XMM's Electrical Power Subsystem

B. Jackson
XMM Project, ESA Directorate for Scientific Programmes,
ESTEC, Noordwijk, The Netherlands

The XMM Electrical Power Subsystem (EPS) is 'conventional' in design in that it follows the ESA Power Standard. This article describes the subsystem's main features and performances, and the steps that have been taken in implementing the lessons learnt from the highly successful SOHO spacecraft recovery efforts, which were described in detail in ESA Bulletin No. 97.

EPS design drivers

The principal mission and spacecraft characteristics influencing the design of XMM's Electrical Power Subsystem (EPS) were:

- Power requirements: 1600 W in Sun, and 600 W in eclipse
- Spacecraft geometry: Two separate Power Distribution Units (PDUs) to reduce harness mass, and to simplify system testing and assembly, integration and verification (AIV), the Service and Focal-Plane Assembly Modules have dedicated PDUs
- Mirror stability: Leading to a dedicated Mirror Thermal Control Unit (MTCU)
- Orbit: (a) 255 eclipses, with a maximum duration of 1.7 h. The eclipses occur when the spacecraft is close to perigee, when the instruments are turned off. This leads to a lower eclipse power requirement and allows smaller batteries to be used; (b) Incomplete ground coverage requires the design to be failure tolerant. It was also a design requirement that the spacecraft should be able to survive for up to 36 h with a failure and without ground intervention.
- Lifetime 10 years: Component total radiation dose levels of 75 krad were taken as a reference, and maximum use was made of spacecraft shielding and rad-hard components. Solar-array degradation is also affected by radiation.
- Telescope: Telescope pointing requirements result in a maximum off-pointing of the solar array from the Sun of 28 deg.

In addition, during the XMM Service Module design phase, the technical requirements were harmonised with those of ESA's gamma-ray observatory spacecraft Integral, leading to a common Service Module design for the two spacecraft.

The resulting EPS design is comprised of the following elements:

- A fixed two-wing deployable solar array for power generation.
- Two nickel-cadmium batteries.
- A Main Regulator Unit (MRU) providing a main bus regulated to 28 V.
- Two independent Power Distribution Units: one for the Focal-Plane Assembly (FPA-PDU) and the other for the Service Module (SVM-PDU).
- A Mirror Thermal Control Unit (MTCU) dedicated to the thermal control of the mirror platform, Mirror Modules and Reflection Grating Assemblies.
- A Pyrotechnic Release Unit (PRU) for automatic activation of the release mechanisms for the solar arrays, and for the initial transmitter and AOCS activation.

XMM's EPS

The overall layout of the EPS is shown in Figure 1, where it can be seen that the Main Regulator Unit (MRU) controls the electrical power generation from the two solar-array wings, the energy storage within the two batteries, and provides the main bus regulation. The power distribution to the various 'users' is performed by the SVM-PDU for the Service Module (SVM) and Mirror Support Platform (MSP) units, and by the FPA-PDU for the units located in the telescope's Focal-Plane Assembly (FPA). Each PDU user outlet is protected by either a Latching Current Limiter (LCL) for switchable users, or by a Foldback Current Limiter (FCL) for permanently powered users.

The MTCU provides thermal control for the three Mirror Modules and the two Reflection Grating Assemblies. In addition, to meet the spacecraft autonomy requirements, it provides a self-monitoring function, which in the event of a failure to respect critical temperature limits performs an automatic switchover to the redundant channel.

The PRU activates the solar-array deployment mechanisms and spacecraft pyrotechnics, as well as automatically configuring the spacecraft into a predefined status at launcher separation.

