the configuration of the test article, and how each test contained in the verification plan and test matrix will be implemented.

The procedures shall contain all details which are necessary to make meaningful interpretations of the results, such as test parameters, instrumentation monitoring, data collection, post-test verifications (alignment) ....

It shall also address safety and contamination control provisions.

8.3.4. Test reports

After each experiment verification test activity has been completed, a test report shall be submitted. The report shall describe the test results and the correlation with mathematical models. Comparison of the measured parameters with requirements or allocations and the degree to which the test objectives have been met shall be shown.

In addition all as run test procedures, test and analysis data shall be retained for review. Every failure shall be reported through an NCR (non-conformance report) in accordance with QA provisions.

8.3.5. Model philosophy

The development approach and model philosophy for the Orbiter payload instruments (and otpionally for landers) shall satisfy :

(i) their specific (instrument level) development and qualification needs,

(ii) and the overall validation and development logic at spacecraft level.

The validation and model philosophy needs at system level require that the instrument model philosophy includes a minimum of deliverable models as summarised hereafter :

- structural dummies for mechanical qualification
- an Engineering Model for electrical and functional validation
- a Flight Model for integration on the spacecraft flight specimen

The model philosophy of the Orbiter payload instruments shall therefore be compatible with the requirements expressed above. Flexibility will be left to the PI's to define the appropriate development approach and associated model philosophy insofar as the above requirements are met, and as the Orbiter instruments are proven to be fully qualified before delivery to the Spacecraft Contractor.

The required deliverable models of instruments to fulfil system needs shall not prejudge from other models being developed for Orbiter instruments qualification purposes for example.

It is understood that the model philosophy retained by the Principal Investigators will depend on the design maturity of the relevant instruments, as existing (e.g. flight spare from other programme) or slightly modified instruments may be available at flight standard very early.

However, the model philosophy and associated delivery / availability dates shall be defined in collaboration with the Spacecraft Contractor with the objective to minimise the risks at system / mission level. They will be finally incorporated in the payload Design Development and Verification Plan to be approved by the Spacecraft Contractor.

In addition, mathematical models of the Orbiter payload instruments shall be provided for integration in the spacecraft mathematical models themselves, as used for engineering analyses (and design verifications) to be correlated with spacecraft test results.

8.3.6. Qualification and acceptance

The qualification shall demonstrate that the item will function within the performance specifications under simulated conditions more severe (and with well-identified margins) than those expected from ground handling, launch and orbital operations. This demonstration shall be achieved by testing, analysis, similarity assessment.

The acceptance shall demonstrate that the item is acceptable for flight .

The minimum required tests to achieve these objectives are shown in the Qualification & Acceptance summary test matrix :

Test requirement	Qualification	Acceptance	Paragraph of requirement definition
Visual inspection	X	X	8.5.3.
Dimensions	X	X	8.4.1.3.3.
Physical properties	X	X	8.4.1.3.3.
Alignment	X	X	TBD
Quasi-static load	X		8.4.1.3.4.
Low level sine	X	X	8.4.1.3.5.
Sine vibration	X	X	8.4.1.3.5.
Random vibration	X	X	8.4.1.3.5.
Acoustic vibration	X	X	8.4.1.3.6.
Shock	X		8.4.1.3.7.
Thermal vacuum	X	X	8.4.1.3.7.
Thermal balance	X		8.4.2.3.2.
Proof pressure	X		8.4.1.5.1.
Leakage	X	X	8.4.1.5.2.
EMC conducted	X		8.4.3.3.4.
EMC radiated	X		8.4.3.3.4.
Functional & performance tests (*)	X	X	8.4.5.

(*) The instrument shall have been independently characterised and calibrated before delivery

Table 8.3/1 : Qualification & acceptance summary test matrix

8.3.7. Re-certification

The re-certification shall demonstrate that modified or repaired units are acceptable for flight. It is applicable for any unit which has been disassembled from the spacecraft after the system environmental testing, is refurbished in any way and then reintegrated to the flight spacecraft.

The re-certification is a limited acceptance certification and serves also as a quality control.

8.4. Detailed Verification Requirements

8.4.1. Mechanical Verification Requirements

8.4.1.1. Requirements applicable to all units

A series of mechanical analyses and tests shall be conducted to :

- qualify the design of the hardware,
- to verify the stiffness requirements,
- to demonstrate specified factors of safety and compliance with associated launch authority (either Soyouz or Delta 2) safety requirements,
- to prove the nterface compatibility (all models),
- to check workmanship (in particular for the flight model)

The tests shall be nominally conducted on a unit configuration representative of the anticipated flight situation corresponding to these environment (as an example : vibration test with deployable cover ON, thermal vacuum test with cover OFF)

8.4.1.2. Mechanical mathematical models and analyses

Structural analysis of experiment units shall be performed in order to show that the unit design is able to meet the stiffness and mechanical environment requirements as described in sections 2.4.1 & 3.2.1. The structural analysis shall include stress and dynamic (through model) analysis.

8.4.1.2.1. Detailed stress analysis

The stress analysis shall demonstrate that the experiment unit can withstand the limit loads factored by the appropriate safety factors. The analysis results shall be compiled in a technical note with at least :

- a description of the configuration analysed 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,
- a 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.

8.4.1.2.2. Dynamic analysis

The dynamic analysis shall be performed to demonstrate that the experiment unit meets the stiffness requirements.

Furthermore, it shall be detailed enough to predict their stiffness below 100 Hz, when applicable, and the dynamic loads :

- sizing the structural elements, the interface loads and any notching necessary in the dynamic tests input spectrum,
- provided by the PI in the frequency range [0, 10 Hz] to study at system level the stability and the performances of the control laws of the Spacecraft attitude control system.

When mechanisms are part of the unit, several models shall be developed to account for the various configurations (stowed, fully deployed ...).

The analysis results shall be compiled in a technical note with at least :

- a description of the configuration analysed with reference to interface control drawings,
- a description of the mathematical finite element model and/or the definition of the assumptions and 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 eigenfrequencies with relevant mode type and associated modal effective masses,
- plots and listings of eigenvectors.
- when necessary :
  - frequency response analysis
  - acoustic response analysis
  - shock response analysis.

8.4.1.2.3. Structure finite element mathematical model detailed requirements

In order to perform system analysis, the PI shall provide the Spacecraft Contractor with finite element models (FEM) of experiment units, representative of the best estimate. The models shall be updated according to dynamic test results. The delivered models shall be either the original ones or simplified representative FEM, depending on the FEM size specification and criticality of the items.

For system stability and thermo-elastic strength analyses, a thermo-elastic model representative of the experiment major items is required. It shall include all structural components which may have a potential effect on the thermo-elastic stability. A single model which meets both the dynamic and thermo-elastic requirements may be delivered.

The requirements for the preparation, and the delivery of the finite element mathematical models of the instrument units in order to incorporate them into the spacecraft mathematical model are defined here.

Non conformance with any requirement must be agreed by the Spacecraft Contractor and ESA. Two types of mechanical models must be provided: a dynamic physical finite element or matrix model and a thermo-elastic model (physical model with thermo-elastic data). A single mathematical model for both aspects is acceptable.

a) Software

The model shall be prepared with MSC/NASTRAN version 68 or at least be compatible with this code.

b) Units

S.I. units are to be used :

- Newtons (N) for force
- Kilogrammes (Kg) for mass
- Meters (m) for length
- Seconds (s) for time
- Degrees Celsius (°C) for temperature

c) Coordinate system

Each assembly may has its own ‘primary’ coordinate system. All coordinate systems shall use the PLM coordinate system as the reference frame. Each part can be moved by changing only the primary coordinate system. Definition and displacement system for each grid (and also elements which need one) must be defined thanks to a coordinate system dependent on the primary coordinate frame of the assembly. In one assembly one can define, as many as necessary, different coordinate systems but they must depend on the primary coordinate system.

d) Model identity/comment cards

The model name, issue and date must be included on a comment card at the beginning of the structural data. Sufficient comment cards must be included to ensure complete understanding of the model. Interface grid points with their associated coordinate systems must be shown using comment cards. The boundary conditions for these interface points shall be clearly defined.

e) Model size and identification numbering

Bulk data cards :

Each Nastran bulk card will have a unique identifier number in the ranges for each model. Continuation card identifiers should contain letters from the card name plus the card identification number (eg +PB12 for a PBAR card). It is requested that no duplicate number among Nastran Bulk data cards exits in each model.

Excluded bulk data

All data cards used to define default values for a set of cards are excluded and also :

PARAM Bailout

PARAM K6ROT

PARAM ERROR...

The unique PARAM authorized to affect set of cards is PARAM, AUTOSPC, YES, but the listing of degrees of freedom affected must be delivered.

Numbering system (Grids and elements) : TBD

f) FEM check

The following checks has to be performed on all finite element models prior to delivery.

Mass properties check

The Grid Point Weights Generator (GPWG) should yield the correct masses and moments of inertia values and the location of the centre of gravity (COG).

Conditioning checks

The model must be free from ill conditioning and mechanisms. This can be demonstrated using NASTRAN SUPORT card. Strain energy (SE) and conditioning numbers ε shall be : SE < 10^{-2} Joule and $\varepsilon < 10^{-8}$. The maximum ratio accepted for a model is MAXRATIO > 10^{5}. If this cannot be achieved, then an explanation on the reason for the low stiffness for the degrees of freedom indicated must be provided. From matrix models, the maximum strain energy shall be 1 Joule (tbc).

Static loads checks

A static analysis shall be performed applying unit acceleration loading (1 g) in each of the three axes with the model constrained at its interface. This analysis shall be shown the following :

- interface forces should summate to the weight of the structure as given by the grid point weights generator.
- studying the reaction loads for all points should reveal no constraint forces at points other than legitimate boundary conditions locations (for Nastran, use the case control command SPCFORCES = ALL).

g) Dynamic matrix model

In case a dynamic matrix model is delivered, the following conditions apply :

- the stiffness and mass matrix shall be written in OUTPUT4 format (double precision). The degrees of freedom shall be in external sort.
- the DMIG matrix for these degrees of freedom shall be provided,
- the GRID cards and associated coordinate systems shall be supplied,

- a series of tests shall be conducted on the matrix model in order to demonstrate that it provides results similar to the initial model, and that it is free of ill-conditioning (Section 1 to 6 requirements are applicable),
- the main dynamic characteristics shall be provided (eigenfrequencies, plots of eigenmodes, effective masses).

h) Specification for thermo-elastic model

The zero-stress test has to be applied to validate the finite element model with the assumptions:

- the model is isostatically supported,
- all the materials used in the model are replaced by a fictitious homogeneous and isotropic material whose properties are: Young modulus = 10^{11} Pa and Poisson's ratio = 0.3
- an uniform elevation of temperature of $\Delta T = 100°C$ is applied throughout the model :
- in all the model maximum stresses and induced rotations have to be smaller than : VON MISES stress : < 10^3 Pa and Theta > 10^{-9} rd.

i) Deliverable information

The following information must be delivered to the Spacecraft Contractor :

1) a document describing the models and containing as a minimum :

- description of the reference frames (a drawing is required),
- grid point, element numbering and connectivity co-ordinate systems,
- modelling assumptions/boundaries conditions,
- plots of the underformed shape,
- discussion of compliance with this document (through matrix if possible),
- mass breakdown and distribution in the models.

2) the output listings including the bulk data echo from eigenvalue analysis for full model, with all six rigid body modes and associated strain energy and conditioning numbers. Associated model plots shall also be provided with a description of the modes and the main dynamic parameters of modal analysis (effective masses).

3) a tape with the following data :

- the complete set of bulk data cards for each model.
- the complete set of bulk data cards for the zero stress test for the thermostatic model.
- the complete mass and stiffness matrices when applicable (ASCII format)

4) a medium format support which can be either :

4.1). 3"1/2 IBM PC compatible disk (file in ASCII format)

4.2). Sun compatible cartridge on Exabyte 8 mm tape "tar" format

4.3). Compatible magnetic tape for VAX COMPUTER of the following specification :

- density : 1600 BPI,
- Record type : fixed length,
- Record length : 80 characters,
- Blocking : 20 records/block,
- ASCII code

5) a copy of the JCL (commands) used to put the data on the tape should also be provided.

8.4.1.2.4. Dynamic mathematical model validation

The unit structural mathematical model shall be updated if necessary according to test results and supplied to ESA/Industry in order to run meaningful system analyses.

This validation may be performed either through a specific modal survey test or a low level sine test whichever is the most practical and efficient.

All the modes observed during the test will be identified by their mode shape, their frequency and their damping ratio. The predicted modes of the Structural Mathematical Model (SMM) shall be correlated / updated against modes whose effective mass is greater than 10% of the rigid body mass. Local modes which require secondary notching shall be included whatever the effective mass. After validation, the SMM shall still describe the main modes up to 150 Hz .

8.4.1.2.5. Mechanisms functional analysis

Each mechanism shall be analysed functionally and the following information shall be at least supplied :

a) a detailed description of the mechanisms, with particular reference to its discrete components (bearings, actuators, switches) and to its operational / safety features

b) a detailed description of the operating modes with reference to ground and orbital actuations

c) 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)

d) a Failure Modes, Effects and Criticality Analysis (FMECA) defining the failure modes and the functional margins of safety against each of them

e) a performance description of the control system which includes the mechanisms.

8.4.1.3. Mechanical verification and testing

8.4.1.3.1. General test requirements

Mechanical verification and testing shall be performed - depending the models - to show that :

- the QM units meet the stiffness and environment mechanical requirements
- that the FM units is acceptable for flight.

For EM type models, their interfaces shall be verified (and traced) as for any other model. In any case, their compliance with the transport environments shall be verified. For QFM or EQM type models, combined requirements apply in following the individual model (QM, FM or EM) approaches.

8.4.1.3.2. Test Level Tolerances

They have to be in accordance with the general ESA standards..

8.4.1.3.3. Geometrical interface verification and physical mechanical properties

The PI shall verify that all units comply with the MICD, that fixations points have the correct position and definition within specified tolerances.

The PI shall provide the following unit properties :

- Mass,
- Dimensions,
- Moments of inertia,
- Position of de centre of gravity.

Except for mass and when tight balancing is required, the other properties may be determined by analysis. When necessary, 3D control results of the unit interface shall be supplied.

8.4.1.3.4. Quasi-static test or strength test

The main objective of this test is to demonstrate that the load carrying structure is able to withstand the flight limit loads without rupture, collapse, damage, permanent deformation or misalignment.

The related amplitude value (g level) is the flight limit loads factored by the qualification or acceptance factor (see part 3).

The test is not required in the following cases :

a) if analysis demonstrate positive margin of safety against the limit loads accounting for the safety factors of 2 and 3 at respective yield and ultimate fore metallic structures, or a safety factor of 3 for non-metallic structures,

b) for hard mounted units, if the test is being covered by the sine vibration test at low frequency.

Parts and subassemblies which do not fall in these cases shall be proof tested at 1.25 times the limit loads.

8.4.1.3.5. Vibration tests (Sinusoidal and Random)

Sinusoidal and Random tests (or Acoustic tests in some cases) are required to demonstrate that the experiment units can survive all stressing events whether during launch, handling, transportation or the Spacecraft AIT program.

8.4.1.3.5.1. Facility

The vibration test facility and procedure shall satisfy the following minimum requirements :

a) The shaker shall have at least 20% margin with respect to the maximum expected interface load.

b) The control equipment shall be able to maintain the specified tolerances.

c) The data handling equipment shall be sized according to the requested instrumentation.

d) In case of unexpected incident, smooth abort shall be programmed.

e) All test incidents shall be reported and fully explained before going on with the test sequence.

f) Blank test using the item fixture is not mandatory but is strongly advised (in particular to verify the cleanliness and related procedures).

Test facility cleanliness

Every precaution shall be taken to avoid contamination by oils, greases ... The test should take place in a class 100,000 clean room or better. A protection shall be used if needed.

Fixture requirement

The Unit shall be hard mounted on a stiff fixture by all its spacecraft attachment points. The PI will be responsible for the definition and procurement of the test fixture. The design of the fixture shall guarantee that the major modes of the unit are not modified (As a typical value, frequency shifts should be less than 5% on the lower frequency modes).

Configuration

The unit shall be vibrated in the launch configuration, except possibly for thermal hardware (MLI) when it does not contribute to the test article stiffness. All non-flight items shall be removed except for optical cubes which may be needed to monitor the unit alignment, and any other low weight instrumentation control equipment.

Vibration and control equipment

♦ Sinusoidal vibration

To control the vibration level applied to the test specimen, at least 2 three-axis accelerometers shall be rigidly attached on the test fixture near the specimen / fixture interface and shall be aligned with the excitation axis.

Accelerometers shall be calibrated for frequency response in the range 5-2000 Hz.

♦ Random vibration

Gaussian random vibration shall be applied with peaks clipped at three times the RMS acceleration.

The control accelerometer response shall be equalised such that the spectral density values are within + / - 3 dB. The filter roll-off characteristics above 2000 Hz shall be at a rate of 40 dB / octave or greater.

♦ Acoustic test

Input shall be controlled by at least 2 microphones located at about 0.5 to 1m from the test specimen.

Monitoring through 1/3 octave band should be preferred to octave band.

Recording instrumentation

All tests shall be fully recorded and records be properly labelled. All accelerometers shall be calibrated and show linear response in the range 5-2000 Hz for amplitudes up to 1.25 times the maximum expected during the tests.

♦ Sinusoidal :

Some carefully selected accelerometers will be used for automatic notching and abort in order to protect the test article.

♦ Random :

During the random vibration tests, the signal from the control accelerometer shall be passed through an analyser, the display time of which has been adjusted to suit the duration of the test. Where a narrow band swept filter analyser is used, the filter bandwidth shall be as narrow as allowed by the testing time and the length of spectrum to be traversed.

Vibration test levels and duration

♦ Sinusoidal :

- The sweep rate for the qualification test shall be 2 octaves / minute.
- The sweep rate for the acceptance test shall be 4 octaves / minute.
- The acceptance levels are 1.5 times smaller than the qualification levels (TBC).

♦ Random :

- The duration of the test shall be 2 minutes per axis for qualification test.
- The duration of the test shall be 1 minute per axis for acceptance test.
- Acceptance levels are 2.25 times smaller than qualification levels (TBC)

Notched input spectrum

If vibration tests induce global interface loads which are larger than the flight limit loads (or qualification loads), « primary » notchings will be allowed in the input spectrum. Notchings shall be defined in order that the qualification / acceptance test interface loads are not larger than the qualification / flight loads. For random tests, the modes with large effective masses will be notched at first.

Other notchings or so-called « secondary notchings » may be granted to protect sensitive equipment - mainly appendices which have large amplification factors - on a case by case basis. Acceptance of such notchings will however require the results of satellite coupled loads analysis.

In all cases a request for waiver shall be raised for such input spectrum modifications . The RFW shall describe :

a) The item which is under consideration and the reasons which support the request

b) The acceptable level for the unit (ie the requested notched input spectrum) and the underlying assumptions (damping ratio).

c) The frequency and mode of concern

Formal acceptance will in general require the results of system analyses (Satellite testing prediction) and will rely on satisfactory model predictions. After the low level sine test (see test sequence), the final qualification / acceptance input spectrum shall be defined accounting for measured amplification factors and estimated or measured interface loads.

Functional test verification / alignment

Before and after each test sequence, the specimen shall be functionally checked and / or alignment verified.

