# X-RAY MULTI MIRROR MISSION

## Experiment Interface Document

## Part A

**RS-PX-0016**
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## LIST OF ACRONYMS

| | |
|---|---|
| AC | Alternating Current |
| AD | Applicable Document |
| AIT | Assembly, Integration and Testing |
| AIV | Assembly, Integration and Verification |
| AMA | Absolute Measurement Accuracy |
| AOCS | Attitude and Orbit Control System |
| APD | Absolute Pointing Drift |
| APE | Absolute Pointing Error |
| APID | Application Identification |
| BB | Broad Band |
| BCP | Broadcast Pulse |
| BOL | Beginning of Life |
| CCD | Charge Coupled Device |
| CCDS | Consultative Committee for Space Data Systems |
| CCS | Central Checkout System |
| CDMU | Control and Data Management Unit |
| CMOS | Complementary Metal Oxide Semiconductor |
| CoG | Centre of Gravity |
| CRC | Clean Room Class |
| CTU | Central Terminal Unit |
| CVCM | Collected Volatile Condensable Material |
| DBI | Digital Bus Interface |
| DBU | Data Bus Unit |
| DC | Direct Current |
| DH | Data Handling |
| ECR | Engineering Change Request |
| EED | Electro-Explosive Device |
| EGSE | Electrical Ground Support Equipment |
| EID | Experiment Interface Document |
| EMC | ElectroMagnetic Compatibility |
| EMI | Electro Magnetic Interference |
| EOBB | Electro-Optical Breadboard model |
| EOL | End of Life |
| EPIC | European (X-ray) Photon Imaging Camera |
| EQM | Engineering Qualification Model |
| ESA | European Space Agency |
| ESD | ElectroStatic Discharge |
| ESOC | European Space Operations Centre |
| ESTEC | European Space research and Technology Centre |
| EUT | Equipment Under Test |
| FM | Flight Model |
| FMECA | Failure Modes, Effects and Criticality Analysis |
| FOP | Flight Operations Plan |
| FOV | Field of View |
| FPP | Focal Plane Platform |
| FPPR | Focal Plane Platform Reference cube |
| FPT | Full Performance Test |
| FS | Flight Spare model |

| | |
|---|---|
| GSE | Ground Support Equipment |
| HK | HouseKeeping |
| IAR | Instrument Angular Reference |
| ICD | Interface Control Drawing |
| IPR | Instrument Positional Reference |
| IS | Instrument Station |
| KAL | Keep Alive Line |
| LEOP | Launch and Early Orbit Phase |
| LET | Linear Energy Transfer |
| LISN | Line Impedance Simulation Network |
| LPT | Limited Performance Test |
| LTMS | Local Time Management System |
| MSP | Mirror Support Platform |
| MSPR | Mirror Support Platform Reference cube |
| MCC | Mission Control Centre |
| MGSE | Mechanical Ground Support Equipment |
| MOC | Mission Operation Centre |
| MoI | Moment of Inertia |
| MOS | Metal Oxide Semi-conductor |
| MLI | Multi-Layer Insulation |
| MM | Mirror Module |
| NB | Narrow Band |
| OBDH | On-Board Data Handling |
| ODE | OM Digital Electronic unit |
| OGSE | Optical Ground Support Equipment |
| OM | Optical Monitor |
| PA | Product Assurance |
| PCB | Printed Circuit Board |
| PDU | Power Distribution Unit |
| PI | Principal Investigator |
| PLM | PayLoad Module |
| QM | Qualification Model |
| RAM | Random Access Memory |
| RD | Reference Document |
| RF | Radio Frequency |
| RGS | Reflection Grating Spectrometer |
| RH | Reference Holes |
| RML | Recovered Mass Loss |
| ROM | Read Only Memory |
| RPE | Relative Pointing Error |
| S/C | Spacecraft |
| SCET | Spacecraft Elapsed Time |
| SDB | System DataBase |
| SEU | Single Event Upset |
| SI | Système International |
| SOC | Science Operation Centre |
| STM | Structural Thermal Model |
| SVT | System Validation Test |
| S/W | Software |
| TAG | Technical Advisory Group |

| | |
|---|---|
| TBC | To Be Confirmed |
| TBD | To Be Defined |
| TC | TeleCommand |
| TCT | Telescope Central Tube |
| TM | TeleMetry |
| TML | Total Mass Loss |
| UCS | Unit Coordinate System |
| UTC | Universal Time Coordinate |
| UUT | Unit/Equipment Under Test |
| XMM | X-ray Multi-mirror Mission |

# 1. INTRODUCTION

## 1.1 Structure and Content of the Document

The Experiment Interface Document (EID) forms the sole formal agreement on engineering, ground operations, flight operations and programmatic and management aspects for the XMM Project between ESA and each Principal Investigator (PI).

The document consists of three parts:

**Part A** contains the ESA requirements of general nature with which each of the three instruments shall comply.

Each **Part B** contains both the PI's response to the requirements placed in Part A specifying, in detail, the Instrument's technical interface information and the Instrument's requirements.

**Part C** contains the programmatic and management aspects governing the relations between ESA and the PI's.

Parts A, B and C of the EID will be put under formal configuration control by the ESA XMM Project Office and, as such, will be a binding agreement between ESA and the PI.

In cases of conflict between Part A and Part B of this document, the agreement or definition in Part B shall take precedence.

As a single point of control of the technical interfaces between ESA and the PI, the EID:

- Defines the resources allocated to the instruments
- Defines the system requirements
- Defines the interfaces between the instrument and the spacecraft
- Defines the design and construction requirements for the instruments
- Defines the verification programme which shall be implemented to demonstrate the instrument's compliance with the ESA requirements
- Defines the operational interface applicable during the ground phase of the mission, the post-launch support and the deliverable items.
- Defines the programmatic and management aspects

### 1.2 Objective of the Document

The EID helps to ensure that:

- The PIs design, build and verify their instruments within the constraints imposed by the payload, spacecraft and launch vehicle.
- The spacecraft Prime Contractor designs, builds and verifies the spacecraft in such a manner that the instruments can be successfully integrated into the system.
- The spacecraft system can be successfully launched and operated to achieve the scientific objectives of the XMM programme.

## 2. APPLICABLE AND REFERENCE DOCUMENTS

The following is a list of applicable documents:

| | | |
|---|---|---|
| AD 1 | RS-PX-0017 | Product Assurance Requirements for XMM Experiments |
| AD 2 | PSS-05-0 | ESA Software Engineering Standards |
| AD 3 | PSS-04-0 | Space Data Communications |
| AD 4 | PSS-04-106 | Packet Telemetry Standard |
| AD 5 | PSS-04-107 | Packet Telecommand Standard |
| AD 6 | TTC-B-01 | Spacecraft Data Handling Interface Standard |
| AD 7 | PSS-03-301 | ESA Pyrotechnic Requirements |
| AD 8 | RS-PX-0032 | XMM Packet Structure Definition |
| AD 9 | PX-RS-00392 | SOC Implementation Requirements Document |
| AD10 | RS-PX-0028 | XMM Operations Interface Requirements |
| AD11 | XM-IF-DOR-0002 | OBDH Bus Protocol Requirement Spec. |

The following is a list of reference documents:

| | | |
|---|---|---|
| RD1 | deleted | |
| RD 2 | XM-IF-DOR-0001 | XMM Electrical I/F Requirements |
| RD 3 | TN-PX-0001 | Instruments' Mass Budget Definition Document |
| RD 4 | TN-PX-0012 | Power Budget Definition |
| RD 5 | PSS-01-609 | The Radiation Design Handbook |
| RD 6 | PX-TN-0540 | Data Error Protection for XMM |
| RD 7 | XMM-PS-GM-01 | Description of the basic modes of the Experiments on Board the XMM Observatory |
| RD 8 | XM-PL-DOR-0018 | EMC Control and Verification Plan |

# 3. MISSION AND SPACECRAFT DESCRIPTION

## 3.1 Mission Description

### 3.1.1 Overall Mission Description

The X-ray Multi-Mirror Mission (XMM) is a high throughput X-ray spectroscopy mission which is the second cornerstone of the ESA long term scientific plan. The XMM is a facility type observatory open to the worldwide astronomical community. The scientific payload forms an integrated mutually complementary package optimised to fulfil the scientific aims of the mission and fully exploit the ESA supplied X-ray optics.
The XMM observatory will offer a major step forward in the field of X-ray astrophysics in the 21st Century. It is envisaged as a long duration facility class mission aimed at performing detailed imaging spectrophotometry of a wide variety of X-ray sources.
The observatory will be placed in a 48 hour highly eccentric inclined orbit.

### 3.1.2 Operational Orbit Description

The baseline orbit parameters for XMM are:

| | |
|---|---|
| Apogee | 114 000 km |
| Perigee | 7,000 km |
| Inclination | 70 degrees. |
| Period | 47,8 h |

## 3.2 Spacecraft Description

The spacecraft consists of a service module which carries the payload module.

### 3.2.1 Spacecraft Coordinate System Definition

The Spacecraft Coordinate System is a right-handed, orthogonal coordinate system, defined as follows:

- The origin, $O_s$, is located at the centre of the interface plane between the spacecraft and the launch adaptor

- The $X_s$ axis is aligned with the launch vehicle thrust axis, pointing positively from the interface plane towards the spacecraft

- The $Z_s$ axis coincides with the sun-solar array line, pointing positively towards the sun

- The $Y_s$ axis completes the coordinate system.

Figure 3.2.1 illustrates the overall configuration.

### 3.2.2 Payload Module Description

#### 3.2.2.1 Introduction

The payload module consists of the following elements:

- The Focal Plane Assembly (FPA) carrying the focal plane instruments (EPIC and RGS)
  NB: The EPIC radiation monitor is located on the service module
- The Telescope Tube (TT)
- The Mirror Support Platform (MSP) carrying the Mirror Modules (MM), the Optical Monitor (OM) instrument, the RGS grating units and the star trackers

The TT provides a stable connection of the FPA to the MSP over the focal length of 7.5 m and will provide the primary load path from the service module to the FPA.

Fig. 3.2.2.1 shows the payload module and its main components.

**Fig. 3.2.1. : Overall Configuration**

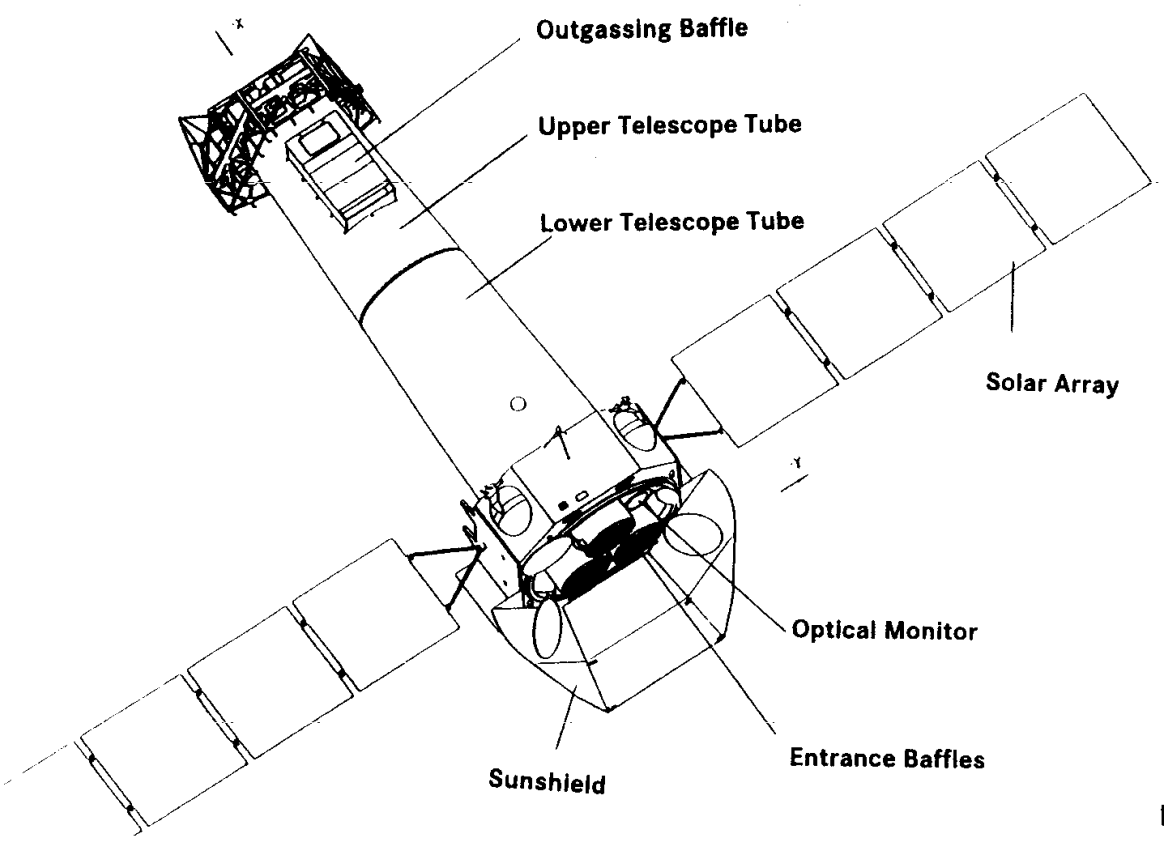
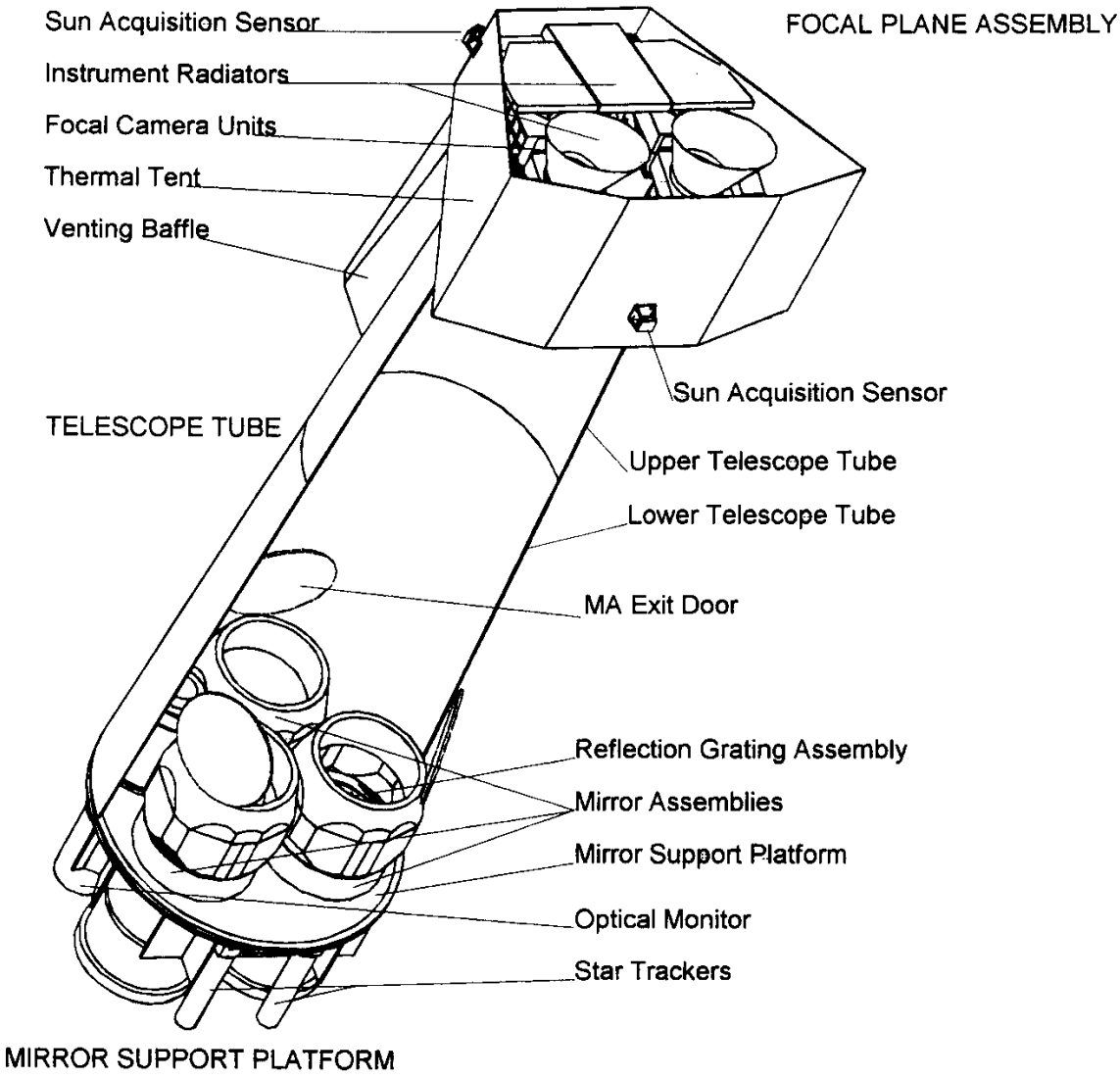


**Fig. 3.2.2.1: Payload Configuration**



#### 3.2.2.2 Mirror Module Functional Coordinate Frame

There are three Mirror Modules. The Functional Coordinate Frames are defined identical for all 3 modules.

The optical axis is defined as the theoretical common axis of revolution of the 58 mirror shells which form each Mirror Module.

The on-axis focus (F) of each Mirror Module is defined as the image point of an X-ray beam, parallel to the optical axis.

The Functional Coordinate Frame of each Mirror Module, MMi (1,2,3)-O-f, is defined as follows :

- the origin is located on the optical axis in the common hyperboloid/paraboloid separation plane of the 58 mirror shells
- the MMi-X-f-axis coincides with the optical axis, pointing positively towards the focus
- the MMi-Z-f-axis coincides with spoke # 1
- the MMi-Y-f-axis completes the coordinate frame

#### 3.2.2.3 Mirror Module Description

Each Mirror Module (MM) consists of:

- 58 mirror shells (hyperboloid, paraboloid) and a spider to support and maintain alignment of the mirror shells

- An interface structure to connect the MMs to the Mirror Support Platform (MSP)

- Thermal hardware including MLI blanket, heaters, thermistors and associated harness for the thermal control of the MM

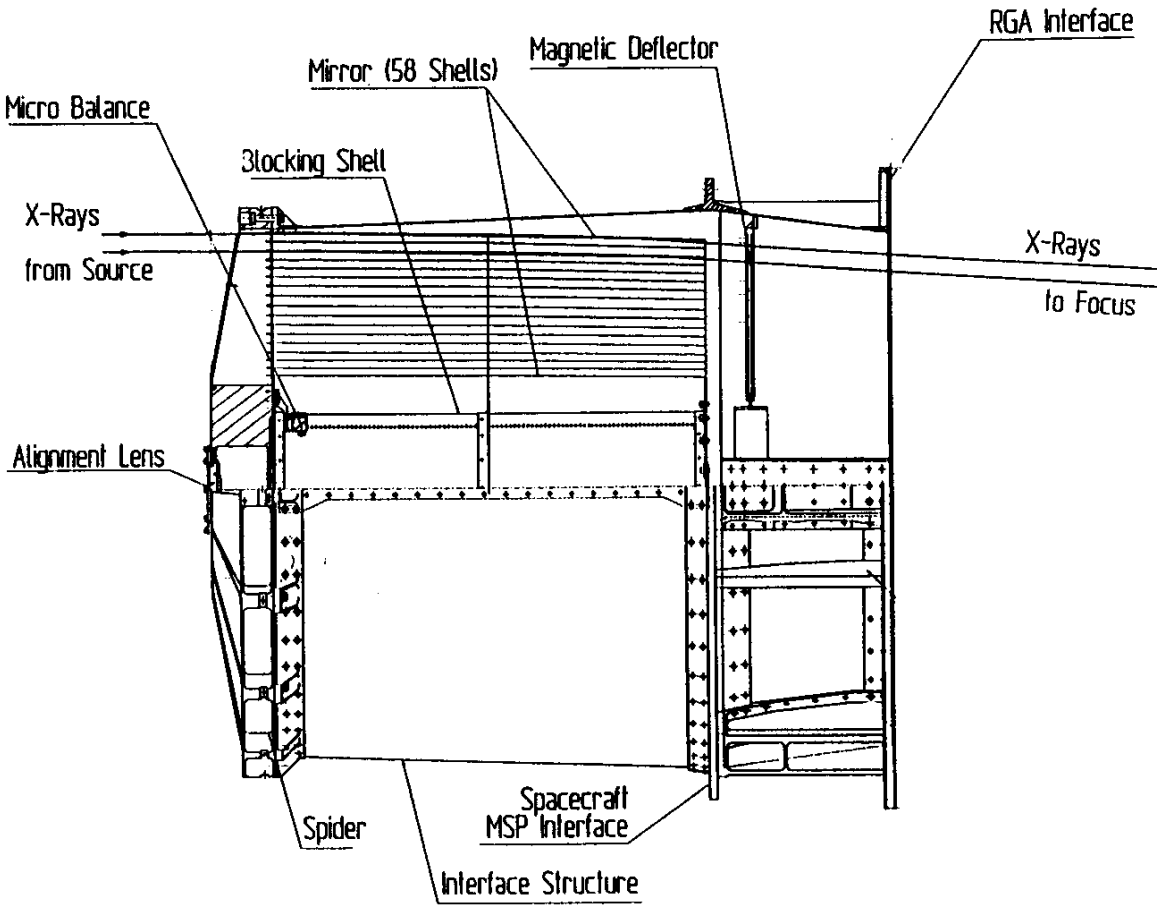
- Alignment devices (cubes, lens) for definition of the MM optical axis and for measurement of the alignment.

- A magnetic deflector to reduce the electron background.

The baffling and thermal control of the Mirror Module is part of the Mirror Support Platform functions.

The optical design layout is shown in Fig. 3.2.2.3.

**Fig. 3.2.2.3: Mirror Module Layout**



#### 3.2.2.4 Scientific Instruments' Description

The three instruments, namely the European Photon Imaging Camera (EPIC), the Reflection Grating Spectrometer (RGS) and the Optical Monitor (OM), are described in Chapter 2 of the respective Experiment Interface Documents, Part B (EID-B).

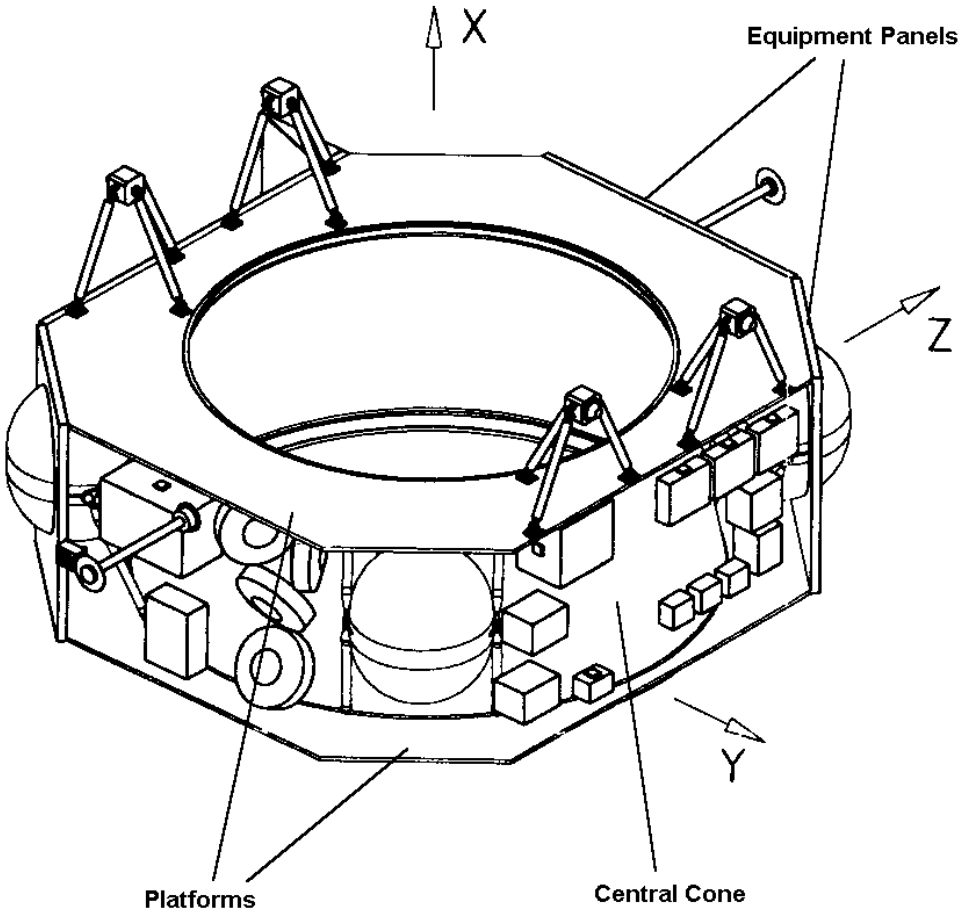
### 3.2.3 Service Module Description

The service module consists of the following elements:

- The centre thrust cone to provide the structural interfaces to the launcher and the MSP
- Platforms and equipment panels carrying the subsystems
- The supporting structure for the solar array

Figure 3.2.3 shows the service module configuration

**Fig. 3.2.3: Service Module Configuration**



# 4. SYSTEM REQUIREMENTS AND PERFORMANCES

## 4.1 Orbit Environment

### 4.1.1 Pressure Environment

- Launch:
  The slope of pressure variation is 2000 Pascal/sec.

- Space Vacuum:
  During orbital life, the equipment is exposed to a pressure of
  $1.3 \times 10^{-8}$ Pascal.

  Instruments shall survive operation at any pressure from ambient to space vacuum.

### 4.1.2 Radiation Environment

#### 4.1.2.1 General

XMM will traverse the Earth's radiation belts twice per orbit. During passages through the most intensive parts of these belts, telescope operation will be impossible.

XMM will also be exposed to energetic protons and heavy ions from solar flares and cosmic rays.

For instruments, the principal anticipated radiation effects are:

- Degradation of electronic components, detectors and materials (dose effect)
- Interference with detector operation (background)
- Cosmic ray-induced upsets
- "Latch-up".

More information on the radiation environment and its effects can be found in RD 5.

#### 4.1.2.2 Radiation Dose

Instruments shall be designed to withstand the doses shown in Fig. 4.1.2.2-1 and table 4.1.2.2.-1.

These doses are relatively high and are likely to cause problems to instrument and associated electronics. It should be noted, however, that the maximum value will be significantly reduced for most components due to spacecraft and box screening. Specific values will be calculated during the spacecraft development programme. Initial values taking into account the instrument preliminary design shall be agreed with the XMM Project Office.

To allow an assessment of the shielding requirements for CCD detectors the mission proton fluence is given in Fig. 4.1.2.2-3.

Fig. 4.1.2.2-2 shows the proton fluence per orbit.

#### 4.1.2.3 Cosmic Ray Effects

Fluxes of cosmic ray heavy ions (CR) produce a low-level background in detectors which is virtually irreducible (except by "vetoing"). These fluxes fluctuate and show a systematic variation with the solar cycle; with 4 mm aluminium shielding, CR proton fluxes will be below 7/cm$^2$/sec for about 90% of the time, with average solar minimum and solar maximum values of about 4/cm$^2$/sec and 2/cm$^2$/sec respectively.

It can be assumed that all penetrating cosmic rays hitting a detector will induce a count. Therefore, the background rate is found to be:

$$r = F S / 4$$

where S is the detector surface area and F is the integral omni-directional flux given above.

CR particles also disturb electronic components (see below). Evaluation of this problem requires testing of components to identify and characterise their sensitivity.