Vibration test sequence

For each axis, the following test sequence shall be performed :

a) Functional verification / alignment

b) Low level sine

c) Sine qualification or acceptance

d) Low level sine

e) Functional verification / alignment

f) Random qualification or acceptance

g) Low level sine

h) Functional verification / alignment

During the qualification test campaign, it is highly recommended to perform intermediate tests before the high level sine or random tests in order to verify the evolution of the amplification factors and modify if needed the notching strategy.

8.4.1.3.6. Acoustic tests

On those units which have a large surface and low weight over area ratio, an acoustic noise test shall be performed since these structures are more sensitive to direct acoustic launch excitation than to structure borne-inputs.

Configuration

The unit shall be tested in the same configuration as for the sine an random vibration test.

Test set-up

The test article shall be suspended in the test chamber by means of low stiffness bungees which will enable full and uniform acoustic excitation of the specimen located at the center of the test facility. The natural frequency of the suspension loaded by a rigid mass equivalent to the specimen shall be 5 Hz or less. The acoustic chamber shall be large enough so that the acoustic field will be spatially uniform in the frequency range of interest.

In case a different configuration is considered by the PI (eg test article fixed on a trolley), Spacecraft Contractor and ESA approval shall be required.

Vibration and control equipment

Input shall be controlled by at least 3 microphones located at about 0.5 to 1m from the test specimen.

Monitoring through 1/3 octave band should be preferred to octave band .

Requirements for accelerometers are similar to sine or random vibration.

Test description

The overall sound pressure level shall be introduced into the chamber with the spectrum specified. The test level shall be monitored through the field microphones. The specified spectrum shall be shaped at approximately 6 dB below the specification in order not to overstress the specimen.

The test sequence shall include the following tests :

a) Functional verification / alignment

b) Low level acoustic test (-6 dB)

c) High level test (+0 dB)

d) Low level test (-6 dB)

e) Functional verification / alignment

8.4.1.3.7. Shock tests

Satisfactory compliance with shock requirements is necessary to ensure that the unit design and workmanship is adequate to withstand the shock environment to be encountered during launch, separation and deployment.

Some units which include mechanisms may require verification in both the stowed and deployed configuration. Unless specified otherwise, verification through analysis will be accepted.

Shock test set-up

The equipment shall be attached to the shock test machine by means of a rigid fixture . During test, the mode of operation of the unit shall be representative of its normal mode at the moment the event is supposed to occur. Instrument monitoring and checkout equipment shall be utilised to judge the performance of the unit during the pre-test, test , and post-test periods.

At least one control accelerometer shall be fixed on the test fixture close to the unit in order to record the input shock. The shock response spectrum of the control accelerometer output shall be made.

Recording accelerometers shall be calibrated for frequency response and amplitude linearity characteristics to values 1.25 time the maximum expected to be recorded during the tests.

Shock test level

The shock test level will be defined by ESA / Prime Contractor either as a half sine input with specific amplitude and duration or as a frequency dependent amplitude level.

8.4.1.3.8. Thermal and thermal stability tests

Thermal test requirements are defined in section 8.4.2 . It is the PI responsibility to verify that the unit structural integrity will not be endangered by the required thermal tests. Thermal stability tests are quite specific to each equipment and related structural requirements are defined in section 8.4.2.

8.4.1.4. Mechanism Test Requirements

8.4.1.4.1. Mechanisms Verification

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.

Tests shall be performed to check mechanisms performance in both launch and operational configurations. 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 test sequence are applicable:

a) Functional tests,

b) Mechanical environment tests,

c) Thermal vacuum functional test,

d) Functional tests,

e) Life tests.

A standard test methodology can be proposed to verify the disturbance sources models. A dynamometric platform can be used to measure forces and moments generated by an instrument when mounted to a rigid interface. The test set-up characteristics must be such that the measurements are valid in a broad frequency range (1 Hz, 150 Hz at least). To achieve a good representativeness of these tests, the configuration of the test specimen must be "as close as possible" to the flight one.

8.4.1.4.2. Mechanisms Lifetime Tests

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. 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	x4 (the minimum number to be used is 10)
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. In order to determine the lifetime to be demonstrated by test, an accumulation of actuations multiplied by their individual factors shall be used.

8.4.1.5. Pressurised and/or Sealed Units Test Requirements

8.4.1.5.1. Proof Pressure Tests

Satisfactory verification of compliance with pressurised items requirements is necessary to ensure that the unit design and workmanship is adequate to withstand the launch and in-orbit environment without degradation and potential hazard to the S/C.

Proof pressure test set-up

TBD.

Proof pressure test level

The shock test level will be defined by ESA / Spacecraft Contractor according to the design requirements of section 3.2.2.

8.4.1.5.2. Leakage Tests

Satisfactory verification of compliance with pressurised items requirements is necessary to ensure that the unit design and workmanship is adequate to withstand the launch and in-orbit environment without degradation and potential hazard/contamination to the S/C.

Leakage pressure test set-up

TBD.

Leakage test level

The shock test level will be defined by ESA / Spacecraft Contractor according to design requirements of section 3.2.2.

8.4.2. Thermal Verification Requirements

8.4.2.1. General requirements

An appropriate set of tests and analyses shall be conducted to demonstrate the following experiment capabilities :

- the satisfactory performances of all the experiment units in vacuum within the qualification or acceptance operating temperature range,
- the capability of the experiment units to withstand without degradation, the qualification or acceptance non-operating temperature range and switch-on minimum temperature,
- the ability of the internal thermal design of the experiment units to guarantee the required temperatures on internal critical items from the specified interface temperature at reference point and thermal environment of each experiment unit taking into account the apportionment of heat sources,
- the quality of the workmanship and the selected materials to pass thermal cycle test.

These tests shall cover at least the realistic flight situations (in terms of environments and configurations) which are derived from mission analysis.

In addition, the test must show the adequacy of the thermal mathematical model with the hardware thermal behaviour and the accuracy of the thermal predictions.

8.4.2.2. Thermal models and analysis

Thermal modelling

Each experiment unit must be modelled by a Thermal Mathematical Model (TMM). All the thermal mathematical models are delivered to be incorporated in the general spacecraft thermal mathematical model. The THERMICA (or ESATAN / TBC) software shall be used for all the thermal mathematical models. The degree of detail (i.e. number of nodes and conductors, etc.) shall be at least sufficient to show that all interface requirements are met. The model shall allow to perform transient analyses, hence a thermal capacity must be allocated to each node.

Heat dissipation shall be identified, in each operating mode. In case of an experiment unit with a large number of operating modes, the number of these operating modes can be reduced for thermal modelling purposes, in agreement with ESA / Spacecraft Contractor.

The PI shall give a description of the configuration, the interface control drawing as well as all the assumptions and simplifications used for modelling. The accuracy of all the thermal mathematical model shall be demonstrated by the PI using simple reference thermal loads (type of test TBD by PI with ESA / Spacecraft Contractor agreement).

A Simplified Thermal Mathematical Model (STMM) with TBD thermal nodes must be provided by the PI on request from ESA / Spacecraft Contractor.

The TMMs must be validated on elementary tests (check of material conductance, infrared emissity, solar absorptivity, ... etc) before delivery to ESA / Spacecraft Contractor.

The TMMs of the experiment units whose interface temperature control is under ESA / Spacecraft Contractor responsibility will be verified by PI during thermal vacuum tests on the experiment units.

The TMMs of the units whose thermal control is under PI will be used by PI to demonstrate that the design is compliant with the requirements defined in section 5.

8.4.2.2.1. Analysis Cases

The contractor responsible for the thermal design shall establish the analysis design cases to be analysed to demonstrate the acceptability of the thermal control design throughout the various mission phases. As a minimum, the following shall be considered :

- BOL and EOL,
- Maximum and minimum environmental conditions,
- Steady states and transient analyses,
- Stowed and deployed.

8.4.2.2.2. Thermal Interface Models

The Instrument Supplier shall provide interface models for System analyses. The purpose of these interface models shall not be to predict instrument temperatures but to provide thermal interface data to be used by Instrument Suppliers in their thermal design and analysis. These interface models shall consist of mathematical models and geometric models and shall :

- enable both steady state and transient analyses to be performed,
- have optical properties for both BOL and EOL analyses,
- be representative of all interface heat fluxes,
- where appropriate, enable different configurations of the instrument (e.g. stowed and deployed) to be analysed,
- and provide heater switching logic for all phases of operations.

8.4.2.3. Thermal test requirements

The thermal interface requirements for all experiment units are defined in section 5 (type of mounting, radiative and conductive interfaces,...etc). The PI shall test the units in respect with these requirements.

8.4.2.3.1. Thermal vacuum test

The thermal vacuum test is required to evaluate and demonstrate the functional performance under vacuum of the experiment units under the extreme and nominal modes of operation, with temperature conditions for the experiment more severe than the maximum and minimum temperatures predicted for the mission, namely within the acceptance or qualification temperature range.

Cycling between temperature extremes has the purpose of checking performance at other than stabilised conditions and of causing temperature gradient shifts, thus inducing stresses intended to uncover incipient problems. A minimum of four (4) cycles shall be performed for acceptance of proto-flight testing, and eight (8) for dedicated qualification.

8.4.2.3.1.1. Test arrangement

As a general requirement, each experiment unit must be tested with conductive and radiative interfaces as close as possible with the spacecraft interfaces in orbit. In particular :

- each experiment unit shall be connected to the test mounting panel using identical fixation interfaces as its fixation onto the spacecraft structure (insulation washers, interface filler,...),
- the test mounting panel temperature must be representative of the flight mounting panel temperature (margin to be added in test),
- its external thermoemissive properties must be identical to the flight ones,
- the radiative environment must be as close as possible with the flight one margin to be added in test).

Because qualification and acceptance temperature range encompass in orbit temperature range, the test interface temperature and/or power must be such to insure the adequate margin on temperature equipment specified on margin policy. The thermal vacuum test arrangement must be designed to give the required qualification or acceptance temperatures on the equipment with approximately representative heat flows to and from the environment.

A possible test set-up and method is depicted in figure 8.4/1. The in flight mounting panel is replaced by a thermally controlled conductive support frame used as heat sink. The equipment is directly mounted to this heat sink. A shroud surrounding the experiment unit simulates the radiative environment. The temperature of the experiment unit can be controlled both radiatively by adjusting the shroud temperature and conductively by adjusting the temperature of the fluid circulating in the mounting frame.

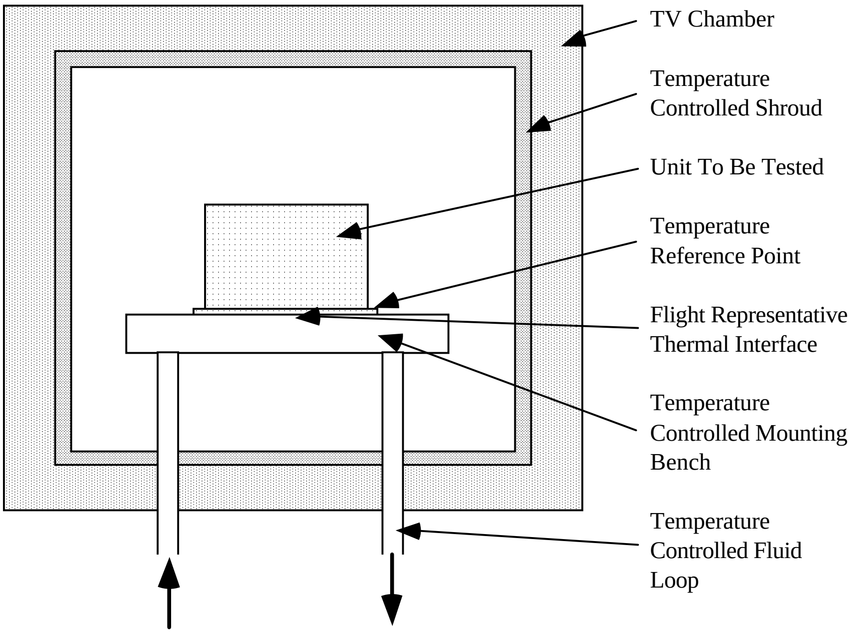


Figure 8.4/1:Equipment thermal vacuum test arrangement

For all electronics units located internal to the spacecraft, the shroud temperature is maintained around the ambient temperature during the test. The conductive heat sink temperature is controlled to achieve the required temperature level on the experiment unit baseplate as specified in section 5.

The experiment units whose thermal control is performed by PI, will have a thermal balance. The thermal vacuum test of these units will be performed in the configuration of the thermal balance test (see chapter 8.4.2.3.2).

8.4.2.3.1.2. Test conditions and instrumentation

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 begin 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,
- stabilisation is achieved when the equipment temperatures have been maintained within tolerance and have not changed by more than 1°C during the previous one hour period.

The PI shall be responsible to define the adequate test instrumentation in order to demonstrate that temperature levels are achieved and to validate the experiment unit thermal mathematical model. This test instrumentation shall include as a minimum :

a) for the shroud temperature, the instrumentation necessary to allow accurate temperature control by using fluid loop or/and electrical resistance heaters,

b) for the experiment unit, at least one temperature sensor on each unit casing wall, and one temperature sensor on each unit foot,

c) for the mounting plate, at least one temperature sensor close to each unit mounting foot and four temperature sensors to derive the lateral gradients inside the mounting panel.

8.4.2.3.1.3. Test level

The equipment test levels shall be in accordance with figures 8.4/2 and 8.4/3.

a) Design verification, qualification and protoqualification test

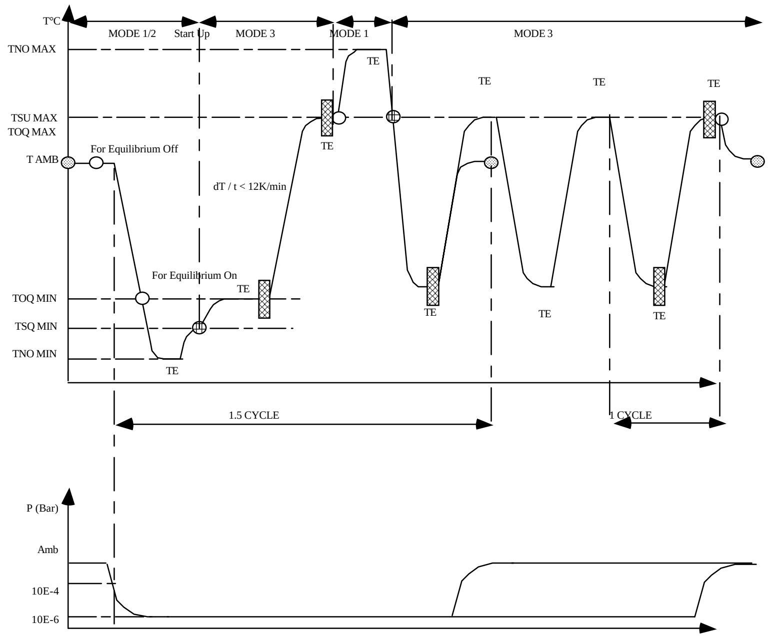
During testing, the same item shall be tested, in the normal post-lift-off sequence, to the thermal environments appropriate to non-operating, switch-on (start-up) and operating qualification temperature limits. The performance of the test, i.e number of cycles, temperature ranges and duration are shown in figure 8.4/2. If preferred, the temperature cycle profile can be changed to give a hot phase first. The temperature gradient dT/dt must be <2°C/mn for electronics boxes inside the SVM. For instrument units mounted externally of PLM and SVM, higher gradients can be defined in the equipment specification (tbc),

b) Heat dissipations

The PI shall demonstrate the units dissipate not more than the maximum values defined in section 5.

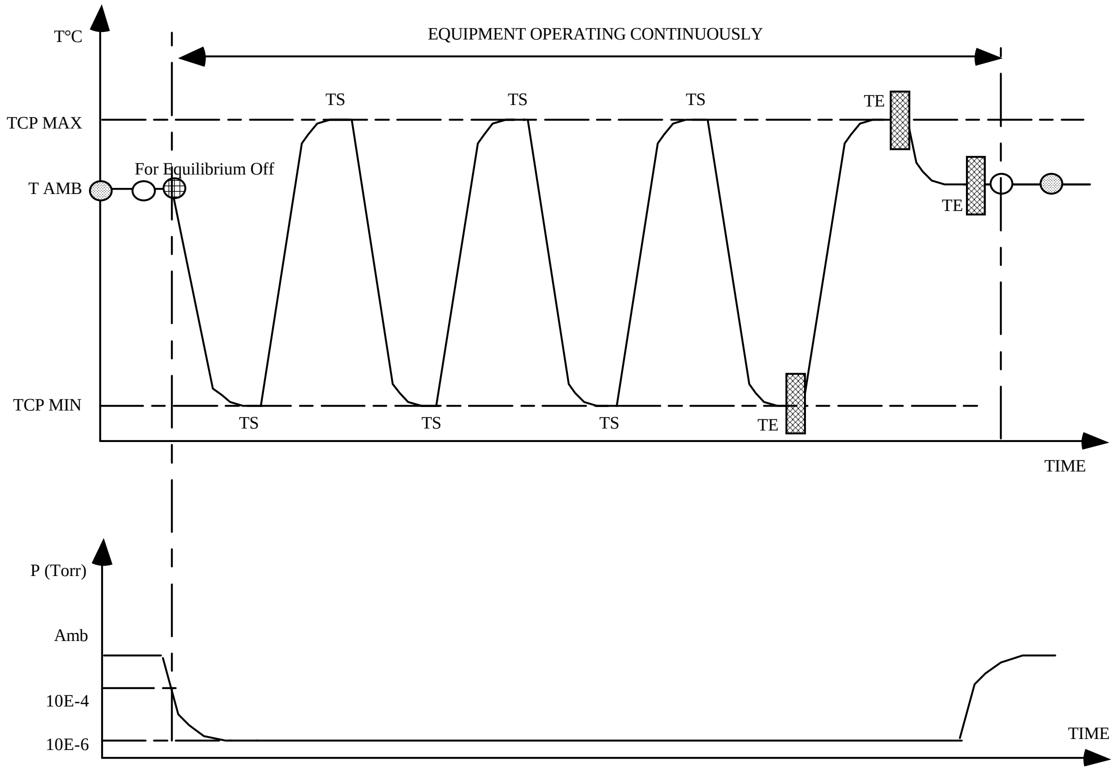
c) Acceptance test

In order to ensure the application of the maximum stress condition, the unit shall be operated continuously throughout the test, which shall comprise 4 cycles, with full functional testing at the last 2 extremes, and adequate monitoring during the remainder of the test. The required sequence is shown in figure 8.4/3. If preferred, the temperature cycle profile can be changed to give a hot phase first.