**Fig. 4.1.2.2-1: Total Radiation Dose for 10 years Against Shielding Thickness**

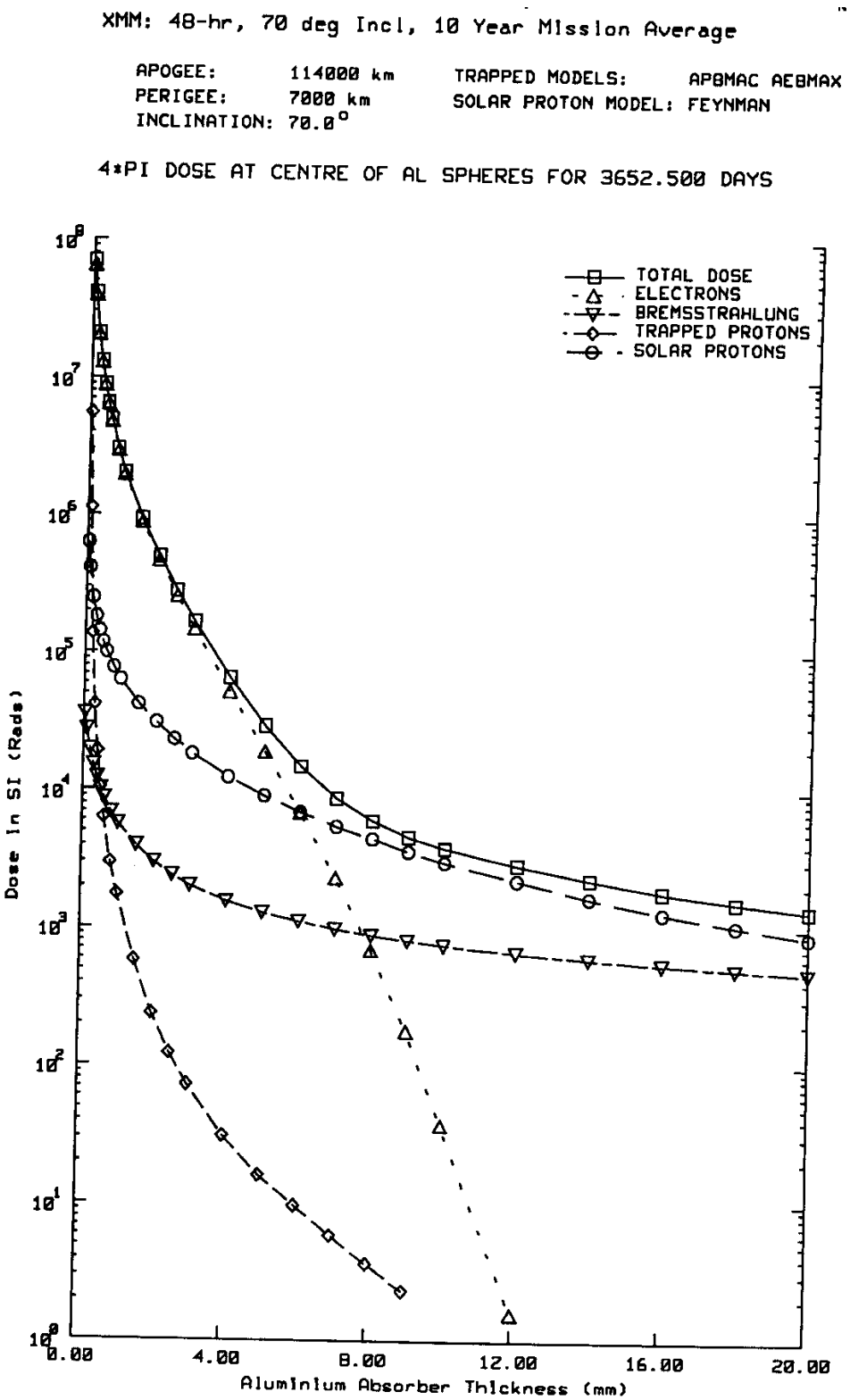


**Table 4.1.2.2.-1 : Total Radiation Dose against shielding thickness**

4*PI DOSE AT CENTRE OF ALUMINIUM SPHERES

| Z (MM) | Z (G/CM2) | TOTAL | ELECTRON | BREMS | TRP PROT | SOL PROT |
|---|---|---|---|---|---|---|
| 0.050 | 0.014 | 0.704E+08 | 0.643E+08 | 0.356E+05 | 0.550E+07 | 0.625E+06 |
| 0.100 | 0.027 | 0.407E+08 | 0.392E+08 | 0.271E+05 | 0.112E+07 | 0.408E+06 |
| 0.200 | 0.054 | 0.208E+08 | 0.204E+08 | 0.193E+05 | 0.137E+06 | 0.248E+06 |
| 0.300 | 0.081 | 0.130E+08 | 0.128E+08 | 0.150E+05 | 0.416E+05 | 0.180E+06 |
| 0.400 | 0.108 | 0.893E+07 | 0.876E+07 | 0.121E+05 | 0.191E+05 | 0.142E+06 |
| 0.500 | 0.135 | 0.646E+07 | 0.632E+07 | 0.101E+05 | 0.103E+05 | 0.117E+06 |
| 0.600 | 0.162 | 0.487E+07 | 0.475E+07 | 0.867E+04 | 0.625E+04 | 0.100E+06 |
| 0.800 | 0.216 | 0.301E+07 | 0.292E+07 | 0.680E+04 | 0.296E+04 | 0.772E+05 |
| 1.000 | 0.270 | 0.203E+07 | 0.196E+07 | 0.564E+04 | 0.175E+04 | 0.630E+05 |
| 1.500 | 0.405 | 0.934E+06 | 0.888E+06 | 0.393E+04 | 0.577E+03 | 0.419E+05 |
| 2.000 | 0.540 | 0.496E+06 | 0.462E+06 | 0.298E+04 | 0.237E+03 | 0.309E+05 |
| 2.500 | 0.675 | 0.282E+06 | 0.257E+06 | 0.239E+04 | 0.122E+03 | 0.232E+05 |
| 3.000 | 0.810 | 0.168E+06 | 0.148E+06 | 0.201E+04 | 0.726E+02 | 0.183E+05 |
| 4.000 | 1.080 | 0.660E+05 | 0.519E+05 | 0.156E+04 | 0.309E+02 | 0.124E+05 |
| 5.000 | 1.350 | 0.294E+05 | 0.190E+05 | 0.129E+04 | 0.162E+02 | 0.912E+04 |
| 6.000 | 1.620 | 0.150E+05 | 0.690E+04 | 0.112E+04 | 0.976E+01 | 0.699E+04 |
| 7.000 | 1.890 | 0.883E+04 | 0.233E+04 | 0.989E+03 | 0.594E+01 | 0.550E+04 |
| 8.000 | 2.160 | 0.609E+04 | 0.696E+03 | 0.893E+03 | 0.367E+01 | 0.450E+04 |
| 9.000 | 2.430 | 0.466E+04 | 0.180E+03 | 0.818E+03 | 0.232E+01 | 0.366E+04 |
| 10.000 | 2.700 | 0.386E+04 | 0.375E+02 | 0.759E+03 | 0.512E+00 | 0.306E+04 |
| 12.000 | 3.240 | 0.294E+04 | 0.160E+01 | 0.671E+03 | 0.000E+00 | 0.226E+04 |
| 14.000 | 3.780 | 0.230E+04 | 0.331E-01 | 0.606E+03 | 0.000E+00 | 0.169E+04 |
| 16.000 | 4.320 | 0.186E+04 | 0.000E+00 | 0.556E+03 | 0.000E+00 | 0.131E+04 |
| 18.000 | 4.860 | 0.157E+04 | 0.000E+00 | 0.514E+03 | 0.000E+00 | 0.106E+04 |
| 20.000 | 5.400 | 0.136E+04 | 0.000E+00 | 0.479E+03 | 0.000E+00 | 0.880E+03 |

**Fig. 4.1.2.2-2: Proton Fluence Per Orbit**

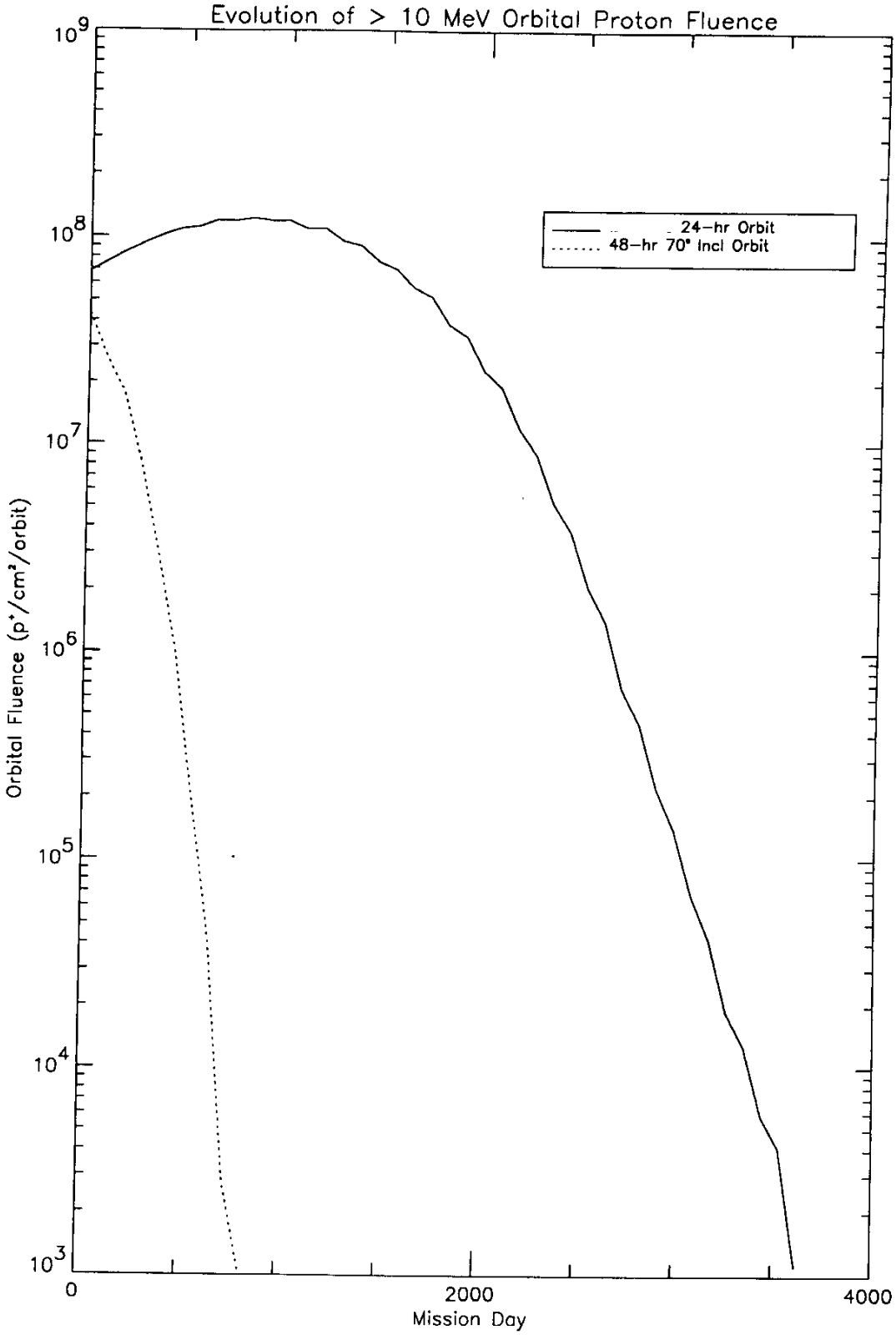
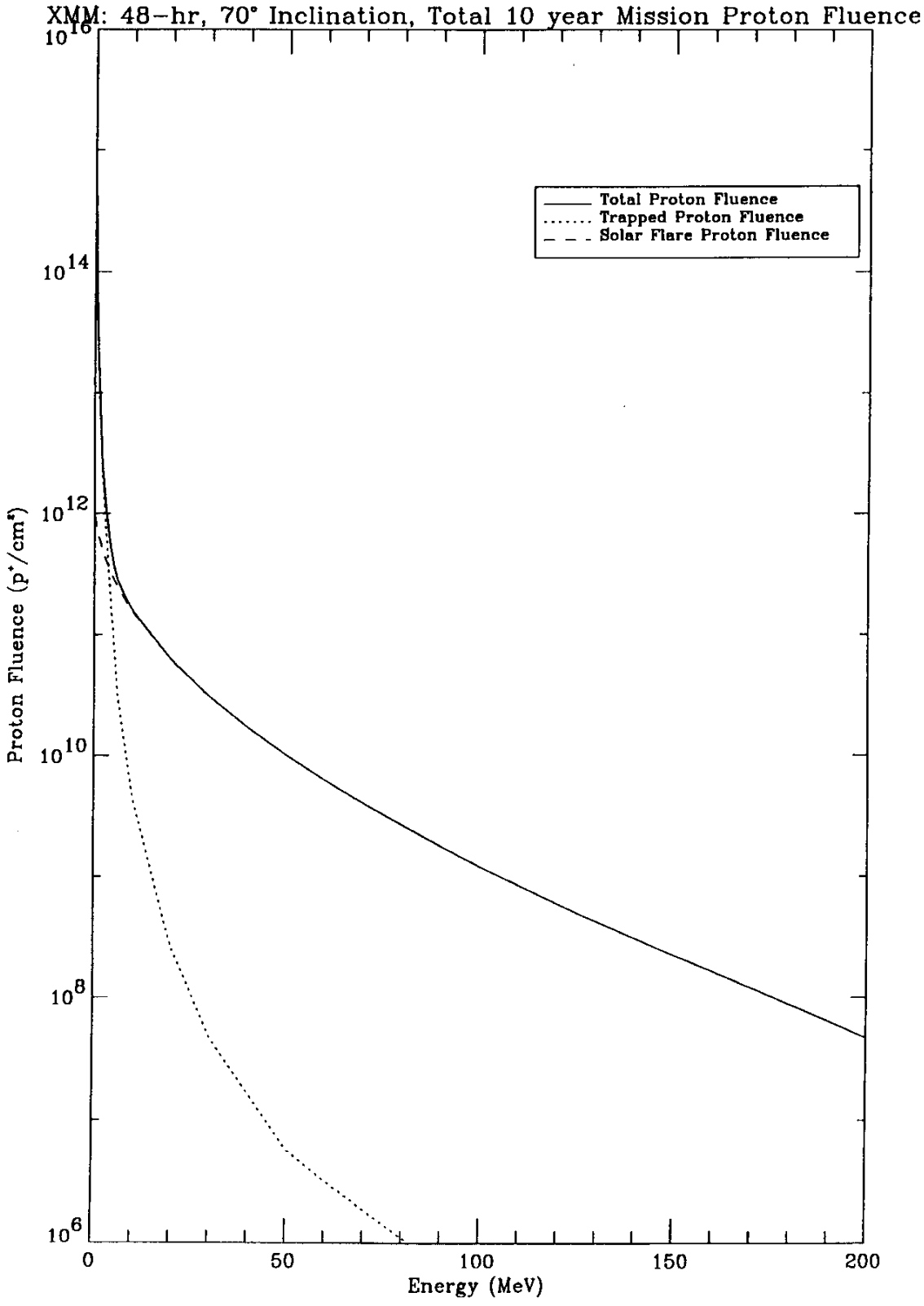


**Fig. 4.1.2.2-3: Solar Proton Integral Fluence**



#### 4.1.2.4 Components and Materials

##### 4.1.2.4.1 Total Dose Effects in Electronic Components

Total dose effects in electronic components are generally dominated by ionisation effects causing charge generation and trapping in surface layers. This charge trapping may cause threshold voltage shifts and increased leakage currents in MOS devices leading eventually to a loss of functionality. Ionisation damage in bipolar devices generally leads to a reduction in transistor gain and increased leakage currents, again leading eventually to a loss of functionality.

The sensitivity to total dose varies widely according to component technology. Certain complex CMOS devices show a tolerance of only a few kilorads, bipolar technologies are generally tolerant to the 100 kilorad range.

Bulk damage in the form of lattice displacement is not normally a problem in the space environment.

##### 4.1.2.4.2 Single Event Upset (SEU)

Devices such as memories containing bistable elements may be sensitive to Single Event Upset caused by cosmic rays. SEU is characterised by a soft error when the charge disposited by a heavy ion is greater than the critical charge required to change the state of a cell. No permanent damage is caused and the cell can be re-written.

Important parameters in defining SEU sensitivity are the "Threshold LET" (linear energy transfer) which is the lowest LET (stopping power) required to cause upset, and the limiting cross section which is the saturated value of the upset rate as a function of LET.

Upset rates in space can be significant, particularly in large memories, and consideration shall be given to error detection and correction schemes.
As a minimum, packet error control and single error detection shall be implemented (RD 6).

##### 4.1.2.4.3 Latch-Up

Cosmic rays may cause Latch-Up in certain technologies, primarily CMOS. Latch-up is caused by thyristor action in the substrate initiated by the passage of a heavy ion.

Latch-up is permanent and potentially destructive and protection must be provided when a device shows high sensitivity to latch-up. Such protection generally comprises a method for detecting latch-up (watchdog timer), a current limiting system and a re-set system which removes all power from the device in order to achieve de-latch.

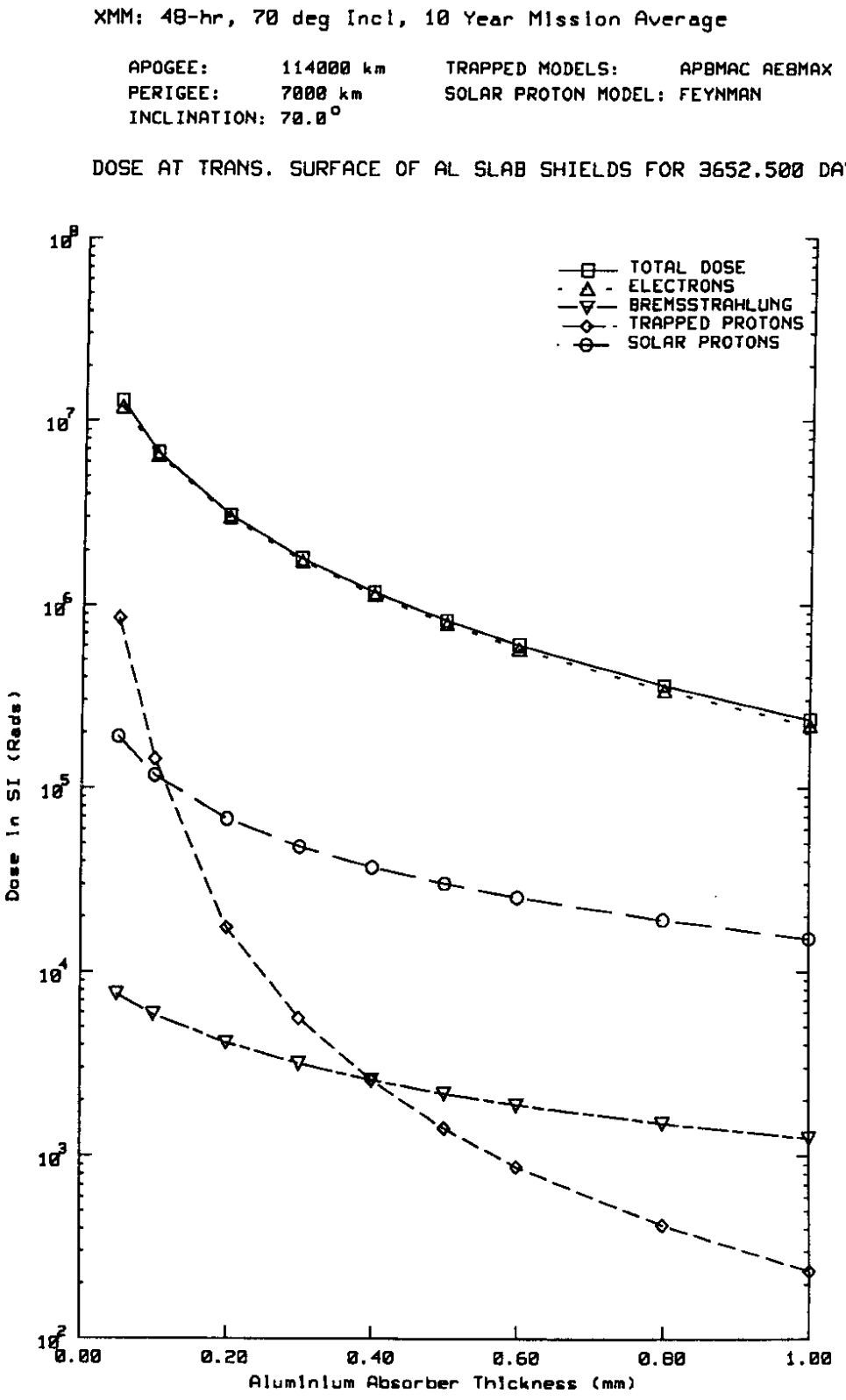
Technologies which use epitaxial silicon or silicon on insulator substrates are generally free of latch-up and are to be preferred (AD 1).

##### 4.1.2.4.4 Materials

Figure 4.1.2.4.4-1 shows the radiation dose/thickness curves for a flat surface with an exploded scale between 0 and 1 mm. These doses are not critical for most materials except for thin transparencies/windows and thermal finishes. The PI shall list all his

external surfaces (e.g. thermal blanket material, thermal paints, transparencies/windows, etc.) and demonstrate that any degradation of properties during the 10 year life is acceptable from a performance viewpoint.

**Figure 4.1.2.4.4-1: Total Plate Radiation Dose Against Depth of Material**



## 4.2 Science Observation Requirements

### 4.2.1 Observation Modes

There are two observation modes for the mission:

a) Pointing to a target fixed in inertial space for a minimum time of 1 hour and a maximum time of 1 orbit.
An average fixed pointing time will be 3 hours. All instruments will be switched on and collecting data during every pointing. The allocation of telemetry will depend on the type of target and the priority instrument for the observation. The instrument configuration will be dependent on the type of target and the priority instrument. The instrument configuration will be dependent on the type of observation but many configuration changes from one target to the next and within a single observation can be envisaged. As an example, these will include such general configuration issues as:

1. Filter wheel position/sequence
2. Calibration source sequence
3. CCD clock configuration
4. Data handling configuration.

b) Slewing between two targets.

The payload will remain switched on during slewing between targets. No reconfiguration of any kind of the payload will occur during such a slew.

### 4.2.2 Sky Coverage

To limit the heat input to the CCD radiators and the straylight input to the telescope, the following constraints apply:

- The line of sight of the telescope shall remain at 90þ ± 20þ away from the sun direction

- The roll angle will remain within 20°

- Mission planning will ensure that the line of sight of the telescope will not be closer than 45þ to the earth limb

Experiments shall, however, consider the violation of these constraints in the failure mode effects and criticality analysis.

The Experiments will receive commands, generated in the on-board data handling subsystems, in case the attitude constraints are violated.

### 4.2.3 Attitude Measurement and Pointing Performances

#### 4.2.3.1 Terminology

**Absolute Pointing Error**: **(APE)** The Absolute Pointing Error is the angular separation between the actual direction and the intended direction of an axis.

**Relative Pointing Error**: **(RPE)** Relative Pointing Error is the angular separation between the actual direction of an axis and a reference axis specified for a given time interval.

**Absolute Pointing Drift**: **(APD)** Absolute Pointing Drift is the change of the angular separation between the actual direction and the intended direction of an axis over a given period of time.

**Absolute Measurement Accuracy**: **(AMA)** Absolute Measurement Accuracy is the angular separation between the actual direction and the reconstructed direction (a posteriori) of an axis.

**Observation Target Coordinate Frame:**

This is a frame defining the actual orientation of the target (X-ray source) to be observed. It is inertially fixed in orientation for each observation period, regardless of the actual attitude of the spacecraft.

The observation target coordinate frame, SC-O-t, is defined as follows :

- its origin shall coincide with the origin of the functional coordinate frame to which it shall be associated
- the SC-X-t-axis coincides with a line from the centre of the observation target towards the frame origin
- the SC-Z-t-axis is such that the SC-XZ-t- plane and a plane defined by the SC-X-t-axis and the centre of the sun is separated by an angle specific to each observation, pointing positively towards the sun
- the SC-Y-t-axis completes the coordinate frame

**RGS Grating Assembly Unit Functional Coordinate Frame**

The Functional Coordinate Frame of each of the two RGS Grating Assembly Units, RGAi (1,2)-O-f, is defined as follows :

- the origin is located in the "G" point of the grating assembly unit
- the RGAi-X-f-axis is parallel to the x-axis of the Unit Coordinate System
- the RGAi-Z-f-axis is perpendicular to the line groves, pointing positively in the anti-dispersion direction
- the RGAi-Y-f-axis completes the coordinate frame

**RGS Focal Camera Unit Functional Coordinate Frame**

The Functional Coordinate Frame of each of the two RGS Focal Camera Units, RFCi(1,2)-O-f, is defined as follows :

- the origin is located in the centre of the reference pixel of CCD number 5 (M point)
- the RFCi-X-f-axis is perpendicular to the plane of the CCD containing the vertex, pointing positively into the CCD
- the RFCi-Z-f-axis coincides with the centre line of the CCD along the dispersion direction, pointing positively towards the decreasing dispersion direction (see EID-B, para 2.1)
- the RFCi-Y-f-axis completes the coordinate frame

**EPIC Boresight Coordinate Frame**

The Boresight Coordinate Frame of each of the three EPIC Focal Camera Units, EBi (1,2,3)-O-f is defined as follows :

- the origin is located in the vertex of the Focal Camera Unit (see relevant ICD in the EID-B)
- the EBi-X-f axis coincides with the boresight of the relevant Mirror Module
- the EBi-Z-f axis is parallel to the spoke # 1 of the relevant Mirror Module
- the EBi-Y-f axis completes the coordinate frame

**EPIC MOS and pn Functional Coordinate Frames**

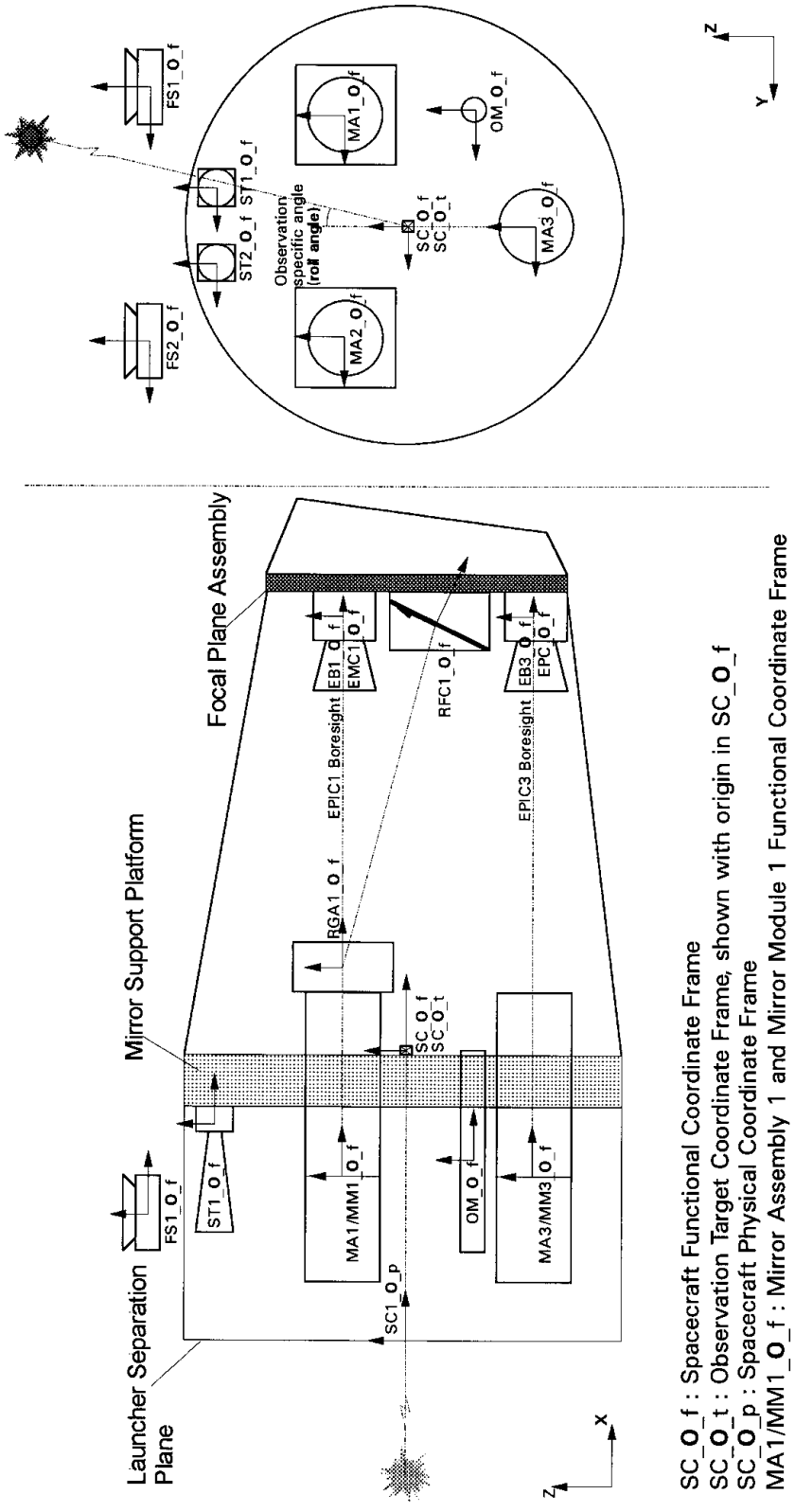
The Functional Coordinate Frame of each of the two EPIC MOS Focal Camera Units, EMCi (1,2)-O-f, is defined as follows :

- the origin is located in the vertex of the Focal Camera Unit (see relevant ICD in the EID-B)
- the EMCi-X-f axis is perpendicular to the plane of the CCD containing the vertex, pointing positively into the CCD
- the EMCi-Z-f axis is parallel to a side of the CCD, containing the vertex
- the EMCi-Y-f axis completes the coordinate frame

The Functional Coordinate Frame of the EPIC pn Focal Camera Unit, EPC-O-f, is defined as follows :

- the origin is located in the vertex of the Focal Camera Unit (see relevant ICD in the EID-B)
- the EPC-X-f axis is perpendicular to the plane of the CCD containing the vertex, pointing positively into the CCD
- the EPC-Z-f axis is parallel to side of the CCD, containing the vertex
- the EPC-Y-f axis completes the coordinate frame

**Figure 4.2.3.1-1 : Coordinate Frame Definitions**




SC_O_f : Spacecraft Functional Coordinate Frame
SC_O_t : Observation Target Coordinate Frame, shown with origin in SC_O_f
SC_O_p : Spacecraft Physical Coordinate Frame
MA1/MM1_O_f : Mirror Assembly 1 and Mirror Module 1 Functional Coordinate Frame
OM_O_f : Optical Monitor Functional Coordinate Frame
EMC1_O_f : EPIC MOS Camera Unit 1 Functional Coordinate Frame
EPC_O_f : EPIC PN Camera Unit Functional Coordinate Frame
EB1_O_f : EPIC Boresight 1 Functional Coordinate Frame
RGA1_O_f : RGS Grating Assembly Unit 1 Functional Coordinate Frame
RFC1_O_f : RGS Focal Camera Unit 1 Functional Coordinate Frame
ST1_O_f : Star Tracker 1 Functional Coordinate Frame
FS1_O_f : Fine Sun Sensor 1 Functional Coordinate Frame

#### 4.2.3.2 Pointing Performances

The pointing performance during the observational phase, for the observation target coordinate frame is specified below.

All figures are given for a confidence level of 95%, single axis rotation.

| | Y,Z | X |
|---|---|---|
| Absolute Pointing Error (APE) | 60" | 30' |
| Absolute Pointing Drift over 1 hour (APD)<br>Absolute Pointing Drift over 16 hours | 5"<br>45" | 6'<br>18' |
| Relative Pointing Error over 2 min (RPE) | 5" | 6' |
| Absolute Measurement Accuracy (AMA) | 10" | 6' |

**Table 6.5-1: System Pointing Performance for XMM**

Performances apply to periods without actuation of instrument mechanisms.

### 4.2.4 Slewing Performance

The slew rate is assumed to be þ 20þ/hour. The maximum number of slews per day will be 6. Stable pointing can be achieved within 10 minutes of target acquisition after a slew (including search and tracking of a new guidestar).

## 4.3 Payload Optical Performance

The in-orbit image quality of XMM over the nominal mission life in the centre of the EPIC camera field of view will be (half energy width):

- Better than 30 arcsec at 8 keV

- Better than 20 arcsec below 3 keV

The on-axis effective area of the XMM mirror module at the end of nominal mission life will be:

- Larger than 1475 $cm^2$ at 1,5 keV
- Larger than 580 $cm^2$ at 8 keV.

The grating units of RGS will vignet 60% of the X-ray beam from the mirror assembly.

## 5. DESIGN AND CONSTRUCTION REQUIREMENTS

### 5.1 Metric Standard

Drawings, specifications and engineering data shall use the Internation System (SI) Metric Standard. Dimensions shall be specified in millimetres and angles in degrees.

### 5.2 Workmanship

All design and manufacturing operations shall comply with the space industry standards and practices and shall comply with the process specifications and standards as defined in the document AD 1.

### 5.3 Failure Propagation

#### 5.3.1 Interface Failure

The equipment shall be designed in such a way that any failure cannot be propagated towards other equipment, through electrical harness or by thermal or mechanical effects.

#### 5.3.2 Internal Failure

The equipment shall be designed in such a way that any failure cannot be propagated inside the unit (e.g. to redundant chain).