T	Temperature
TNOMAX	Maximum non-operating qualification temperature (non-functioning)
TNOMIN	Minimum non-operating temperature
T AMB.	Ambient temperature
TOQMAX	Maximum qualification temperature (functioning)
TOQMIN	Minimum qualification temperature (functioning)
TSUMAX	Maximum Start-up Temperature (the highest design temperature of the equipment, at which the equipment may be switched on)
TSUMIN	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 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
t_E	Minimum equilibrium temperature Time, dwell time
⊗	Switch-on (Start-up)
O	Switch-off

Figure 8.4/2: Thermal vacuum qualification test at experiment unit level



T	Temperature
T AMB.	Ambient temperature
TOQMAX	Maximum specified acceptance temperature
TOQMIN	Minimum specified acceptance temperature
ts	Minimum time for which equipment must be controlled a temperature extreme: 1hour
tE	Minimum time for which equipment must be controlled a temperature extreme prior to starting performance test: 2 hours
Å	Initial and final performance test
X	Intermediate reduced performance test
t_E	Minimum equilibrium temperature Time, dwell time
⊗	Switch-on (Start-up)
O	Switch-off

Figure 8.4/3: Thermal vacuum qualification test at experiment unit level

8.4.2.3.2. Thermal balance tests

A thermal balance test at experiment level shall be conducted on experiment units whose thermal control is under PI responsibility. The results of the thermal balance test are used :

- to verify that the instrument operates nominally under realistic mission conditions
- and to correlate and update the experiment thermal mathematical models.

The thermal balance test simulates extreme and nominal conditions to verify the experiment unit thermal control system. The number of energy balance conditions simulated during the tests shall be sufficient to verify the thermal design. The exposure shall be sufficient enough for the test item to reach stabilisation so that the temperature distribution in steady state condition may be verified.

8.4.2.3.2.1. Test method

The experiment units must be tested in conditions close to the flight but the interface conditions can be slightly different of the flight ones especially the sink temperatures. In that case, test predictions must be made from the thermal mathematical model with test thermal interface conditions; the thermal mathematical model is then updated from the test results and the updated thermal mathematical model is used to make flight prediction.

The test configuration must encompass the conditions necessary for evaluating thermal design and correlation of the analytical model :

- in steady state :
    - minimum external absorbed flux and internal power dissipation
    - maximum external absorbed flux and internal power dissipation
    - nominal flux and internal power dissipation
- in transient case :
    - minimum to maximum external absorbed flux and internal power dissipation

Thermal instrumentation shall be attached at locations where the data may be correlated with specific points on the analytical thermal model produced by the PI. The set-up shall include means to determine the different heat flow in order to correlate mathematical models.

The duration of the thermal balance test will depend on the unit design and operating modes as well as time to reach stabilisation. The verification criteria of the unit thermal design including the allowable difference between predicted and measured temperatures will be determined by the PI thermal analyst and agreed by ESA/Prime Contractor.

8.4.3. Electrical

8.4.3.1. General Electrical Verification Approach

The soundness of the Mars Express instruments electrical interface design shall be based on the following points :

- ➢ margin management and allocation — high margins being allocated to parameters on which higher design criticality is expected,
- ➢ early analyses and predictions, implying if needed the use of simulation tools and models,
- ➢ breadboarding of critical interfaces,
- ➢ EICD and budgets management,
- ➢ identification of non-compliances,
- ➢ acceptance testing.

8.4.3.2. Analyses

8.4.3.2.1. Compliance Matrix

The instrument supplier shall provide a Compliance Matrix, aiming at defining instrument compliance with respect to the PID. In case of non-compliance, a justification shall be provided, so as to state on the non-compliance acceptability.

8.4.3.2.2. Electrical Interface Control Document

The instrument supplier shall provide an Electrical Interface Control Document, as defined in Section 6.

8.4.3.2.3. Budgets

In addition to the data provided within the Electrical Interface Control Document, the following budgets shall be provided as part of the instrument supplier data packages :

- ➢ line budgets, defining the number of signals in each category required by the instrument,
- ➢ TM/TC budgets, defining the commands and telemetry required by the instrument, as well as their main characteristics.

8.4.3.2.4. Analyses

In case a numerical simulation tool and model is used in the framework of instrument electrical interface analyses, model validation test shall necessarily be undertaken in order to demonstrate the soundness of the simulation results.

The following analysis shall be performed for all electrical interfaces during the design phase :

- Part Stress Analysis, in accordance with ESA PSS-01-301 (*Derating Requirements Applicable to EEE Components for ESA Space Systems*),
- Worst Case Analysis, aiming at demonstrating that all interface requirements are met in the worst case.

8.4.3.3. Tests

8.4.3.3.1. Electrical Tests

Electrical tests shall be performed with an average power bus voltage of 28 V, except where explicitly notified otherwise.

For redunded instruments, tests shall be performed with nominal and redundant channels, and all cross-strapped configurations shall be envisaged.

The following electrical tests shall be performed during instrument acceptance tests :

- continuity and insulation tests,
- compliance with input voltage specification, including anomalous regimes,
- power budgets, considering lowest and highest operational input voltages,
- Deployment Devices interface tests,
- protections and automatic devices tests,
- Science Data collection interface tests,
- Command & Control interface tests,
- Discrete Telemetry interface tests,
- Configuration Commands interface tests,
- Clocks and Synchronisation Lines interface tests,
- High Voltage tests,
- Conducted EMC tests,
- Radiated EMC tests,
- ESD tests as applicable.

8.4.3.3.2. Test Connectors

Any connector giving access to the satellite via paths other than the ones used for in-flight operation (i.e. RF radiation or other physical phenomena), particularly during the tests listed below, is known as a test connector.

Test connectors are used :

- *as inputs, to feed in simulation or stimulation signals (e.g. sun sensor signals),*
- *as outputs, for pulses to be time-stamped, or in order to monitor traffic on the data buses, etc.*
- *to simulate equipment that cannot be activated during tests (pyrotechnic devices, Solar Array, thrusters, etc.) or which are not present on the satellite,*
- *to ensure the safety of personnel and equipment (connector linking satellite ground to local and test facility earths),*
- *to stimulate automatic protection devices (disjunction devices, etc.).*

As a minimum, test points shall be protected against operator errors and ground facility failures by one of two means :

- protection resistors,
- diodes against polarity inversion.

The electrical interfaces between the EGSE and the equipment under test shall not affect in any way the requirement for grounding the reference point in a single point. This implies the use of differential or insulated interfaces.

These interfaces shall be defined within the units Electrical Interface Control Documents (EICD).

Wherever possible, test connectors shall be connected in parallel, rather than in series, with the relevant electrical circuits.

Interface circuits shall be compatible with cable lengths of up to 40 m between equipment and EGSE.

8.4.3.3.3. High Voltage Tests

Proper discharge tests shall be performed so as to prove the hardening of the Instrument against high voltage breakdown.

Any high power microwave Instrument shall be tested such that a 6 dB margin between the maximum expected power levels and the onset of multipaction is established.

In case an Instrument has components operating at a voltage in excess of 100 V, it shall be demonstrated during the Instrument thermal vacuum test that corona discharge and arcing do not occur for pressure between ambient and flight conditions.

The susceptibility of high voltage Instrument to arcing shall be characterised by test.

8.4.3.3.4. EMC Tests

Test Site

Unless there is some major reason not to do so, these tests shall be performed in a shielded room meeting the following requirements :

a) The internal dimensions shall be such that the antennae are always at 1 meter from each wall, except in the case of dipole and rod antennae for which the minimum distance shall be 30 cm.

b) Power sources shall be filtered so that the residual interference is 6 dB or more below the specification limits. Ambient electric fields shall be 6dB or more below the specification limits.

c) For these measurements, the power source shall power a load offering the same impedance as the test article with the relevant measuring instrument and test facilities operating.
If, at the instant when the measurements are recorded, the overall interference level (ambient interference plus interference due to the UUT) is below the specified limit, the article is deemed to pass the test.

d) The shielded room shall include a ground plane at least 2 m^2 in area (and at least 0.75 m wide), meeting the requirements of MIL-STD-462. The ground plane may take the form of a sheet of copper, brass or lightweight alloy with a minimum thickness of 0.7 mm.

e) The ground plane shall be linked to the shielded room at least once every meter by suitable bonding leads. These leads shall preferably take the form of copper straps at least 0.7 mm thick and 1/5th as wide as they are long.

f) The use of absorbent materials similar to those used in anechoic chambers is recommended.

Performing tests outside a shielded room

Choose a test site that is free of obstacles and pointed objects. It should preferably be close to an earth ground or ground connector. The ground plane shall be connected to the previously mentioned earth ground or ground connector.

The ground plane may take the form of a sheet of copper, brass or lightweight alloy with an area of at least 2 m^2.

Ambient electromagnetic interference shall be monitored in the way defined immediately above.

Test Facilities

The minimal characteristics for measuring receivers are :

➢ Input impedance	: 50 Ω	
➢ Input VSWR	: up to 200 MHz	: <5
	: from 200MHz to 18 GHz	: < 2

➢ Specified analysis bands :

FREQUENCY RANGE	BANDWIDTH
30Hz - 10 kHz	10 Hz
10kHz - 2.5 MHz	100 Hz
MHz - 25 MHz	1 kHz
25 MHz - 1 GHz	10 kHz
1 GHz - 18 GHz	100 kHz

If necessary, the analysis bands may be narrowed to reduce measurement noise.

➢ Recommended detection modes :

- peak
- rms
- average

➢ Frequency measurement accuracy : 1%

➢ Voltage measurement accuracy : 2 dB

Note : a spectrum analyser may be used as the measuring receiver. The resolution bandwidths used for the measurements shall be similar to those recommended here above.

A current probe shall be used to measure the conducted interferences over the range 30 Hz to 100 MHz.

The probe factor is defined as the ratio of the voltage measured across its output loaded by a pure 50 Ω resistive load and the current flowing through the clamped leads.

Thus $PF = \frac{V_S}{I_p}$

The probe factor curve versus frequency shall be joined in annex to any EMC test report.

The following types of measuring antennae are recommended in the following frequency ranges :

➢ below 30 MHz : 1 meter rod antenna

➢ from 20 MHz to 200 MHz : Dipole

: Biconical antenna

- from 200 MHz to 1 GHz : Dipole
  : Conical logspiral
  : Logperiodic
  : Horn antenna, Double ridged guide
- from 1 GHz to 12.4 GHz : Double ridged guide
- above 10 GHz : Horn antenna, parabola

In order to reproduce the system power bus impedance and to standardise the measurement conditions used in different test sites, conducted emissions measurements on primary power lines shall be performed on inserting a Line Stabilisation Impedance Network (LISN) between the EGSE power supply and the unit under test.

Before starting any CE test, the source impedance connected to the unit shall be measured and compared to the specified line impedance.

All measuring instruments shall carry a label indicating the last date of calibration, which shall be less than one year before the testing activities.

Although this requirement does not apply to passive instruments (e.g. current probes), the corresponding calibration curves shall be available.

Test Set Ups

The test conditions and set-ups shall be in accordance with MIL-STD-462 where applicable.

When testing the conducted susceptibility of the satellite modules or units on the power lines, the sinewave signals shall be injected using one of the two methods detailed in Figure TBD.

The set-up dedicated to each unit shall reproduce the unit's nominal operating conditions as closely as possible. In particular, set-ups for testing complete items of equipment shall meet the requirements listed here below :

- grounding conditions identical to those of installed equipment
- cables of same type and offering the same immunity to interference as those used with the installed equipment.
- a detailed description and definition of test set-up cabling and wiring supplied
- each test unit transmit and/or receive antennae replaced by dummy antennae or shielded loads offering an impedance equivalent to that of real antenna.

The test conditions and the test set-ups shall be described in detail in the EMC test plan dedicated to the testing of each unit or module.

UUT Operational Mode

Conducted and Radiated Emission tests shall be performed with the Unit Under Test in a configuration which is demonstrated to maximise emissions.

Conducted and Radiated Susceptibility tests shall be performed with the Unit Under Test in a configuration which is demonstrated to maximise the susceptibility of the unit.

The primary power voltage shall be set at 28 V for all EMC tests.

Test Report

Raw results shall be presented in the following form :

- either as XY plots with a resolution sufficient to allow easy exploitation,
- either as oscilloscope or spectrum analyser photographs.

Data gathered during emission tests shall be acquired by doing an automatic or manual continuous frequency sweep.

Results shall be presented, after exploitation, superimposed on the templates of this specification, in order to allow a quick and easy analysis.

All complementary information, such as sensitivity, bandwidth, antenna factor, etc. shall be provided together with the plots or the photographs.

8.4.3.3.5. EGSE EMC Requirements

Grounding and Isolation

The grounding of all EGSE shall follow the safety rules listed below :

a) Load currents shall not be allowed to flow in any part of the ground networks,

b) The ground networks and EGSE grounding straps shall be sized to carry worst case faults currents that could occur,

c) All EGSE enclosures shall be connected to the EGSE ground star point,

d) On-board hardware interfacing power shall be grounded in a way to simulate the satellite grounding configuration (i.e grounded at the instrument EGSE ground star point),

e) The integration facility shall provide a dedicated ground bus bar where no other users are connected to,

f) The facility ground rail shall be located in such a way to enable the connection of EGSE/Instrument star point to the facility ground rail with a resistance of less than 10 MΩ (length 10 m maxi).

AC Line Power

All EGSE shall be supplied by dedicated isolation transformers which shall provide galvanic isolation of the EGSE use AC line power (isolated primary power) from the AC line facility provided power (non-isolated primary power) and safety earth.

Adequate safety protection devices for the dedicated EGSE isolation transformers shall be provided.

The neutral of the AC line facility provided power (non isolated primary power) shall not be directly connected to the facility ground bus bar.

Galvanic isolation shall be maintained between the earth connection of the non isolated primary power and the EGSE isolated primary power ground reference point.

EGSE Secondary Power

For EGSE secondary power, including on-board hardware interfacing power supplied by converters, galvanic isolation shall be maintained between :

- on-board hardware interfacing power and non-isolated primary power
- secondary power and non-isolated primary power

In each case the above defined isolation shall be verified by test.

Isolation shall be confirmed by test prior to the connection of the dedicated grounding connection for each new EGSE hardware configuration.

On Board Hardware and Signal Interfaces

The Isolation between the on-board hardware ground reference and the EGSE ground reference must be galvanically ensured. This isolation requirement is not applicable to the RF signal interfaces using a carrier frequency (or a repetition frequency) larger than 10 MHz.

For these RF signals, coaxial cables or wave guides shall be used.

Harness Definition

The harness configuration between the unit under test and the EGSE shall be representative as close as possible to the flight standard one.

The representativity must be ensured in terms of :

- cable configuration : (TP , TSP , COAXIAL..)
- cable characteristics : (balancing impedance Zc, Transfer impedance Zt, AWG)
- shields termination’s and bonding
- height above grounding planes or GRRs.

The Spacecraft Contractor will communicate these information's to the subcontractor and to the co-contractor in charge of EMC testing together with EMC test procedure approval.

Bonding

Each EGSE rack shall provide an internal ground reference point.

Each EGSE rack shall provide a bonding stud. The bonding stud shall be connected to the internal ground reference point by less than 5 mΩ.

The EGSE rack housing shall be grounded to the internal ground reference point by not more than 5 mΩ.

The safety wire of the non isolated primary power shall be galvanically isolated from the rack internal ground reference point.

Note : the use of a removable strap is permitted if required.

Such a strap, if used shall be handled as a red tag item.

Each EGSE shall provide an EGSE star point, to which all associated EGSE Equipment bonding studs can be connected to allow a single connection to the METOP ground star point with less 10 mΩ.

Prior to the attachment of the dedicated ground connections to the EGSE ground star point, all EGSE racks shall maintain mutual galvanic isolation.

This shall be confirmed prior to the connection of the dedicated grounding connections and prior to the harness connection.

Note : It must be noted that for higher level test configurations (i.e. satellite level or RFC test) the EGSE may be formed into two ground star groups : it is the responsibility of the instrument supplier to ensure that this configuration can be facilitated.

Cable Shields

The grounding of the shields on interfacing directly with onboard hardware and EGSE shall be in accordance with the following rules:

- the outer shields shall be bonded to the chassis ground at both ends, so EGSE side and onboard hardware side,
- the outer shields shall be bonded to the structure ground at the location where the harness pass through the shielded wall between the EGSE room and the EMC test chamber.

(This applies also to the RF coax-cables and waveguides.)

For EGSE, interfacing with onboard hardware, the requirements of cabling and connectors are applicable.

For each unit tester/test Equipment a grounding diagram shall be established by the EGSE / Instrument supplier.

For each electrical / functional test a grounding diagram showing the grounding of the EGSE and the item under test shall be established and form part of the test procedure.

EGSE RE/RS Performances

The EGSE shall not compromise the objective of the EMC test.

For EGSE interfacing directly with onboard hardware the relevant conducted emission requirements minus a 6 dB safety margin are applicable.

These requirements shall be verified by test prior to the EMC test sequence with the onboard hardware switched off and the EGSE switched ON.

For EGSE interfacing directly with on board hardware the relevant conducted susceptibility requirements are applicable.

These requirements shall be verified by design review.

For EGSE harness interfacing directly with the on board hardware, the relevant radiated emission requirements minus a 6 dB safety margin are applicable to the harness part routed in the EMC test chamber.

These requirements shall be verified by test prior to the EMC test sequence with the onboard hardware switched off, the EGSE switched ON and the interfacing harness nominally connected at both ends, so to the EGSE and onboard hardware interfaces).

For the EGSE Equipment located in the EMC test chamber during the radiated tests, the applicable requirements are the following:

- the relevant radiated emission limits minus a 6 dB margin,
- the relevant susceptibility requirements.

The tests to be carried out on these EGSE Equipment are :

- Narrow band radiated emissions from 150 kHz to 18 GHz with the EGSE powered and the interface harness connected to the EGSE and to the device under test (ambient measurement prior to RE tests with the onboard hardware switched off and the EGSE switched ON).
- Radiated susceptibility to the Mars Express RF transmitter frequencies with interface harness connected to the EGSE and to the device under test.

8.4.3.3.6. Antennas

For antennas, the following tests shall be performed :

- antenna pattern measurement for TBD elevation angles and in TBD cuts in azimuth,
- VSWR measurement.

8.4.4. Software verification and validation

The ESA PSS-05-0 standard is fully applicable for the verification and validation of the payload software. Special attention shall be paid to the interface with the spacecraft software.

All the documents defining the payload software architecture, design, development, coding, tests and validation shall be made available to the Spacecraft Contractor.

For new payloads, the prime contractor shall be associated to the development and validation of all aspects related to software interface with the spacecraft.

On the Spacecraft Contractor side, functional tests will be performed to validate the interface between the spacecraft software and the payload software. As both software communicate via TM/TC, two kinds of verification tests will be performed at spacecraft level :

- Sending of TC from the S/C software towards the payload software with checking of the correct command reception at payload level. Depending on the implementation of the payload command/control, checking of the correct reception by the S/C software of the associated acknowledgement TM will be performed.
- TM generation at payload level and checking of the correct reception of this TM at S/C level (this including science data).

From a general standpoint, all mission modes (through realistic scenarii) will be tested at spacecraft level in order to finally accept the payload software.

8.4.5 Performances Verification

8.4.5.1 Definition and Implementation

By definition, the performances verification is the determination either by test, analysis, or a combination of both, that the complete flight payload (or any payload unit) can operate as intended throughout its pre-defined mission. The detailed implementation of these actions is to be described in the Performances Verification Plan (PVP).

These basically include the proof (by analysis, test, or both) that :

- the design of the complete payload (or payload unit) has been qualified
- and that the complete flight payload has been accepted.

This acceptance means that the payload (or payload unit) has been built according to the qualified design, and is ready for any forthcoming satellite ground and flight operation.

Part of the acceptance is to be obtained through proper (full or limited) performance testing.