#### 5.3.3 Redundant Failure

The equipment shall be designed in such a way that nominal and redundant functions are physically and electrically separated (i.e. both functions shall use separate electrical components).

### 5.4 Single Point Failure

The design shall avoid, as far as possible, single point failure. If single point failure exists, it shall be clearly identified together with all the necessary precautions required to minimise its consequences.

## 5.5 Mechanical Design

### 5.5.1 General Requirements

The instrument shall be designed to withstand the environment it will encounter during its lifetime maintaining nominal performance and without detrimental influence to the spacecraft or other instruments' performance. The mechanical loads induced by these environment include:

- Fabrication and assembly loads (e.g. welding, interference fitting)
- Handling and transportation loads
- Test loads (including thermal stresses)
- Launch loads including ascent depressurisation
- Attitude and orbit control induced loads
- Thermally induced loads.

### 5.5.2 Mechanical Safety Requirements

The spacecraft and its payload has to comply with the launcher safety regulations.

Whenever hazardous items are incorporated in the design of the instrument or are employed during testing or check-out, the experimenter has to provide appropriate documentation covering all safety-related activities: component choice, safety and warning devices, fault trees for catastrophic events, and, in general, all data enabling evaluation of the risk.

The following elements are considered hazardous items:

- Electro-explosive devices
- Pressurised systems
- Batteries
- Radioactive sources.

The instrument has to obtain approval for these items from the XMM Project Office. In order to obtain this approval, the Principal Investigator has to demonstrate that his equipment and its utilisation comply with the provisions of the Safety Regulations (AD 1).

### 5.5.3 Materials' Requirements

The selection of materials for the design of the instruments shall be performed by the PI under careful consideration of the lifetime requirements including 3 year ground life. In particular, the PI shall specifically address the material performance under long exposure to space environment and potential performance degradation over the lifetime due to the operational loads (creep, ageing effects, etc.).

The material performance data assumed in the design and the corresponding reference source shall be specified in the design verification documentation.

### 5.5.4 Stiffness Requirements

Each instrument unit shall have a first global resonance above 120 Hz in hard-mounted conditions.

Resonances of internal items (e.g. PCBs, CCD benches, etc.), should be higher than 200 Hz to avoid coupling with inputs from the launcher.

### 5.5.5 Strength Requirements

The payload structure shall be designed to withstand limit loads as defined in the table below.

The loads are to be applied as "worst combination" for any different part of a unit. Loads along the x axis can have any sign. Lateral loads, along the y and z axes, can have any direction, see Fig. 5.2.5.

| Item | Axial (g) | Lateral (g) |
|---|---|---|
| Focal Plane Platform mounted camera heads | 15 | 15 |
| Mirror Support Platform mounted instruments (OM telescope & RGS gratings) | 15 | 8 |
| Electronic Units | 25 | 25 |
| Focal Camera Radiators incl. supports | 50 g. in any direction | |

The following factors of safety on limit levels against the various failure modes are to be used:

Ultimate: 2.0 - against rupture or yield leading to loss of function

Yield: 1.5 - against permanent deformation

Buckling: 3.0 - against collapse, using realistic imperfections of the critical elements

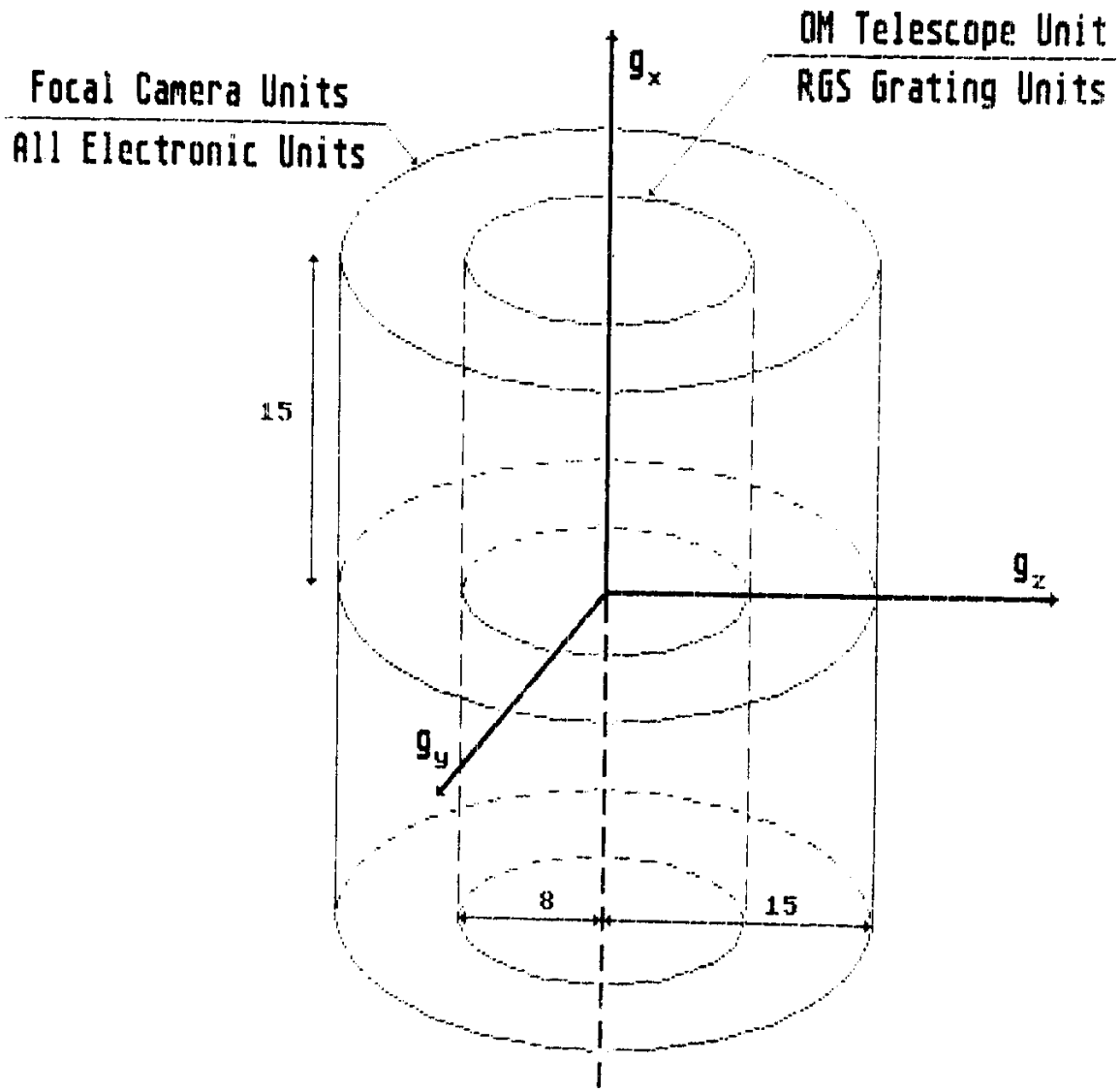
Ultimate: 3.0 - for non-metallic materials
3.0 - for joints

The design shall have a *positive* margin of safety which is defined as:

$$\left( \frac{\textit{Allowable load or stress}}{\textit{Limit load or stress x factor of safety}} \quad 1 \right) > 0$$

Note: Above mentioned levels cover the instrument global design. However, resonances can lead to important responses under random test conditions. This shall be covered by a local sizing for the specified random vibration test levels.

**Fig. 5.5.5: Loads' Envelope**



### 5.5.6 Unit Coordinate System Definition

The Unit Coordinate System ($O_u$, $X_u$, $Y_u$, $Z_u$) shall be fixed relative to the unit geometry. It shall be a right-handed, orthogonal coordinate system defined as follows:

- The origin ($O_u$) shall be located at the centre of the Reference Hole (see definition in para. 5.5.9) at the level of the mounting interface plane.

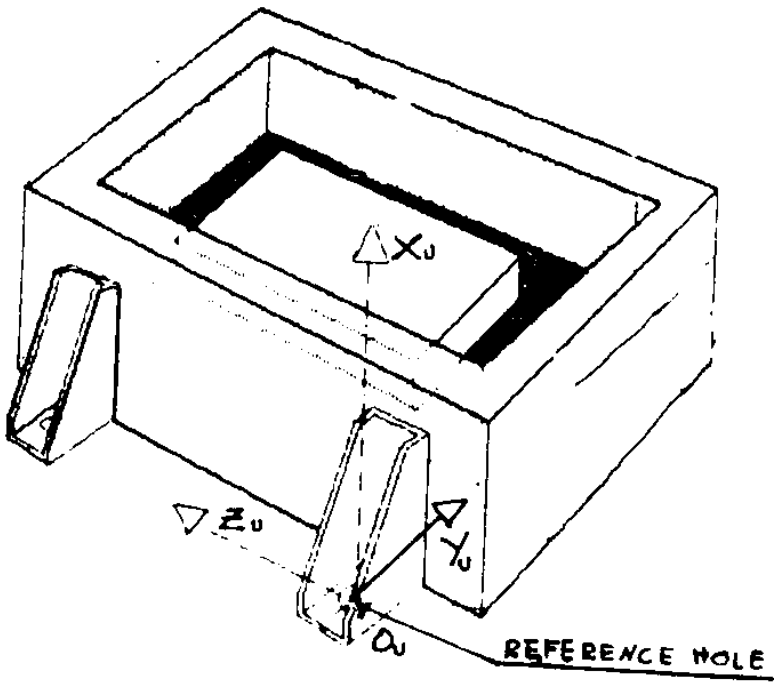
- The $X_u$ axis shall be perpendicular to the mounting interface plan, pointing positively towards the unit.

- The $Y_u$ and $Z_u$ axes shall be oriented such that the unit will be included inside the $+Y_u/+Z_u$ quadrant of the mounting interface plane. Moreover, if the unit has a rectangular shape, the $+Y_u$ and $+Z_u$ axes shall be parallel to the sides of the unit.

The Unit Coordinate System shall be clearly shown on the Interface Control Drawing (ICD) of the corresponding unit.

Fig. 5.5.6 illustrates the Instrument Unit Coordinate System.

**Fig. 5.5.6: Instrument Unit Coordinate System**



### 5.5.7 Mounting Requirements

#### 5.5.7.1 General Functional Requirements

The attachment points of the unit shall be designed to guarantee the compliance to the following general functional requirements:

- Controlled mechanical connections between the unit and the spacecraft throughout the ground and orbital lifetime of the satellite

- Ease of accessibility with standard tools to the attachment bolts during (de)integration of the unit of the spacecraft, or related covers and/or deployment of any unit-related device.

Where this cannot be fulfilled, the PI shall provide a kit of tools to allow proper torquing of the bolts from an accessible position.

Besides the mechanical design of the mounting attachments shall contribute to a proper thermal control of the unit, by taking into account the thermal loads encountered throughout the mission lifetime, and shall contribute to a good electrical bonding to the spacecraft structure.

#### 5.5.7.2 Standard Mounting Requirements

These requirements apply to units which do not require alignment.

Hard-mounted (i.e. fixed to the spacecraft by means of potted inserts), rectangular-shaped units whose mass is þ 2 kg shall not have more than 4 (four) attachment points (lugs).

Hard-mounted, rectangular shaped units, with a mass higher than 2 kg shall have at least 4 (four) lugs, whose relative distance shall comply with the requirement specified in Table 5.5.7.2.

Exact number of lugs and distance between them shall be agreed with the XMM Project Office on the basis of structural, dynamic and thermal control considerations; as a rule, one lug shall not carry more than 1,5 kg.

The design of the lugs for hard-mounted rectangular shaped units shall meet the general requirements specified in Table 5.5.7.2.

Fig. 5.5.7.2 illustrates the geometry parameters for the standard attachment lugs.

### 5.5.7.3 Special Mounting Requirements

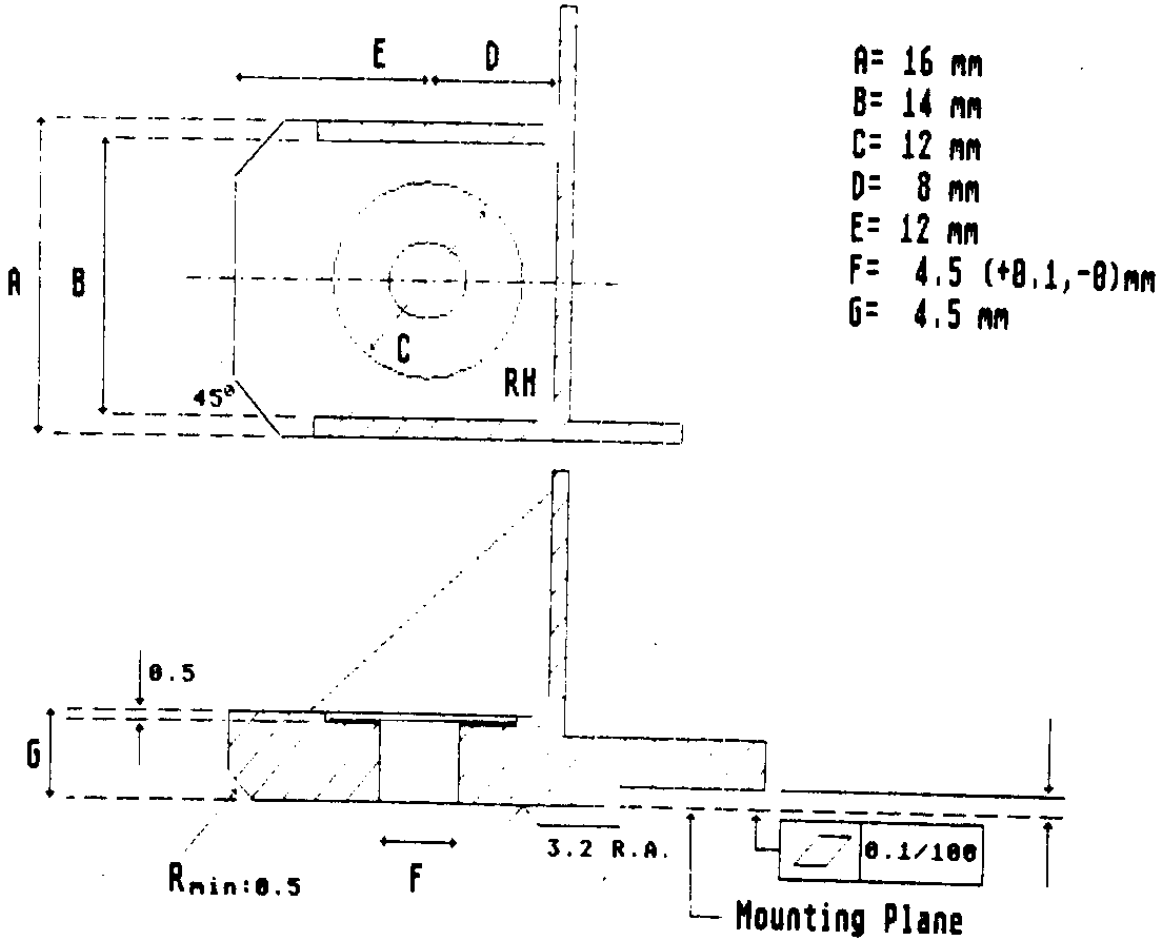
These requirements apply to units which require alignment during integration.
Since the final position of each focal point will only be determined after spacecraft integration, the PI shall provide sufficient adjustment capability of the detector position in all three axes. The PI shall provide means for fine adjustment in displacement/tilting angles.
If a tight mounting precision is required by the Principal Investigator, a request for higher positional precision will be considered on the basis of clear scientific/technical justification of the limitations and on the basis of system level investigation of the overall consequences of the requested mounting/alignment precision. It should be noted that the reflection grating box mounting plane will be separated from the central plane of the mirror module by TBD mm (reference value 600 mm). This value depends on the final design layout of the mirror platform and the mirror module support structure and will only be defined later in the programe.
A specific design for precise positioning and locking of the detector units, mounted on the FPP, and of the OM, mounted on the MSP, shall be proposed by the PIs and agreed with the XMM Project Team.

**Table 5.5.7.2**

| | | |
|---|---|---|
| Attachment bolt | M4 | (1) |
| Attachment bolt material | Stainless Steel (TBC) | |
| Bolt torque (T) | 2.0 þ T þ 2.5 Nm | |
| Washer dimensions | Typical Diameter: 9.0 mm<br>Typical Thickness: 0.8 mm | (1) |
| Attachment hole diameter | 4.5 (+0.1, -0.0) mm | |
| Distance (d) between attachment holes (lugs) | Typically 100 mm | |
| Tolerance distance between centre of attachment holes (w.r.t reference hole, R) | þþþþþþþþþþþþþ<br>þ þ þØ 0.2 þ R þ<br>þþþþþþþþþþþþþ | |
| Diameter of Spot Face Area for bolt head and washer | 12 mm | (2) |
| Attachment lugs: | | |
| - dimensions | See Fig. 5.5.7.2 | (2) |
| - roughness | þ 3.2 microns | |
| - flatness | þ 0.1/100 mm | |
| - coplanarity | þ 0.1/100 mm | |
| - edge radius | þ 0.5 mm | |
| Minimum distance between centre of attachment hole and unit side wall | þ 8.0 mm | |
| Clearance between unit baseplate and attachment lugs | þ 1.0 mm | |

(1) Spacecraft provided
(2) The lug's footprint in contact with the unit baseplate and the spot face area shall not be painted, but only treated with a surface finish.

**Fig. 5.5.7.2: Standard Attachment Lug Geometry**



### 5.5.8 Mechanisms Requirements

#### 5.5.8.1 General Requirements

Mechanisms shall be designed to meet the mechanical and electrical interfaces and performance requirements. They shall withstand the predicted environment during handling, transportation, testing, storage, launch and orbital lifetime without losing nominal performance.

Structural parts of mechanisms shall be designed with the same criteria, loads and safety factors as all other structural items.

Redundancy concepts shall be selected to minimise single point failures and to optimise reliability for the mission and life requirements.

Release and latching devices shall be redundant to assure positive opening and locking capabilities in the mechanism. Failure of a mechanisms shall not cause obscuration of the field of view of a mirror module or another instrument.

Mechanisms' materials, parts and components shall be selected according to the ESA approved qualification procedures and they shall be compatible with the requirements specified in the relevant applicable documents. The PI shall specifically address their performances under long exposure to space environment and potential performance degradation over the ground life.

Mechanisms shall be compatible with the EMC and cleanliness requirements.
The mechanism design shall be compatible with operations in ambient and thermal vacuum conditions. They shall be representatively tested on ground according to the mechanism verification test programme.

#### 5.5.8.2 Lifetime Requirements

The lifetime of a mechanism shall be demonstrated by test in the appropriate environment, using the sum of the prediced 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 Prediced Cycles**

| | |
|---|---|
| Ground Testing | x4 |
| - number of on-ground test cycles (the minimum number to be used is 10) | |
| In-orbit predicted cycles: | |
| - 1 to 10 actuations | x10 |
| - 11 to 1000 actuations | x4 |
| - 1001 to 100000 actuations | x2 |
| - >100000 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.

Example:

- predicted ground test cycles: 15 — 15 x 4 = 60
- predicted in-orbit actuations: 100
  - þ 10 actuations — 10 x 10 = 100
  - þ 90 actuations — 90 x 4 = 360

Total test number = 520

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

**5.5.8.3 Torque Margin Requirements**
Actuators (electrical, mechanical, thermal and other) shall be sized to provide throughout the operational lifetime actuation torques/forces which are at least 2 times the combined worst case resistance torques/forces predicted. The following minimum factors shall be applied in the design to the resistance components:

| | | | |
|---|---|---|---|
| Inertia | (I) | 1.1 | |
| Spring | (S) | 1.2 | |
| Friction | (F) | 3.0 | # |
| Hysteresis | (Hy) | 3.0 | # |
| Others (e.g. Harness) | (Ha) | 3.0 | # |

Hence the required torque/force, in worst case, shall be defined by the equation:

$$T \geq 2.0 \times (1.1 \times I + 1.2 \times S + 3.0 \times F + 3.0 \times Hy + 3.0 \times Ha)$$

The factors marked by # may be reduced to 1.5 providing that the worst case measured torque/force resistant components to which they refer are determined by measurement according to an ESA approved test procedure.

**5.5.8.4 Pyrotechnic Devices**
All pyrotechnical and other one shot devices (e.g. thermal knives, memory metal actuators, etc.), 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 fulfil to the requirements of AD 7.

Any proposed pyrotechnic actuator must have been qualified to the levels defined by XMM System Requirements and all materials and processes used shall meet the requirement of AD 1.

As a minimum, properties such as strength, shock impulse, redundancy, performance output, sealing, and operation at temperature extremes under vacuum must have been demonstrated.

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.

#### 5.5.8.5 Pyrotechnic Interfaces

All Electro-Explosive Devices (EEDs) required by an instrument will be initiated via a spacecraft dedicated pyrotechnic unit. This unit will incorporate the inhibits compatible with the ARIANE safety requirements for hazardous devices.

The electrical characteristics of the qualified initiators shall meet the requirements specified in para. 5.8.3.

### 5.5.9 Reference Hole Definition

One of the attachment holes of the unit shall be chosen by the PI as the Reference Hole which shall be identified by an engraved letter "R" on the unit.

The Reference Hole shall be clearly indicated also on the Interface Control Drawing.

All the interface dimensions of the unit shall be specified with respect to the Unit Coordinate System, whose origin shall lie at the centre of the Reference Hole at the level of the mounting place between the unit and the spacecraft.

### 5.5.10 Grounding Point Requirements

All the units shall be equipped with a grounding stud which will ensure the correct electrical bonding of the unit to the spacecraft structure.

A stainless steel stud (M4 x 6) shall be located on the unit, 20 - 30 mm above the mounting plane.

The stud shall be easily accessible when the unit is integrated; a torque of 3 Nm will be applied during integration.

### 5.5.11 Handling Provisions Requirements

Units weighing more than 10 kg shall be equipped with handling points (e.g. threaded bushes) which will enable the connection to special handles, provided by the PIs, for helping 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, to make them doubly protected).

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

### 5.5.12 Aperture Covers Requirements

Units that incorporate a detector aperture shall be delivered with a metallic dust/protective cover over the aperture.

These covers shall be either removable before flight or be part of the instrument and be deployed/retracted in-orbit.

#### 5.5.12.1 Removable Covers (Non-Flight Items)

All removable covers shall be:

- Normally removed during system test where flight configuration is mandatory, i.e. thermal vacuum testing or vibration testing

- Accompanied by a detailed procedure for their removal by ESA Project personnel or their Contractors during integration

- 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. They shall also be clearly marked as non-flight items on the relevant Interface Control Drawing.

#### 5.5.12.2 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 as a mechanism and the deployment/retraction mechanism shall meet the design requirements of Section 5.5.8.

The variations of C.o.G. and M.o.I. of the experiment produced by the deployment shall be minimised.

The type of cover deployment used shall take into account the field of view and other requirements of adjacent instruments and/or units and it shall therefore be agreed with the XMM Project Team during the design phase.

### 5.5.13 Accessibility Requirements

The design of the units and the positioning of the relative attachments points to the spacecraft shall be such to provide sufficient accessibility to enable the mounting and removal of the units themselves, the related covers and alignment devices with normal standard tools.

Where this requirement cannot be fulfilled, the PI shall provide a kit of tools as part of the instrument MGSE, such that the mounting bolts can be torques from an accessible position.

### 5.5.14 Physical Properties

#### 5.5.14.1 Mass
The mass allocated by the Project to each instrument, shall include the total experiment hardware that is intended for flight.

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

Such STM/QM and FM estimated/measured masses shall be, in any case, less than the allocated mass.

In order to ensure interchangeability between the FM and the respective FS units, the *measured* mass of each FS unit shall be within 0.1 kg or 1% of the measured mass of the FM, whichever is larger.

#### 5.5.14.3 Allocated Mass
The maximum allowed masses for each instrument are specified in the EID-B.

#### 5.5.14.4 Centre of Gravity
The estimated Centre of Gravity (CoG) coordinates of each deliverable unit shall be computed and specified with respect to the Unit Coordinate System.

Any variation of CoG coordinates due to consumables or appendages deployment shall be reported to the Project.

The CoG values of the Instrument interconnecting harness, which are considered themselves separate deliverable units, need not be computed.

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

#### 5.5.14.5 Centre of Gravity Tolerance Requirements
The tolerance between the measured CoG coordinates of each STM/QM and FM unit and the respective estimated coordinates, specified by the PI on the relative Interface Control Drawing, current at the time of the STM/QM and FM delivery to the Project, shall be within a sphere of 5 mm radius.

In order to ensure free interchangeability between the FM and the respective FS units, the *measured* CoG coordinates of the FS unit shall be within a sphere of 5 mm with respect to the FM *measured* CoG coordinates.

**5.5.14.6 Moments of Inertia (MoI)**
The estimated Moments of Inertia of each deliverable unit (excluding non-flight items) shall be specified with respect to a coordinate system parallel to the Unit Coordinate System and with its origin in the CoG of the unit.

Any variation of MoIs due to consumables or appendages deployment shall be calculated.

MoIs of harnesses need not be specified.

**5.5.14.7 Moments of Inertia - Tolerance Requirements**
The difference between the measured MoIs of each STM/QM and FM unit and the MoIs, specified on the Interface Control Drawing, current at the time of the STM/QM and FM delivery to the Project, shall be less than 10%.

In order to ensure interchangeability between the FM and the FS units, the *measured* MoIs of the FS units shall differ less than 10% from the *measured* MoIs of the FMs.

**5.5.14.8 Change of Mass Properties**
No movement of an instrument mechanism or combination of movements of instruments mechanisms shall change the CoG coordinates of the spacecraft by more than ± 1 cm, during all mission phases.

No movement of an instrument mechanism or combination of movements of instruments mechanisms shall change the principal MoI values of the spacecraft by more than 20 kg $m^2$, during all mission phases.

The maximum torque impulse applied to the spacecraft during movements of an instrument mechanism or combination of movements of instruments mechanisms shall not exceed 0.02 Nm $s^{-1}$.

**5.5.14.9 Unit Dimensions and Tolerances Requirements**
Each unit shall lie inside the respective envelope specified by the Project, and their overall dimensions, d, shall be specified to a tolerance smaller than:

| | |
|---|---|
| + 0.5/-0.0 mm | for d < 500 mm |
| + 1.0/-0.0 mm | for 500 mm < d < 2000 mm |

### 5.5.15 Unit and Connector Identification

Each unit shall be unambiguously identified by a Unit Identification Code.

This code shall be either engraved directly onto the unit or onto a Unit Identification Label, which shall then be rivetted or glued onto the unit.

The surface treatment and the thermal coating of the label shall be in line with the related general requirements.

Its location shall be such to render it clearly visible (legible with an unaided eye from a distance of 0.5 m) during or after the integration of the unit on the spacecraft (preferably on the top of the unit).

The location of the Unit Identification Code shall be shown on the Interface Control Drawing (ICD).

#### 5.5.15.1 Unit Identification Code Definition

The code shall be alphanumeric, consisting of the experiment acronym (EPIC, OM, RGS) and a letter/number combination.

The last two characters shall identify the MODEL of the experiment unit and shall be as follows:

- SM for the Structural Thermal Model
- QM for the Qualification Model
- FM for the Flight Model
- FS for the Flight Spare Model.

Identical units within one model category shall be distinguished by a serial number.

The complete list of units and their codes shall be part of the EID-B.

#### 5.5.15.2 Connector Identification Code Definition

Each connector unit shall be identified by a Connector Identification Code.

The code shall be either engraved directly onto the experiment unit or onto a Connector Identification Label in aluminium which shall then be attached onto the experiment unit.

The code shall be adjacent to the connector, but avoid mechanical interference and/or visual obstruction with connector back shells.

The surface treatment and thermal coating on the aluminium label shall be in line with the related general requirements.

The location of each connector identification code shall be shown on the Interface Control Drawing (ICD).

The code shall be alphanumeric consisting of 3 (three) characters.

The first character shall identify whether the connector is a unit fixed (hard mounted) connector or a harness mounted connector. In the former case the first character shall be "J", in the latter a "P".

The last two digits shall identify the connector NUMBER and shall be a two digit sequential number, starting from "01". Each unit shall be treated individually in this respect (i.e. the connector identification code of each unit shall start at "J01" for unit hard mounted connectors and the corresponding mating harness mounted connector shall start at "P01").

This rule for the identification of harness mounted connectors shall be applied to both spacecraft or experiment supplied interconnecting harness or EGSE, should connection to test connectors on the unit be necessary.

The Connector Identification Code alone is insufficient to identify unambiguously a unit connector. In general, for "full" identification of a connector, the code shall be preceded by the Unit Identification Code (example, connector "J03" of the EPIC MOS CCD analogue electronic unit will be identified by the full code "EPIC 2A J03").

For unit fixed connectors the "full" identification code is unnecessary, as the unit identification code is already specified on the unit.