8.4.5.2 Full Performance Test

The Full Performance Test (FPT) is a detailed demonstration that the hardware and software meet their performance requirements within allowed tolerances. It shall nominally demonstrate this satisfactory performance, when the payload is fed with appropriate stimuli, in all the operational modes.

The test shall demonstrate the proper operations of all redundant circuitry.

Any initial FPT shall serve as a baseline against which the results of all the later FPT's can be unambiguously compared. This may be in particular the case for :

- the various payload (or spacecraft) level FPT's before and after the environmental test sequence
- or the last payload level FPT and the first spacecraft level one (if practicable).

A minimum of two FPT's are required at payload level :

- the first one just after the payload complete integration
- and the second one just before delivery (after the overall test sequence).

8.4.5.3 Limited Performance Test

The Limited Performance Test (LPT), subset of the Full Performance Test (FPT), shall demonstrate that the performances of selected hardware and software payload functions are within allowed limits.

As for the FPT, any initial LPT shall serve as a baseline against which the results of all the later LPT's can be unambiguously compared. This will be in particular the case during and after the various individual environmental tests, as appropriate, in order to demonstrate that the instrument functional capability has not been degraded.

These limited tests may also be used in cases where a comprehensive performance testing is unwarranted or impracticable. This may be the case, at satellite level, for :

- some specific optical performances testing of any orbiter instrument
- or the performances testing of the landers after integration on the satellite.

The nominal payload LPT (runned at payload level) shall be implementable at satellite level.

8.5 Payload Final Acceptance

8.5.1. General Principles

The acceptance shall demonstrate that the payload has been fully verified in terms of :

- scientific performances (including calibration and characterization)
- behaviour versus environmental conditions (including EMC)
- all the functional interfaces

The acceptance of the payload will be performed in the following staged fashion :

- completion of acceptance tests, including calibration / characterization at the payload supplier premises, in order to verify that the payload together with ground suport equipment meet all interface specifications and that the payload is ready for integration into the satellite
- delivery of the payload together with ground support equipment (including software) to the satellite AIV site, and performance by the payload supplier of an incoming inspection
- delivery and review of the acceptance data package.

Following successful completion, the payload will be released for integration onto the satellite. Notwithstanding with the mandatory instrument level tests, the payload software acceptance will only be effected after S/C level tests.

8.5.2 Acceptance Procedure

The acceptance procedure will check and ascertain the following topics :

- visual inspection
- compliance of the interfaces measurements (overall)
- availability of a complete set of functional performances data (using both the Limited Performances Test and Full Peformances Test procedures)
- availability of calibration and characterization data
- any ground support equipment relevant characteristics

After successful demonstration of performance, a Certification of Compliance will be issued and the shipment of the payload and associated deliverables to the satellite AIT site authorized.

8.5.3 Incoming Inspection

The payload incoming inspection includes visual and electrical functional tests :

- the initial inspection includes the verification of the payload mechanical properties as well as visual inspection for any physical damage (quality control aspect)
- the electrical incoming inspection will be performed with the payload dedicated tools and ground support equipment in order to demonstrate that no degradation has occurred as a result of transportation.

For the later, the functional performances will be verified in ambient conditions in using the Limited Performances Test (LPT) procedure.

The time slot required for all incoming activities is expected to be max. TBD working days.

Table of Contents

9. PRODUCT ASSURANCE

9.1 Introduction

9.1.1 Scope.

This part addresses the specific MARS EXPRESS payloads Product Assurance Requirements.

While all spacecrafts procured by the European Space Agency are required to conform to the general Agency's Product Assurance Requirements (as set out in the ESA-PSS series of documents), it is not the intention to impose (from the pure Spacecraft Contractor viewpoint) this full series of requirements on the MARS EXPRESS payloads.

The following sections cover requirements and guidelines (together with their rationale) which are aimed to enhance the overall mission integrity. They are derived from the general ESA requirements and have been tailored and limited to the relevant items contributed by the Principal Investigators.

It is to be understood that this tailoring exercise (reduced to the critical interfaces / MMS scope of responsibility) needs to be approved by ESA to be extended to either the overall payload or the scientific mission aspects (ESA scope of responsibility) and performances.

9.1.2 Documents.

The documents quoted hereafter are listed either for application when the so-called « critical interface » is concerned, or are kept as guidelines for the whole instrument elesewhere.

ESA PSS-01-20 iss.2 Feb 91	Quality assurance of ESA spacecraft and associated equipment
ESA PSS-01-30 iss.2 Mar 92	Reliability assurance requirements for ESA space systems
ESA PSS-01-40 iss.2 Feb 88	System safety requirements for ESA space systems
ESA PSS-01-60 iss.2 Nov 88	Component selection, procurement and control for ESA space systems
ESA PSS-01-70 iss.4 Jan 94	Material, mechanical part and process selection and quality control for ESA space systems and associated equipment
ESA PSS-01-201 iss.1 Aug 83	Contamination and cleanliness control
ESA PSS-01-202 iss.1 Jun 83	Preservation, storage, handling and transportation of ESA spacecraft hardware
ESA PSS-01-204 iss.1 Sep 88	Particulate contamination control in claen rooms by particle fallout measurements
ESA PSS-01-301 iss.2 Apr 92	Derating requirements applicable to EEE components foe ESA space systems
ESA PSS-01-302	Failure rates for ESA space systems
ESA PSS-01-303 iss.1	Requirements for FMECA

ESA PSS-01-401 iss.1 Mar 89	ESA fracture control requirements
ESA PSS-01-402	Design safety requirements for ESA space systems
ESA PSS-01-403	Hazard analysis requirements and methods
ESA PSS-01-603 iss.2 Jun 85	ESA preferred parts list
ESA PSS-01-605 iss.1 Jul 86	Capability approval programme for hermetic thick film hybrid microcircuits
ESA PSS-01-606 iss.1 Jul 86	Capability approval programme for hermetic thick film hybrid microcircuits
ESA PSS-01-608 iss.2 Apr 87	Generic specification for hybrid microcircuits
ESA PSS-01-700	The technical reporting and procedures fot materials and processes
ESA PSS-01-701	Data for the selection of space materials
ESA PSS-01-702	A thermal vacuum test for the screening of space materials
ESA PSS-01-703 iss.1 Oct 82	The black anodising of aluminium with inorganic dyes
ESA PSS-01-704 iss.1 Oct 82	A thermal cycling test for the screening of space materials and processes
ESA PSS-01-708	The manual soldering of high reliability electrical connections
ESA PSS-01-726	The crimping of high reliability electrical connections
ESA PSS-01-728	The repair and modification of printed circuit boards and solder joints for space use
ESA PSS-01-733	The application of the thermal control paint Pyrolac PSG 120 FD
ESA PSS-05-0	ESA software engineering strandards
ESA/SCC QPL	current issue
ESA-RD:01	Outgassing and thermo-optical data for spacecraft materials
ECSS-E-30-01 draft Nov. 97	Fracture control
ECSS-Q-60A	Space product assurance. E.E.E. components.
MIL-HDBK-217F	Reliability prediction of electronic equipment
MIL-STD-975	Standard parts list for flight and mission essential ground support equipment
FED-STD-209E	Airborne particulate cleanliness classes

9.1.3 P.A. requirements applicability.

9.1.3.1 Approach.

The programmatic constraints of Mars Express are not allowing for the exhaustive application of the general Agency's Product Assurance Requirements (as set out in the ESA-PSS series of documents). But, owing to the complexity and interrelation of the spacecraft and the various payloads :

- the entire European Space Agency's Product Assurance requirements will be generally given as applicable to the payload to spacecraft so-called « critical interfaces »,
- while a restricted number of these are placed by the Spacecraft Contractor on the overall payloads to ensure the success of the programme.

The approach followed for the definition of the applicable requirements has been to trace out in an « applicability matrix » those requirements which are applicable to the « critical interfaces » only, and those applicable to the « overall payload ». Then, this « minimum spacecraft requirements » approach shall begin with :

- first, the definition of the term « interface » shall be properly given and understood by every party
- and second the proper identification - in term of payload internal hardware - of the « interfaces ».

The later shall be proposed by the Principal Investigator in due time in order to permit an examination and mandatory agreement by ESA and the Spacecraft Contractor.

This hardware definition will be in particular the result of an early stage and extensive FMECA (from the P.I.) directed onto the identification of those P/L items which - if defective - may generate a defect or an anomaly on the spacecraft.

In case where the definitions are considered to be unclear, or the proper identification of the critical interface are judged to be not sufficient or too constraining (cf completeness / thoroughness of the FMECA work), either the P.I. or the Spacecraft Contractor may come back to the original and complete application of the P.A. requirements.

Despite this context, safety requirements imposed by the Launcher Authorities are to be applied on all flight hardware and software and hence the relevant requirements are applicable for all payload elements.

9.1.3.2 Definition of the critical interface

In this document we call a « critical interface » (or simply « interface ») any part of either the payload or the spacecraft able to interfer with its counterpart (then of the rest of the spacecraft or the payload, respectively). Then, we will consider that an « interface » may be :

- either the result of a physical contact of a part of the payload with a part of the rest of the spacecraft (examples : surface of contact of an equipment, harness connections, ...)
- or the result of a functional interference characterised by the fact that, if a part of the payload is not properly performing, then a part of the rest of the spacecraft is not performing as well (example : electrical failure propagation, outgassing)

The physical interface between the instrument and the spacecraft is the part of the payload which is physically in contact with a part of the spacecraft.

The critical interface between one instrument and the rest of the spacecraft is defined by the entire set of elements which can affect the spacecraft assembly (ex : the loss of a thermal protection of one instrument can affect the thermal state of the spacecraft).

All these elements can be sorted in classes : mechanical, thermal (conductive and radiative), electrical (power and signal), radiative (electromagnetic), optical (straylight, field of view), contamination (particulate, molecular, biological cleanlinesses). For every class, an inventory of faults must be performed in order to identify all the risks and to insure that each one is cancelled by an appropriate mean.

Mechanical / dynamic critical interface :

All element which contribute to the mounting, fixation, position of the instrument, a sub-assembly, a device or part of them and which can, by its failure or faulty operation, damage another part of the platform or other instrument shall be considered as being a part of the mechanical critical interface between the instrument and the spacecraft.

A failure can occur by :

absence or accidental removal or looseness (blocking rod)

abnormal dimension or feature

unpredicted movement or speed change (release mechanism)

breakage (bracket , baffle)

distortion, wear

operation or misoperation (pyro device)

field of view obstruction by moving part (thermal blanket)

Electrical critical interface :

All element (harness or electronic component) which can be a source of any possible overvoltage or undervoltage, overcurrent or under current consumption (versus nominal design) or any unpredicted variation (even in the nominal ranges of voltage and current) of an electrical signal of the interface circuits, liable to create any degradation to the electrical circuit characteristics or to the operation performance of the platform or any other instrument shall be considered as being part of the electrical critical interface between an instrument and the S/C-E.

Thermal critical interface :

All element which can cause any unexpected temperature or thermal change liable to generate a major disturbance of the thermal balance with degradation of the payload or platform performances (functional or by heating power increase) shall be considered as part of the thermal critical interface.

Radiative electromagnetic critical interface :

All element which can cause any disturbance of the platform or another instruments by electromagnetic effect shall be considered as part of the electromagnetic critical interface. Include optical generated disturbances.

Optical critical interface :

All element which can cause any disturbance of the platform or anther payload by generation, reflexion, absorption, filtration, biasing or modification (straylight) of the optical flux to a sensor, detector or from a source shall be considered as a part of the optical interface.

Contamination critical interface :

All element which can cause any disturbance of the platform or the other instruments by molecular or particulate contamination shall be considered as part of the critical interface. In fact, for this type of risk, the whole instrument (excepted the totally hermetic volumes) shall be considered as critical interface.

9.1.3.3 Applicability matrix.

The applicability of each chapter of the P.A. requirements (« interface » only or « overall ») is given by the right column in the tables 9.1.3.3 / 1 and / 2 hereafter.

Paragraph	title	page	applicability
9.1	INTRODUCTION	1	
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9.1.3	P.A. requirements applicability	3	O
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9.1.3.2	Definition of the critical interface	4	O
9.1.3.3	Applicability matrix	5	O
9.1.4	Implementation	7	O
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9.2.1	General	9	O
9.2.2	Organisation	9	O
9.2.3	Product Assurance Plan	9	O
9.2.4	Contractor and Supplier Surveillance	10	O
9.2.5	Status review, facility review	10	O
9.2.6	ESA and S/C contractor participation in inspections and tests	10	O
9.2.7	Product Assurance progress reporting	11	O
9.2.8	Instrument qualification	11	O
9.3	QUALITY ASSURANCE	12	
9.3.1	General	12	
9.3.2	Procurement control	12	I
9.3.3	Incoming inspections	13	I
9.3.4	Surveillance of manufacturing/integration, MIPs	13	I
9.3.5	Test witnessing. Pre-test/post-test review	14	I
9.3.6	Logbooks and traceability	15	O
9.3.7	Cleanliness and contamination control	15	O
9.3.8	Non-conformance control	16	O
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9.3.8.2	Major / minor NCRs.	17	O
9.3.8.3.	Non-conformance report	18	O
9.3.8.4	ESA and S/C contractor involvement with Major NCR's	19	O
9.3.8.5	NCR close-out	20	O
9.3.9	Metrology and calibration	20	O
9.3.10	Handling, storage, packing, marking and labelling, transportation control	20	O
9.3.11	Alerts	21	O
9.3.12	Software Quality Assurance	21	O

Table 9.1.3.3 / 1 : PA Requirements Applicability Matrix

Notes : the **letter O** means that the requirement is applicable to the **overall payload**,

the **letter I** means that it is applicable only to the **critical interface**

Paragraph	title	page	applicability
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9.4.1	General	22	I
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9.4.3	Part Stress Analysis (PSA)	25	I
9.4.4	Worst case analysis (WCA)	25	I
9.4.5	Critical Life-limited items	25	I
9.4.6	Dependability Critical Items List (CIL)	25	I
9.5	SAFETY ASSURANCE	26	
9.5.1	General	26	O
9.5.2	Applicable requirements	26	O
9.5.3	Safety assurance tasks	27	O
9.6	PLANETARY PROTECTION	28	O
9.7	COMPONENTS QUALITY, SELECTION AND PROCUREMENT	29	I
9.7.1	General	29	I
9.7.2	Component management	29	I
9.7.3	Prohibited materials and components	30	I
9.7.4	Radiation sensitive components	30	I
9.7.5	Components derating, component drift and degradation	30	I
9.7.6	Component approval	31	I
9.7.7	Preferred components, non PPL listed components	31	I
9.7.8	Non-qualified components	32	I
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9.7.10	Component screening and burn-in	33	I
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Table 9.1.3.3 / 2 : PA Requirements Applicability Matrix

Notes : the **letter O** means that the requirement is applicable to the **overall payload**,

the **letter I** means that it is applicable only to the **critical interface**

9.1.4 Implementation.

The applicability will be determined with the following steps :

- At the latest for the Preliminary Design Review the Principal Investigator shall prepare and submit to the Spacecraft Contractor a Failure Modes, Effects and Criticality Analysis (may be combined with a Fault Tree Analysis which is a top-down analysis as opposed to the bottom-up approach in the FMECA) with a proposal of what elements of the instrument are considered to interface or not interface with the spacecraft or other payloads.
- Then, the Spacecraft Contractor will review the analysis and proposal and will determine with the Principal Investigator and ESA on which element of the Payload element only the limited Product Assurance requirements apply.
- Finally, the agreed Product Assurance requirements baseline will be fully implemented by the Principal Investigator on the critical interface elements of the instrument whereas the baseline will be applied as guideline to the maximum possible extent to the other elements with a view on minimizing the possibility of instrument anomalies.

If specific rules or procedures defined in the various sections developed hereafter are not relevant or impracticable or inefficient for implementation on a particular element, the Principal Investigator may propose alternative procedures which will achieve the same objective. These procedures are subject to agreement with the Spacecraft Contractor on a case by case basis. The same policy also applies to the « on the shelf » elements.

9.2 General Product Assurance Requirements and Management

9.2.1 General

The Principal Investigator shall establish and implement an effective product assurance programme for the payload element in support of the product assurance activity for the total MARS EXPRESS programme. It shall provide for the prevention or early detection of actual and potential deficiencies, system incompatibilities, marginal reliabi1ity or quality or any other conditions which could result in unsatisfactory performance.

The main features of this P.A. programme shall be described in a payload P.A plan which will contain the rules to be applied all along the development.

9.2.2 Organisation

A product assurance representative for the instrument shall be designated who will manage the product assurance activities and who will coordinate these activities with other organisational elements and ESA.

Should the Principal Investigator not already have suitable facilities and experienced personnel, such facilities should be provided, and training of personnel be implemented, for processes and operations for which insufficient experience exists, or which are new or specifically critical for the project. The use of National Agency resources, consultants and contractors should be considered for specific tasks for which inhouse expertise is not available and where the investment may not be possible.

9.2.3 Product Assurance Plan

A Product Assurance Plan shall be prepared and submitted to ESA and MMS as part of the P.I.D., Part B. The plan shall provide :

- a description of the Product Assurance responsibilities within the project organisation and outside facilities or personnel which may be used
- a description of how the Product Assurance Programme will be implemented including critical areas pertinent to the instrument such as magnetic or optical cleanliness, deployable items, planetary protection and safety items. Each section of these PA requirements is to be addressed in the Product Assurance Plan or in separate sub-plans.

A thorough planning of the various tasks shall be performed with identification of responsibilities, related inputs and outputs, start and end dates and the necessary manpower effort. The basic planning shall be incorporated into the Product Assurance Plan and the planning shall be updated and provided at least for every major instrument review or if significant changes occur.

9.2.4 Contractor and Supplier Surveillance

Whenever the Principal Investigator uses contractors to supply equipment or services he shall impose on his contractors and suppliers a set of product assurance requirements corresponding to the requirements listed herein and tailored to the criticality of the products or services being provided.

The Principal Investigator. shall carry out an effective survey of the activities carried out by contractors and suppliers. Status/problem reporting shall be included in the regular progress reports.

9.2.5 Status Review, Facility Reviews

A review of the status and the results of the Product Assurance programme shall be included in the project reviews. A programme of Facility Reviews shall be organised and implemented on inside and outside organisations at key points of the programme, such as before the start of manufacturing activities and before the start of qualification or acceptance tests, to review acceptability of material, facilities, tools, equipments, instruments, calibration, services, operational procedures and documentation with applicable requirements. These reviews should not need major manpower support, they should address specific points agreed with the Spacecraft Contractor and be completed usually in less than a day. Follow-up reviews shall be made to verify that required revisions have been implemented effectively. Separate and unplanned reviews shall be carried out on subjects or organisations which are or become subject of particular concern and these will be discussed on a case-by-case basis. ESA shall be invited to participate in critical reviews carried out by the Principal Investigator.

9.2.6 Spacecraft Contractor and ESA Participation in inspections and tests

For the purpose of product assurance and technical coordination the Spacecraft Contractor must have the Principal Investigator agreement to participate in or carry out together with or without ESA participation the audits, surveys, source inspections, test observations or witnessing, mandatory inspection etc., at the facilities of the Principal Investigator. and his contractors and suppliers.

ESA or Spacecraft Contractor participation shall not in any way replace or relieve the Principal Investigator of his responsibility but will be more to identify problem areas and/or provide indication of satisfactory progress.