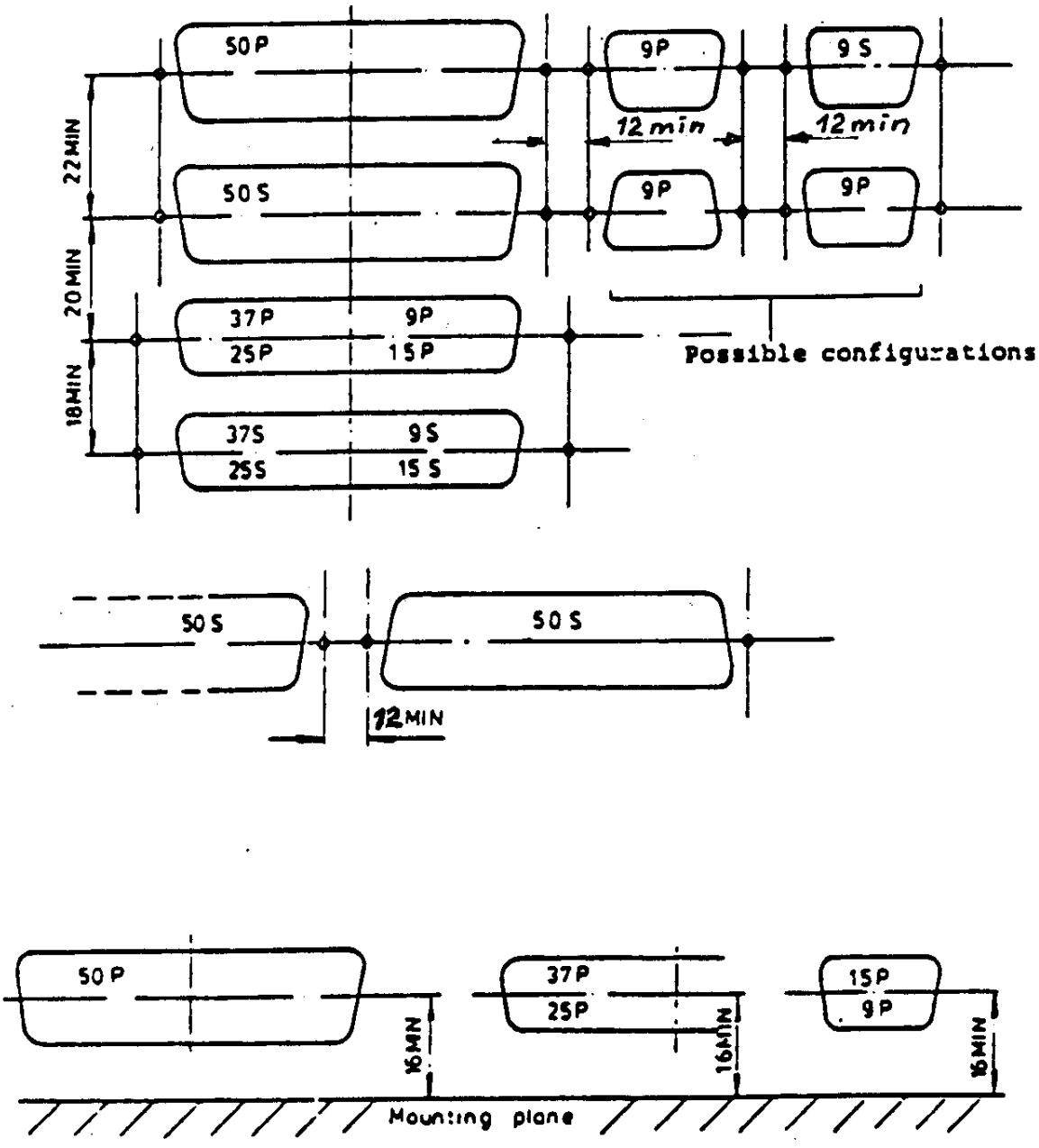
#### 5.5.15.3 Connector Mounting

Equipment and bracket mounted connectors shall be located in easily accessible positions. The physical position is to be indicated on the Interface Control Drawings and must be compliant with the minimum distances between connectors and mounting plane as described in Fig. 5.2.15.3.

Mechanical methods in conjunction with identification markings shall be employed to prevent incorrect mating of connectors. Colour coding may be used as an aid to identification.

Where mechanical methods cannot be used, or visual identification is impossible, an electrical continuity loop may be envisaged to establish correct mating.

**Fig. 5.2.15.3: Connectors Mounting Requirements**



**5.5.16 Interface Control Drawing (ICD)**

For each unit an Interface Control Drawing is required in order to establish the interfaces with the spacecraft structure, harness and Ground Support Equipment (GSE).

All Interface Control Drawings shall be drawn in 1st angle projection and unit the INTERNATIONAL SYSTEM (SI) Metric Standard. Dimensions shall, however, be specified in millimetres and the angles in degrees.

The Interface Control Drawing shall be 1:1 scale, unless otherwise agreed by ESA and it shall be drawn on tracing paper, if necessary on more than one sheet.

The Interface Control Drawing shall clearly identify the unit it represents and the responsible authority. It shall bear a proper number, which shall uniquely identify the unit and it shall be subject to a properly controlled issue/revision updating system, with the date and signature of the experiment developer.

Each issue/revision of the drawing shall specify, on the drawing itself, or on an accompanying sheet, a list of all the changes which have been incorporated since the previous revision.

The Interface Control Drawing shall contain, as a minimum, the following unit interface/configuration data characteristics:

- Identification of the Unit Coordinate System
- Identification of the Reference Hole
- Shape, dimensions (including thickness of external walls) and associated general tolerances
- Mechanical envelope necessary for: integration, articulation and check-out of extendable/deployable appendages of the unit (mechanisms actuation, door opening), where applicable.
- Estimated mass at the time the drawing is issued (cfr. "estimated mass" definition in ANNEX 1).
- Centre of Gravity (CoG) coordinates in launch configuration and relative tolerances specified with respect to the unit coordinate system. Variation of the CoG, if extendable/deployable devices and/or consumable are used.
- Moment of Inertia (MoI) values and relative tolerances computed with respect to the coordinate system parallel to the unit coordinate system and with its origin in the CoG. Variation of the MoI values if extendable/deployable devices and/or consumable are used
- Unit material(s) (e.g. housing, shielding, screws, washer, etc.).
- Mounting screws/washers data (e.g. type, dimension, torque, etc.) if provided by the PI

- Mounting hole pattern, hole diameter, centring tolerance, cut out area, dimensions and thickness of each mounting foot
- Spotfaced area for seating of the mounting screw washers where applicable
- Dimensions and location of dowel pins, where applicable.
- Location of Unit Identification Label and Connector Identification Labels
- External harness fixation devices, where applicable: types and locations.
- Identification and specification of the total contact area with the mounting plane (mounting feet and/or baseplate).
- Location, type and function of all connectors.
- Maximum allowed torque for connector fixation screws.
- Units interconnecting harness: type, preferred routing, length, mass.
- Roughness of contact area
- Flatness of contact area.
- Coplanarity of mounting feet.
- Identification of non-flight hardware
- Location of and interface mechanical mounting for thermal straps, where applicable.
- Identification of purging interfaces, where applicable.
- External surface finish.
- External thermal finish.
- Thermal properties of external surfaces (i.e. emissivity absorptivity).
- Location of venting holes, where applicable.
- Heater(s) location, where applicable.
- Blanket(s) location and performance.
- Spacecraft powered thermistor location.
- Mean specific heat.
- Details of any special thermal requirements, where applicable.

- Optical entrance aperture interfaces.
- Field of view, where applicable (origin and viewing direction).
- Location of any alignment device and specification of its reference plane, where applicable.

Additional peculiar unit interface characteristics not covered by the above list may be included on the drawing upon ESA agreement.

## 5.6 Optical Design

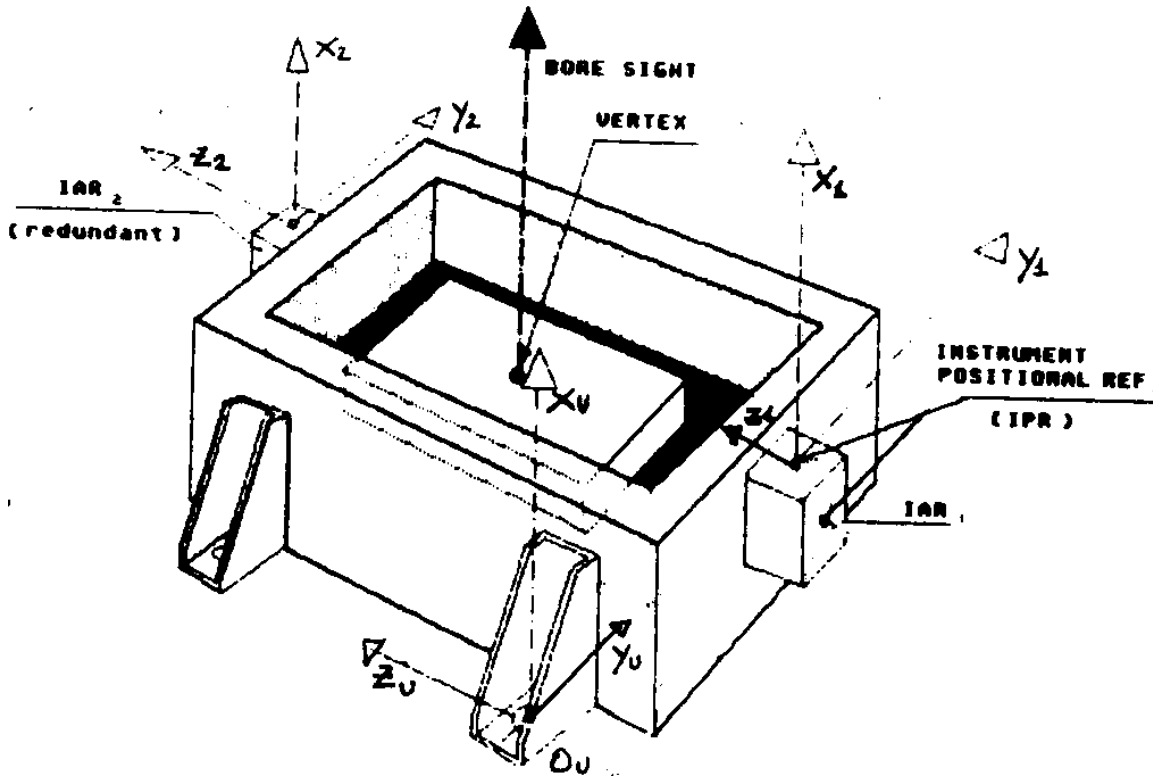
### 5.6.1 Instruments Alignment

#### 5.6.1.1 Field of View, Boresight and Vertex Requirements

Where relevant, each instrument shall specify its field of view as a cone referenced to the unit coordinate system. The requirements on the field of view shall, if necessary, consider different types of limitations for different zones of the field of view, clearly justifying the scientific/technical bases for the different limitations. In some zones, for instance, the requirement could be absence of other bodies, in a second zone the requirements could be limited to absence of highly reflective bodies (e.g. for the Optical Monitor); in a third zone the requirement could prohibit the presence of specular reflecting surfaces and so on (e.g. for CCD radiator plates).

The EPIC vertex is a three dimensional reference position inside the focal camera unit which has to be positioned in the actual on-axis focus of the mirror module.

**Fig. 5.6.1.1: Boresight, Vertex, IAR, UCS**



#### 5.6.1.2 Optical Reference Requirements

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

##### 5.6.1.2.1 Angular Reference Requirements

Whenever an angular reference is necessary for alignment measurement purposes, two mirror cubes shall be mounted. The normal to each of the reference surfaces shall be parallel to the boresight axis. The minimum size of its faces shall be 15 x 15 mm and the quality of its surface compatible with autocollimation techniques (planarity better than lambda /4).

Attention is drawn to the criticality of this reference hardware, and therefore to the reliability of its fastening and the possibility of providing a redundant reference (optical or mechanical).

The instrument reference cubes shall be mounted on a fixed part of the instrument structure. Their positions shall be agreed with the XMM Project Office in order to account for System level integration constraints. The position with respect to the internal detectors shall be determined with an accuracy of 0.1 mm.

##### 5.6.1.2.2 Positional Reference Requirements

Whenever an optical reference for positional alignment is required, a cross shall be marked on the alignment cube reference surface.

#### 5.6.1.3 Internal Alignment Requirements

##### 5.6.1.3.1 Boresight Requirements

The instrument boresight shall be determined during the instrument level AIV activities and shall be referred to two reference systems:

1. **The Instrument Angular Reference (IAR)**

   Two IAR mirror cubes mounted externally to the instrument shall be the only reference used for alignment after instrument delivery.

   The instrument logbook shall contain the calibration bias angles between the boresight and the IAR axes ($X_i$, $Y_i$, $Z_i$) with tolerances, to be used during System level alignment to ensure boresight positioning within the required tolerances. The bias angles shall be minimised and expected tolerances shall be known at an early stage.

2. **The Unit Coordinate System (UCS)**

The magnitudes of the bias angles between instrument boresight and the UCS ($X_u$, $Y_u$, $Z_u$) directly influence the range of adjustment of the instrument, i.e. the greater the bias angles allowed between boresight and the Unit Coordinate System, the greater the range of adjustment necessary to bring the boresight parallel with an external reference.

These bias angles can be minimised (to TBD values) by alignment adjustment capabilities built within the experiment.

These values and the means implemented in the experiment design to compensate for them (i.e. shims, screws, eccentrics, etc.), shall be agreed with the XMM Project Office.

#### 5.6.1.3.2 Vertex Requirements

The instrument vertex shall be determined during instrument AIV activities and shall be referred to two reference systems.

1. **The Instrument Positional Reference (IPR)**

The IPR shall be marked on the two IAR reference cubes and shall be the only reference used for positional alignment after instrument delivery.

The instrument logbook shall state the relative positions of the vertex and the IPR.

2. **The Origin ($O_u$) of the Unit Coordinate System**

The magnitude of the error between the nominal positions of the instrument vertex and $O_u$ directly influence the *range* of adjustment of the instrument, i.e. the greater the error allowed between vertex and $O_u$, the greater the range of adjustment necessary to bring the vertex in the correct position with respect to the external reference.

The maximum allowed errors and the means of adjusting the vertex position shall be agreed with the XMM Project Office.

#### 5.6.1.4 Instrument Alignment at System Level

TBD.

### 5.6.2 Straylight Requirements

The straylight requirements shall be specified in the EID-B.

## 5.7 Thermal Design

### 5.7.1 Introduction

The equipments shall be designed to withstand the temperature extremes which may be experienced during transportation, testing and which are expected during launch and orbital life.

In order to achieve compatibility of unit design with the satellite thermal control, each unit shall be designed according to the following requirements:

- To withstand the unit qualification tests

- To comply with operating, non-operating, minimum start-up and qualification level temperatures.

Each unit shall be designed so that all internal heat sources have the required insulation paths or heat flow to the external surfaces of the unit, when the average temperature of these surfaces is within the applicable temperature range.

The temperature distribution within the unit and hot spots on the external housing due to the point heat sources are the responsibility of the PI.

The PI is responsible for the analysis of this internal temperature distribution.

In designing the unit and ascertaining the optimum heat flow paths, the PI shall take due account of the mounting method and the exchanges with the environment by both conduction and radiation.

The unit design shall be such that the following conditions are respected for the junction temperatures of all solid state devices and for temperatures of other components with similar limitations:

- These temperatures must be below the associated maximum rated value when the unit is at the maximum acceptance temperature limit defined in the unit specifications

- These temperatures must not exceed the associated maximum derated value when the equipment is at maximum in orbit operating temperature.

The PI shall carry out a thermal analysis of the unit in sufficient detail to demonstrate that the above two requirements are met for all thermally critical components, even under single failure conditions.

Geometrical, thermo-optical and thermal properties, contact area, dissipated power for different modes shall be defined in EID Part B.

Thermal elements for which dismountability is required during integration and verification shall be removable without degradation of their thermal characteristics.

The method of application of any thermal coating shall be such that in the event of damage occurring to any thermal surface, the re-work/repair could be carried out in situ.

### 5.7.2 Temperature Reference Point Definition

The temperature reference point of each unit shall coincide with its reference hole. All temperatures are defined for this point (test or in orbit).

### 5.7.3 Temperature Definitions

**Operating Temperature**

This temperature range will be maintained whilst the instrument is operating. This temperature shall be independent of instrument operating modes.

**Non-Operating Temperature**

This temperature range will be maintained when the instrument is not operating. This temperature will be maintained on-ground and during flight.

**Switch-On Temperature**

The minimal temperature that the unit shall withstand without permanent damage when power is supplied.

**Temperature Range**

Temperatures given in the table below are the temperatures which the units may experience in flight.

| Unit | Operating (deg. C) | Non-Operating (deg. C) | Switch-On (deg. C) |
|---|---|---|---|
| All | -10 +40 | -25 +55 | -25 |

**Table 5.7.3: Temperature Ranges for Payload Units**

### 5.7.4 Temperature Measurement

The PI may be requested to incorporate into the instrument, temperature sensors which are spacecraft powered and included in the spacecraft housekeeping data, to provide essential data for spacecraft/payload thermal operational control under all conditions of payload status. The positions and quality of these sensors shall be agreed on a case-by-case basis with the XMM Project Office. Conditioning of these sensors shall be the responsibility of the spacecraft.
In addition, temperature monitoring of selected points within an instrument shall be provided by the PI and embedded in the instrument housekeeping data to cover the following cases:

a) Instrument health and safety monitoring
b) Instrument operational temperature and performance monitoring.

### 5.7.5 Thermal Coatings and Finish

These shall be selected from the advised list of Coatings and Finishes. All external surfaces shall meet the EMC requirements. Specularly reflecting external surfaces shall be avoided where possible.

As a general rule, external thermal coating shall have a high emissivity (þ > 0.8), unless otherwise specified. Use of chemglaze Z306 paint is strongly recommended.

As a general rule, no surface treatment is necessary for stainless steel, beryllium, fibreglass or carbon fibre, except for the needs of thermal control (if applied to external surfaces). Any other material shall be surface treated.

Surface treatment of materials must be approved.

Cadmium plating is not acceptable.

Metal couples with a maximum of 0.25 V difference in general - or 0.5 V in controlled airtight atmosphere - in electrochemical potential are permitted.

### 5.7.6 Radiators

Heat rejection from detectors containing cold sensors shall be by means of dedicated radiators forming part of the focal camera units.

### 5.7.7 Simplified Thermal Schematic

In addition to a Thermal Mathematical Model (ref. para. 6.2.2) for their detector units and the gratings, the PI shall provide in the EID-B simplified thermal schematics of the camera heads and the grating showing the main links between the nodes, heat inputs, conductive and/or radiative links and values, between nodes and with the spacecraft.

## 5.8 Electrical Design

### 5.8.1 Power Interface

#### 5.8.1.1 Spacecraft Power Supply Main Characteristics

The spacecraft power supply subsystem will provide:

- A main bus transferring the spacecraft regulated solar array power and/or the battery power to the users
- The distribution and switching of power to the users
- The distribution of a low power "keep alive" voltage line to the users.

#### 5.8.1.2 Power Supply Voltages

The instruments shall be designed to operate with nominal performance within the following steady state voltage limits:

| Line | Minimum Voltage | Maximum Voltage |
|---|---|---|
| Power Bus | 26 V | 30 V |
| Keep Alive Line | 7 V | 9 V |

**Table 5.8.1.2: Equipment Operating Voltage Limits**

In addition, all the users of these power lines shall safety survive any standing or fluctuating voltage in the full range 0 V to maximum voltages defined in Table 5.8.1.2.

In particular, equipment shall survive an instantaneous short circuit occurring on the external power line.

#### 5.8.1.3 Power Distribution

Power is switched and distributed to instruments on protected output lines.
Power is available for the full orbit except during eclipses.

#### 5.8.1.4 "Keep Alive" Line (KAL)

This line is constantly powered independent of the power bus and is isolated from ground on the source side. Failure of this line shall not result in the loss of essential instrument functions.
The maximum power per line shall not exceed 1 Watt. A total of 10 Watts is available for experiments.

#### 5.8.1.5 Bus Characteristics and Requirements

##### 5.8.1.5.1 Bus Output Impedance

The bus impedance is specified in the chapter 6.5.2.

##### 5.8.1.5.2 Voltage Transients and Ripples

The equipment shall operate with nominal performance when subject to the perturbations specified in Section 5.9.4.1.

##### 5.8.1.5.3 Average Power Demand

The average power is defined for an equipment as the maximum average power drawn from its dedicated power lines in the worst case conditions of Table 5.8.1.2.

The maximum average is defined as the average during a period of 5 minutes shifted to any point in time where this average will yield a maximum and does not include peak power defined hereafter.

##### 5.8.1.5.4 Peak Power Demand

Peak power demand is defined for an experiment as the maximum drawn from its power lines, in the worst case conditions of Table 5.8.1.2.

a) **Long Peak**

To be defined as a long peak, the power demand shall last less than 5 minutes per orbit (cumulated duration of individual peaks if any) and more than 100 ms.

The peak value is defined as the integral mean during a period of 100 ms shifted to any point in time where the integral will yield a maximum.

b) **Short Peak**

To be defined as a short peak, the power demand shall last less than 100 ms.

The peak value is defined as the integral mean during a period of 1 ms shifted to any point in time where the integral will yield the maxmimum.

#### 5.8.1.5.5 Inrush Current

Inrush current shall be limited to 1.5 times the peak value, specified in the EID-B.

The duration of the inrush peak shall not exceed 0.5 msec. The rate of change of the inrush current shall not exceed 1A/þsec.

#### 5.8.1.5.6 Load Transients

Voltage transients on the power supply lines generated during nominal mode changes or switching on/off shall comply with the transient requirements outlined in para. 5.9.4.1.1.

#### 5.8.1.5.7 Initial Electrical Status

After being switched OFF during a minimum of 10 sec and when switch on, equipments shall have an initial electrical status (except for latching relays if used), which is reproducible and identified in the EID Part B.

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

#### 5.8.1.5.8 Experiment Converters Synchronisation

Experiment converters are required to be synchronised by a spacecraft supplied clock with a frequency of 131.072 KHz ± 0.1% and operate at half the synchronisation frequency.

In the absence of a clock signal the converters shall be free running at nominal frequency 65.536 KHz ± 10%.

The synchronisation signal (squarewave, 5 V) is provided ground free (transformer coupled).

The experiment input impedance is assumed to be 5 k ohm in parallel to 200 pF.

### 5.8.2 Overall Power Allocation

The power allocated to each instrument is specified in the EID-B.

### 5.8.3 Pyrotechnic Interfaces

All Electro-Explosive Devices (EEDs) required by an instrument shall be initiated via a spacecraft dedicated pyrotechnic unit.

The electrical characteristics of these devices shall meet the following requirements:

- þ Firing command pulse duration: 22 ± 4 ms
- þ Firing current per filament: $3.5\ A < I_f < 4.0\ A$
- þ Max. no fire current: 1 A for 5 min.

þ Input resistance (bridgewire + experiment harness): $R_{in}$ < 1.3 Ohm

þ Isolation resistance between filaments and EED case after and before firing: > 100 M Ohm at 500 $V_{dc}$ for 30 s.

The design shall provide good accessibility for installation and removal of pyro initiators and charges without requiring the removal of other equipment, and can be performed at the launch site within less than one hour.

### 5.8.4 On-Board Data Handling

#### 5.8.4.1 Introduction

The acquisition and telecommanding for the spacecraft and its payload is based on the standard ESA On-Board Data Handling (OBDH) system. Telecommanding (i.e. uplink) is based on AD 05 which comprises the command packet service.

Telemetry (i.e. downlink) is based on AD 04, which comprises the telemetry source packet service.

Telemetry and Telecommand with the experiments is via DBU's connected to the OBDH bus.

Telemetry and telecommand packet types and data structures are specified in AD 8. The protocol used to exchange these packets between the OBDH and the experiments is specified in AD 11.

#### 5.8.4.2 Overall Data Handling Concept

The entire data acquisition shall be performed with telemetry source packets acquired from each instrument and transmitted to ground in a transparent manner.

- Each instrument shall format its data into Source Packets. The Source Packet structure shall comply with the ESA Packet Telemetry Standard. The maximum length of a Source Packet including the header shall be 518 octets.

- The acquisition is performed under the control of the Control and Data Management Unit (CDMU) sequentially from instrument to instrument on a single packet basis. If the addressed instrument has a packet available, this packet will be acquired.

- The CDMU generates from these packets, the AOCS packets, and from the spacecraft housekeeping packets a stream of a telemetry frames adding idle packets and frames as required.

The data acquisition is perfomed as shown in table 5.8.4.2.1 within the packet allocation given in table 5.8.4.2.2.

One extra programmable Telemetry mode and one extra programmable packet allocation will be available during operations.

| **Telemetry mode** | **OM kb/s** | **RGS kb/s** | | **EPIC**(*) **kb/s** | | |
|---|---|---|---|---|---|---|
| | | RGS1 | RGS2 | MOS1 | MOS2 | (p-n) |
| 1 | 4 | 6 | 6 | 4 | 4 | 40 |
| 2(**) | 4 | 6 | 6 | 16 | 16 | 16 |
| 3 | 4 | 12 | 12 | 4 | 4 | 28 |
| 4 | 4 | 12 | 12 | 12 | 12 | 12 |
| 5a | 4 | 2 | 22 | 4 | 4 | 28 |
| 5b | 4 | 22 | 2 | 4 | 4 | 28 |
| 6 | 8 | 6 | 6 | 8 | 8 | 28 |
| 7 | 8 | 6 | 6 | 16 | 12 | 16 |
| 8 | 8 | 12 | 12 | 8 | 8 | 16 |

**Table 5.8.4.2.1: Telemetry modes' bit rate allocation**

(*) The EPIC radiation monitor bit rate, not includedin this table, is taken from the EPIC allocation.
(**) Telemetry mode 2 is the default.

The sum of the experiments bit rate is 64000 bps and includes any packet type.

The maximum number of packets acquired by the OBDH every second is shown in the following table 5.8.4.2.2.

| | **OM** | **RGS1** | **RGS2** | **MOS1** | **MOS2** | **PN** |
|---|---|---|---|---|---|---|
| **Packets per second** | 2 | 6 | 6 | 4 | 4 | 10 |

**Table 5.8.4.2.2: packet allocation**

The total number of bytes sent by each instrument within 1 second time interval is monitored by the OBDH.
If an instrument exceed its allocated bit rate as given in table 5.8.4.2.1 in any 1 second time interval, it will not be polled until the end of this time interval and the allocation on the following 1 second time interval will be reduced by the excess.

It shall be possible from the OBDH side to halt the instrument's processor or to force the instrument's processor to a basic mode. In the basic mode, it shall be possible to dump and load all memory of the instrument.

### 5.8.4.3 On-Board Data Handling Interfaces

The instrument shall interface to the OBDH bus via the Digital Bus Interface (DBI), specified in RD 2. On the spacecraft side, the DBI will be realised by a Digital Bus Unit (DBU) providing connection to main and redundant OBDH bus.

The spacecraft Control and Data Management Unit (CDMU) controls which OBDH bus is active. The instrument determines which DBU branch is active by powering the selected DBI branch. The DBU will automatically determine which OBDH bus is active.

The DBU is a part of the OBDH system.

The DBU shall be powered by the instrument.

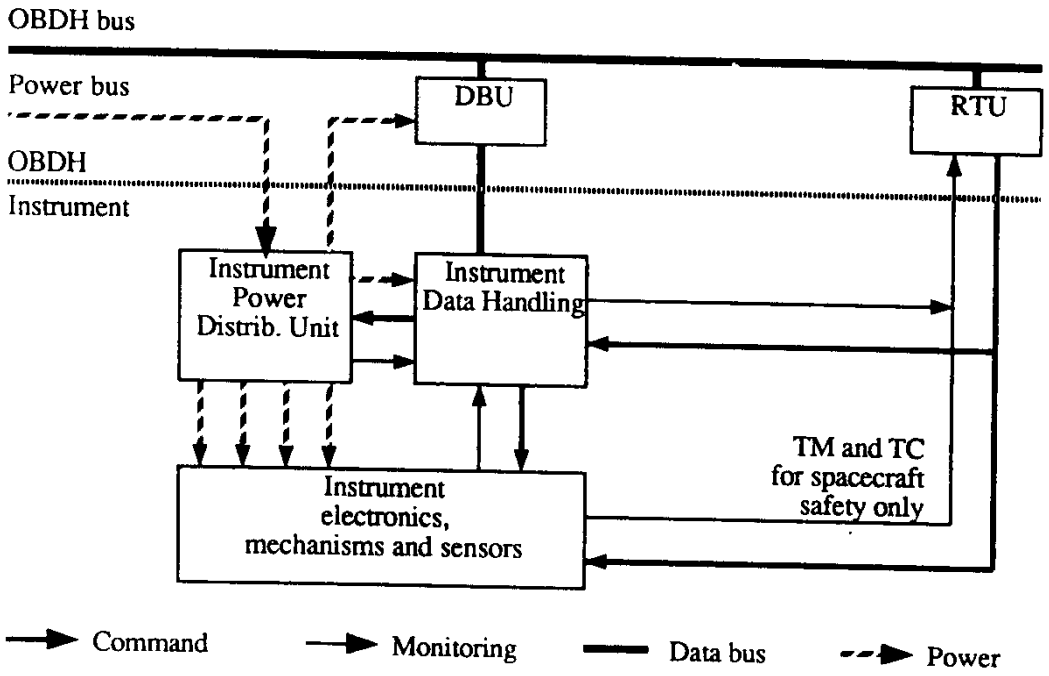
The protocol between the OBDH system and the instrument is specified in AD 11.

The nominal way to command the instrument during operations shall be through telecommand packets. Likewise, the nominal way to monitor the instrument shall be through housekeeping telemetry packets, generated by the instrument.

The electrical characteristics of these command and monitoring interfaces shall comply with AD 6.

The spacecraft interface is shown in Fig. 5.8.4.3.

**Fig. 5.8.4.3: OBDH Interfaces**



#### 5.8.4.3.1 DBI Electrical Interfaces

The DBU inputs have the following characteristics:

- Data rate: 524.288 kHz ± 5 KHz

- Voltage thresholds:
  - Logical one: þ 4.2 V
  - Logical zero: þ 1.35 V

- Input impedance: þ 40 kOhm in parallel with 40pF max

- Leakage current: þ 5 þA

The DBU outputs have the following characteristics:

- Data rate: 524.288 kHz ± 5 kHz

- Voltage thresholds:
  - Logical one: þ 4.5 V þ 5.5 V
  - Logical zero: þ 0 V, þ 0.4 V

- Output impedance: þ 1 kOhm in parallel with 40pF max sufficient to drive a load of 120 pF in parallel with 40 kOhm with a rise/fall time less than 270 ns (measured between 10% and 90%).

The DBU power input:

- Nominal voltage: 6 V ± 5%

- Maximum voltage: 7 V

- Nominal average current: 150 mA

- Maximum average current: 200 mA

**5.8.4.3.2 Digital Serial Data - RTU Interface**

Digital Serial Data (MLC, TM) between the RTU and the experiments use differential drivers and receivers with the following characteristics.

Driver Circuit

| | |
|---|---|
| Type | differential, DC coupled |
| Zero reference | signal ground |
| Low Level | 0 to 0,5 V |
| High Level | 4,5 to 5,5 V |
| rise/fall times | 0,1 to 1 µsec when connected to load |
| drive capability | 50 mA |

Receiver Circuit

| | |
|---|---|
| Type | differential, DC coupled |
| Differential Input | Low Level þ - 1V |
| | High Level þ+ 1V |
| Hysteresis | 2 to 3V |
| Input Impedance | DC þ10 kþ |
| (differential) | AC 120 þ in series with 100 pF |
| Fault voltage tolerance | -3 to +10V |

Harness

Twisted shielded pair, shield connected to signal ground on source and receiver side

Timing Characteristics

As in AD6, time period $t_1$ = 61 µsec

#### 5.8.4.4 On-Board Timing

The electrical user interface for onboard time distribution is implemented in the Digital Bus Interface, DBI. The time distribution of XMM shall make use of the built-in time functions of the RBI component.

The reconstructed CDMU clock frequency is provided on the DBI signal RIR-CLK. The clock signal is derived from the OBDH bus clock which in turn is driven by the central clock in the CDMU of the OBDH system. The frequency of this signal will be 524288 Hz ($2^{19}$ Hz).

Two BroadCast Pulses, BCPs, of the interrogation format have been allocated for synchronization. The BCPs are recovered from the interrogation formats by the RBI function. BCP2 provides a 1 Hz signal to which the subsystems synchronize when BCP2 is preceeded by a pulse on BCP3.

The OBDH sends a BCP4 to verify that the subsystems are synchronized to the central clock. The subsystems should on detection of a BCP4 pulse sample the clock register and generate a time verification packet.

The time synchronization and verification procedures are specified in AD 11.

#### 5.8.4.5 Instruments Temperature Monitoring

For units which need temperature monitoring during phases when the instrument is powered off, the spacecraft can provide temperature monitoring.
Instrument requirements shall be specified in the EID-B.

### 5.8.5 Data Handling Requirements

#### 5.8.5.1 Housekeeping Requirements

The instrument housekeeping data shall include, as a minimum:

- All parameters indicating health and safety

- Digital status monitors of the complete instrument configuration including status and condition of any software, i.e. which sequence is operating, which is waiting, and all error coding.

#### 5.8.5.2 Spacecraft Resources

The Principal Investigator shall define in the EID-B the resources requested from the data handling subsystem.

The maximum number of high level ON/OFF channels shall be 8 per instrument chain (EPIC = 3 chains, RGS = 2 chains and OM = 1 chain).

For each ON/OFF channels the instrument shall incorporate a separate monitoring channel of "relay status" type.

#### 5.8.5.3 TM and TC Interface Requirements

The APID (application identifier) of the TM packet header will specify the source of the packet. The APID of the TC packet will specify the sink of the packet. Packets shall include a data field header as specified in AD 8.

The data type of the data field header defines how the packet shall be interpreted.

All data items representing a number shall have the most significant bit in the position closest to the packet header.

Any parameter that quantifies/qualifies another parameter shall precede it in the packet.

Packet grouping (according to AD80, specific time formats for datation, special parameters format and data compression can be used for Science Data Packets TM (15,X).

All information and algorithms required for the processing and reconstruction of science data shall be provided in the Experiments User Manuals.

### 5.8.6 Redundancy Definitions

Circuit Redundancy is illustrated in Fig. 5.8.6.1 for the OBDH bus interface. Each DBI interface connects the user to the prime or redundant OBDH bus.

Circuit and Line Redundancy is illustrated in Fig. 5.8.6.2 for the power interface.

The EID-B for each experiment shall specify the redundancy requirement for each interface.

**Fig. 5.8.6.1: Circuit Redundancy**

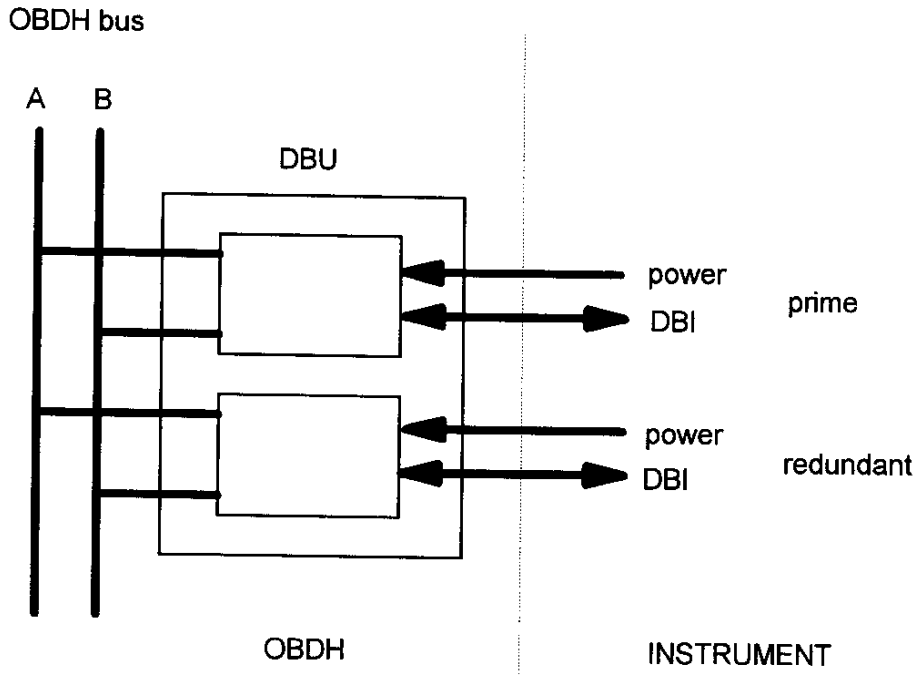
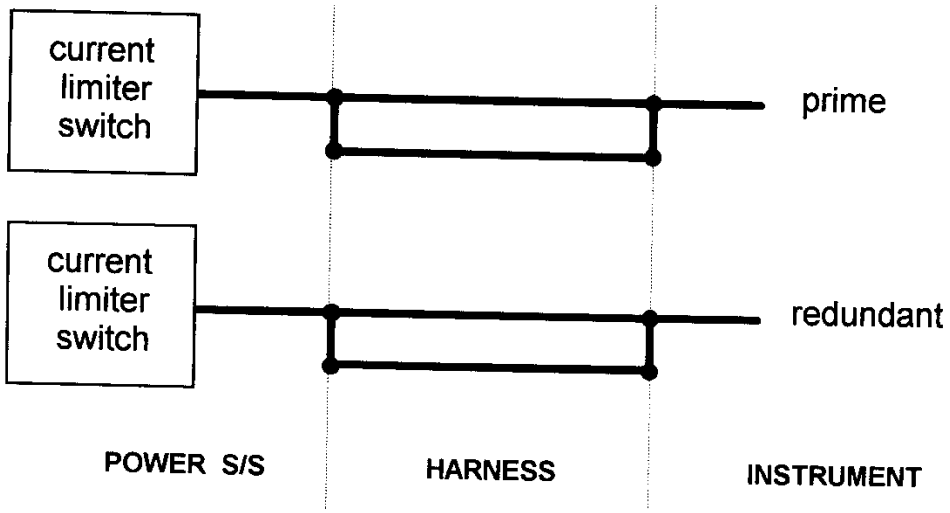



**Fig. 5.8.6.2: Circuit and Line Redundancy**



### 5.8.7 Software Requirements

Flight software development shall be in accordance with AD 2.

Flight software shall be resident in a separate ROM package and transferred into RAM for operation.

Flight software in RAM shall be modifiable by telecommand and shall be readable by telemetry.

The mechanical design shall allow for easy and late installation (up to the close out of the Flight Readiness Review) of ROM containing the flight software.

## 5.9 Electromagnetic Cleanliness Requirements

### 5.9.1 System EMC Concept

The EMC requirements are to ensure the proper function of a system which is characterised by low electrical sensor signals and extensive use of computer high rate digital equipment. The spacecraft/payload EMC requirements cover the following system-imposed aspects:

- Design requirements, which ensure a coherent satellite system design

- Performance requirements, which quantify the applicable emission and susceptibility levels to ensure the required margin at system level.

### 5.9.2 EMC Control and Verification

An EMC Control Plan at system level (RD8) defines by what means the EMC control will be implemented for the spacecraft and its subsystems. For each instrument an EMC control and verification plan as part of the Instrument Verification Plan shall be established.

The EMC design requirements will be verified by inspection and design reviews.

The verification of the EMC performance requirements is part of the respective test programmes.

### 5.9.3 EMC Design Requirements

#### 5.9.3.1 Grounding and Isolation

A distributed single point ground scheme shall be applied, i.e.:

- The primary DC power is grounded to structure in one point within the power subsystem only. All return lines shall be isolated from the structure

- Each user shall generate his own secondary power isolated from the input power. The secondary power return shall be connected to the structure in one point which serves as the signal reference ground.

- In the case that several units are supplied from the same DC/DC converter secondary output, the distribution shall be a starpoint system for power lines and returns where practical

- Between electrical units, all signal driver outputs shall be referenced to ground and all signal receiver inputs shall be isolated from ground. Receivers shall provide common mode rejection capability. Balanced differential signals are preferred.

The grounding and isolation concept is shown in Fig. 5.9.3.1.

An instrument grounding diagram shall be part of the EID-B.

The isolation between the primary power lines and equipment structure shall be 1 Mþ minimum with a capacitance of 20 nF maximum. The isolation between primary and secondary power lines shall be 1 Mþ minimum with 1 nF maximum (secondary ground disconnected). The use of a static shield between primary and secondary windings of transformers is recommended.

#### 5.9.3.2 Electrical Bonding and Case Shielding

Each unit shall be housed in a non-magnetic metallic case which shall form an electromagnetic shield. The case shall not contain any apertures other than those essential for sensor viewing or outgassing vents. If outgassing vents are required they should be as small as possible (less than 5 mm in diameter) and should be located in the case surface which is closest to the equipment shelf (spacecraft structure ground).

Joint faces between metallic parts shall be flat and clean before assembly; the only permitted surface finishes for joint faces are:

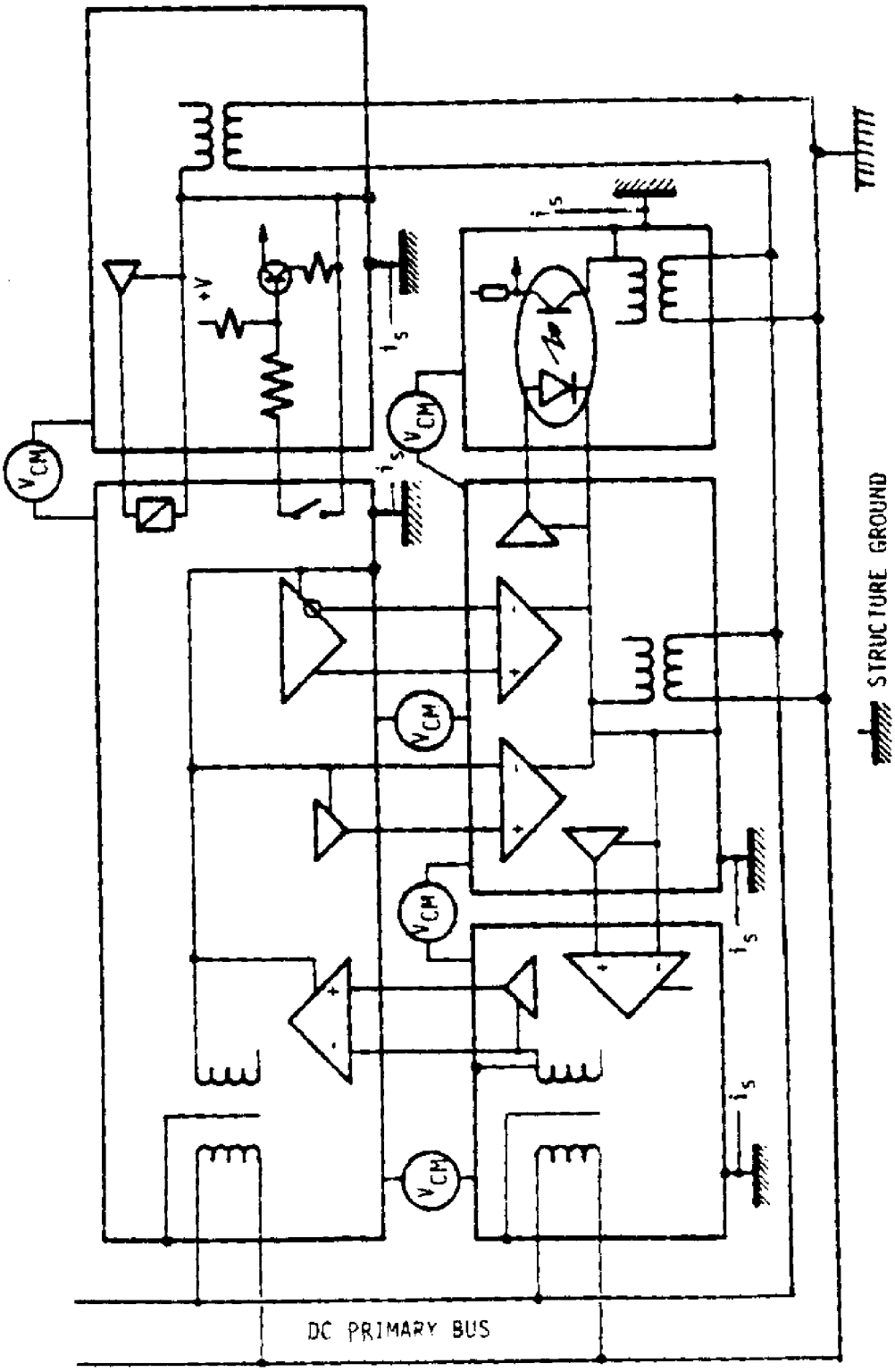
- Clean metal, except magnesium
- Gold plate on the base metal
- Alodine 1200 or equivalent.

Any other anti-corrosion finish, e.g. anodising, shall be removed from the joint faces before bonding.

Electrically dissimilar materials shall be avoided for electrical bonds.

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

**Fig. 5.9.3.1: Distributed Single Point Ground System**



The case of each unit will be grounded to the spacecraft structure.

The DC resistance across the electrical bond between any two adjacent parts, including connector shells, shall not exceed the following limit for both test polarities:

- 10 mOhm (test at 100 mA).

The DC resistance between the grounding point and any structure point shall be less than 10 mOhm.

One pin at least of each connector of any electronic unit shall be connected to the structure of this unit by means of an internal strap of 5 cm maximum length to provide a low inductive link. The DC resistance between those pins and the chassis ground of the unit shall be less than 10 mOhm (test at 100 mA).

Non-metallic conductive structure parts shall be electrically bonded to the metallic structure reference in order to avoid differential charge build-up.

All outer surfaces exposed to space shall be conductive (R < 10 kþ/square) in order to avoid differential charge build-up resulting in the risk of electrostatic discharge (ESD).

#### 5.9.3.3 Cable Shielding and Separation

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

Class 1: Power lines
Class 2: Digital lines (high frequency, noisy, Converter Synchronisation Signals)
Class 3: Pyro
Class 4: Analogue lines (low frequency, sensitive)
Class 5: RF

All spacecraft interface cabling shall connect to a single experiment unit.

Lines of different EMC classes shall be separated by at least 5 cm and shall be routed through separate connectors. Where this is not possible, separation shall be implemented by a row of grounded pins.

Lines of Class 2 to Class 5 shall be shielded.

These rules shall also be applied within electrical units.

The active wire(s) shall be twisted around the return wire. The twisted wires shall be routed through a connector on adjacent pins to minimise the wire loop.

Cable shield shall not be used as the return path for signal or power.

Harness and connector layout shall permit the termination of cable shields at both ends.

Shields shall be terminated on connector pins adjacent to their signal pins; the unshielded length of the internal connection shall be minimised. The unshielded length of any single cable shall not exceed 2.5 cm.

The resistance between any harness shield termination pin and the grounding point (para. 5.5.10) shall be less than 10 mOhm.

### 5.9.4 Performance Requirements

The instrument units shall operate with nominal performance when submitted to any of the signals detailed below.

The conducted interference limits defined below apply when lines are powered from an impedance representing the power bus characteristics.

Both intentional and unintentional levels are contained in the emission requirements below. The levels, are given in peak amplitude, unless otherwise specified.

The conductive emission requirements are applicable up to 50 MHz, except for equipment with no internal functional signal above 1 MHz for which the range is reduced up to 10 MHz.

The requirements shall be met for any operating mode of the instruments.

The assumed bandwidths of the receiver are as follows for a narrow band (NB) and broadband (BB) recordings.

| EMISSION | FREQUENCY RANGE | BANDWIDTH |
|---|---|---|
| NB | 30 Hz - 10 KHz | 30 Hz |
| NB | 10 kHz - 2.5 Mhz | 300 Hz |
| NB | 2.5 MHz - 25 MHz | 3 kHz |
| NB | 25 MHz - 1 GHz | 30 kHz |
| NB | 1 GHz - 10 GHz | 300 kHz |
| BB | 10 kHz - 2.5 MHz | 3 kHz |
| BB | 2.5 MHz - 25 MHz | 30 KHz |
| BB | 25 MHz - 1 GHz | 300 KHz |

*Distinction* between NB and BB shall be made as follows:

1) **NB** requirement level shall be applied when the measured level increases *less* than 10 dB when the bandwidth of the receiver is switched from NB to BB as above. For f < 10 kHz, the level with 40 Hz bandwidth shall be measured.

2) **BB** requirement level shall be applied when the measured level increases *more* than 10 dB when the bandwidth of the receiver is switched for NB to BB as above.

It is understood that the bandwidth values are instrument dependant. Any deviation from above affects the measured levels and shall be approved by ESA.

#### 5.9.4.1 Conducted Emissions

##### 5.9.4.1.1 Power Lines (Including Keep Alive Lines)

- **Single Line Noise**

  In the frequency band 30 Hz-50 MHz, the noise appearing on any primary power line or return shall not exceed the following limits. The limits apply to groups of wires (e.g. from a single connector) sharing a common return rather than to individual wires.

  ***Narrow band*** current limits are defined in Fig. 5.9.4.1.1-1. The noise current amplitude at low frequencies shall be smaller than 5% of the average DC current value.

  The limits are applicable to all operating modes of the instrument.

- **Common Mode Emissions**

  The narrow band limit is set to 100 þAp from 30 Hz to 50 MHz (power line together with return).

- **Time Domain Transients**

  The instantaneous rate of current change shall not exceed $5.10^4$ A/sec.

  Conducted voltage transients on the primary power bus, appearing during nominal mode switching (excluding on/off) shall be < 1 Vpp.

  At unit power switch-off the voltage transients on the primary power bus shall not exceed the envelope shown in Fig. 5.9.4.1.1-2.

**Fig. 5.9.4.1.1-1: Conducted Emission on Power Lines**

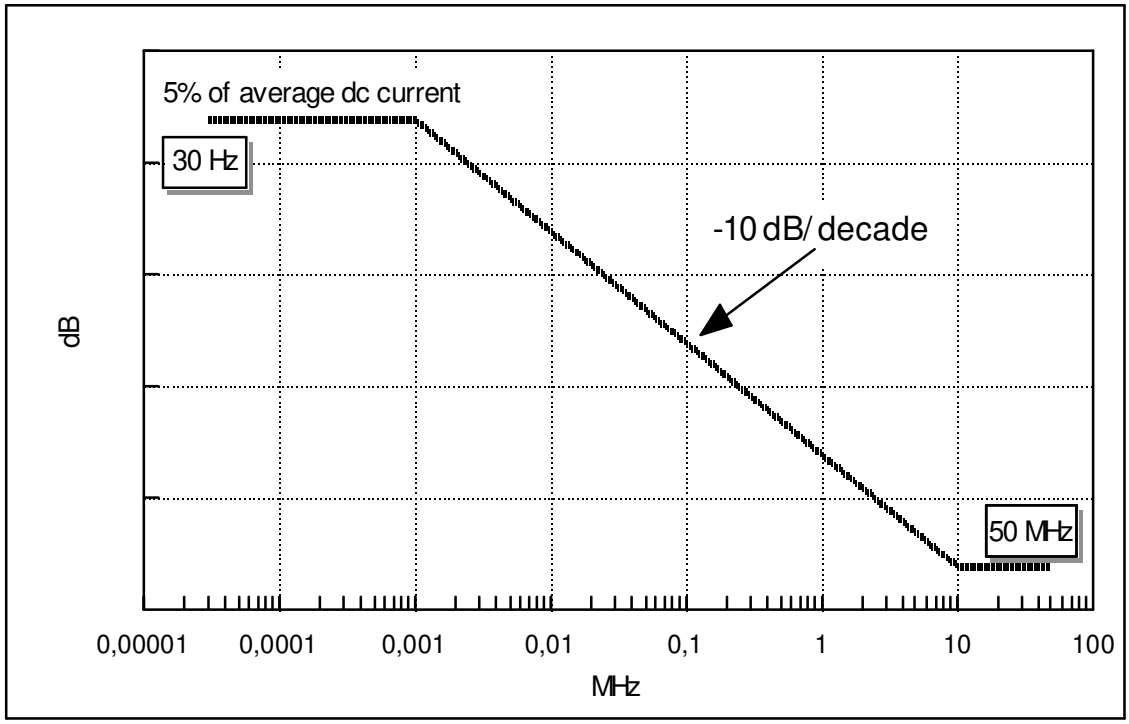
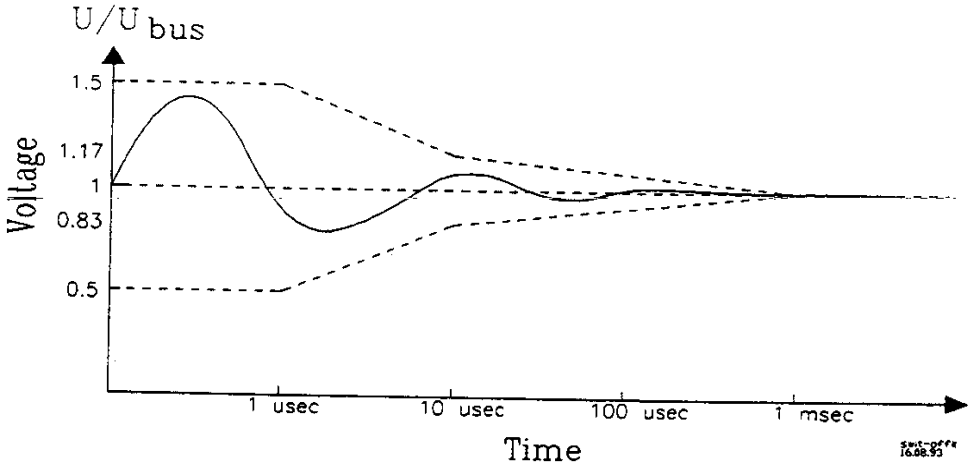


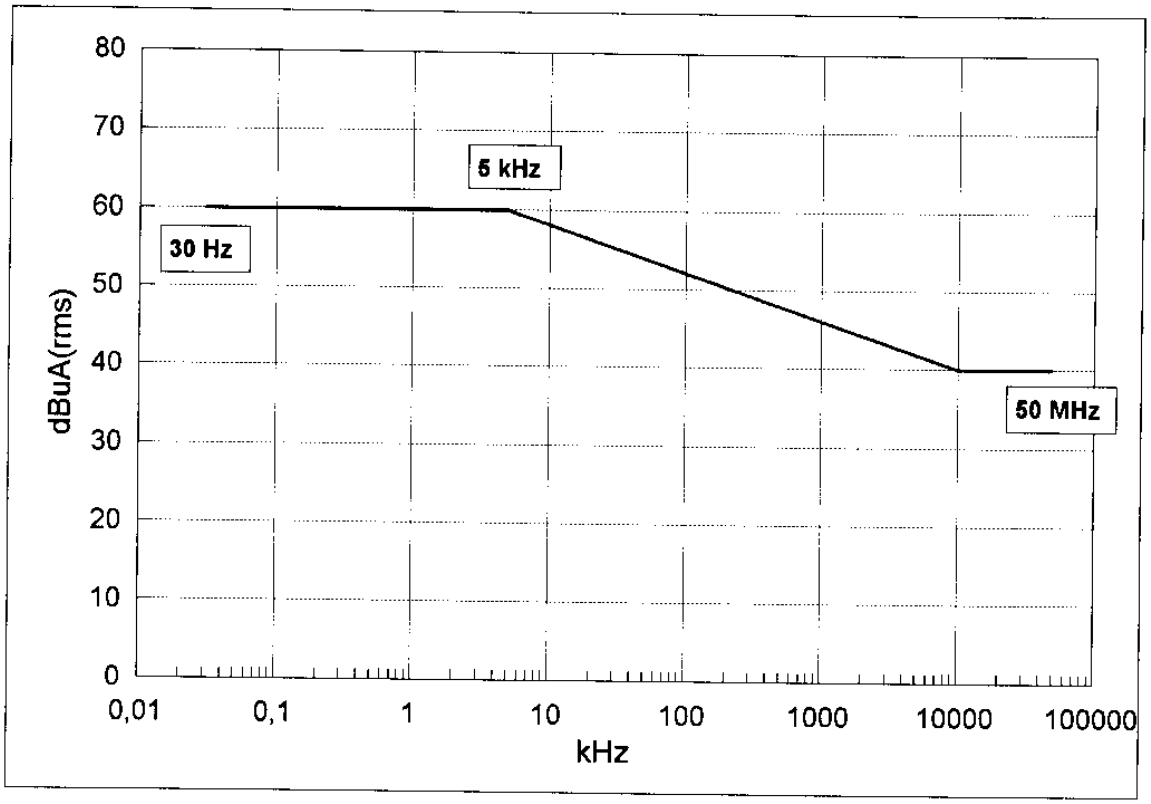

**Fig. 5.9.4.1.1-2: Switch-off Voltage Transient Characteristics**



#### 5.9.4.1.2 Conducted Emission on Signal Lines

Conducted narrow band (NB) current emissions in the frequency range 30 Hz - 50 MHz appearing on the unit signal lines (common mode) shall be within the limits of figure 5.9.4.1.2-1.

**Fig. 5.9.4.1.2-1 Conducted Emissions on Signal Lines**



#### 5.9.4.2 Conducted Susceptibility

##### 5.9.4.2.1 Power Lines

The unit or subsystem under test shall keep nominal performance when subjected to the following perturbation.

**Differential Mode**:

- Continuous sinewave from 30 Hz to 50 MHz:

  - 28 V lines: 0.5 Vrms
  - KA lines: 150 mVrms

- Squarewave from 30 Hz to 1 MHz:

  - Amplitude: 2 Vpp (for 28 V lines)
    0.6 Vpp (for KA lines)
  - Rise time: þ 1 µsec for f þ 50 kHz
    þ 100 nsec for f > 50 kHz
  - Duty cycle: 50%

The sweep speed of the sinewave and the squarewave tests shall be less than 1 octave/minute.

For continuous signals (sinewave, squarewave), the injected current shall be monitored and limited to 1 ampere peak, on their input power lines.

- Voltage transients:

  The pulse characteristics shall be as follows:

  - Trapezoidal shape: Rise time, tr < 4 µsec
  - On 28 V lines: E = 3 V þ = 800 us
  - On KAL lines: E = 2 V þ = 200 us.

(þ): duration at half amplitude.

The shape of the transient is shown in Fig. 5.9.4.2.1.

A minimum of ten pulses of each polarity shall be experienced, applied alternatively, with a repetition rate less than 10 Hz.

Both polarities shall be tested, i.e. a positive pulse at the upper supply voltage limit resp. a negative pulse at the lower supply limit.

**Common mode:**

- Squarewave from 30 Hz to 1 MHz

- þ Amplitude: 4 Vpp
- þ Rise time: < 5% of the period
- þ Duty cycle: 50%
- þ The sweep speed shall be less than 1 octave/minute.

- Transient single event:

  - þ Shape as for differential mode, see Fig. 5.9.4.2.1
  - þ Amplitude: E = 28 V
  - þ Rise time: tr þ 10 ns
  - þ Duration: þ þ 150 ns

Both polarities shall be tested (10 pulses minimum).

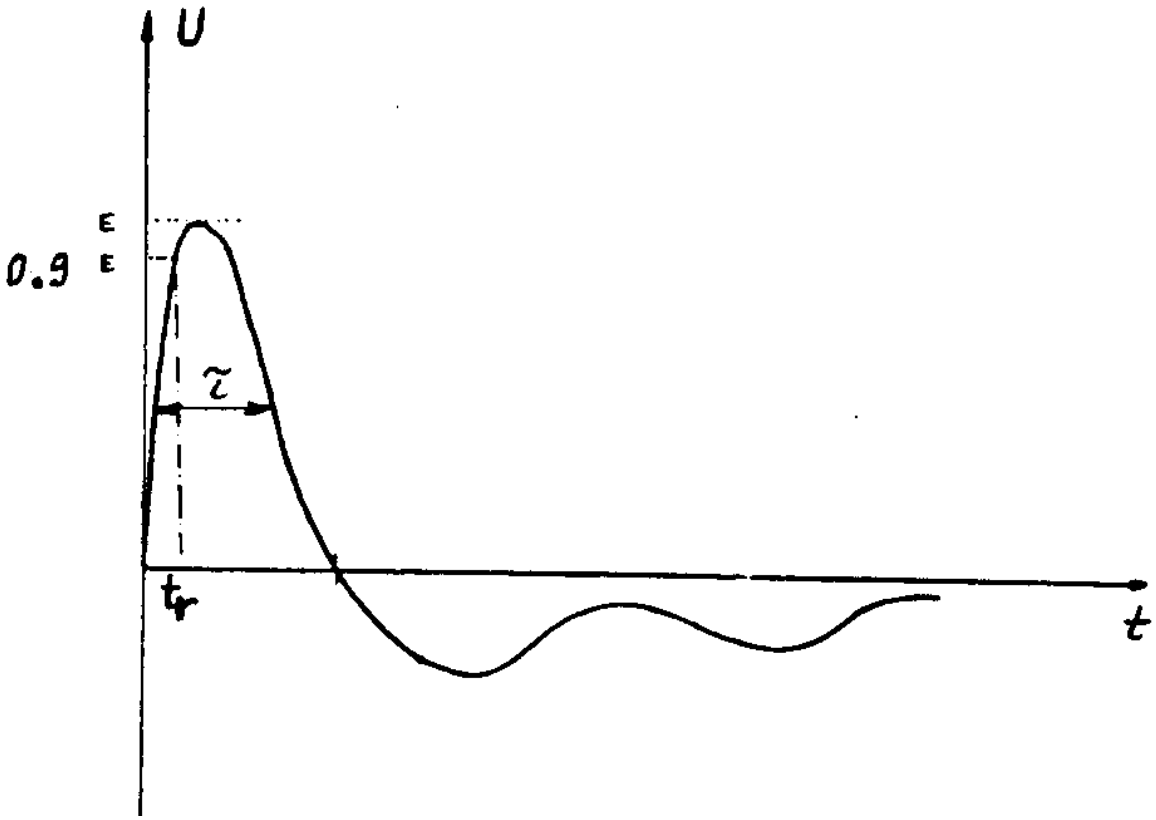
In common mode the spurious signal is injected between the power return wire and the unit case.

#### 5.9.4.2.2 Conducted Susceptibility on Signal Lines

Receiver circuits shall not respond to the below specified voltages injected into the signal lines (common mode):

| | |
|---|---|
| 1) Transient voltages | 360 mV |
| 2) Rise/fall time | 100 nsec |
| 3) Duration | 500 nsec |

**Fig. 5.9.4.2.1: Transient Shape**



#### 5.9.4.3 Radiated Emissions

##### 5.9.4.3.1 Radiated Magnetic Field Emission

Narrowband magnetic field emissions from individual units, shall not exceed the limits defined in Fig. 5.9.4.3.1.

DC magnetic field: 106 dBpT (200 nanotesla) at 1 m distance.