Arrangements shall be made to permit designated ESA and Spacecraft Contractor personnel free access to all technical and programmatic documentations, areas and operations within the facilities of the Principal Investigator, and his contractors and suppliers in which work related to the MARS EXPRESS programme is being performed.

Proprietary rights of the Principal Investigator. and of third parties will be fully respected.

9.2.7 Product Assurance Progress Reporting

Reporting on the progress and status of the P.A. programme shall be included in the regular project progress reports. As applicable for the various phases of the programme the following items are to be addressed :

- Progress and accomplishments of each major product assurance task.
- Resolved and new problems
- Status of FMECA and Hazard Analyses
- Status of EEE parts programme (lists, approval status, status of qualification programmes as applicable, procurement problems).
- Status of materials and process control programme (lists, approval status, status of qualification programmes as applicable)
- Status list for Deviation and Waiver Requests
- Status list for major non-conformances/failures
- Overview of major events/activities in the forthcoming reporting period.

It is anticipated that this report would be short and follow a standard format.

9.2.8 Instrument qualification.

9.2.8.1 Qualification of "on the shelf" instruments.

The Principal Investigator shall give evidence of the qualification of the instrument versus the MARS EXPRESS requirements.

The Spacecraft Contractor P.A. will ensure that this qualification meet the requirements of MARS EXPRESS for all the aspects of the critical interface as defined in this document.

9.2.8.2 Qualification of newly designed or modified instruments.

The Spacecraft Contractor will ensure that all the instrument dependability and safety requirements, are taken into account by the Principal Investigator along the development of the instrument for the critical interface as defined in this document.

9.3 Quality Assurance

9.3.1 General

For quality assurance requirements, ESA specification PSS-01-20 had been used for the definition of the following summarised PA baseline to be applied for the MARS EXPRESS elements.

These requirements are applicable to:

- flight models and spares
- hardware subjected to or participating in design verification/qualification testing with respect to the properties relevant for those tests.
- portions or the GSE which interface directly with flight hardware or which can have an impact on safety (e.g. testing/documentation for lifting slings).

The quality assurance activities shall be tailored by the contractors according to the complexity of each element and to the needs to assure compliance to formal requirements. They shall be described in sufficient detail in the Q.A. section of the P.A. Plan.

9.3.2 Procurement Controls

For the procurement of equipment, components, parts, materials and services the investigator shall evaluate and select manufacturers, suppliers or contractors who are capable of supplying the items with the required properties and the necessary quality levels. Quality assurance provisions shall be defined in purchase orders and contracts which are adequate to ensure and to verify/document that all requirements of the procurement specification are met. This shall include as appropriate:

- in-process inspections and tests (non-destructive and/or destructive on a sample basis. e. g. strength of bonded joints) and associated records (the purchase documentation must define what records are to be kept by the supplier and what data records are to be delivered with the end item)
    - end item inspections and tests both for qualification and for acceptance purposes with associated records (deliverable documentation is to be defined).
- provision of a Certificate of Conformance in which the supplier formally declares that the item meets all requirements of the applicable specification/contract (with the exception of agreed Requests for Waivers or Deviations).

The Principal Investigator shall also ensure that the supplier applies adequately calibrated instrumentation, and operates a non-conformance control system which eliminates deficient items and assures corrective actions as necessary.

The implementation of these requirements by the supplier shall be checked as appropriate for the criticality of the procured item by surveys and/or audits, witnessing of specifically critical processes, inspections or tests, and review of inspection and test results.

9.3.3 Incoming inspections

Incoming Inspections on the items procured from outside sources shall be performed to check compliance with applicable requirements by one or a combination of the following activities depending on the criticality of specific parameters for the application of the item and the quality assurance provisions already carried out by or with the supplier :

- review of the Certificate of Conformance and of deliverable documentation with inspection/test results.
- visual inspections for completeness and freedom from obvious damage or deficiencies (also check for lifetime of life limited items).
- sample testing or testing on all items for compliance to the most essential parameters (e.g. interface dimensions of a housing).
- inspection/test of all applicable interface and performance parameters (e.g. on a complete mechanism or sensor).

9.3.4 Surveillance of Manufacturing/Integration, Mandatory Inspection Points (MIP's).

Surveillance of manufacturing and assembly activities by the designated quality assurance personnel shall be provided depending on the complexity of the operations and their potential effect on the properties and integrity of the end product.

Manufacturing / assembly flow charts shall be established and mandatory / key inspection points included at places where the maximum visibility of quality of the hardware is given. The purpose of the inspection is to assure compliance to requirements and satisfactory workmanship.

The flow charts shall be provided to ESA and Spacecraft Contractor at the time of the Instrument Preliminary Design Review for identification of proposed points for Mandatory Inspections which will be carried out with ESA participation. The MIP where ESA participatlon is envisaged will be agreed with the Principal Investigator. These will at least include the processing or installation of safety critical items. Besides medium term planning, ESA is to be notified about the exact date for the agreed ESA Mandatory Inspection Point at least one week in advance. Important deficiencies which can indicate degradation of inherent safety or non-compliance to formal requirements or significant schedule impacts and which the instrumenter detects during his own inspections, shall be reported to ESA within 48 hours.

9.3.5 Test witnessing. Pre-Test/Post-Test Review

Critical development tests and formal qualification and acceptance tests shall be monitored or witnessed by quality assurance personnel to ensure that applicable procedures are followed without errors, that adequate records of the activities and test results are taken, and to document any deficiencies and non-conformances which are encountered and to initiate corrective and preventive actions according to the rules given in section 9.3.8.

At least before the start of formal qualification and acceptance tests a Test Readiness Review should be held with attendance of quality assurance personnel to determine the following:

- the as-built configuration status of the test specimen conforms to the released design baseline or potential differences are acceptable and documented.
- status and acceptability of previous non-conformances/ failures, Requests for Waivers/Deviations, open work
- availability and approval status of applicable test procedures
- test facility (e.g. cleanliness) and test equipment (e.g. calibration status checked)
- assignment of responsibilities during the test

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 should be held to determine that :

- all required data records are complete and at least a first assessment has been made to determine whether the parameters were within required limits
- non-conformances/failures have been recorded and at least initial dispositions affecting continuation/completion of the test have been made by the appropriate Material or Failure Review Board
- all deviations from or modifications to the initial test procedure which had to be made during the test were properly authorised
- all portions and steps of the applicable procedure have been completed, the test specimen and test equipment have been brought into a safe condition and the test set up can be dismantled.

The Spacecraft Contractor intends to monitor or to witness at least some of the formal qualification and acceptance tests and to participate in some Test Readiness Reviews and Post Test Reviews. The Spacecraft Contractor shall be notified at least one week in advance of the Test Readiness Review at the start of environmental tests, EMC tests and interface verification tests.

9.3.6 Logbooks and Traceability

Equipment log books shall be established for all operations and tests, starting with the final inspection of the hardware after the manufacturing/assembly phase and they shall include:

- historical record sheets with :
    - dates of operation/test/transport

  * name of operation/test/transport from/to
  * applicable procedure and/or report
  * responsible organisation and signature for entry
  * remarks e.g. on NCR's or unplanned events
- operating time/cycle record for limited life items
- connector mating records for the formally identified as critical connection
- age sensitive items records
- pressure vessel history log
- temporary installations record- open work/deferred work records.

The log books shall accompany the hardware whenever it is placed under the custody of another organisation and this organisation shall update and maintain these records. The log books will form part of the Acceptance Data Packages which are to be delivered for every item at the time of acceptance. (see also sections 9.8).

9.3.7 Cleanliness and Contamination Control

The Principal Investigator shall prepare and implement a plan (part of the PA/QA Plan or separate) covering the criteria and tasks for contamination control :

After establishing the cleanliness criteria for the element to obtain its own cleanliness and to avoid contamination of the other payloads or the spacecraft, the provisions, activities and verification method necessary to achieve that goal must be identified for 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 must be defined (in the Operation Manual) and the Spacecraft Contractor shall be informed so that the necessary arrangements can be made in time.

Potential contamination hazards and causes shall be considered, the necessary provisions included in facilities and procedures and their implementation shall be controlled. They include at least :

- lack of degreasing of raw materials
- residues from cleaning agents, fluxes or machine lubricants
- insufficient curing and bake-out of materials
- handling of flight hardware with bare hands or dirty tools
- inadequate clean room clothing
- inadequate discipline of personnel in clean rooms
- condensation of moisture or contaminants on cold surfaces during tests or transportation
- suitability (no PVC !) and cleanliness of packing and packaging materials
- biological and organic material deposit.

It has to be noticed that the environment encountered during the integration and launch preparation phases of the spacecraft (usually class 100 000, Fed.-Std-209E) does not necessarily conform to the high cleanliness standards which can be achieved in a laboratory where sensitive equipment is assembled. Therefore protection devices, and as far as feasible also provisions for cleaning sensitive areas at later integration phases (if that would prove to be necessary) shall be incorporated in the design. As well, purging equipments and interfaces shall be delivered together with the payload. Generally, all precautions shall be notified in the P/L User Manual.

To this respect, the bake-out in vacuum at elevated temperatures before integration of contamination related items into the instrument shall be considered as an effective method to reduce the risk of contamination in orbit.

9.3.8 Non-conformance Control

The Principal Investigators and their contractors and suppliers shall operate a Non-conformance Control System including all manufacturing, integration and test phases which will provide a disciplined approach to the identification, segregation, reporting, review, disposition, analysis, corrective action, re-verification, and prevention of recurrence of confirmed or suspected non-conformances or failures.

9.3.8.1 Decisions by Materials review Board/Failure review Board.

When a non-conformance or failure is detected during an inspection or test or during any other activity, it shall be recorded in a suitable form and a disposition made with respect to stopping or continuing of ongoing activities, use-as-is, rework to original requirements, scrapping of defective items or other appropriate dispositions to be determined by a Material Review Board (MRB) or a Failure Review Board (FRB). The Material Review Board or Failure Review Board shall consist at least of one representative of the Product Assurance Organisation and one representative of the Engineering Organisation. Specialists may be invited and consulted and in cases defined below, representatives of other organisations may also participate as voting members in the MRB and FRB.

The nomenclature used by various contractors and organisations differs : the MRB and the FRB have basically the same function, however in some places the MRB has only limited decision authority like "scrap" or "rework to standard repair procedure" on "minor non-conformances" whereas the FRB has full decision authority also for "non-standard repairs" and "use as is" on "major non-conformances" or "real" failures. To make it simple, reference will be made in the following text only to "MRB" which acts both on "minor" and on "major" non-conformances.

The MRB shall determine:

- the cause of the discrepancy, as necessary with the help of experts or outside organisations
- the disposition with corrective and preventive actions including :

  « scrap »

  "use as is". If a formal specification requirement remains violated, preparation and acceptance of a Request for Waiver/Deviation (usually for a one-time case only) or a specification change can be recommended. They are both subject to approval by the appropriate "Change Control Board", see Configuration Control procedures.

  « repair » (standard or non-standard methods to be defined)

  « change/modify the design » (Engineering Change Requests are subject to separate approval)

  preventive and corrective actions which may also be necessary for other models or similar items

- re-verification to be performed after repair or modification which may consist of re-inspection, re-test (a late modification may also affect the validity of previous qualifications tests) and updating of previously established design analyses.

The cause of the discrepancy and the dispositions and actions agreed by the MRB are to be documented on the Non-conformance Report or in associated MRB minutes. Quality Assurance personnel shall verify the completion of all actions and re-verification defined by the MRB and when that has been achieved successfully, the NCR can be "closed out" with reference to re-verification reports or updated documents and QA signature on the NCR form.

9.3.8.2 Major/Minor NCR's

The definition of MAJOR and MINOR non-conformances shall be as follows :

MAJOR non-conformances are non-conformances, or failures, which may affect :

- Higher contractor approved design requirements with respect to form, fit, function, performance, materials and safety as specified in applicable design requirement specifications.
- Approved configuration baselines
  - Higher contractor approved test requirements and procedures (which includes formal qualification and acceptance tests with vibration, TV, EMC)
- Higher contractor approved Interface Control Documents.

A MINOR non-conformance is a non-conformance, which does not affect any points given in the definition of the major non-conformance. It is of inconsequential nature as regards the requirements and does not influence fitness for use and safety, or is trivial with regard to workmanship criteria applicable to deliverable items.

For COMPONENT (= Electrical, Electronic, Electromechanical parts = EEE-parts) non-conformances, the following shall apply :

If a lot or batch of high reliability components is rejected during manufacturing, screening or testing at the manufacturers the non-conformance shall be classified as MAJOR if :

- it is proposed to accept the rejected lot as is or to continue processing, rework or testing although the lot does not comply with the specification requirements.

All component non-conformances after delivery from the manufacturer shall be classified as MAJOR except at incoming inspection where the following non-conformances may be classified as minor :

- random failures where no risk related to reliability or quality exists,
- the form, fit or function of the accepted EEE-part is not affected,
- minor inconsistencies in delivered documentation.

All DPA (Destructive Physical Analysis) rejections shall lead to non-conformances which shall be classified as MAJOR.

SOFTWARE non-conformances shall be dispositioned and processed as hardware non-conformances although the nomenclature used may be different. Non-conformances found during formal acceptance testing of flight and checkout software shall be regarded as MAJOR non-conformances.

Non-conformances found during formal acceptance testing of deliverable GSE shall be regarded as MAJOR non-conformances if they cannot be corrected and verified again before the end of the acceptance tests.

9.3.8.3 Non-Conformance Report

Reporting and dispositioning of MAJOR non-conformances shall be as follows :

A notification report shall be sent to the higher contractor by telefax (or equivalent) within two working days after occurrence or observation of the non-conformance with at least the following informations :

- unique NCR number
- identification of the non-conforming item
- date and time of occurence or observation
- inspection or test (name and number of procedure) and test step and detailed environmental and operational condition at which the non-conformance was observed
- description of the non-conformance
- causes of the non-conformance as far as already known
- immediate/intermediate disposition and actions taken or proposed corrective actions and reverification if the cause of the non-conformance could already be sufficiently identified.
- remarks e.g. on potential effects, schedule impact, required changes, closeout if applicable.

The full non-conformance report shall in addition include :

- final failure analysis and identification of the cause of the non-conformance
- all actions to be taken for correction of the non-conformance and prevention of similar occurences including necessary document changes or recommendation to apply for RFW (Request for Waiver or Deviation) if necessary
- all reinspection and retesting to be performed.

The NCR shall be submitted to the Spacecraft Contractor within two working days after it was established by the appropriate local MRB. Important results from intermediate steps of failure investigations and follow-on MRB's shall be reported to the Spacecraft Contractor as they become available.

Upon request by the Spacecraft Contractor, notifications and other documentation on major NCR's shall be sent in parallel to other parties on the MARS EXPRESS programme, e. g. the Prime Contractor.

The contents of the MINOR non-conformance reports shall be the same as for MAJOR non-conformance reports. They shall be dispositioned by local MRB and kept under Q.A. control. Minor NCR's shall be made available to the Spacecraft Contractor for review as requested, e.g. at the times of Mandatory Inspections, Test Readiness Reviews or Acceptance Reviews.

9.3.8.4 ESA and Spacecraft Contractor involvement with Major NCR's

Within three working days after receipt of a notification of a major non-conformance the Spacecraft Contractor will respond to the initiator if further information, evaluation, or analysis is required, if the Spacecraft Contractor disagrees with the proposed corrective actions and reverification, or if the Spacecraft Contractor wants to participate in follow-on Material Review Boards (MRB) either in meetings to be arranged or by further telecommunication. In urgent cases a faster Spacecraft Contractor response may be requested.

The Spacecraft Contractor reserves the right to participate as voting member on MRB's for major non-conformances and to invite inside and outside experts to participate in failure analysis and MRB meetings.

9.3.8.5 NCR Close-out

When all required corrective actions and reverifications have been completed successfully, the close-out shall be recorded (wlih reference to re-test report etc.) and certified by Q.A. on the NCR and a copy supplied to the Spacecraft Contractor.

9.3.9 Metrology and Calibration

Calibrated instruments shall be used at least for all parameters of the instruments which are to be verified against interfaces with the spacecraft or associated items.

A programme shall be established which ensures that calibrated instrumentation with the accuracy, stability and range appropriate to the intended application will be available when needed in the various phases of manufacturing, integration and tests.

Calibration of the instruments shall be traceable to national standards. Re-calibration shall be performed at intervals on the basis of the stability, purpose and use of the instrument calibration labels attached to the instruments shall indicate the last and next date of calibration and they shall allow traceability to the applicable calibration records.

The total error resulting from a calibration and measurement process attributable to the instrument, personnel, procedures and environment, shall not exceed a significant amount (e.g. 10%) of the tolerance for the parameter to be measured. Where practical limitations do not allow measurement with the required accuracy, an estimate of the cumulative calibration and measurement error has to be provided.

9.3.10 Handling, Storage, Packing, Marking and Labelling, Transportation Control

Procedures and instructions shall be made available and be used for handling, storage, packaging and transportation which ensure that the integrity of the item and tolerable environmental conditions including cleanliness, humidity, pressure, temperature, radiation, vibration and shock will be maintained to prevent deterioration and damage. Effective implementation of applicable procedures and instructions shall be verified by the qua1ity assurance activity.

Appropriate marking and labelling for packing, storage, and transport shall be applied including the following :

- nomenclature of the item and serial number
- cleanliness level and decals or labels to permit ready detection of loss of packaging integrity or exceeding of environmental limits which could have deteriorated the item
- applicable caution/warning notes for handling transportation and unpacking.
- applicable caution/warning notes for dangerous or toxic contents
- life expiration dates
- package orientation arrows, weight and centre of gravity, handling and lifting points
- conditions and instructions for handling and unpacking
- name, address, phone number of sender and recipient for transport and shipment and shipping documents. Compliance to the requirements shall be verified by the quality assurance organisation responsible before and after transport or shipment.

9.3.11 Alerts

On 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), ESA and NASA are operating an alert system. These alerts will be screened when they are received by the ESA for a first assessment or potential applicability to MARS EXPRESS. If it is suspected or if it cannot be excluded that an instrument may be affected, the alert will be forwarded to the Contractor with a request to evaluate the alert and to take corrective actions as necessary to assure that the reported problem is avoided or eliminated on the element. Within 15 days after receipt of a formal alert a response shall be provided to the ESA Project Office either indicating its non-applicability or the appropriate actions (to be) taken.

9.3.12 Software Quality Assurance

For software (flight and test/checkout software) interfacing with the spacecraft system, the investigator shall prepare and implement a quality assurance programme including the following :

The responsibilities for software development and verification and the relationship to other organisational elements shall be clearly defined.

Software standards and specifications shall be checked to assure completeness of performance - and interface requirements and of all operational and environmental constraints. Software verification shall be carried out including reviews, audits and formal acceptance testing in which compliance to all applicable requirements shall be demonstrated.

The principal Investigator shall insure that the payload software is developed according to the ESA Standard PSS-05-0. He shall insure that the critical TC (category 1 & 2 according to the paragraph 9.2.2) are identified and that the relevant rules about instrument commandability are implemented in the software design and development.

Configuration control shall be exercised on requirements-specifications, design documentation, source listings and test -plans, -procedures and -reports and it shall include labelling and version control of software carriers.

A non-conformance control system shall be implemented equivalent to the one defined in section 9.3.8.