##### 5.9.4.3.2 Radiated Electric Field Emission

Electric field emissions shall not exceed the limits as defined in Figures 5.9.4.3.2 and 5.9.4.3.3.

DC electric field: No stray field is allowed. All external surfaces shall be at ground potential and conductive.

#### 5.9.4.4 Radiated Susceptibility

##### 5.9.4.4.1 Magnetic Field Susceptibility

DC requirement: 170 dBpT (0.3 mTesla).

**NB**: This is to ensure nominal operation for any orientation in the earth magnetic field.

Units shall operate with nominal performance when exposed to irradiation along each orthogonal axis by continuous waves field swept at less than one octave per minute with the characteristics defined in Fig. 5.9.4.4.1.

##### 5.9.4.4.2 Electric Field Susceptibility

Unit shall operate with nominal performance when exposed to an electric field of:

1 V/m in the frequency range from 14 kHz to 10 GHz,

5 V/m from 2.0 to 2.4 GHz for units mounted inside the S/C, and 12 V/m from 2.0 to 2.4 GHz for units mounted on outer surfaces of the S/C.

The sweep speed shall be less than one octave per minute and the signal shall be amplitude modulated to 30% by a 1 kHz squarewave.

**Fig. 5.9.4.3.1: Limits for Radiated Magnetic Fields at 1 metre from Equipment**

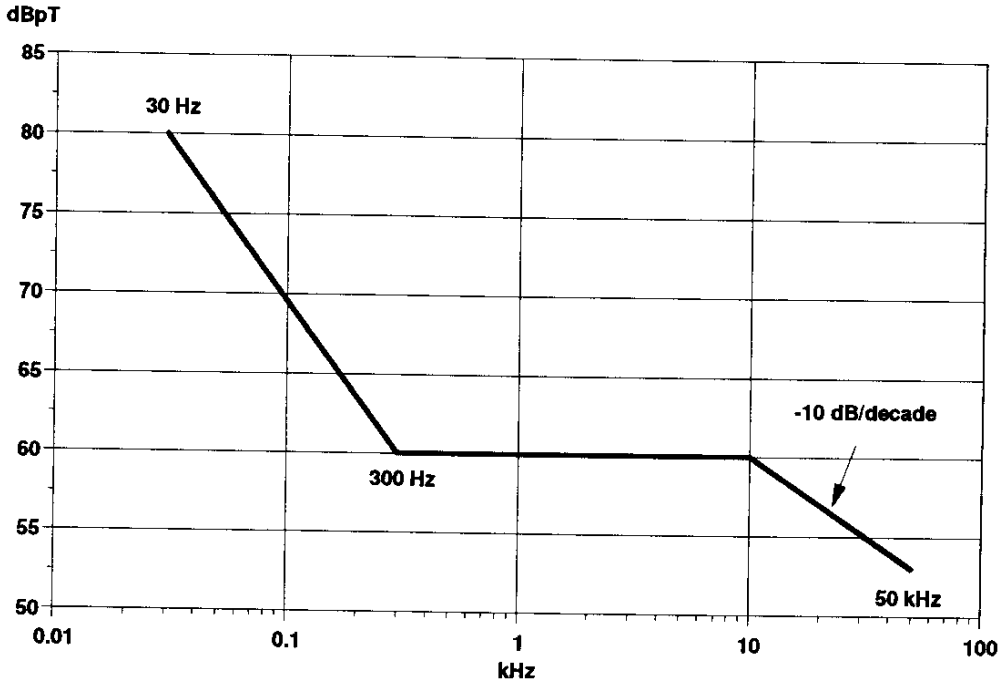



**Fig. 5.9.4.3.2: Radiated Broadband Limits for Electric Field Emission**

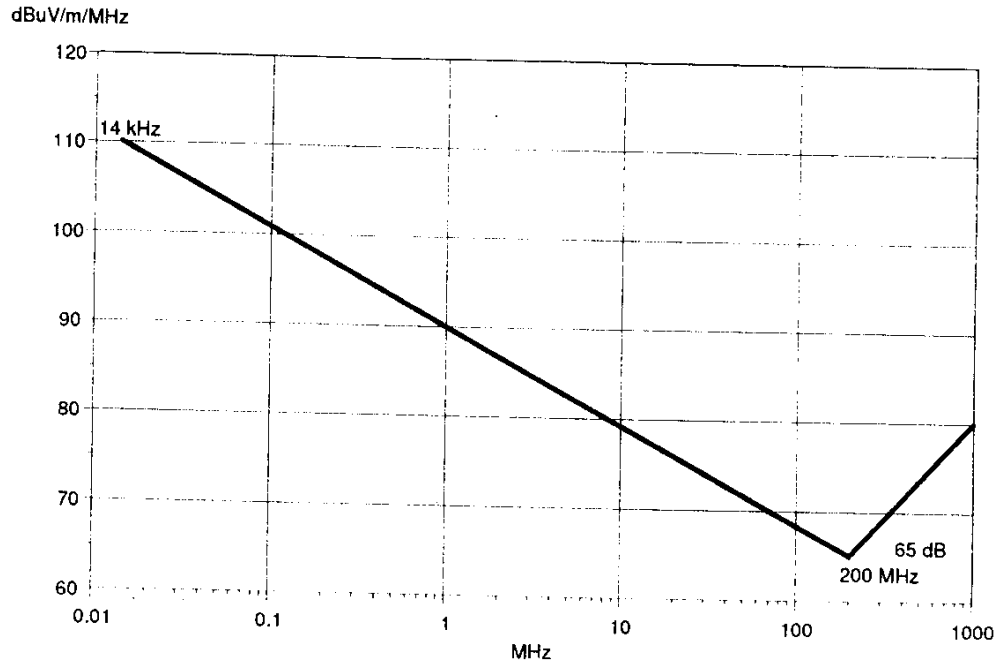


**Fig. 5.9.4.3.3: Radiated Narrowband Limits for Electrical Field Emission**

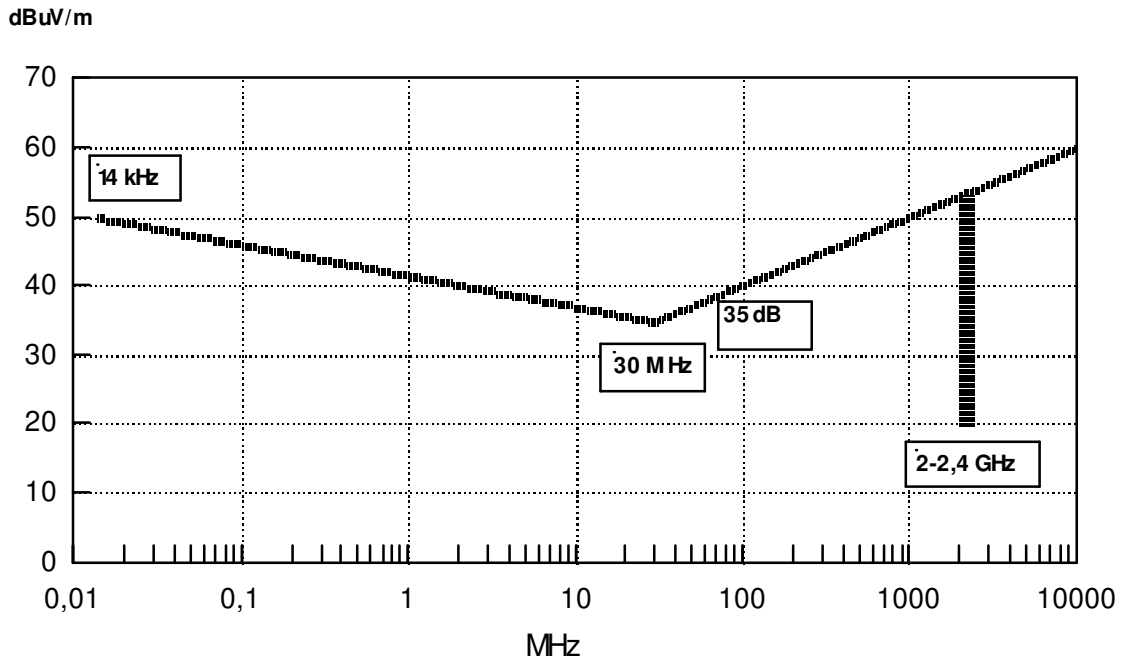
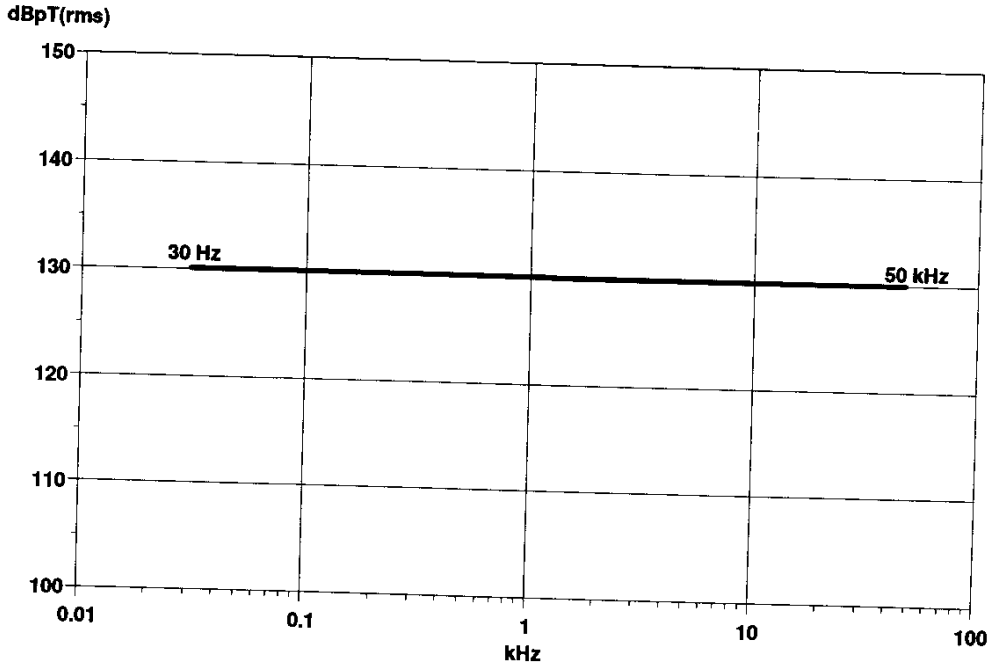


**Fig. 5.9.4.4.1: Radiated Susceptibility Limits - Magnetic Fields**



**5.9.4.5 Electrostatic Discharge**

Each instrument unit shall operate with nominal performance under electrostatic discharges with the following characteristics:

- Conducted ESD (current injected in the structure of the equipment):

  - Imax: 25 A
  - Rise time: < 5 ns (10-90%)
  - Duration: 30 nsec. (at half amplitude).
  - Repetition rate: 0.1 Hz ÷ 10 Hz

- Radiated ESD:

  - Spark gap discharge at 30 cm of the unit and harness
  - Energy: 15 mjoule
  - V = 10 kV
  - Repetition rate: 0.1 Hz ÷ 10 Hz

## 5.10 Molecular and Particulate Contamination

### 5.10.1 Contamination Environment and Requirements

#### 5.10.1.1 General Concept

The cleanliness requirements are mainly dictated by the instruments using detectors and/or optical surfaces sensitive to molecular as well as particulate contaminants. Sources of contamination are outside and inside the instruments.

Outside sources of contamination are the other instruments, the satellite, ground support equipment used during on ground activities (integration and testing) plus the facilities and the people that operate the facilities, the ground support equipment, the satellite and the instruments.

Inside sources of contamination are the materials (outgassing, manufacturing debris) and processes (potting, sealing, painting). Effects of contamination are scattering of light (due to particulates), reflection loss (due to thin layer deposit absorption) and corona arcing (due to low gas pressure at high voltages).

Requirements on chemical deposits and particulates relate to each of the sources at all levels. However, the effects can only be quantified as allowable degradations of the instruments.

Each component can be source and receptor of contamination: these two aspects shall be taken into account by the PI during the design. For the source aspects by optimising use of materials and processes in terms of outgassing, for receptor aspects by design of protective devices, use of cleaning procedures, etc...

This concept requires a cleanliness control which includes budgeting and monitoring of contamination levels at each stage of manufacturing, integration, test and usage of the instruments.

Therefore, the PI shall define the sensitive items and the foreseen methods to monitor their performance degradation throughout the lifetime. He shall also identify methods of cleaning of the sensitive surfaces as a function of the stage of integration of the instrument. This information will be input to the Cleanliness Control Plan that will be established and which will be applicable to all the contributors at all levels and throughout the lifetime, starting with manufacturing and finishing with the end-of-life (EOL) in orbit.

#### 5.10.1.2 Chemical Cleanliness Requirements

Mirrors, gratings, and detectors shall be clean and protected, not to be degraded by a layer of chemical deposits of more than $2 \times 10^{-7}$ g/cm$^2$ both at launch and at end of nominal mission life.

#### 5.10.1.3 Particulate Cleanliness Requirements

Mirrors, gratings, and detectors shall be clean and protected, not to be degraded by a collection of particulates of more than 300 ppm both at launch and at the end of nominal mission life. It has to be assumed that the collection of particulates follows the distribution function as specified in Mil. Standard 1246 A. Note that the level of contamination increases approximately linearly with time in a given class of cleanroom. The simplified relation between the cleanroom class (CRC) and obscuration factor in ppm/day is 0.116 * $(CRC)^{2/3}$. The PI shall propose cleaning methods.

#### 5.10.1.4 Materials and Processes Selection

The PI shall establish and implement a standardisation programme to optimise the use of space approved materials and processes in line with AD 1.

#### 5.10.1.5 Specific Design Requirements

- All honeycomb type of materials used shall be perforated to allow outgassing.
- All thermal super-insulation hardware shall be designed and manufactured to resist the rapid depressurisation during launch.
- Parts made of polymeric material shall not have a direct view to satellite surfaces and sensors with the exception of thermal insulation.
- All electronic and detector units that are not sealed (leak rate > $10^{-4}$ Pascal $sec^{-1}$) shall be designed with venting holes. For the initial design, a vent area of Ø = 3 mm should be assumed per $dm^3$ box volume.
- Wet lubricants shall not be used.
- The apertures of sensors, which are sensitive to the contamination environment, shall be protected by either in-orbit deployable covers or covers removable prior to launch.
- The instrument shall be designed to allow cleaning at the latest possible moment before launch which is allowed by the satellite design (late access).
- The PI shall assess the necessity for protection against chemical contamination by purging and/or maintaining sensitive areas in a dry nitrogen environment during on-ground operations.

Sensitive areas like optics and dectector that are used (also on-ground) at low temperatures, like -100þC, behave like contamination traps, since the molecular mobility is much lower at these temperatures than at ambient temperature. Special provisions, like heaters and/or covers, or dedicated design of the instrument may be required.

## 6. DESIGN VERIFICATION REQUIREMENTS

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

The PI shall prepare a Verification Plan which defines the tests and analyses and that collectively demonstrates that hardware and software complies with the requirements.

The Verification Plan shall show the overall approach to accomplish the instrument qualification and acceptance programme. When appropriate, the interaction of the test and analysis activity shall be described.

### 6.1 Definitions

**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.

**Functional Tests**: The operation of a unit in accordance with defined operational procedures to determine that performance is within the specified requirements.

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

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

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

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

**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.

## 6.2 Analyses

### 6.2.1 Mechanical Mathematical Models and Analyses

#### 6.2.1.1 Detailed Stress Analysis

This shall include at least:

a) A description of the configuration analysed with reference to interface controlled drawings

b) A description of the mathematical finite element model and/or of the assumptions taken to verify the structure

c) A description of all possible loading cases and an identification of the design driving load cases or load combinations

d) Detailed description of the most loaded elements listed with relevant stresses, and the loading cases that generated them

e) A list of the materials and structural components with characteristics data sheets (including long-life effects under space environment)

f) A set of tables showing, for each structural element, the maximum value on each type of stress or combination of them with the allowable value, and the load case that determines it, together with its margin of safety.

#### 6.2.1.2 Dynamic Model

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

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

The model shall fulfil the requirements of the Design Verification Requirements, cfr. Chapter 6, when compared to test results.

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

All mathematical models shall be maintained in current configuration.

#### 6.2.1.3 Dynamic Analysis

This shall include at least:

a) A description of the configuration analysed with reference to interface controlled drawings

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

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

d) A list of eigen-frequencies with relevant mode type and associated modal effective masses

e) Plots and listings of eigen-vectors.

#### 6.2.1.4 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 activations

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 that the mechanisms form a part of.

### 6.2.2 Thermal Mathematical Models and Analysis

The thermal performance of the instrument shall be calculated by means of thermal mathematical models. The PI shall use models for design purposes and shall also provide models for use during the spacecraft design.

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, along with indication of the appropriate operating mode. The number of operating modes considered for thermal modelling purposes may be a reduced set of the total number.

The PI shall give a description of: the configuration analysed with reference, if necessary, to the interface control drawing, the finite element model used and the assumption/reductions taken in the analysis.

### 6.3 Testing

#### 6.3.1 Test Procedures
For each test defined in the verification plan (e.g. EMC, vibration, electrical, thermal, etc.), the PI shall provide a detailed procedure. The PI shall provide a final report for each test.

#### 6.3.2 Test Matrix
In addition to the Verification Plan, the PI shall prepare a Test Matrix that summarises all the tests that will be performed on each instrument unit. The purpose of the 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 seen environmental exposures that are sufficient to demonstrate acceptable workmanship. In addition, the matrix shall provide traceability of the qualification heritage of hardware. All flight hardware, spares (and prototypes when appropriate), shall be included in the matrix.

#### 6.3.3 Test Sequences
No specific environmental test sequence is required, but the test programme should be arranged in a way to best disclose problems and failures associated with the characteristics of the hardware and the mission objectives. It is strongly recommended that the vibroacoustic test precede the thermal vacuum test unless there is an overriding reason to reverse that sequence.

### 6.4 Test Conditions

#### 6.4.1 Test Environment
Unless otherwise specified herein, all system, module or equipment fabrication, handling and tests shall be made at ambient conditions as specified below. These environmental conditions shall be controlled and recorded:

- Temperature: 22þC ± 3þC
- Humidity: 40% < RH < 60%
- Pressure: 900 to 1100 mBar
- Cleanliness: class 100 K or better

For transportation and storage, the following environment applies:

- Humidity: < 60% for units, dry nitrogen for satellite and modules
- Pressure: 200 to 1100 mBar
- Temperature: -10 +45þC

### 6.4.2 Test Level Tolerances

The test tolerances, unless otherwise specified are:

| | | |
|---|---|---|
| - | Temperature:<br>-55þC to +150þC | Tolerance<br>Tmax 0 + 3þC Tmin 0 -3þC |
| - | Pressure:<br>Equal or above 0.1 mbar<br>Below 0.1 mBar | <br>10%<br>50% |
| - | Relative humidity: | ± 5% |
| - | Sinusoidal vibration:<br>Acceleration, amplitude<br>Frequency above 50 Hz | <br>± 10%<br>± 2% |
| - | Random vibration: | |
| þ | Power spectrum density (50 Hz or narrower)<br>20 to 500 Hz<br>500 to 2000 Hz<br>Overall g rms | <br>± 1.5 dB<br>± 3.0 dB<br>± 1.5 dB |
| - | Static force: | ± 5.0% |
| - | Acoustic: | ± 1 dB |

**Electromagnetic Compatibility**

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

**Magnetic Properties**

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

**Mass Properties**

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

### 6.4.3 Test Equipment Accuracy and Calibration

The accuracy of the instrument and test equipment used to control or monitor the test parameters shall be verified periodically by calibration procedures.

## 6.5 Test Requirements

### 6.5.1 Electrical Functional Test Requirements

#### 6.5.1.1 Full Performance Test

The Full Performance Test (FPT) shall be a detailed demonstration that the hardware and software meet their performance requirements within allowed tolerances. The test shall demonstrate operations of all redundant circuitry. It shall also demonstrate satisfactory performance in all operational modes. The initial FPT shall serve as a baseline against which the results of all later FPTs can be readily compared.

The test shall also demonstrate that, when provided with appropriate stimuli, performance is satisfactory and outputs are within allowed limits.

#### 6.5.1.2 Limited Performance Tests

The Limited Performance Tests (LPT) shall be a subset of the FPT and shall be performed before, during and after environmental tests, as appropriate, in order to demonstrate that functional capability has not been degraded by the environmental tests. The limited tests may also be used in cases where comprehensive performance testing is unwarranted or impracticable. Specific items on which it is intended that Limited Performance Tests will be performed shall be listed in the Verification Plan. The Limited Performance Tests shall demonstrate that the performance of selected hardware and software functions is within allowed limits.

#### 6.5.1.3 Limited Life Time Elements

A life test programme shall be performed on electrical or electromechanical elements that have limited lifetimes. The Verification Plan shall address the life test programme, identifying the electrical or electromechanical elements that require such testing, describing the test hardware that will be used, and the test methods that will be employed.

### 6.5.2 EMC Test Requirements

The requirements laid down in para. 5.8.1.5 and 5.9 shall be verified by tests.

#### 6.5.2.1 Special Requirements on Test Equipment

Grounding of Test Equipment:

Test instruments shall use an isolation transformer on the AC power lines and a separate ground cable to the central ground point. The ground cable shall consist of braided cable.

Power Adjustment:

The primary power for XMM instrument and unit level tests shall be adjusted to the worst case condition, ie 26 Volts for emission and susceptibility tests.

Line Impedance Simulation Network (LISN):

For all conducted emission and susceptibility tests on subsystem and unit level a LISN, simulating the XMM primary power bus impedance, shall be used. Figure 6.5.2.1-1 shows the circuit diagram of the LISM. The corresponding bus impedance is shown in figure 6.5.2.1-2. The LISN has to be designed and provided by the party responsible for the test. Prior to test the network impedance shall be measured in the relevant frequency range and attached to the test report.

**Fig. 6.5.2.1-1: DC Power Line Impedance Simulation Network (LISN)**

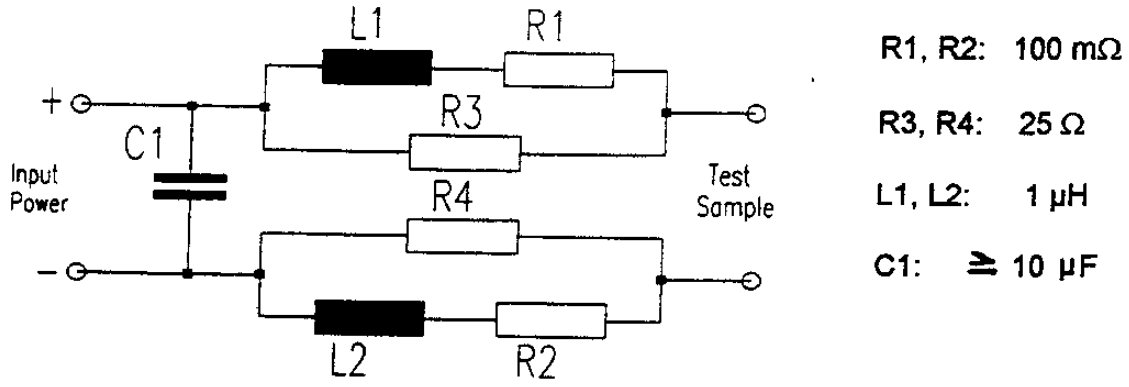
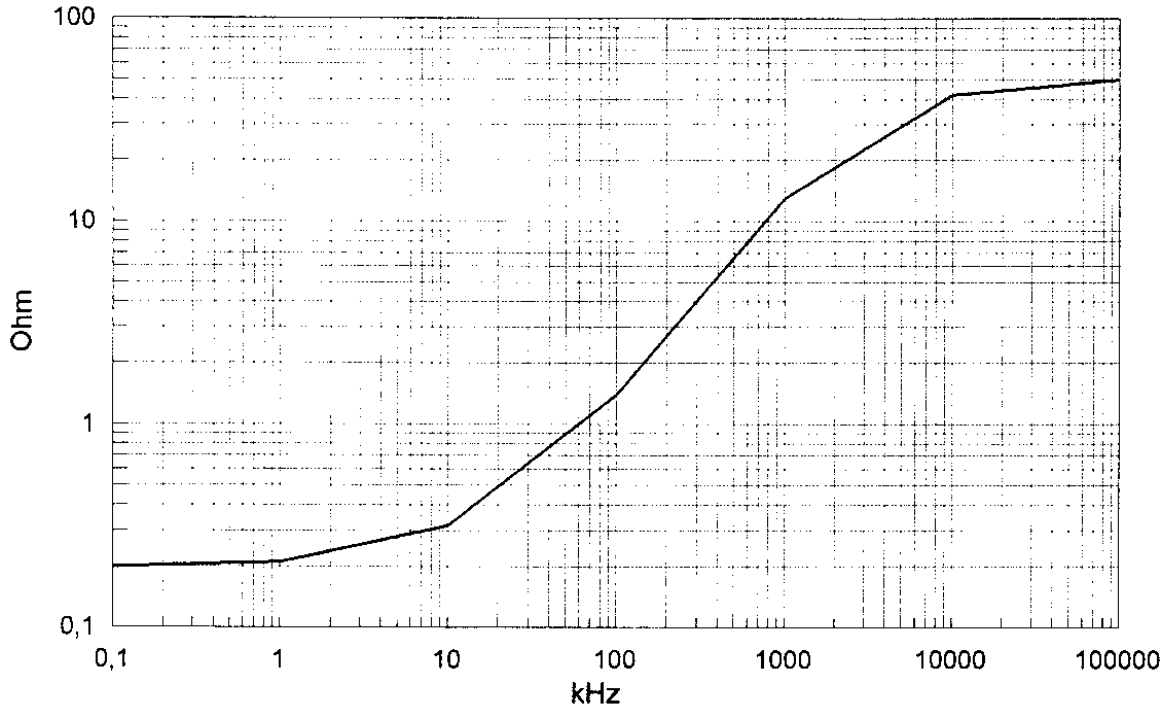



**Fig. 6.5.2.1-2: LISN Bus Impedance**




Operational Modes for Units under Test:

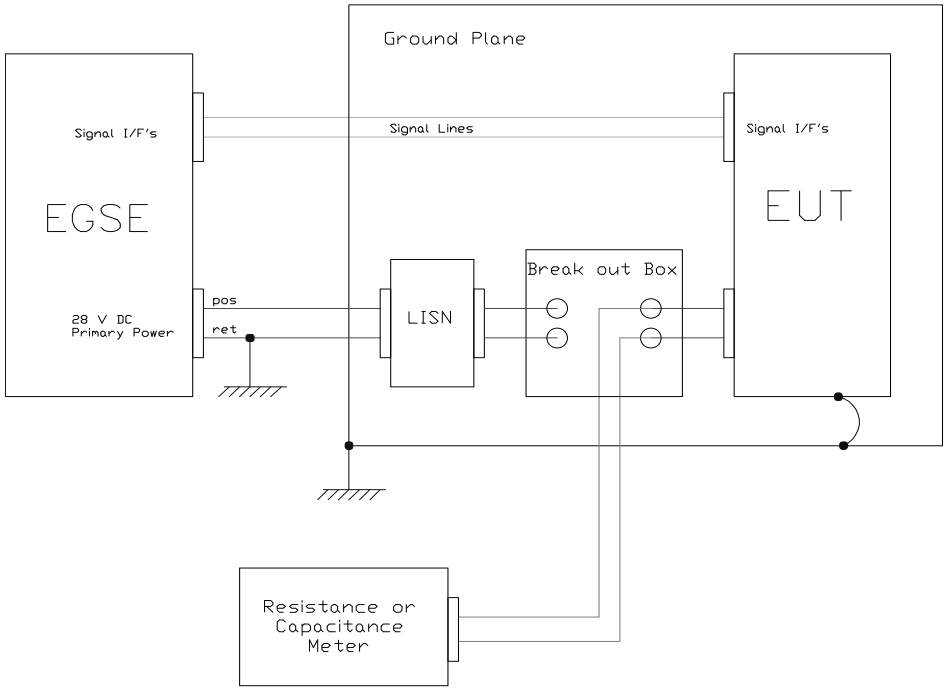
For emission tests the operational mode of a unit under test ( UUT, EUT ) shall be selected to maximize the level of interference appearing on power or signal lines, while remaining within the nominally expected operating range. The mode selection for susceptibility testing shall be such as to maximize the response of the UUT to the noise environment being created.

### 6.5.2.2 Grounding and Isolation Test Requirements

The bonding test shall be performed with the unit under test (UUT) switched off. The measurements shall be done with the four wire method and with both directions of polarity.

Figure 6.5.2.2.1 shows the principle test set-up for isolation measurements.

**Fig. 6.5.2.2-1: Test Set-up for Isolation Measurements**




### 6.5.2.3 EMC Test Configurations

#### 6.5.2.3.1 Conducted Emission Tests

UUT operation shall be selected in order to maximise the level of conducted interference appearing on the power lines, while remaining within the normally expected operating range.

The test set-ups are shown in figure 6.5.2.3.1-1 (frequency domain) and 6.5.2.3.1-2 (time domain).

Ripple and transient interference tests on the power lines shall be performed using standard current probe measurement techniques for each DC power lead.