Documentation to be supplied with the software for acceptance shall include Configuration Status Lists, Design Descriptions, Test Reports, Software Manuals, Waiver Requests, Non-Conformance Reports and Identification of open work, temporary modifications and open tests.

9.4 Dependability Assurance

9.4.1 Genera1

This section is based on ESA specifications PSS-01-30, PSS-01-301, PSS-01-302 and PSS-01-303 which are tailored here and concentrated on dependability assurance requirements applicable to interfaces which the payloads have with other elements of the spacecraft.

The scope and purpose of the dependability assurance activities are to verify compliance of the design with the fault tolerance requirements, fault isolation requirements and fault detection requirements which are defined in the technical requirements and dependability specifications for MARS EXPRESS.

To achieve these goals, thorough Failure Modes Effects and Criticality Analysis, Part Stress Analysis and Worst Case Analysis (which may be complemented by Fault Tree Analyses) shall be performed on all elements of the payload which have interface with other MARS EXPRESS elements.

In order to assure that dependability is designed and built into a system it is imperative that systematic analyses are prepared already in the conceptual design phase and in the design and development phase of the programme in close cooperation with engineering and safety assurance functions and that resulting conclusions and recommendations are implemented if feasible.

The relevant section of the PA Plan shall describe how the preparation of FMECA will be linked and coordinated with ongoing engineering and safety activities. The various steps for initiation, updating and finalising the dependability analyses shall be identified in the plan.

9.4.2 Failure Modes Effects and Criticality Analyses (FMECA)

Comprehensive Failure Modes, Effects and Criticality Analyses shall be prepared on all critical interfaces of the instrument (but excluding structural elements whose integrity will be assessed with stress analyses and fracture mechanics analysis as necessary).

Interfacing elements of instrument GSE delivered for spacecraft AIV shall also be evaluated to demonstrate that no single failure within the GSE can propagate to MARS EXPRESS spacecraft.

A FMECA shall be carried out for all operational modes of the instrument during orbital phases, launch phase and also for ground testing if not yet covered by analyses of the other phases.

Design specifications, design descriptions, functional diagrams and descriptions, drawings, circuit diagrams, reliability diagrams and associated analyses which are used for the preparation and the understanding of the FMECA shall be attached or referenced including the applicable issue of those documents.

The logical sequence of the FMECA task shall include the following steps :

- identify the item under consideration and its function
- identify the assumed failure modes for that item

- analyse and describe, the qualitative and quantitative effects of each assumed failure mode from item level to instrument level
- identify observable symptoms for each assumed failure mode taking into account the relevant automatic hardware protections (EDAC, current limiter...) and software associated actions (triggering of parameters monitoring, algorithm...)
- establish for each failure mode what provisions are inherent in the design :
  - ∗ to compensate the effect of the malfunction (e.g. switching to redundant unit, automatically or by ground telecommand)
  - ∗ to isolate the fault or to switch to contingency operational modes.
- identify for each failure mode the criticality category of the failure effect according to the following definition :

∗ Category la :	the failure could result in a catastrophic hazard (see definition in Safety Assurance section),
∗ Category 1b :	the failure could result in a critical hazard (see definition in Safety Assurance section),
∗ Category 1c :	the failure could result in failure propagation to spacecraft or other instruments
∗ Category 2 :	loss, degradation or suspension of operational capability for related instrument;
∗ Category 3 :	no or minor effect

  The following suffixes shall be added to the criticality number :

  - ∗ R : if the design contains redundant hardware which can perform the same function
  - ∗ D : if the failure results in anomalous thermal dissipation
- provide for each failure mode remarks and recommendations if applicable or necessary or desirable modifications for the design or operations (e.g. elimination of single point failures).

The FMECA shall be performed on the basis of the lowest level of design definition which is available at the successive steps of the design and development process (e.g. for an electrical critical interface : functional failure modes when no actual design is available, component failure modes later on).

The following functional failure modes shall be considered in the FMECA as a minimum :

- ∗ premature operation
- ∗ failure to operate at a prescribed time
- ∗ failure to cease operation at a prescribed time
- ∗ failure during operation
- ∗ degradation or out of tolerance operation

For analysis at component level, failure modes depicted in PSS-01-302 and functional impact of SEU on sensitive parts shall be considered.

Updated/revised FMECA shall be submitted in the instrument data packages as stated in section 10.10.

Further details for the preparation of FMECA can be found in ESA specification PSS-01-303.

9.4.3 Part Stress Analysis (PSA)

Electrical, Electronic and Electro-mechanical (EEE) parts used in instrument circuits interfacing with spacecraft elements shall be subjected to a stress analysis to assess conformance with the derating requirements of PSS-01-301.

The analysis shall be performed with the highest steady state and transient stress levels resulting from the specified operational and environmental requirements on the assembly.

Updated/revised PSA shall be submitted in the instrument data packages as stated in section 10.10.

9.4.4 Worst Case Analysis (WCA)

Electrical, Electronic and Electro-mechanical (EEE) parts used in instrument circuits interfacing with spacecraft elements shall be subjected to a worst case analysis to show that these circuits keep their specified performance during the intended lifetime even under the most unfavourable conditions which might exist. The conditions to be considered shall be the following :

- Initial tolerances of component parameters
- Drift of component parameters due to ageing
- Drift of component parameters due to cosmic radiation
- Drift of component parameters due to varying temperatures
- Varying electrical interface conditions, e.g. extreme values of supply voltages or loads
- Single event phenomena

Parameter variations of electronic components which shall be taken into account in the analyses are defined in PSS-01-301. Other values have to be substantiated with support from test data (e. g. end of long-term life test limits from qualification tests).

Updated/revised WCA shall be submitted in the instrument data packages as stated in section 10.10 (it may be included as an identifiable section in other design analyses).

9.4.5 Critical Life-limited Items

For all instrument life-limited items whose failure might affect spacecraft performances, the lifetime qualification status shall be provided to the spacecraft contractor.

9.4.6 Dependability Critical Items List (CIL)

All critical items identified through the various Dependability analyses shall be incorporated into a CIL and subjected to periodic management and control.

This list shall cover :

- FMECA criticality 1a, 1b & 1c items
- parts not meeting the derating requirements
- functions not meeting their performances in worst case conditions
- critical limited-life items

A Rationale for Retention shall be given for each Critical Item describing the constraints which prevent its elimination or prevention, the reasons why the associated risk could be regarded to be acceptable and the means which are or will be applied to reduce the failure probability (e.g. by special component selection and screening programme).

This rationale shall be subject to spacecraft contractor & customer approval.

9.5 Safety Assurance

9.5.1 General

A safety assurance programme shall be implemented to assure compliance to specified safety requirements and to identify potential hazards, eliminate them or reduce them to acceptable levels.

This shall cover the design, fabrication, testing, transportation, ground operations, launch and orbital operations.

The Principal Investigator shall identify the responsibility in his team and identify a contact person for safety related aspects.

Description and planning of the activities shall be included in the Product Assurance Plan.

9.5.2 Applicable Requirements

The design of the instrument and its associated GSE and their operation shall conform to :

- the national safety standards and regulations in the country of origin
- the TBD selected launchers range safety regulations
- PSS-01-40

In particular, the design of the instrument and its associated GSE and their operation shall conform to the two following failure tolerance requirements :

- No combination of 2 failures, 2 operator errors, 1 failure + 1 operator error shall have catastrophic consequences (loss of life, life threatening or permanently disabling injury or occupational illness)
- No single failure or operator error shall have critical consequences (temporarily disabling but not life threatening injury, temporary occupational illness, loss of or major damage to flight systems or ground facilities, short term detrimental environmental effects)

As these requirements have been established to cover different and complete launcher payloads, they are naturally very comprehensive. Instruments form part of the launcher payload and therefore these requirements must be made applicable. However only portions of them will really affect the design and operation of the instrument and the tasks to be performed by the Principal Investigator form only a part of the total safety assurance programme for the system.

The Spacecraft Contractor will act as "Payload Authority" for the launcher and 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 Spacecraft Contractor will identify and control the detailed safety requirements to be met by the payload.

This will be implemented by an iterative process of :

- identifying potential hazards inherent in the design and operations of an instrument and checking the applicability of established requirements and regulations

- eliminating potential hazards or reducing them to acceptable levels by definition and implementation of requirements on the design and operations and verification by analyses, inspections or tests.

The structural design of the payloads shall be based on the "fail safe" principle.

A reduced Fracture Control Programme shall be implemented per ECSS-E-30-01 and the associated Fracture Control Plan shall be submitted for spacecraft contractor approval.

Structural bolts and fasteners shall be procured to national standards and shall be subject to NDI. Fasteners shall comply with the requirements of PSS-01-746.

It is pointed out that fracture control will also require specific efforts for quality assurance on procured raw materials and mechanical parts and on process- and inspection - procedures. The corresponding tasks for procurement and process controls are addressed in more detail in Section 9.7.

The safety requirements imposed on the by the relevant Launcher Authorities are not yet finalised. The above requirements are based on what is currently known and may be subject to later revision even on a case by case basis.

9.5.3 Safety Assurance Tasks

As a first step, the investigator shall prepare and submit a Preliminary Hazard Analysis (supported by FMECA, see paragraph 9.4.2) and an applicability matrix for the safety requirements relevant for his element.

As far as possible the Principal Investigator shall identify also the safeguards he intends to implement into his design and operational procedures to reduce potential hazards and also the verification methods (analyses, inspections, tests) which he will apply to assure compliance to the requirements.

In response to the data submitted by the Principal Investigator, the Spacecraft Contractor will, together with Launcher Authorities, investigate the system effects and will subsequently define more specific implementation and verification requirements for each instrument (e. g. definition of inhibits for a pyrotechnic function which must be implemented on the payload side and those which will be implemented on the spacecraft side). Safety related test and operating procedures which shall be submitted (at the appropriate time) for formal approval by ESA will be identified.

The hazard analysis and the definition of verification methods for all safety design requirements and operational procedures shall be completed as the programme activities progress. Updated versions of the hazard analysis, compliance/verification documents, and Residual Hazard Sheets shall be provided.

All safety verification activities to be performed by analyses or tests for design qualification shall be finalised and the results provided for review.

Completion of verification activities (acceptance testing, inspections, certifications) on deliverable items and acceptability of residual hazards will be subject to formal review during the Acceptance Review before delivery.

The Principal Investigator shall support, as needed, the preparation of data packages and safety review meetings at spacecraft and spacecraft/launcher level and for integration and launch operations.

The Spacecraft Contractor will define the detailed procedures and forms to be used for safety analyses, Residual Hazard Sheets, and safety data submissions.

9.6 Planetary protection assurance

Because of the optional presence of any lander, and in particular of any exo-biological one, no specific measure will be taken to reduce - at payload level - the biological load.

Anyway, and still depending of the overall ESA approach, it could be proposed (TBC) to measure the effective biological load at payload AIT level in using samples chosen in the most suitable payload material (example : carbon materials as the most sensitive).

9.7 Component Quality, Selection and Procurement

9.7.1 General

The parts quality assurance contributes essentially to the overall chance of success for the spacecraft mission and therefore suitable parts shall be used (use of high reliability parts is not the general rule / TBC) at least for those circuits of the payloads which interface with other elements of the spacecraft.

The quality level of these parts shall be as defined by the levels of screening, lot acceptance testing and evaluation/qualification specified in sections 9.7.7 to 9.7.11 below.

This applies to flight standard hardware and to components coming into direct contact with flight standard hardware like 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.

The Principal Investigator is encouraged to apply high reliability parts also in those sections of his instrument (flight model) which do not interface with other parts of the spacecraft. However, the quality level should at least correspond to good military standards or manufacturer with in-house high reliability grades.

9.7.2 Component Management

As deficiencies in identifying the needed components, potentially long delivery times, and the necessary preparation 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.

The Principal Investigator shall define the responsibility for the component engineering and procurement activities within his team and he shall nominate a contact person for coordination with outside organisations (e.g. Spacecraft Contractor and ESA).

A section of the Product Assurance Plan shall describe how the component programme will be carried out with identification of tasks which will be carried out by team members of the Principal Investigator, or by procurement agents, test houses or consultants as applicable.

9.7.3 Prohibited Materials and Components

Components containing materials which may constitute a safety hazard or can cause contamination are prohibited from being used without prior approval by ESA. Examples are components containing

a - Beryllium-oxide

b - Cadmium

c - Zinc

d - Mercury

e - Radioactive materials

f - PVC

Special precautions may be required if such materials are used, e.g. a warning label on a box if Beryllium-oxyde washers are inside).

Use to components with known instability shall be avoided unless specifically approved. Examples of unstable components are :

a - Wet electrolytic capacitors (except CLR 79 type)

b - Plastic encapsulated semiconductors

c - Hollow core resistors

d - Variable resistors

The requirements of this paragraph apply to the whole of the instrument, not only to critical interface circuits.

9.7.4 Radiation sensitive components

The radiation environment expected for MARS EXPRESS leads to select components with a sufficient radiation proven hardness (the total mission dose is evaluated to 6 kRads, assuming a 2 mm thickness aluminium structure).

The components are considered suitable if they have been successfully tested to 12 kRads. The other components shall be either submitted to a specific Radiation Verification Test or demonstration shall be given that their environment (S/C and instrument structure, dedicated protection) will procure them with a margin factor of at least 2 with respect to their proven hardness.

The components shall be latch-up immune (LETth > 80 $Mev.cm^2/mg$) or, if not, will be accepted if they are fully characterised and if any propagation to the platform is avoided. A same rule (or demonstration) applies for the Single Event Upset.

9.7.5 Component derating, component drift and degradation

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 derated values specified in PSS-01-301.

Drift and degradation of performance parameters (e. g. increase of leakage currents of diodes) as specified in PSS-01-301 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 sections 9.4.3 and 9.4.4.

It is strongly recommended to apply these rules to all circuits of the instrument.

9.7.6 Component approval

Components used in flight standard hardware in the critical interface circuits of a payload as defined above are subject to the Spacecraft Contractor approval following the procedures as outlined below and in paragraph 9.7.12 and 9.7.13.

Component types will be approved by ESA if at least one of the following criteria applies :

a) they have been qualified according to the requirements of the applicable SCC specification or to equivalent requirements.

b) they have successfully passed the component evaluation and approval programme as outlined in para 9.7.8 below.

c) they have received circuit type approval as outlined in ESA PSS-01-608 (for hybrid integrated circuits).

The 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 (see para 9.7.12). 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.

Approvals can be given by signature on the Part Approval Document (PAD) (see para 9.7.12) or by letter, telex, or in meetings with signed minutes, if sufficiently detailed information is provided on the Declared Components List (DCL) (see para 9.7.13).

An approval reference should be entered on the DCL to maintain traceability of ongoing work.

9.7.7 Preferred components, non PPL listed component

The ESA Preferred Parts List (PPL) ESA PSS-01-603 and the ESA/SCC Qualified Parts List and the GSFC-Preferred Parts List and MIL-STD-975 shall be used as the primary basis for component selection.

The Principal Investigator shall prepare and submit at the latest at the Instrument Preliminary Design Review a first issue of the Components List for his element. 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.

For each component selected which is not listed in one of the ESA- or NASA-PPL/QPL's a detailed justification and supporting information shall be provided on a Part Approval Document.

The selection of Non-PPL-listed components shall be based on the knowledge regarding technical performance, qualification status or history of previous usage in similar applications. Preference shall be given to components from sources which would necessitate the least evaluation/qualification effort.

9.7.8 Non-qualified Components

In case a valid and acceptable qualification cannot be demonstrated, a component evaluation and qualification test programme shall be implemented. This programme shall cover the following elements :

- Design and application 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 organisation, 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 PSS-01-60 and ECSS-Q-60A.

Experienced consultants or procurement agents may have to be used by the investigator to perform these tasks.

If applicable, the evaluation/qualification programme and the test results for a specific component to be qualified for use on MARS EXPRESS shall be provided with the Part Approval Document (see para 9.8.12) which is subject to ESA approval.

9.7.9 Hybrid Circuits

Hermetic hybrid circuits shall be procured according to PSS-01-608 class C from manufacturers whose capability has been approved per ESA-PSS-01-605 for thin film and per PSS-01-606 for thick film or according to the MIL.PRF 38534 class H. For U.S. parts procurement GSFC specification S-311-200 is regarded to be equivalent.

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 PSS-01-606 or PSS-01-605. All add-on components shall be selected as defined herein and shall meet the requirements of PSS-01-608.

All hybrid circuits with all add-on components shall be entered on Part Approval Documents (PAD).

9.7.10 Component screening and burn-in

The screening of components for the critical interfacing redunded elements of the instrument shall be at least level C as defined in the ESA/SCC specification system for active, passive components and for relays, crystals, inductors. Generally, when no redundancy is provided, the level B is required.

Grade 2 parts as defined in GSFC Preferred Parts List (PPL) and MIL-STD-975, are regarded to be equivalent to level C as defined in the ESA/SCC specification system, as MIL.S.19500 JANTXV for transistors and diodes.

9.7.11 Lot acceptance testing (LAT)

Lot Acceptance Testing shall be carried out as defined in the ESA/SCC-specifications with the following levels :

a) LAT l : if LAT 1 has not been carried out within the previous 24 months then LAT 1 shall be performed

b) LAT 2 : if neither LAT 1 nor LAT 2 has been carried out within the previous 12 months then LAT 2 shall be performed

c) LAT 3: shall be carried out for all cases not included within (a) or (b) above.

Equivalent lot acceptance testing shall be applied for components selected from GSFC-PPL or MIL-STD-975.

9.7.12 Part Approval Document

A Part Approval Document (PAD) shall be prepared and submitted for approval for all parts which are not straightforward from the applicable PPL's and QPL's and for which the complete information on manufacturer, specification, screening and LAT levels cannot be entered on the Declared Components List.

The PAD shall include :

- Non-repetitive PAD number (with revision if needed)

- Identification of instrument/instrument unit for which the part will be applied and numbers used per flight model
- Part number(s), type, family (plus commercial equivalent) Generic specification, detail specification and amendments if applicable (with revisions)
- Proposed manufacturer and back-up if available
- Justification for selection of non-preferred/non-qualified part if applicable
- Radiation hardness data
- Present qualification status (with reference)
- Results of preliminary evaluation
- Proposed delta qualification or full evaluation/qualification programme, if applicable. Test results are to be provided when available.
- Applied screening level
- SEM/Precap inspections if applicable
- LAT levels
- Destructive Physical Analysis.
- Signatures of requesting party and approval signatures

A minimum of 15 working days shall be included in the schedules to allow for the ESA review of the P.A.D.

9.7.13 Declared Components Lists (DCL)

All components to be used on flight standard hardware, shall be listed in a Declared Component List which is to be completed stepwise as the selection of components and the approval process progresses. The final version must be available at the time of the Instrument Critical Design Review. Formal issues are to be submitted to every Design Review, whereby the list submitted for the Preliminary Design Review can be regarded to represent the first choice of components which is subject to further efforts on standardisation and coordination.

The DCL shall identify the instrument/instrument unit and the design status to which it is applicable. The parts shall be grouped according to the families identified in the PPL and the list shall contain the following entries for each part :

- Sequential line item number

- Component/part number

- Equivalent/similar commercial or military type

- Generic and detail specification number with revision index and with amendments if applicable.

- Manufacturer (and back-up if applicable)

- Screening level

- LAT level

- PAD number or approval reference.

Parts applied in the payload sections interfacing with other spacecraft elements may be marked or listed separately.