**Fig. 6.5.2.3.1-1: Test Set-up for Conducted Emission on Primary Power Lines Frequency Domain**

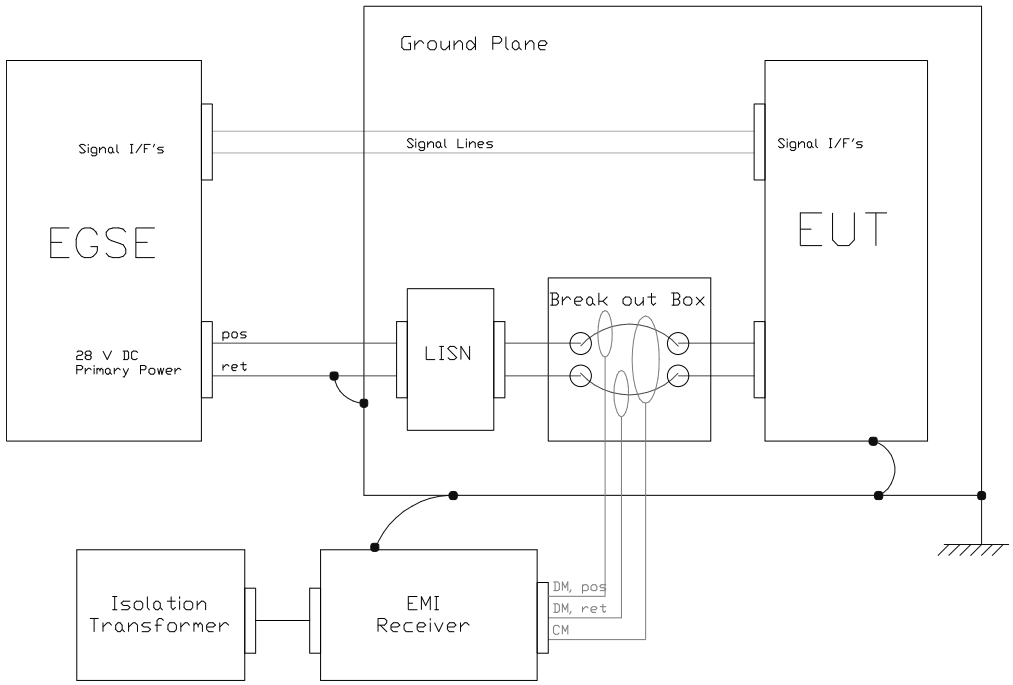



**Fig. 6.5.2.3.1-2: Test Set-up for Conducted Emission on Primary Power Lines, Time Domain**

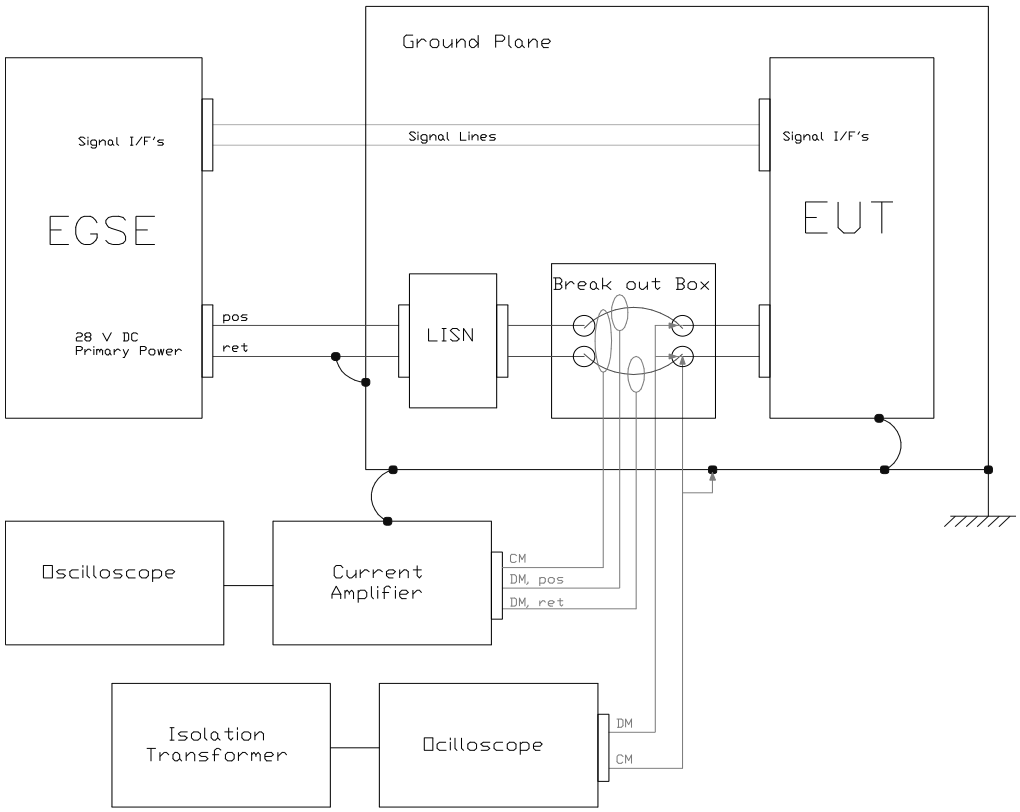
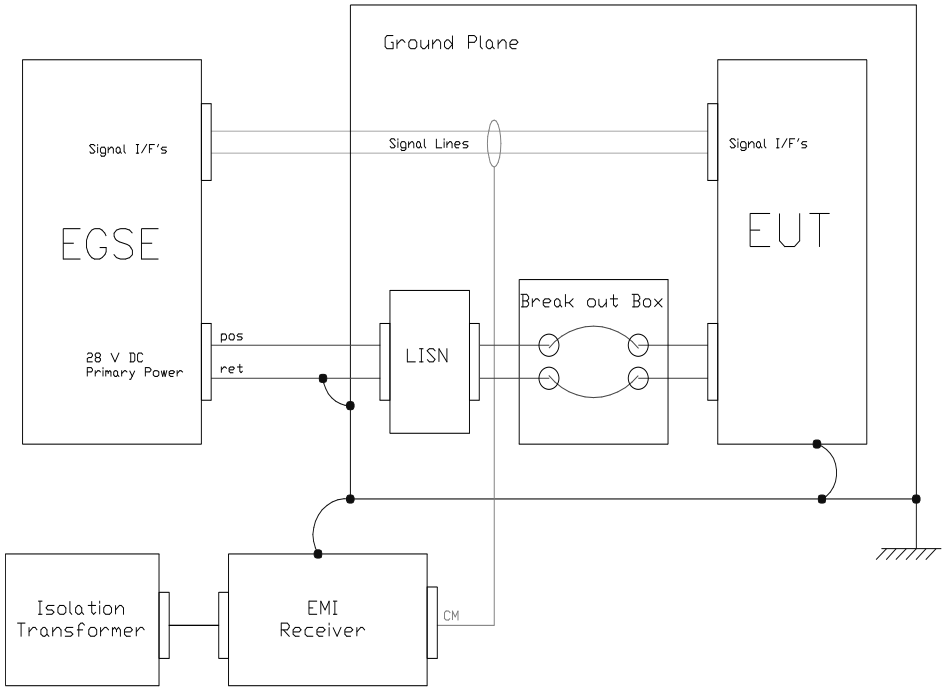

Ground Plane
Signal I/F's
EGSE
28 V DC
Primary Power
pos
ret
Signal Lines
LISN
Break out Box
Signal I/F's
EUT
Oscilloscope
Current
Amplifier
CM
DM, pos
DM, ret
Isolation
Transformer
Oscilloscope
DM
CM

**Fig. 6.5.2.3.1-3: Test Set-up for Conducted Emission on Signal and Control Lines**



#### 6.5.2.3.2 Conducted Susceptibility Tests

**Fig. 6.5.2.3.2-1: Test Set-up for Conducted Susceptibility on Primary Power Lines, 30 Hz - 50 kHz**

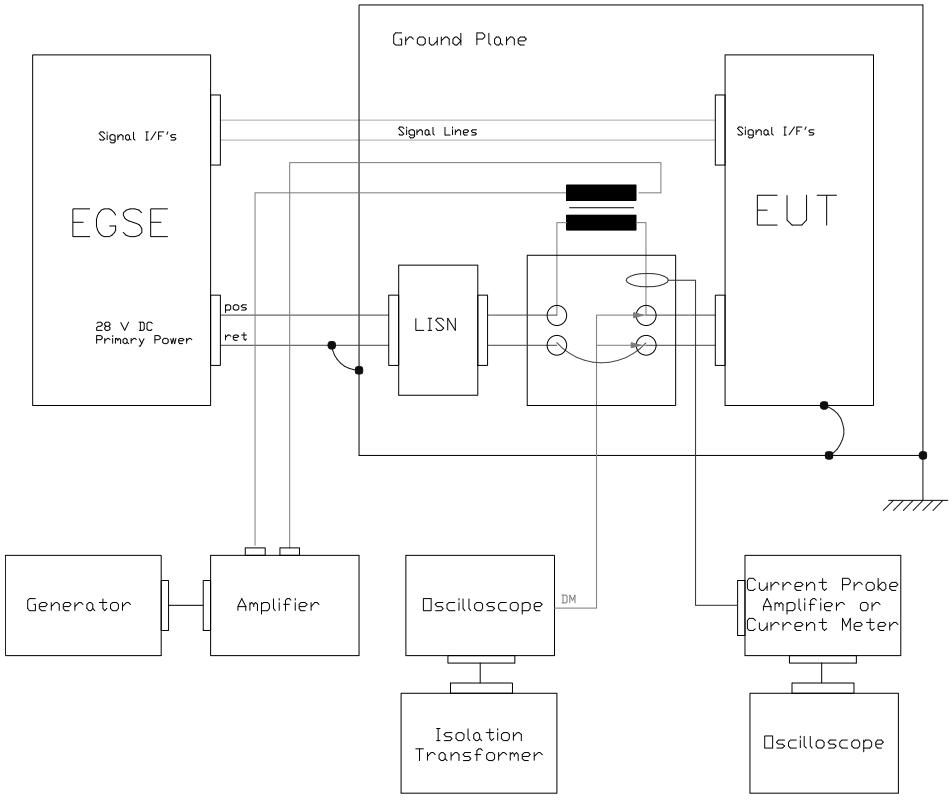


**Fig. 6.5.2.3.2-2: Test Set-up for Conducted Susceptibility on Primary Power Lines, 50 kHz - 50 MHz**

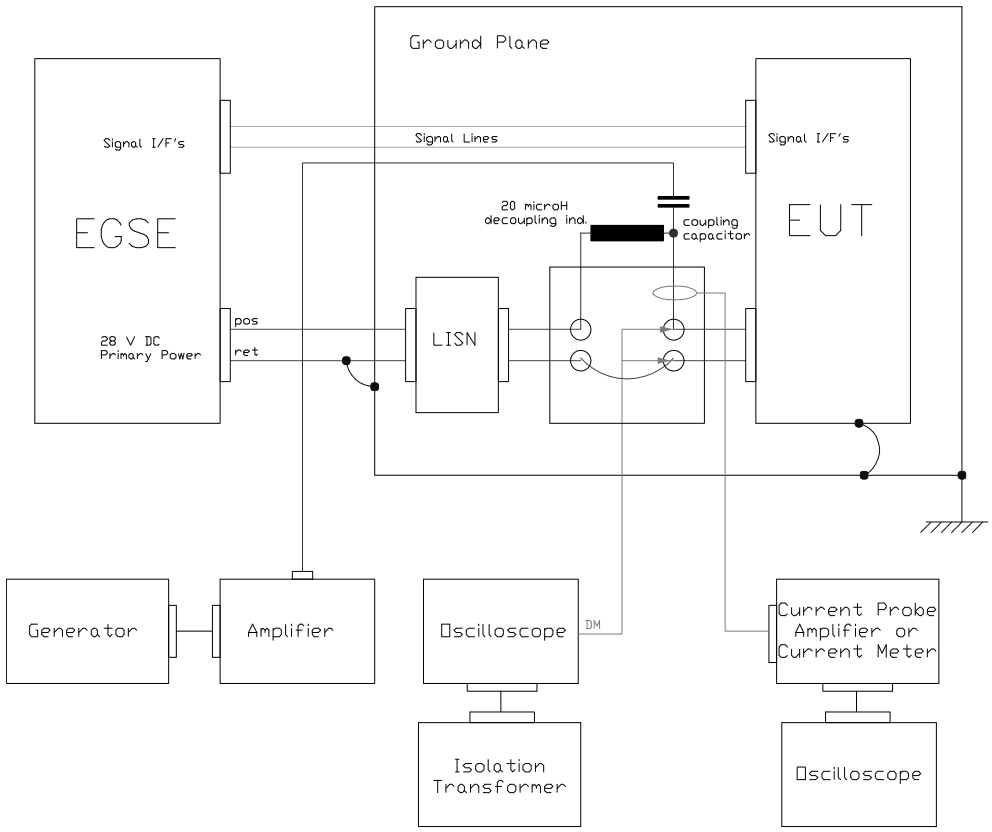


**Fig. 6.5.2.3.2-3: Test Set-up for Conducted Susceptibility on Primary Power Lines, Transients**

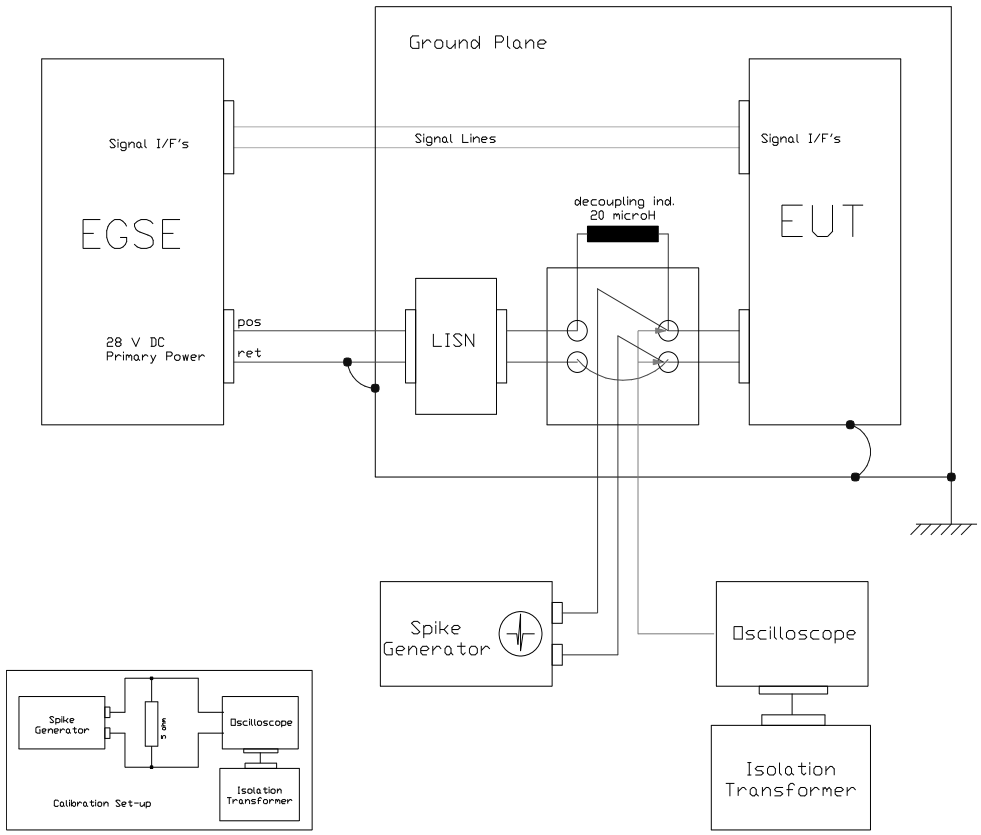
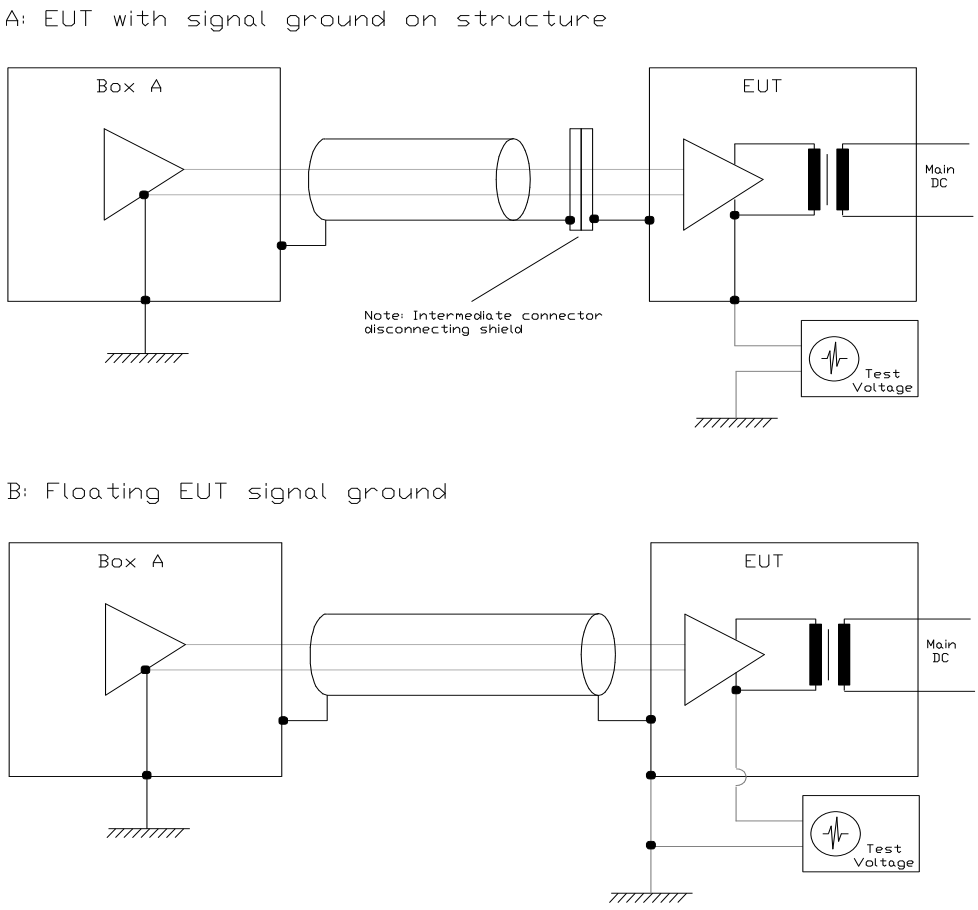


**Fig. 6.5.2.3.2-4: Test Set-up for Conducted Susceptibility on Signal Lines, Transients**



### 6.5.2.3.3 Radiated Emission Tests

**Fig. 6.5.2.3.3-1: Test Set-up for Radiated Emissions, E-Field**

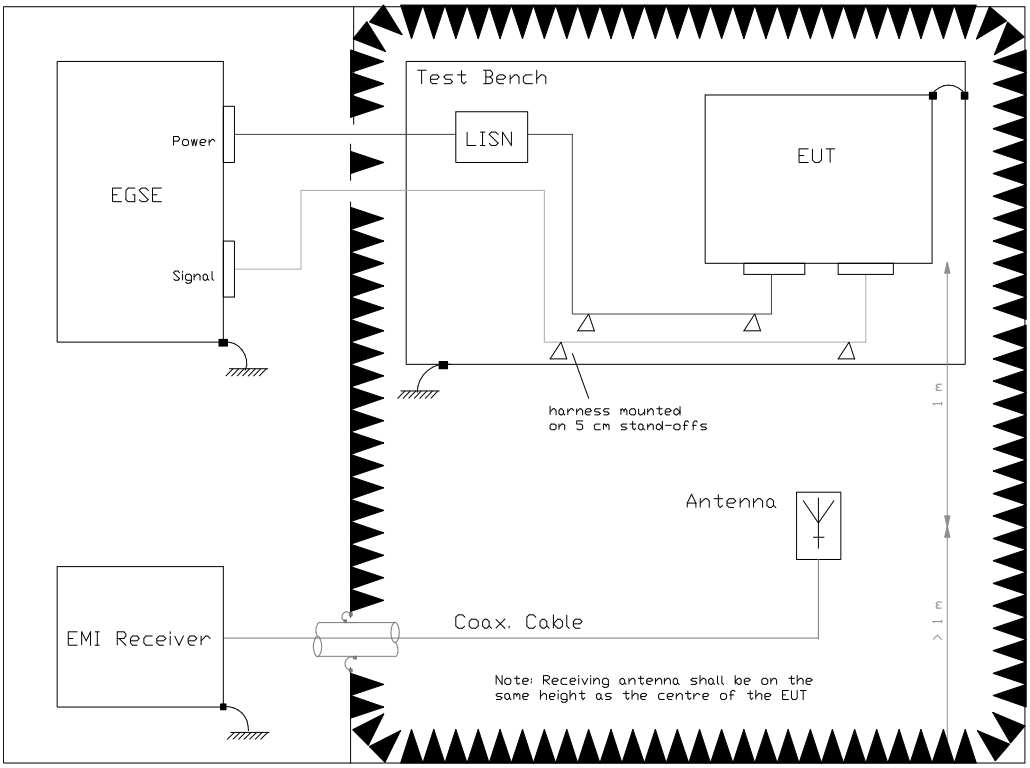
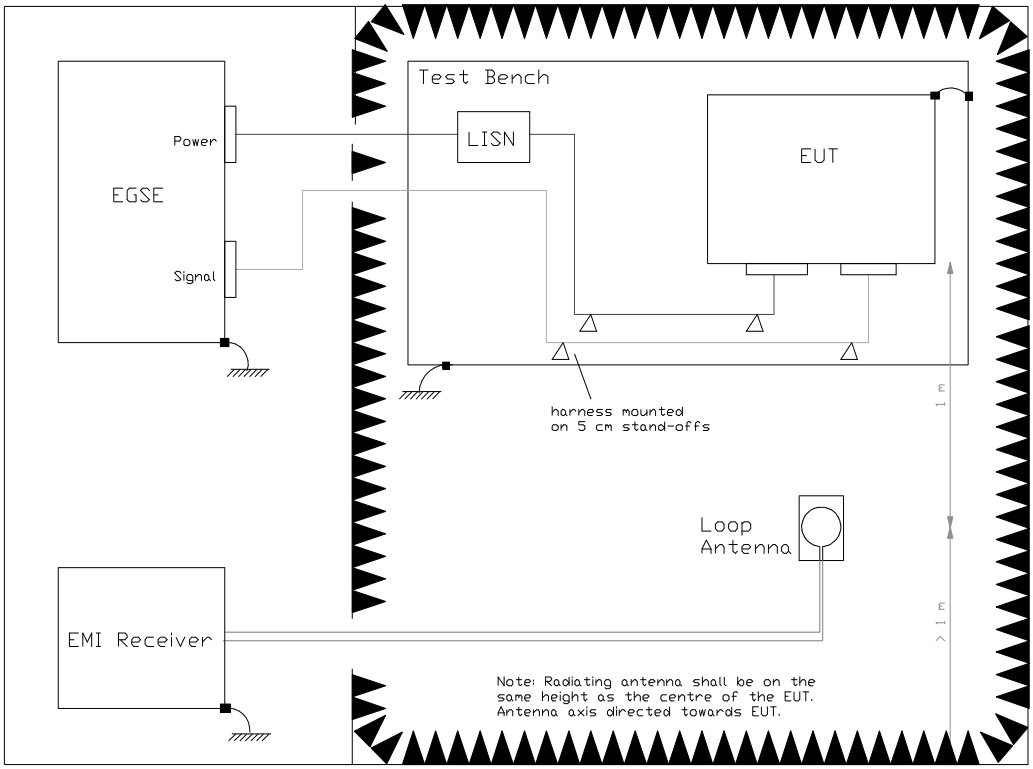


**Fig. 6.5.2.3.3-2: Test Set-up for Radiated Emissions, H-Field**



### 6.5.2.3.4 Radiated Susceptibility Tests

**Fig. 6.5.2.3.4-1: Test Set-up for Radiated Susceptibility, E-Field**

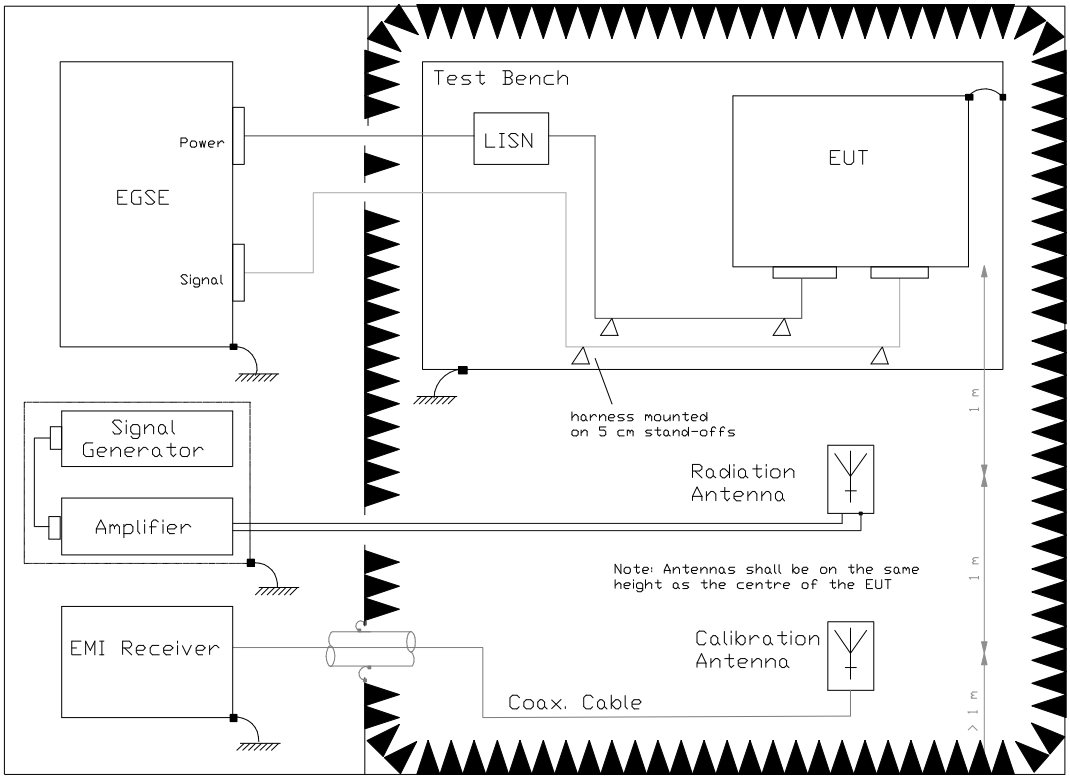
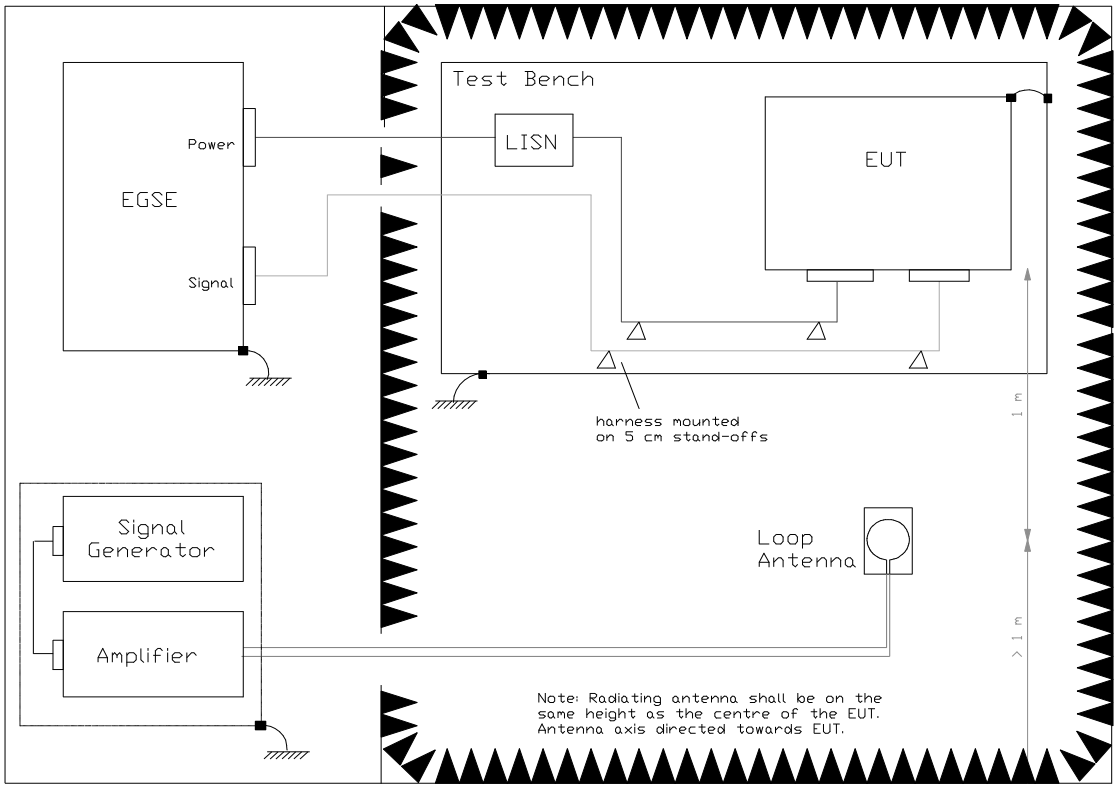


**Fig. 6.5.2.3.4-1: Test Set-up for Radiated Susceptibility, H-Field**



### 6.5.2.3.5 Electrostatic Discharge Tests

**Fig. 6.5.2.3.5-1: Test Set-up for Conducted ESD**

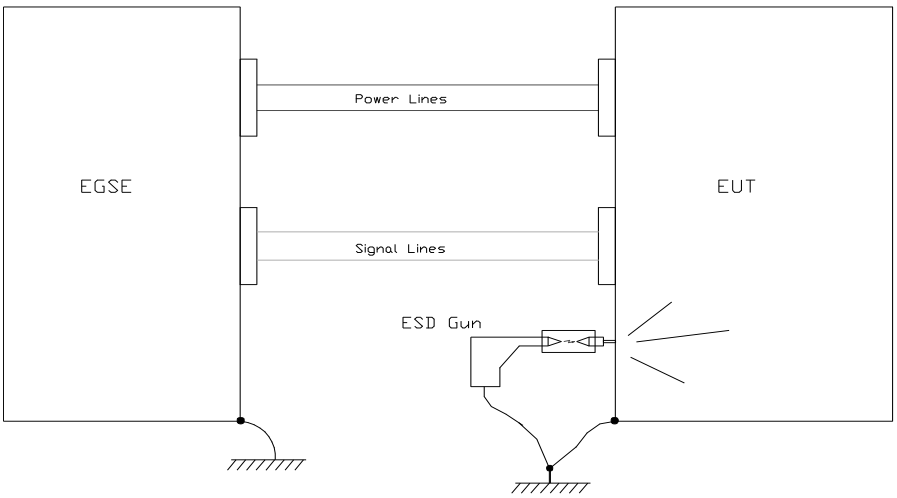
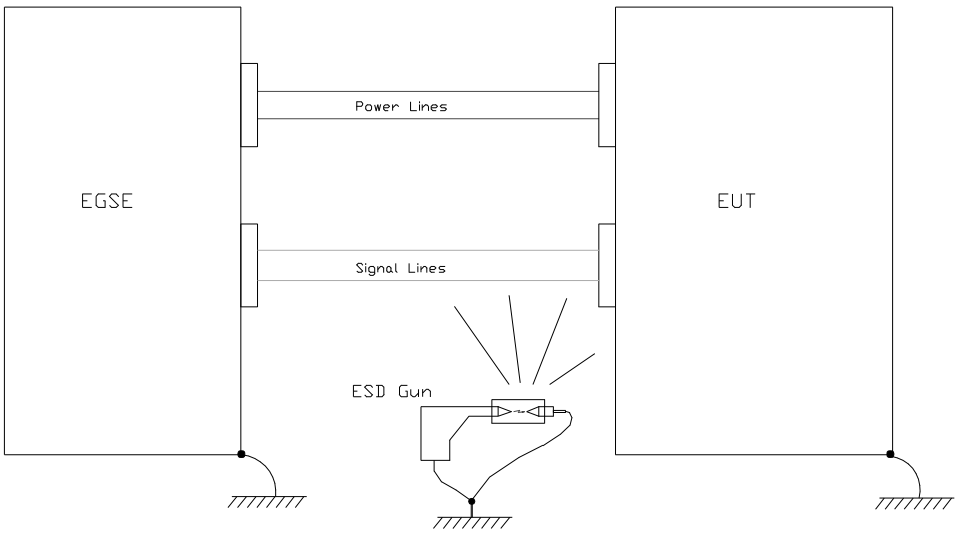



**Fig. 6.5.2.3.5-2: Test Set-up for Radiated ESD**




## 6.5.3 Mechanical and Structural Tests Requirements

#### 6.5.3.1 Introduction

Prior to acceptance by ESA a series of tests shall be conducted to verify the design of the hardware, the interface compatibility, workmanship, and the compliance with the mechanical environment.