9.7.14 Manufacturer Surveillance

Manufacturer surveillance shall be carried out as necessary with audits or participation at critical processing/inspection steps, e.g. with customer (or procurement agent) participation in precap visual inspections or witnessing of some acceptance tests.

9.7.15 Receiving Inspections and Destructive Physical Analysis (DPA)

Receiving inspection of all components shall be carried out by the user or a procurement agent who is independent of the manufacturer and this shall include :

- Review of the Manufacturer delivered documentation
- External visual inspection
- Electrical measurement of critical parameters (if practicable)
- Destructive physical analysis
- Magnetic screening

Destructive physical analysis shall be carried by the user or his procurement agent or an independant laboratory on three samples from each date code of the component types listed below :

- discrete semiconductors
- integrated circuits
- filters
- ceramic capacitors
- relays
- crystals
- hybrids
- switches
- high voltage components
- high frequency components
- opto-electronic components

DPA is not required on components with a valid ESA/SCC qualification. Grouping of DPA for certain families of components (e.g. logic families, capacitor range within one housing) can be considered. For specifically expensive items the number of DPA samples may be reduced upon agreement with ESA.

Three additional samples of each date/lot code shall be procured and supplied to ESA for DPA if so requested with the approval of the part on the PAD or the DCL.

9.8 Material and Process Selection and Control

9.8.1 Basic Requirements

Materials and Process Controls are to be implemented for the payloads at least with respect to hazardous and forbidden materials. outgassing, strength and stress corrosion resistance on structural items including bracketery and on pressurised items.

The following requirements have been derived from ESA specifications PSS-01-70 and PSS-01-700. ESA's Involvement in the selection and control of materials and processes will be aimed at those which could affect the performance of other instruments or spacecraft systems.

Materials and processes for which positive and negative experience has been gathered in previous space projects are listed in ESA PSS-01-701 and JSC 09604. These lists can be used for guidance but suitability for use on the MARS EXPRESS programme is to be evaluated for each application.

Materials shall have a low outgassing rate with TML < 1% and VCM < 0.1% if tested per specification ESA PSS-01-702. ASTM-E-595 and JSC/SPR-0022A can be regarded to be equivalent to PSS-01-702. Documents ESA and NASA JSC 20810 contain data from previous outgassing tests.

Materials or material combinations subjected to thermal cycling in orbit shall be assessed to determine their suitability for the intended application. ESA standard thermal cycling test is described in specification ESA PSS-01-704.

Materials which will be exposed to particle/UV radiation shall be assessed to determine their ability to withstand the type and degree of radiation dosage expected during the mission. Tests to simulate the effects of the radiation environment are described in specification ESA PSS-01-706.

Materials which are sensitive to stress corrosion and which are exposed to long term external (including assembly stresses) or residual internal (frequently present in welded constructions) tensile stresses in the terrestrial atmosphere shall not be used. This requirement shall also apply to GSE lifting devices for loads higher than 30 kg. For the listing of SCC sensitive materials MSFC-SPEC-522B can be regarded to be equivalent to ESA PSS-01-736 and for SCC testing ASTM G44-75 equivalent to ESA PSS-01-737.

As determined by the fracture control programme (see the safety-section) the crack growth properties and initial crack sizes must be controlled for materials in critical structura1 appl1cations.

9.8.2 Materials Control

Material procurement documents and specifications shall include all the technical requirements and adequate quality assurance requirements like inspections, batch-tests, qualification requirements if applicable, verification of chemical composition etc. to assure delivery of a product with properties as needed for the intended application. Procurement specifications shall be made available upon request to ESA for review; proprietary rights will be respected. Receiving inspections and lot/batch testing before use and in parallel to processing of materials in the production flow of assemblies shall be carried out to the degree necessary to assure that variable and significant characteristics are within required limits. Lot/batch acceptance test reports shall be kept at the instrumenter's or contractor's plant for at least 5 years, together with other historical manufacturing/production records for the assemblies. Non-conformances on materials and processes shall be recorded and treated as specified in para 9.3.8 of this document.

The investigators shall be responsible for 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.

Mechanical parts like ball bearings, bolts, nuts, inserts, rivets, springs and actuators (if not covered by equipment specifications) shall be included in one section of the materials list or listed separately. Mechanical parts (for bolts/nuts at least for size M4 and larger) are to be covered by procurement specifications including all technical requirements and adequate quality assurance provisions. These specifications shall be made available to ESA upon request.

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. The name of the source/manufacturer shall be entered in the list together with the name of a back-up source for critical procurements.

Material Design Allowables used in mechanical stress analyses shall correspond to ‘’A-values ‘’ as defined in MIL-HDBK-5 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.

9.8.3 Process Control

Process procedures shall include sufficient inspections and controls during and at the end of the processing steps to assure that the characteristics of the product are within the required limits. In the course of previous space projects ESA has prepared a number of detailed process specifications in the framework of the PSS-01-7XX series, which include for instance :

- PSS-01-703 for black anodising of aluminium
- PSS-01-708 for soldering of high-reliability connections
- PSS-01-726 for crimping of high-reliability connections
- PSS-01-728 for repair and modifications on PC-boards
- PSS-01-733 for application of thermal paint PSG120FG

It is not intended to make these specifications formally applicable to the MARS EXPRESS payloads (if no interface parameters are involved). However, they should be used by the Principal Investigators as support-specifications to check the contents and completeness of procedures applied for the manufacturing and assembly of the instruments. Process procedures shall be made available or accessible to ESA upon request (e.g. during the Instrument Critical Design Review) for review that all processing steps are adequately specified and that adequate controls are included.

9.8.4 Materials and Process Identification and approval

At the beginning of the programme, the investigator shall identify all materials and processes which need to be evaluated and qualified before they can be applied for the production of flight standard hardware. The corresponding planning shall be refined and updated as necessary to assure timely completion of these activities. Process Identification Sheets/Process Approval Requests for processes requiring evaluation/qualification and for the processes which can have an effect on the structural integrity of the instrument (like bonding of inserts into honeycomb panels) shall be submitted identifying all materials involved (also cleaning agents), all major processing steps, inspections and tests during and at the end of the process flow, and the detailed test programme for evaluation/qualification or the procedure and qualification plan are to be attached. At the end of the evaluation/qualification programme, a report shall be submitted to ESA upon which the approval of the process will be based.

The Principal Investigators shall submit for approval Declared Materials Lists (DCL) and Declared Process List (DPL) which contain all materials and processes used on an instrument. Items applied by co-contractors should be integrated into a consolidated list.

The Declared Materials List shall contains :

- sequential non-repetitive item number
- commercial identification/name/number of the material
- chemical nature, type of product, form and condition
- procurement specification, manufacturer
- summary of processing parameters (e.g. mix and cure)
- application and location
- size/weight code
- test/approval references, remarks

The Declared Process List shall contain :

- sequential non-repetitive item number
- identification/name of the process
- specification/procedure number (and issue) description/identification
- use and location

- associated DML items
- approval status/references, remarks

Lists for a suitable grouping of materials and processes can be found in the Annex of PSS-01-700.

A first issue of the Materials and Processes Lists is to be submitted in the conceptual design phase for ESA comments and guidance on the selection of suitable and replacement of unacceptable materials and processes.

Declared Material and Process Lists shall be updated to reflect the degree of definition of the design in the ongoing phases of the programme and revisions shall be provided for each of the project design reviews. The review/approval activities and all necessary evaluation/qualification programmes for materials and processes shall be scheduled such that they will be finalised at the Instrument Critical Design Review (start of manufacturing of qualified flight hardware).

9.9 Configuration Control

9.9.1 Configuration Control Procedure

The Principal Investigator shall set up a Configuration Control procedure in such a manner that the status of all aspects of his instrument such as the design and manufacturing of hardware and development of software can be unambiguously defined and documented at any time.

The control procedure shall allow :

- to identify the configuration baseline definition and requirements for the instrument through a set of approved specifications, interface documents, applicable documents, manufacturing and test process description together with associated data as relevant, and,
- to control and document any change to the configuration baseline definition and requirements which are put under configuration control,

so that ESA and/or the Spacecraft Contractor could conduct a configuration audit at any point in the programme in order to obtain the up-to-date status of the instrument or a part thereof.

The Principal Investigator shall ensure that all affected organisations and parties are cognizant of applicable baselines and of the impact of changes and that they will participate as necessary in the decision making process on baselines which are to be agreed and any changes thereto.

The Product Assurance organisation shall be responsible of the correct implementation of the configuration management system and shall ensure its reliable operation.

The Principal Investigator shall also impose configuration management and control requirements on contractors and suppliers as appropriate for the items being provided to the instrument and compliance to these requirements shall be monitored by the P.A. organisation.

9.9.2 Configuration Items Data List

Configuration Item Data Lists (CIDL) listing all the documents and their applicable issues and revisions which define the configuration baseline shall be established and maintained.

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.

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.

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10. MANAGEMENT AND SCHEDULE REQUIREMENTS

10.1 Organisation, Responsibilities, and Formal Management

10.1.1. ESA

Mars Express is an element of the revised ESA Horizon 2000's plan for space science missions which was approved in principal, subject to confirmation of funding, in June 1997. The plan introduces the concept of smaller more quickly implemented missions. The spacecraft, an Orbiter carrying remote sensing instrumentation, is planned for a launch during the Mars flight opportunity in May/June 2003.

Mars Express will be operated as a Principal Investigator (PI) type mission. The Orbiter spacecraft is developed, operated and fully funded by ESA, with the exception of the scientific instruments.

ESA shall be responsible for :

- the selection of the payload instruments in consultation with the Spacecraft Contractor,
- the payload instruments scientific performance,
- the procurement of the Orbiter spacecraft,
- the mission analysis,
- the launcher interface,
- the launch preparation and procurement,
- the mission and science operations.

10.1.2. Principal Investigator

Each Orbiter investigation shall be led by one Principal Investigator (PI), the Instrument Team Members participate as Co-Investigators. The Principal Investigator shall act as the single point formal interface for the experiment. He shall have the full responsibility (i) vis-à-vis ESA for all scientific aspects of the investigation through launch and subsequent in-orbit operations including scientific data analysis and archiving, and (ii) vis-à-vis the Spacecraft Contractor for all technical interfaces, day to day progress monitoring verification of performance (excluding scientific performance under ESA authority) and timely delivery of the instrument.

It is the overall responsibility of the PI to ensure that the complete experiment is implemented and executed in a manner that the science objectives are achieved within the mission constraints of the Mars Express programme.

The PI shall be responsible for ensuring that adequate funding and budgetary control procedures are in place for all aspects of the investigation.

The Experiment Manager shall be responsible for meeting the technical and schedule requirements.

The Experiment Manager within the PI organisation will have the responsibility of the design, procurement, integration, testing and provision of the relevant instruments. He shall implement an efficient management scheme, with appropriate hardware, software, and procurement expertise, especially when many institutes are involved. He shall exert adequate control over all aspects of the programme, including the required financial resources.

10.1.3. Spacecraft Contractor

The Spacecraft Contractor will be responsible for the instruments to spacecraft interfaces, the provision of the necessary Spacecraft ressources, the timely delivery of the instruments, the day to day progress monitoring and the administration of changes and non conformances. The proper performance of the instruments to achieve the scientific objectives will be under ESA/PI responsibility. The Spacecraft Contractor will directly follow-up and monitor the instrument development for all interface aspects. The Spacecraft Contractor will participate in technical and progress meetings, reviews, boards. He will have the approval authority for all interface aspects covering the interface design and verification, delivery schedule, and verification activities requested by the PI to be made at system level.

10.1.4. Formal Management Implementation

The following management scheme applies :

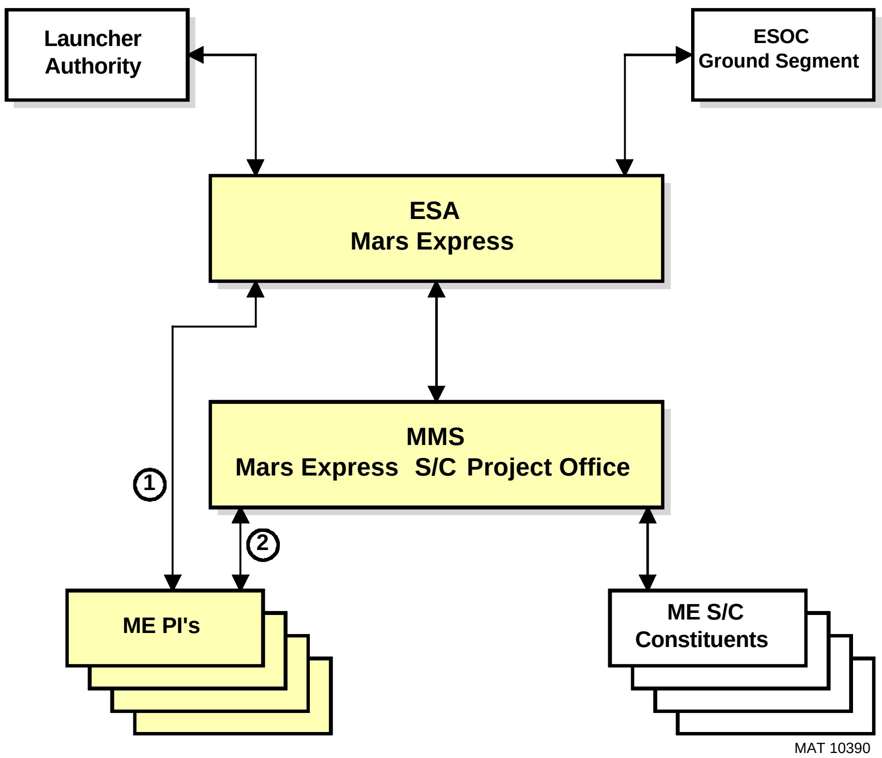

Figure 10.1/1 : formal relationship between ESA, the PI's and the Spacecraft Contractor

For Mars Express, the PI will have formal relationship simultaneously with ESA and with the spacecraft contractor :
Link 1 - The formal relationship between the PI and ESA covers the scientific performance of the instrument
Link 2 - The formal relationship between the PI and the spacecraft contractor covers the procurement of the instruments.

The present document concerns the Link 2 only.

The formal management and control of the instruments by the S/C contractor will be established through :

- Payload interface documents (PID/A and PID/B),
- Payload Master Schedule,
- Memorandum of understanding (TBD).

10.2. Funding

The PI's shall at all times be responsible for the funding arrangements of their investigation and the management thereof. They shall ensure that adequate funding is available at the required time(s) for all aspects of the experiment and its support. A financial reserve should be provided, not only to cater for experiment evolution, but also to finance changes deemed necessary by ESA and/or the Spacecraft Contractor, in particular covering accommodation.

The Payload Contractors shall not assume any funding from either ESA or the Spacecraft Contractor for any part of their investigations.

This requirement shall apply up to the point of final acceptance of the experiment by the Agency and Spacecraft Contractor.

10.3. Formal Communications

All formal communications and agreements concerning technical and programmatic aspects, except scientific performance, shall be made between the PI and the Spacecraft Contractor Project Managers. No other party shall have formal authority, without written delegation.

All communications with the Spacecraft Contractor shall be via the Spacecraft Contractor Project Office.

All communications with ESA including formal communication for scientific performances shall be via the ESA Project Office.

10.4. Reporting

10.4.1. Progress Reports

The PI shall submit within one week following the end of the month, a Monthly Progress Report in which the current status of each activity is described and problem areas or potential problem areas are highlighted with identification of proposed remedial action.

The Monthly Progress Report shall include the following topics:

1. Overall summary,

   covering scientific and technical performance, status of design changes and open ECR's

2. Design, Development and Verification,

   covering status of design definition and verification, interfaces, test and calibration, GSE, operations

3. PA,

   covering PA status, NCR's and RFW's

4. Programmatic and schedule,

   covering schedule and milestones reporting, status of deliverable items

5. Problem areas and corrective actions.

The Monthly Progress Report shall be concise (not exceeding 5 pages as an objective) and submitted in a format to be agreed between PI and Spacecraft contractor.

The Monthly Progress Reports submitted will be analysed in conjunction with the overall spacecraft programme by the Spacecraft Contractor. By this monitoring action early alerting to potential conflicts will be communicated back to the PI. In the case of major conflicts the Spacecraft Contractor may call for special schedule meetings to resolve the issue.

10.4.2. Emergency Reports

Any major discrepancy with respect to the formal documents mentioned in § 10.1.4 for the formal management and control of the instruments shall be reported within 48 hours from the date of occurrence of such discrepancy.

10.5. Meetings

10.5.1. Instrument Progress Meetings

Regular instrument progress meetings shall be held on the premises of the PI during the design, development and verification programme of the instrument.

These meetings will be conducted between ESA/Spacecraft Contractor and the PI team with the objective of ensuring that the interface technical design integrity of the instrument, its compatibility with the spacecraft system, and instrument programmatic are proceeding in a manner which will not jeopardise the overall programme.

The meetings shall be held on a monthly basis (to be confirmed). The frequency may be changed on request of ESA/Spacecraft Contractor.

Detailed technical problems occurring on either side of the interface shall be flagged during these meetings and corrective actions, including schedules, agreed and implemented.

These progress meetings shall not relieve the PI from fulfilling the reporting requirements as per §10.3.

10.5.2. Other Meetings

ESA and/or the Spacecraft Contractor may call for other meetings to be held as deemed necessary on managerial, technical, or schedule aspects, and the PI shall be requested to organise and/or participate in such meeting.

10.6. Reviews

10.6.1. General

10.6.1.1 Programme Phases and Reviews

The Mars Express Definition Study phase (phase A) has been accomplished during the 1st half of 1998 with the objectives (i) to define the mission including selection of the payload instruments, (ii) to define a spacecraft reference design concept, and (iii) to elaborate project plans for the procurement phase.

This initial system definition study phase has been combined with the Announcement of Opportunity (AO) phase for Payload Experiments with issue of the AO in December 1997 and ESA selection of the payloads in June 1998.

An Invitation to Tender (ITT) for the spacecraft procurement will be issued at the end of this study phase.

After the payload and spacecraft contractor selection, the overall Mars Express project development logic for the procurement phase B/C/D can be split into several phases :

- the Preliminary Definition phase (phase B), from the project kick-off until the spacecraft Preliminary Design Review (PDR) at the end of 1999. The objectives of this phase are (i) to freeze the System requirements, (ii) to finalise the spacecraft design definition down to lower level constituents specifications, and to select the spacecraft unit suppliers.
- the Detailed Design and Validation phase from the Preliminary Design Review until the spacecraft Critical Design Review (CDR) in the 2nd half of 2001. The detailed design phase down to unit level is authorised following the PDR and the validation is achieved at the spacecraft Critical Design Review (CDR),
- the spacecraft Flight Model integration and qualification from the Critical Design Review until the Qualification Review (QR) at the end of 2002. This phase is authorised following the CDR and will last until the spacecraft qualification.

It is then followed by the launch preparation and launch campaign for a launch in June 2003.

10.6.1.2. Instrument Reviews

All interface aspects of each instrument programme shall be reviewed by ESA and the Spacecraft Contractor.

Instrument Reviews shall be conducted by ESA/Spacecraft Contractor, at PI premises or any other premises to be agreed upon. The objectives will be to ensure that the instrument :

- will achieve the anticipated science objectives as accepted at selection, (ESA only as a reviewer),
- complies with the interface requirements of the PID,
- will achieve the schedule delivery dates requirements,
- will provide a verification status according to the DDVP

The Review Board will be made up of ESA/Spacecraft Contractor personnel and shall be co-chaired by the Spacecraft Contractor Payload Assistant project Manager or his appointed representative together with the ESA Project Scientist. The review process will be implemented through Review Item Discrepancies (or RID's) to be further responded under the responsibility of the PI. To this respect, the PI and his scientific and technical team, shall support the review, its panel and board sessions with appropriate manpower, expertise and decision makers.

Documentation to be reviewed shall consist of Review Data Packages and shall cover both hardware and software aspects with details of any other deliverables such as GSE and documentation. It shall be delivered to the ESA/Spacecraft Contractor at a minimum ten working days prior to the scheduled review date.

The content of the deliverable documentation is detailed in the DRL (and in the DRD / TBD Annex).

The output of the review shall provide agreed recommendations to be implemented by ESA/Spacecraft Contractor or the PI in technical or programmatic areas. Either party shall provide a formal response to such recommendations within one month of review completion.

The reviews planned are :

- the Instrument Requirements Review (IRR),
- the Instrument Preliminary Design Review (IPDR),
- the Instrument Critical Design Review (ICDR)
- the Instrument Qualification and Acceptance Review (IQAR).

Depending on the design maturity of the instrument at the time of selection, the IRR, IPDR and ICDR may be merged.

10.6.2. Instrument Requirements Review

This shall be conducted during the initial stage of the industrial system design phase (Phase B) in parallel or just after the programme level System Requirements Review. The objective of the review shall be:

- to freeze the instrument interface requirements (PID/A)
- to approve the first issue of the instrument interface control document (PID/B),
- to review the instrument qualification status (depending the instrument definition) by assessing the available justification documentation
- to authorise the manufacture of instrument Engineering Model (EM),
- to approve the preliminary draft issue of the Instrument Operation Manuals.

10.6.3. Instrument Preliminary Design Review

This shall be conducted during the final stage of the system design phase (phase B) just before or in parallel to the spacecraft Preliminary Design Review. The objective of the review shall be :

- to freeze the instrument interface control document (PID/B),
- to assess the instrument EM programme,
- to freeze the instrument verification plan, including system level activities,
- to approve the first issue of the Instrument Flight Operations Manual.

Following this review, the interface requirements shall be detailed and frozen.

10.6.4. Instrument Critical Design Review

This shall be conducted after the completion of the instrument design verification programme and before the Spacecraft Critical Design Review. The objective of the review shall be :

- to assess the results of qualification testing at instrument level on EM,
- to approve the FM design

10.6.5. Instrument Qualification and Acceptance Review

This shall be conducted after the completion of the instrument qualification and acceptance verification programme and in parallel to or just following the Spacecraft Critical Design Review. The objective of the review shall be :

- to assess the results of qualification and acceptance testing at instrument level on FM,
- to approve the final issue of the Instrument Flight Operations Manual.

10.6.6. Instrument participation in the System Qualification Review

The Payload Contractor shall participate in the System Qualification Review for what concerns his instrument test review.

10.7. Change Control

10.7.1. Change Control Procedure

To ensure that the controlled interfaces between ESA/Spacecraft Contractor and the PI correctly reflect the up do date status, the PC shall implement a Change Control Procedure.

No change shall be made to any hardware, software or documentation which is under Configuration Control, which has not been submitted to the Spacecraft Contractor on an Engineering Change Request (ECR) format. Furthermore, no change shall actually be implemented before the corresponding ECR has been approved by ESA/Spacecraft Contractor.

A change may be initiated at any time by either the PI or ESA/Spacecraft Contractor by written request (ECR).

Following its receipt, the proposed change is submitted to the Change Control Board (CCB) who shall process the request and take a decision on the change (ECR disposition) within 4 weeks.

The approval and decision to introduce a change is taken by ESA/Spacecraft Contractor.

No activities on the proposed change shall be started prior to the written approval of the ECR. The work to implement the change shall start immediately after a formal authorization to proceed has been given.

10.7.2. Engineering Change Requests

All changes shall be addressed to the Spacecraft Contractor under an ECR, whether they concern an interface aspect or another aspect of the instrument.

All ECR's shall be addressed to the Spacecraft Contractor Project Manager.

The ECR shall be completed with all points in the agreed format, including the proposed classification of interface impact/no impact.

The Spacecraft Contractor shall have the right to override the proposed classification.

Incomplete ECR's will not be processed. Resulting consequences (i.e. schedule, delivery time) will be borne by the PI.

The ECR shall include and define the responsibility of the funding of the modification. PI’s shall ensure that adequate resources and funding are available to them for execution of a proposed change prior to submittal to the Spacecraft Contractor.

10.7.3. Numbering of Changes

Each ECR shall be identified by an individual number. This number shall be used on all subsequent correspondence until the ECR is either rejected, accepted or closed. Whatever the ECR disposition, this number shall not be used again.

The PI shall register and control his own numbering sequence.

The ECR numbering will be defined later on (to be provided by ESA/Spacecraft Contractor).

10.7.4. Handling of Deviations and Non Conformances

The PI shall submit Requests For Waiver (RFW) when deviations from the specification occur during the design. A RFW log shall be set up.

The PI shall submit Non-Conformance Report (NCR) when deviations from the design occur during build and test. A NCR log shall be set up.

The PI shall review all Requests For Waiver and Non-Conformance Reports issued by its Contractors and submit these, with its assessment to the S/C contractor.

The RFW and NCR log shall be included in the progress report.

10.8. Programmatics and Schedule

10.8.1. Overall Approach

The accommodation of the payload instruments with maximum flexibility whilst securing the development schedule will be reached by working out a plan of exchanges between the Payload Contractors and the Spacecraft Contractor ; such plan will be specific to each instrument, thus taking due account of its own development status and complexity :

- Exchange of interface specification (PID/A) and of interface control document (PID/B)
- Exchange of mathematical models (thermal, mechanical, dynamics, stray light, ...)
- Delivery by the PI of instrument engineering models
- Delivery by the PI of structural dummies
- At last, delivery by the PI of the instrument flight model entering the spacecraft flight specimen integration

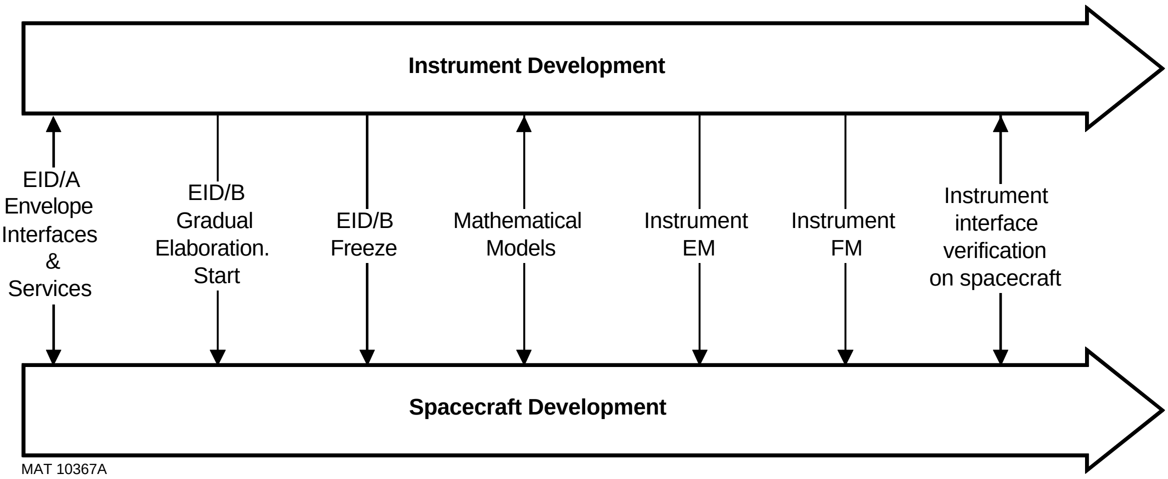

Figure 10.8/1 : development interfaces

10.8.2. Project Control Objectives

The above development interfaces and plan of exchanges will require to apply project control procedures and methods which are strictly observed.

The PI's shall implement as part of their management programme a system of planning, reporting and progress control focused on risk assessment and management.

The objective of this section is to clearly specify the management information required from each PI. Due to the importance of the experiments in the programme, it is essential that each PI supports this scheme with relevant configuration, schedule, work progress and risk analysis information.

10.8.3. Project Breakdown Structures

10.8.3.1. General

In order to clearly identify the experiment or module, the scope of the work and the responsibilities involved, the following structures will be created and maintained by the PI :

- the Product Tree (PT) to break down the instrument into its constituents, both hardware and software,
- the Work Breakdown Structure (WBS) to define the scope of the work and the responsibilities involved.

10.8.3.2. Product Tree

A Product Tree shall be defined and provided by the PI, depicting a product oriented breakdown of the experiment or module into successive levels of detail. The Product Tree shall be submitted to the ESA/Spacecraft Contractor and shall be maintained under configuration control.

10.8.3.3. Work Breakdown Structure

A Work Breakdown Structure shall be defined and provided by the PI, based on its agreed Product Tree and extended to all the development models and support functions necessary to produce all the deliverables.

The WBS shall be submitted to the ESA/Spacecraft Contractor and maintained up to date throughout the project duration.

10.8.4. Schedule Control

10.8.4.1. Master Schedule

The PI shall establish and submit to the ESA/Spacecraft Contractor a Master Schedule covering the entire experiment or module programme activities identified in the Work Breakdown Structure.

All interfaces, internal and external, such as procurement items, long lead items, EEE parts, hardware and software deliveries, reviews, critical activities shall be clearly identified.

The schedule shall reflect the result of detailed task analysis and critical review of all the activities associated with the experiment or module programme. It shall contain all activities, estimated duration, interfaces, and constraints. The schedule shall be constructed so that automatic time analysis of earliest and latest dates for all activities can be performed and critical paths identified.

The Master Schedule shall be put under formal configuration control. Changes to the Master Schedule shall only be made with the approval of the ESA/Spacecraft Contractor.

10.8.4.2. Milestones

From the Master Schedule, the Spacecraft Contractor will use selected milestones for progress control, early alert reporting and trend analysis.

10.8.4.3. Schedule Monitoring

The PI shall continuously record progress achieved and maintain forecasts.

The PI shall consolidate the progress and forecasts of all groups contributing to the instrument and compare schedule performance with respect to the overall Baseline Master Schedule.

Where deviations to the baseline have occurred or are predicted, the PI shall develop and implement corrective actions.

For each milestone, the PI shall maintain a record of the baseline achievement date, the forecast achievement date and the actual date achieved.

10.8.4.4. Schedule Reporting

In order to track the progress, the PI shall provide to the Spacecraft Contractor the schedule report as part of the progress report on a monthly basis.

During the manufacture and test phases, or depending on the design maturity of the instrument, the frequency of schedule reports may be increased, or specific meetings may be called for, should the ESA/Spacecraft Contractor judge progress to be critical.

10.9. Deliverable Items

10.9.1. General

During the course of the Mars Express project, the experiment items specified in this chapter shall be delivered by the PI to the Spacecraft Contractor at the required delivery dates.

Prior to delivery, the instrument deliverable items will undergo formal acceptance by ESA and the Spacecraft Contractor. This acceptance will involve the formal verification of all interfaces between the instrument and the spacecraft together with review of all relevant test reports and supporting analyses and documentation.

Where relevant, acceptance of EGSE software to be used at system level shall also be undertaken.

Following formal signature of the Certificate of Compliance, the experiment item will be accepted for delivery to the spacecraft contractor.

Shipment of the instrument models and any other equipment required by either ESA/Spacecraft Contractor 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.

Any insurance deemed necessary by the PI for his equipment during shipment or whilst on the premises of ESA, the Spacecraft Contractor, their lower tier contractors, or on the launch site, shall be the financial responsibility of the PI.

10.9.2. Mathematical Models

The PI shall deliver a Structural Mathematical Model and a Thermal Mathematical Model of his instrument/lander module units. These mathematical models shall be updated as the design progresses. Other mathematical models may be delivered as proved necessary (dynamics and flexible modes, stray light, power, data flows...). They will serve as input to the spacecraft mathematical models.

10.9.3. Instrument Data Base

The Instrument Data Base (TBD format) will be the core element of the computerized (as possible) tool for processing the P/L ICD data (in particular for TC's and TM's).

10.9.4. Instrument Structural Model

The PI shall deliver a Structural Model (SM) which is fully representative of the physical and mechanical properties of the experiment interfaces (mass, dimensions, CoG, ..). It will serve for the mechanical qualification testing of the spacecraft.

10.9.5. Engineering model

Where relevant, the PI shall deliver an engineering model, as appropriate, which is fully representative of the functional interfaces with the spacecraft. It will serve to undertake advanced system validation.

10.9.6. Instrument Flight Model

The Flight Model shall be of a standard compliant with all the requirements of PID/A and PID/B, and it shall have successfully undergone a full programme of acceptance level testing and verification prior to delivery.

10.9.7. Instrument Flight Spare

The PI shall propose an integration and flight spare philosophy which safeguards the system integration schedule and the experiment embarking on the spacecraft.

Ideally, a flight spare should be available. However, this requirement may be waived provided that the PI accepts that, once delivered, there will be no further chance to modify an instrument, prior to launch, and that if, for any reason, there is a failure, the launch will proceed without the instrument in question.

The Flight Spare Model (if any) shall be of a standard compliant with all the requirements of PID/A and PID/B, and they shall have successfully undergone a full programme of acceptance level testing and verification prior to delivery.

The level of verification and test of this model shall be such as to enable it to support the spacecraft programme by direct substitution during the system level programme at any point up to launch.

10.9.8. Ground Support Equipment

The PI shall deliver the following ground support equipment together with each experiment model:

- Mechanical Ground Support Equipment (MGSE) necessary to transport, handle and integrate experiment hardware together with appropriate documentation and proof load and calibration certificates,
- Optical Ground Support Equipment (where relevant) necessary to stimulate and test the instrument,
- Electrical Ground Support Equipment (EGSE) necessary to stimulate the experiment and to perform quick look analysis of experiment TM during system tests. It shall be designed such that it can be integrated into the system EGSE set-up. The instrument EGSE software to be delivered with the EGSE equipment shall comply with the ESA software standard PSS-05-0.

The instrument ground support equipment shall remain at the spacecraft integration site until launch. However, the PI remains responsible for the maintenance of this equipment.

The PI shall also provide the necessary manpower and expertise support to integrate the instrument EGSE into the system EGSE.

10.9.9. Delivery Dates

The following table recalls the main deliverables (but documentation listed under paragraph 10.10. hereafter) and their delivery dates.

Category	Deliverable Item	Date
Mathematical Models	Structural Mechanical Model	IPDR
	Thermal Mathematical Model	IPDR
Instrument Models	Structural Model	mid 2000 (*)
	Engineering Model	mid 2000 (*)
	Flight Model	mid 2001 (*)
	Flight Spare	with Flight Model
GSE	Mechanical Ground Support Equipment	with each model
	Electrical Ground Support Equipment	with electrical representative models
	Optical Groud Support Equipment (**)	with each model

(*) to be confirmed for each payload

(**) as required by performance measurements at S/C level

10.9.11. Post-Delivery Support

Post-delivery support shall cover all activities after delivery of the experiment and their associated documentation, as required by ESA/Spacecraft Contractor or agreed with the PI. Such activities shall encompass, but not limited to, instrument integration and test support at system level, participation in the System Qualification Review for instrument test review, maintenance, debugging, repair or replacement of defective items, launch operations support, etc..

The PI shall undertake to maintain support facilities, equipment and associated software which may be needed for fault diagnosis, repair or calibration of their experiment throughout the programme.

The maintenance facility shall be valid and available up to and including the launch operations phase.

10.9.12. Shipment

Shipment of the experiment, its associated ground support equipment and any other equipment required by either ESA/Spacecraft Contractor, 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.

The points of delivery of all items will be defined later on.

Any insurance deemed necessary by the PI for his equipment during shipment or whilst on the premises of the Agency, the Spacecraft Contractor, their lower tier contractors, or on the launch site, shall be the financial responsibility of the PI.

Customs regulations for importation of equipment associated with the programme to the various countries involved will apply. Failure to adhere to the requirements with respect to customs regulations shall be the financial responsibility of the PI.

The PI shall develop a logistics plan identifying the location of all deliverable and support hardware at all times.

10.10. Documentation

The deliverable documentation (but monthly progress and schedule reporting) is defined in the table hereafter (in the so-called Documentation Requirements List, or DRL) and the contents of each document - anyway already introduced in the various parts of this PID - are to be further detailed in the Documentation Requirements Description (DRD) Annex (presently TBD).

The project documentation shall be written in English, and nominally delivered in 10 copies (5 for ESA and 5 for the S/C contractor).

REVIEWS / DOCUMENTS	IRR	IPDR	ICDR	IQAR
Review Closure Report (minutes, RID status and actions)	X	X	X	X
Configuration				
As Designed Configuration				
CIDL (including Software)	X	X	X	update
Engineering Change Request Status List		X	X	update
Request For Waiver Status List		X	X	update
As Built Configuration				
Non Conformance Report Status List				X
Set of Major Non Conformances				X
Design				
Payload System				
Design Description	X	X	update	update
P/L Performance Budgets	X	X	X	update
P/L Resources Budgets	X	X	X	update
PID-A Compliance Matrix	X	freeze	update	update
PID-B	X	freeze	update	update
Drawings (*)				
Assembly Drawings	X	X	X	update
Interface Drawings	X	X	X	update
Interface Circuit Diagrams	X	X	X	update
Mathematical Models (*)				
Structural Model		X	X	update
Simplified Thermal Model		X	X	update
P/L Data Base (all TC's and TM's)	X	freeze	update	update

(*) drawings and mathematical models are part of the PID-B

(**) includes the updated instrument master schedule

DOCUMENTS / REVIEWS	IRR	IPDR	ICDR	IQAR
Design (continued)				
Analyses				
Performance Analyses	draft	X	X	update
Mechanical / Dynamic / Structural		X	X	update
Thermal (incl. validation and all predictions)		X	X	X
Electrical (incl. EMC)		X	X	X
Manuals				
P/L User Manual (for ground use)	draft	X	X	update
GSE User Manuals	draft	X	X	update
P/L Flight Operations Manual	draft	X	X	update
Product Assurance				
Product Assurance Plan	X	update		
Contamination Control Plan / Verification Document		draft	X	update
Material and Processes Lists		draft	X	update
Parts				
Procurement Specifications			X	update
Declared Part List		draft	X	update
Requests for Parts Approval			X	update
Dependability				
FMECA		X	update	update
Parts Stress Analysis		X	update	update
Electrical Worst Case Analysis		X	update	update
Dependability Critical Items List		X	update	update
Hazard Analysis / Verification Document		X	update	update
Assembly, Integration and Verifications				
Verification				
Design Development and Verification Plan	draft	freeze	update	update
PID Verification Control Document	draft	freeze	update	update
Qualification status	X (***)	X	X	X
Assembly, Integration and Tests				
test specifications (all tests)		draft	X	update
test procedures (all tests)			draft	X
test reports (all tests, incl. as run test proc's)				X

(***) depending instrument definition (re-use or not of hardware / design) and available documentation assessments