#### 6.5.3.2 Structural Testing

##### 6.5.3.2.1 Sine Vibration Test Levels

Qualification levels during sine vibration tests for all the units except the electronic units are specified in the table below:

| Axis | Frequency (Hz) | Level | Sweep Rate |
|---|---|---|---|
| X | 5-18<br>18-100 | ±11 mm<br>±15 g | 2 Oct/min. |
| Y, Z | 5-14<br>14.100 | ±11 mm<br>±8 g | 2 Oct/min. |

The respective qualification levels for the electronic units are:

| Axis | Frequency (Hz) | Level | Sweep Rate |
|---|---|---|---|
| All | 5-22<br>22-100 | ±11 mm<br>±22.5 g | 2 Oct/min. |

Acceptance levels are to be derived by dividing the qualification levels by a factor of 1.5 and to be applied with a sweep rate of 4 Oct/min.

##### 6.5.3.2.2 Random Vibration Test Levels

Qualification levels during random vibration tests for the *detector units* mounted on the Focal Plane Platform and for *all* the electronic units are specified in the Table below:

| Axis | Frequency (Hz) | Level | g (RMS) | Duration |
|---|---|---|---|---|
| All | 20-100<br>100-500<br>500-2000 | +6 dB/Oct<br>0.2 $g^2$/Hz<br>-6 dB/Oct | 12.7 | 2 min per axis |

Qualification levels for units mounted on the Mirror Support Platform are specified in the table below:

| Axis | Frequency (Hz) | Level | g (RMS) | Duration |
|---|---|---|---|---|

| | | | | |
|---|---|---|---|---|
| All | 20-100<br>100-500<br>500-2000 | +6 dB/Oct<br>0.068 $g^2$/Hz<br>-6 dB/Oct | 7.4 | 2 min per axis |

Acceptance levels are to be derived by dividing the qualification levels by a factor of 2.25 and to be applied for 1 min. per axis.

#### 6.5.3.2.3 Acoustic Test Levels

The levels below are extracted from the ARIANE 5 Users Manual
Reference level: $2 \times 10^{-5}$ Pascal.

| Octave Band Centre Frequency (Hz) | Qualification Level (dB) | Acceptance Level (dB) |
|---|---|---|
| 31.5 | 128 | 124 |
| 63 | 134 | 130 |
| 125 | 139 | 135 |
| 250 | 143 | 139 |
| 500 | 138 | 134 |
| 1000 | 132 | 128 |
| 2000 | 128 | 124 |
| 4000 | 124 | 120 |
| 8000 | 120 | 116 |
| Overall Level | 146 | 142 |
| Duration | 2 min | 1 min |

### 6.5.3.3 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.

Tests shall be performed in all mechanisms hardware models within the following test levels:

- Development testing
- Qualification testing (applicable to EQM)
- Acceptance testing (applicable to FM)

Tests required per each levels mentioned above, shall be defined according to mechanisms design and the XMM models philosophy. All the mechanisms specifications shall include the tests required and the sequence to be followed along the verification test programme.

As a reference, the following tests' sequences are applicable:

- Functional tests (before and after thermal vacuum exposure)

- Mechanical environment tests
- Thermal vacuum functional test
- Life test

### 6.5.4 Thermal Test Requirements

#### 6.5.4.1 Introduction

Units shall be subjected to a thermal vacuum test to the limits and conditions specified in the following paragraphs.

#### 6.5.4.2 Definitions

**Operating Mode**: The unit shall fulfil all its operating requirements when within the operating temperature limits specified in Table 6.5.4.3.1.

**Non-Operating Mode**: The unit is not expected to fulfil its requirements within the non-operating limits, specified in Table 6.5.4.3.1, but shall not show any permanent damage.

**Switch-On Temperature**: The cold switch-on temperature, which is the minimal temperature required to avoid permanent damage when power is supplied to the unit, coincides with the minimum non-operating temperature (cfr. Table 6.5.4.5.1).

#### 6.5.4.3 Qualification Thermal Test Requirements

Thermal qualification shall be achieved in a thermal test under vacuum conditions.

Vacuum temperature tests may include corona tests as the inital portion of the vacuum test.

**6.5.4.3.1 Qualification Temperature Limits**
The unit qualification is obtained after verification of the unit performance for temperature conditions more stringent than those met in-orbit.

Table 6.5.4.3.1 gives the temperature limits for qualification.

| Equipment | Operating Mode | | Non-Operating Mode | | Switch On |
|---|---|---|---|---|---|
| | T1 Mini | T1 Maxi | T Mini | T Maxi | |
| All | -20þC | +50þC | -35þC | +65þC | -35þC |

**Table 6.5.4.3.1**

**6.5.4.3.2 Thermal Cycle Vacuum Tests**
The thermal cycle vacuum tests are required to demonstrate the performance of each unit for operating and non-operating mode.

The performance of each unit shall be measured:

- at a pressure equal to or less than $1.33.10^{-3}$ Pascal

- under the temperature conditions defined in Table 6.5.4.3.1.

The temperature shall be achieved at the temperature reference point.

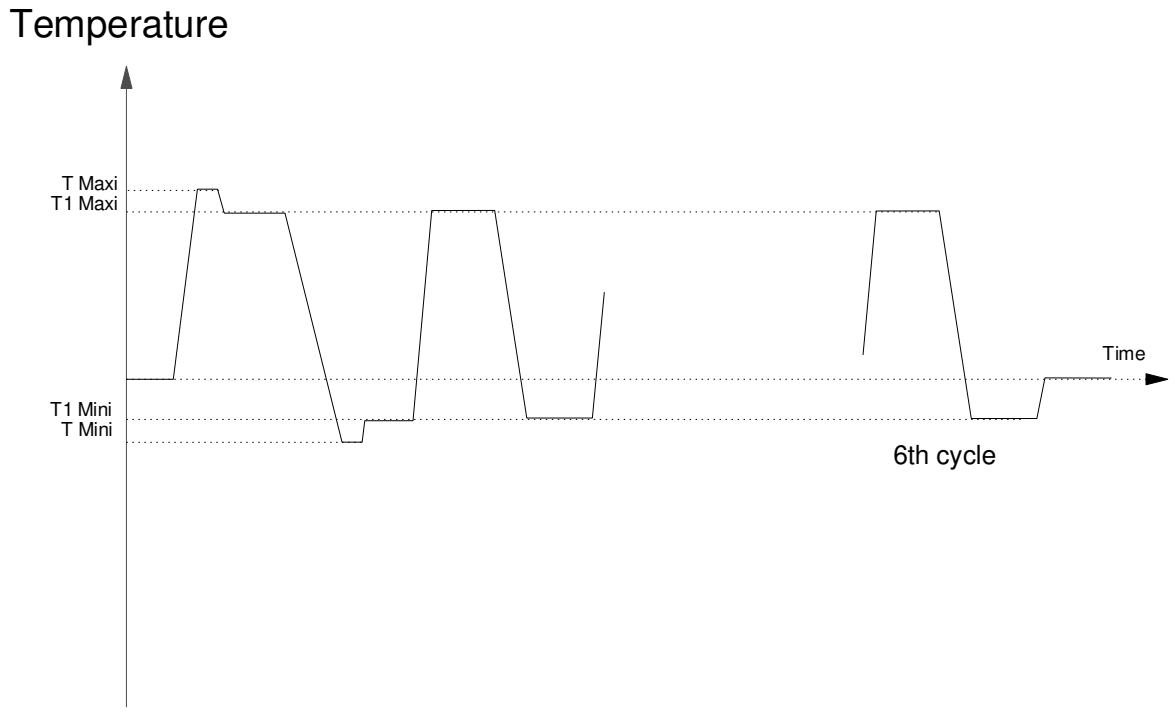
The thermal cycle is represented in Fig. 6.5.4.3.2, six cycles shall be performed.

The variation temperature speed shall be between 0.2þC/minute and 1þC/minute.

For the steady state at the minimum and maximum operating temperature, the performance of the unit shall be tested.

Cold start capability at the minimum switch on temperature shall be demonstrated.

**Fig. 6.5.4.3.2: Temperature Cycling During Thermal Vacuum Test**




### 6.5.4.4 Acceptance Temperature Limits

The table below gives the temperature limits for acceptance. They are 5 degrees below the maximum and 5 degrees above the minimum qualification temperature limits.

| Unit | Operating Mode | | Non-Operating Mode | | Switch On |
|---|---|---|---|---|---|
| | T1 Mini | T1 Maxi | T Mini | T Maxi | |
| All | -15þC | +45þC | -30þC | +60þC | -30þC |

### 6.5.5 Corona Test Requirements

#### 6.5.5.1 Introduction

The purpose of the corona test is to verify that no permanent damaging electrical discharge occurs during transition to vacuum conditions or can be caused by trapped residual partial pressure.
This test is applicable to units with internal voltages of more than 70 Volts.

#### 6.5.5.2 Corona Test

During the corona test the temperature shall be at ambient. The unit under test shall be powered on with high voltages activated and continuously monitored from the start with pump down until $1.33.10^{-3}$ Pascal (pressure of vacuum temperature test) is reached.

The test can be performed as part of the thermal vacuum test.

# 7. GROUND SUPPORT EQUIPMENT

## 7.1 Mechanical Ground Support Equipment

### 7.1.1 Introduction

All mechanical ground support equipment needed to support the instrument during System level AIV operations, shall be provided by the PI, together with written procedures as to its use. The PI shall support the above mentioned activities with trained personnel who are familiar with the instrument and the MGSE equipment.

MGSE shall be supplied in sufficient quantities (one set per instrument model) to ensure efficient integration and test on the various spacecraft.

The type of equipment normally classed as MGSE includes such dedicated items as:

- Instrument handling jigs and/or support frames
- Special tools
- Alignment devices
- Containers or enclosures used to protect the instrument against environmental extremes and contamination during transportation
- Test fixtures, adapters, stands and lifting equipment
- Purging equipment.

### 7.1.2 Test Fixtures and Configuration

Test fixtures required for design qualification of the instrument shall be provided by and be the responsibility of the PI.

Such fixtures shall not contribute to or detract from the test in hand, and shall therefore meet the following requirements:

- The fixture shall be rigid and have a fundamental resonance frequency above 2 kHz

The instrument shall be attached to the test fixture by its spacecraft interface attachment points. Vibration test fixtures shall enable the instrument to be vibrated in all three unit axes.

### 7.1.3 MGSE Cleanliness

The criteria which shall be met by MGSE equipment to ensure that the strict levels of environmental cleanliness and contamination control required can be achieved and maintained are given in AD 1.

### 7.1.4 Transport Containers

The PI shall supply a dedicated container for each instrument model (STM, EQM, FM, FS) and ground support equipment. These containers shall provide adequate protection against the environment, both in storage and in transport.

For sensitive items (e.g. focal camera units, etc.), shock recorders shall be installed during transport.

### 7.1.5 Load Factors for Handling and Transportation

MGSE ultimate load factors shall apply as follows, relative to a loading due to gravity for worst case loading combinations:

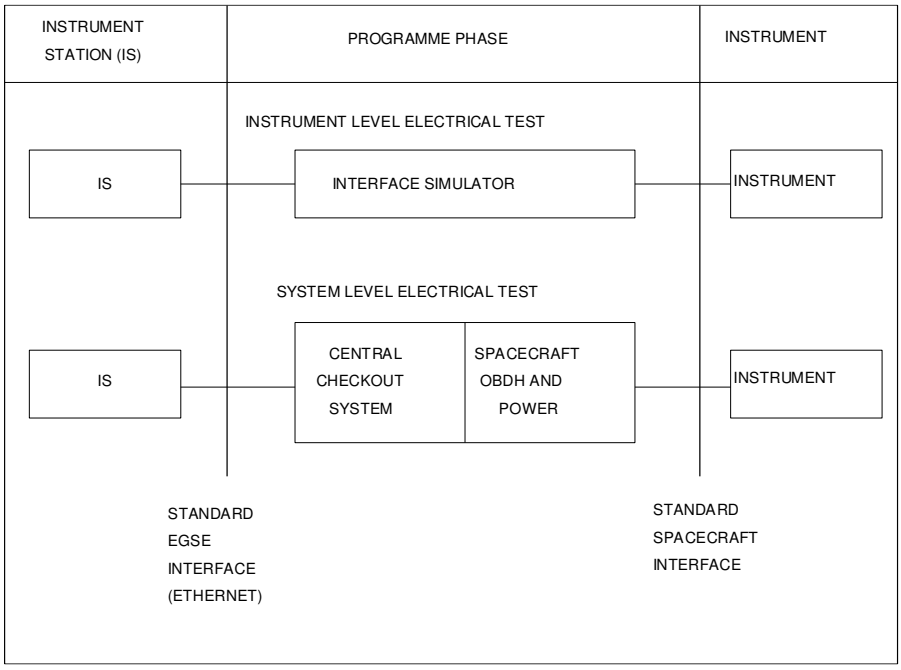
| ITEM | LOAD FACTORS |
|---|---|
| 1. Lifting and tie-down devices | 5 |
| 2. Supporting structures for all flight hardware | 5 |
| 3. Rotating, tilting, turn-over, transporting structure | 5 |
| 4. Metal components other than above | 3 |
| 5. Non-metal components | 10 |
| 6. Workstands | 4 |

## 7.2 Electrical Ground Support Equipment

### 7.2.1 General Concept

The instrument EGSE shall be designed to support the electrical verification programme from instrument level through to system level.

The design baseline is the use of identical interface standards throughout the programme. This will be realised by the specification and imposition of standard protocols on all EGSE elements. The programme can be split into two phases with the following EGSE configurations:



The prime functions of the EGSE in each phase are:

### 7.2.2 Instrument Level Electrical Testing

The Instrument Station (IS) shall be interfaced to the instrument via an interface Simulator (see below). The IS shall perform the following functions:

1) Control of the electrical test using a pre-defined sequence of operations contained in a control file

2) Commanding of the instrument by reading the control file and intepreting it, or by interactive command generation for trouble-shooting

3) Process and analyse instrument housekeeping data to enable the control file to be conditionally sequenced.

4) Process and analyse instrument science data

5) Generate and amend all required control files, housekeeping monitor and telecommand databases.

All the above functions are required to support the acceptance tests.

### 7.2.3 Interface Simulator

The interface Simulator shall be constructed based on interface information provided by ESA.

The Simulator shall simulate the power/data handling interface to the instrument under test and is controlled from the IS.

### 7.2.4 System Level Electrical Testing

For system level electrical testing the IS has to interface to the Central Checkout System via a local area network (ETHERNET).

The IS shall support the real-time evaluation of instrument data at all test levels and shall be operated by instrument personnel.

The IS shall be capable of performing the following functions:

- Accept from the CCS Telemetry via an Ethernet Local Area Network

- De-compose incoming telemetry into various data entities to be stored in a dynamic data base (fixed HK parameters, command verification information, memory dump information, etc.)

- Monitor HK parameters

- Convert raw values to engineering values (when applicable)

- Limit check monitored parameters (when applicable)

- Compute derived parameters (when applicable).

- Perform command verification

- Perform On-Board memory verification

- Display results of parameters monitoring activity on pre-defined alphanumeric displays (display, out-of-limit, alarms, etc.)

- Perform various IS housekeeping functions such as:

  IS initialisation
  IS reconfiguration
  File maintenance and display utilities.

- Maintain a log of all IS activities

- Support the user dialogue defining the man/machine interface required to drive the IS.

- Display of scientific data in graphic form.

### 7.2.5 Central Checkout System

The Central Checkout System (CCS) will support all system level tests of the spacecraft under test. The CCS will sequence all system level tests using only housekeeping data and be the only source of all telecommands to the spacecraft. The CCS will monitor all housekeeping data and use this to give GO/NO-GO results of tests in near real-time. All the telecommands will be issued by the CCS with reference to pre-defined control files held in the CCS for each instrument.

A control file running in the CCS will sequence the test by testing housekeeping values and sending pre-defined commands. Control files will be held on the CCS. The instrument will only be allowed to amend the files after an Engineering Change Request (ECR) has been processed.

The CCS will maintain a full archive of all telemetry, telecommands and test results. This archive will be available for IS to interrogate and copy so as to enable the science data obtained to be verified by the instrument team.

### 7.2.6 Flight Operations

Each PI shall deliver instrument specific software which is required during the flight operations phase. The EGSE software shall form the basis to generate control files, housekeeping monitor and telecommand databases as inputs to the Flight Operations' Procedures and Spacecraft Data Handbook.

The software interfaces will be defined by the ESA Project Office in consultation with the PI.

### 7.2.7 EGSE Interfaces

**Telecommand**

The Telecommand interface differs between bench level using the interface simulator and system level test/operations via the CCS. At bench level there will be a physical interface to the interface simulator specifying commands in near real-time to the instrument. At system/operations level the specifying of commands shall be via a control file given to the CCS/Operations Centre in accordance with the planned schedule. The same interface shall be used for transfer of the HK monitor and command database.

**Telemetry**

The telemetry interface to the IS shall be standard for all phases. It will be in the form of an interrogation of an archive database following CCDS (Consulatative Committee for Space Data Systems) recommendations or standards. Interface bit rates will change depending on the type of connection to the database (local or remote). All telemetry will be in the form of unprocessed whole packets as sent from the instrument.

### 7.2.8 Control Files

The objective of the control files will be to control a test or any action that affects the status of an instrument.

All control files shall be written in an Agency-specified standard high level language (TBD). Standard functions will be supported, for example:

- Send a pre-defined command to the instrument
- Expect the status of the instrument to change within a pre-determined time limit
- Wait for an event or time out
- Test a pre-defined housekeeping or parameter
- Simple loop and branch control.

The objectives of the language will be to allow the PI to specify to the Agency all actions to be performed on the instrument via a simple electronic link, thus avoiding paperwork and duplication of effort.

Control files shall be delivered to the Agency for acceptance at the same time as the instrument. They shall be validated as part of the instrument integration.

Separate control files will be required to perform turn on/off, status changes and mode switching in support of the system level tests and flight operations.

Contingency procedures for instrument responses not covered in a control file shall be provided with the control file documentation.

### 7.2.9 Housekeeping Monitor and Command Database

A housekeeping monitor and command database for each instrument model shall be delivered to the Agency for acceptance at the same time as the instrument. They shall be validated as part of the instrument integration.

The validated database will form the prime input for monitoring and command generation for all flight operations of the instrument.

The database shall be in the form of a standard Agency-defined language TBD.

The database shall be transferred via a simple electronic link to a recognised standard.

### 7.2.10 Hardware

PIs shall deliver identical IS so they can be used as back-up/spares for each other. A suitable configuration shall be agreed with the ESA Project Office. The hardware design shall conform to the following requirements:

- All units shall contain built-in power supplied such that they can be operated from 220/240 volts, 50 Hz mains supply by means of an isolating transformer

- The IS shall distribute mains power between its own units, only one power point for each IS will be supplied by the Agency

- The IS shall supply two spare accessible power outlets for connecting diagnostic equipment

- IS power returns shall be connected to chassis which in turn shall be connected to ground when powered by AC mains supply

- All units shall have protective covers

- The EGSE shall be designed to withstand the prevailing environmental conditions during its usage, storage and transportation.

### 7.2.11 Maintenance

The instrument team shall be responsible at all times for the maintenance of their IS. Diagnostics for all elements shall be available including a real-time programme for data link validation.

### 7.2.12 Stimulation of Instrument

As baseline, all tests at system level requiring stimulation shall be performed using standard telecommands to activate instrument on-board test generators or stimuli sources.

### 7.3 Optical Ground Support Equipment

The OGSE shall include all the ground support equipment needed at all levels for the development, functional performance, qualification and acceptance testing of the instruments. In particular, it will include all equipment necessary for measuring and verifying an instrument's internal alignment and the front end stimulation required for testing and calibrating if this is not part of the instrument.

The OGSE shall include:

- Any equipment, either of standard laboratory type, or
- specifically developed for the XMM Project
- The optical front ends required at relevant test levels
- Software as required
- Any other required facilities.

The OGSE shall:

- Be compatible with and identified for the environment in which it is to be used and/or stores (e.g. vacuum, clean room, transportation, etc.)
- Not contaminate nor damage flight hardware, software nor other GSE (including electromagnetic contamination)
- Be compatible with factors of safety as defined in Section 7.1.5.

The calibration and/or accuracy of critical measuring and/or test equipment shall be substantiated by traceability to National or International standards or by test and analysis methods.

All Optical Ground Support Equipment (OGSE) needed to support the instrument during System level AIV operations or at the system AIV site shall be provided by the Principal Investigator together with written procedures as to its use. The Principal Investigator shall support the above mentioned activities with trained personnel who are familiar with the instrument and the OGSE.

## 8. GROUND OPERATIONS

### 8.1 Delivery of the Instrument to the AIV Site

Following acceptance of the instrument and its associated equipment, software and documentation by ESA, the PI shall be responsible for the shipment of his instrument to the AIV integration site.

Following successful completion of an incoming inspection, the instrument will be released for integration onto the spacecraft.

### 8.2 Integration of Instruments at the AIV Site

Integration followed by Functional Test of the instrument on the spacecraft will be carried out according to a Prime Contractor supplied procedure based on PI inputs supplied prior to integration.

After successful completion of the integration and functional test, the Prime Contractor will issue an Integration Report containing the list of parameters measured and their value.

The PI shall support the integration and functional test phase until satisfactory completion of the task.

### 8.3 Purging Requirements

If it is demonstrated that during system level ground operations purging is absolutely necessary, the PI shall provide and instrument purging plan which shall establish:

- Definition of purging connection

- Detailed purging requirement and procedures (for PI supplied MGSE)

- Assistance in commissioning purging procedures.

During the final phase of launch preparations after mating with the vehicle, the performance of purge operations on a regular basis may violate safety procedures and hence be impossible.

The standard high purity dry nitrogen used during ground operations has the following specification:

- Oxygen — less than 5 parts per million (ppm)
- Water — less than 5 ppm
- Hydrogen — less than 5 ppm
- Argon — less than 1000 ppm

corresponding to $N_2$ type "U" in France or "white spot" in UK.

### 8.4 High Voltage Control Requirements

Instruments generating high voltages which may present a hazard during the ground operation shall provide an enable/disable connector mounted on the relevant instrument unit.

## 9. SCIENCE OPERATION SUPPORT (AD 9)

### 9.1 Mission Phases

The XMM Mission Operations have been divided into the following mission phases:

- Pre-launch phase, starting 30 days prior to launch and terminating at launch minus eight hours.

- Launch and Early Orbit Phase (LEOP), starting at launch minus eight hours, including Ariane lift-off, orbital injection, check-out of essential spacecraft functions, performance of first orbit manoeuvre. This phase will last a few days.

- Switch-on phase will include completion of the spacecraft checkout, further orbit manoeuvres and achievement of the operational orbit, initial instrument testing, and opening of the Telescope door. This phase will last about 2 weeks.

- Calibration phase will include the initial calibration of all instruments as well as further Spacecraft performance testing. This phase will last about 1 month.

- Scientific performance verification phase will include determination and demonstration of the Observatory's Scientific Capability. This phase will last about 2 months.

- Scientific Mission Phase will then commence until mission termination.

- Post mission phase will commence after mission termination.

### 9.2 PI Responsibilities

#### 9.2.1 Development Phase

i) Instrument-specific S/W identified in SOC user requirements for the reduction and analysis of instrument data (for possible re-implementation in SOC) with the corresponding S/W documentation.

ii) Inputs required by the MOC/SOC for the development of the instrument simulators.

iii) The instrument data bases (checkout, calibration, etc.)

iv) Participation in Ground Segment Reviews.

v) Instrument on-board software and documentation for

. validation of simulator
. operations preparation

vi) Instrument on-board software maintenance environment description.

vii) Interface definition for Instrument on-board S/W maintenance systems.

viii) Instrument ground test and calibration raw data and processed results.

ix) Instrument User Manuals.

x) Instrument checkout sequences and procedures.

xi) Inputs to instrument operations procedures.

### 9.2.2 Operational Phase

i) Manpower required to support the listen-in tests and two SVTs (System Validation Tests) and the initial phases of XMM orbital operations (switch on + calibration).

ii) Support in anomaly situations.

iii) Support for maintenance of instrument on board S/W during the initial scientific operation.

## 9.3 Calibration Provisions

The baseline design of the spacecraft assumes that any in-flight calibration function, hardware or softwarem, is specific to the instrument and will be a built-in feature of the instrument design.

The calibrator shall not only be used during flight operations but shall also be exercised as a tool during bench and system testing to establish the status of the instrument and to get a history of the instrument behaviour over its ground life.

It is required that the PI explicitly details in the EID Part B, the provisions for in-flight calibrations and the extent to which the instrument or units can be checked out during ground testing and in-orbit operations. The duration of a calibration cycle shall be stated.

## 10. DELIVERABLE ITEMS

### 10.1 Deliverable Models

During the course of the XMM Programme the following instrument models shall be delivered to the Agency by the Principal Investigator:

1. **Electro-Optical Breadboard (EOBB)**
   The Electro-Optical Breadboard shall include a fully operational detector unit which can be used in vacuum for an end-to-end X-ray performance test with a representative mirror module in an X-ray facility. The EOBB shall be electrically and mechanically representative of the instrument interfaces, although the data handling need not be flight standard and have various functions omitted. For optical units (e.g. reflection grating boxes), the EOBB shall be mechanically representative of the instrument interfaces but need not include all the optical elements.

2. **Structural/Thermal Model (STM)**
   The Spacecraft Structural and Thermal Model will undergo structural and thermal balance verification and environmental tests. The PI shall deliver instrument hardware which shall accurately simulate the structural (mass, first eigenmode) and thermal properties of the instrument, external interfaces and geometry (including stand-off structure, heat pipes, radiators, radiator supports, alignment devices, etc.). It shall include connectors at their prescribed locations. For PIs who propose flying the same generic instrument at the focus of more than one mirror module, complete sets of units shall be supplied.

3. **Engineering Qualification Model (EQM)**
   The instrument EQM shall be, in all respects, representative of the Flight Model except that flight quality components need not be used in its construction. It is therefore permitted to use commercial grade parts instead of hi-rel parts. The manufacturing processes and procedures for these parts, however, have to be identical to the ones used later for flight purposes. Before delivery, the EQM shall have undergone, successfully, a full environmental and functional qualification programme.

   The EQM will be used at spacecraft level primarily for electrical, optical, functional and mechanical compatibility and EMC/ESD verification on the Electrical Model spacecraft.

   For instruments with more than one identical chain on the spacecraft, only one EQM need be delivered. For the other instruments, the PI shall deliver models to a definition agreed with the XMM Project Office.

4. **Flight Models (FM)**
The Flight Models shall be of a standard compliant with all the requirements of EID Parts A and B and shall have successfully undergone a full programme of acceptance level testing and verification prior to delivery.

The instrument will be integrated into the Flight Model spacecraft which will undergo an environmental test programme and full electrical verification prior to launch.

The model shall be fully calibrated prior to delivery.

5. **Flight Spares (FS)**
These units shall be identical to Flight units and shall undergo the same test programme as flight models.

The flight spare philosophy shall be agreed with the ESA Project Office.

### 10.2 Deliverable Ground Support Equipment

#### 10.2.1 Mechanical Ground Support Equipment

The Mechanical Ground Support Equipment to support the instrument during transportation, handling and integration shall be delivered (with each instrument model) together with the appropriate documentation.

Any MGSE which is re-used from one instrument model to another at the integration site shall have been refurbished where appropriate and re-certified for use during the expected duration of the relevant part of the programme.

All MGSE proof load and test certification shall accompany the MGSE at all times.

Other items of mechanical support equipment such as protective devices, alignment devices and purge connections shall be specific to each individual instrument and delivered together with the appropriate instrument model.

In the event that test fixtures are required in order to perform any test on the equipment pre- or post-delivery, it shall be the responsibility of the Principal Investigator to provide them.

#### 10.2.2 Electrical Ground Support Equipment

The PI shall deliver with the Electro-Optical Breadboard (EOBB) a limited set of Electrical Ground Support Equipment (EGSE) and software to support the end-to-end X-ray test on the EOBB in vacuum.

The PI shall deliver with the instrument EQM, one complete set of EGSE and associated software.

The delivered EGSE shall be either replaced or updated at each subsequent model delivery to maintain compatibility. This may require changes to software or hardware as appropriate. The system shall be maintained under full configuration control at all times after delivery.

This equipment shall remain on the spacecraft integration site to support the instrument throughout all subsequent system level activities through to launch.

It shall be the responsibility of the Principal Investigator to provide full spares and provide maintenance facilities on the EGSE such as to minimise any schedule delay at system level caused by EGSE malfunction.

Operation and maintenance of PI-supplied EGSE shall be the responsibility of the Principal Investigator at all times. He shall provide the necessary expert manpower to enable operation of the EGSE and, where applicable, analysis of the data from the instrument at all stages of the system level programme where this is required by the Agency.

### 10.2.3 Optical Ground Support Equipment (OGSE)

Optical Ground Support Equipment to support the instrument during any integration, alignment or test activities shall be delivered with each instrument model together with the appropriate documentation.

Any OGSE which is re-used from one instrument model to another at the integration site shall have been refurbished where appropriate and rectified for use during the expected duration of the relevant part of the programme.

## 10.3 Deliverable Software

The following software-related items shall be delivered by the PI in support of the system level electrical verification programme:

- Software sequences required to perform the instrument level acceptance tests
- Software sequences required to perform the system level tests in real-time or off-line mode
- Auxiliary software sequences necessary to perform investigations, including diagnostics on the instrument, instrument EGSE and interfaces
- Detailed documentation of delivered software (e.g. software specifications, functional flow diagrams, coding, test cases).
- Instrument database (TM/TC definition, test results, HK parameter definition etc.)

The deliverable software and documentation for the support of the SOC is specified in para. 9.2.