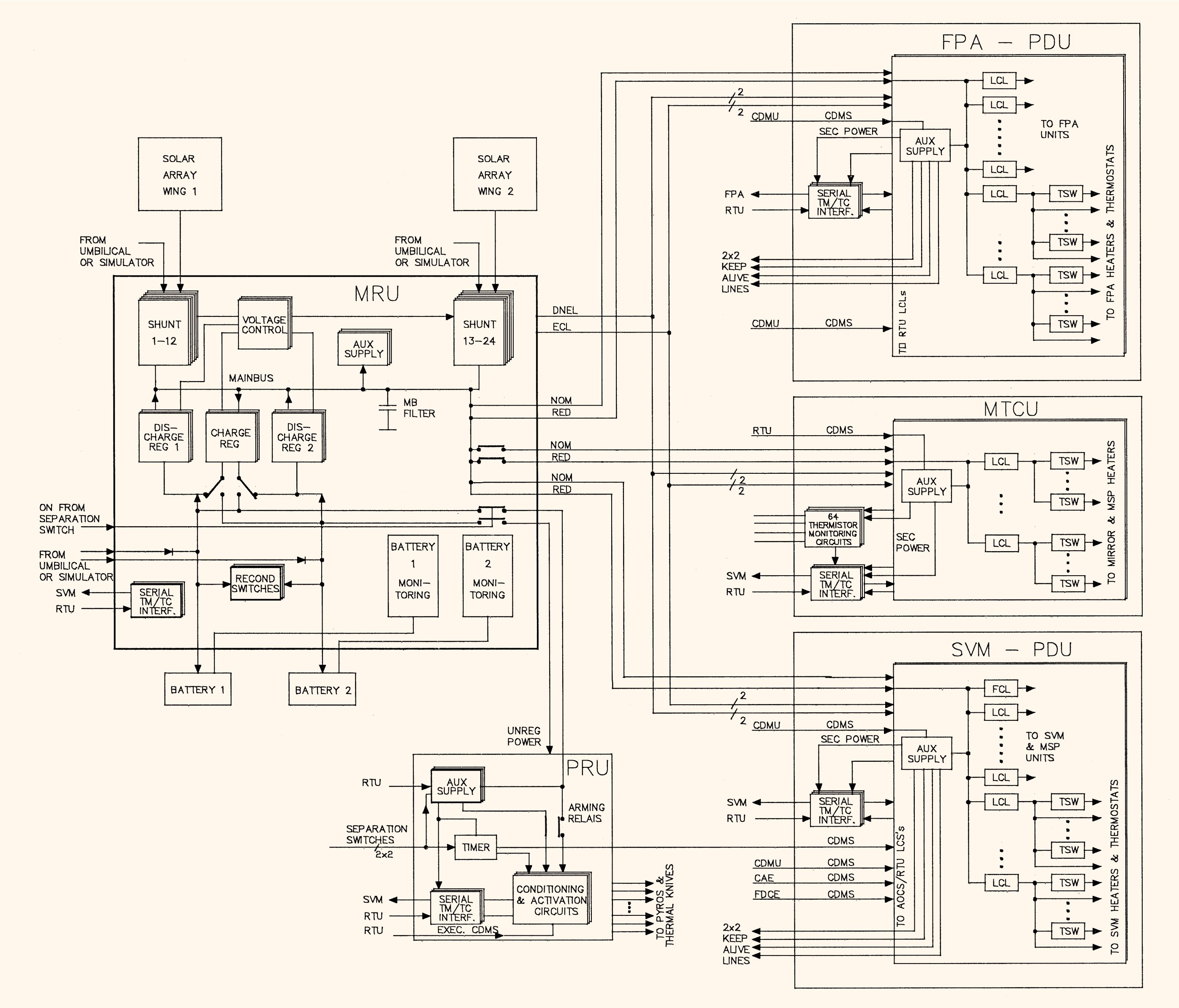

Figure 1. Overview schematic of XMM's Electrical Power Subsystem (EPS)

The solar array
The solar array has two wings (Fig. 2), each with three rigid 1.94 m x 1.81 m panels, giving a total area of 21 m^2 and a mass of 81.4 kg. The two wings are body-fixed and have a Sun incidence angle variation around normal of up to 28 deg. At end-of-life (EOL = 10 years), in the worst case, including one failed section, the solar array is required to provide 1600 W at 30 V at the interface connectors.

During the launch and the early orbit phase, the three panels of each wing are folded and stowed on four hold-down points on the spacecraft. Kevlar restraint cables keep the solar array in this folded position until they are cut on a command from the PRU.

Each wing carries 12 power sections with four sections per panel. The solar-cell strings are interconnected by blocking diodes and discrete panel wiring on the rear of the panel. Power delivery is divided equally between the 24 sections of the array. The 2 ohm.cm silicon Back-Surface Reflector (BSR) solar cells (210 microns), with 300 micron CMX cover glasses, are mounted on CFRP panels. To prevent electrostatic discharges, the cover glasses are coated with conductive indium-tin oxide.

The batteries
XMM has two identical 24 Ah nickel-cadmium batteries (Fig. 3), each with 32 cells. Each 573 x 188 x 222 mm^3 battery weighs 42 kg. To allow for cell short-circuits, the nominal energy budget is calculated with 31, rather than the full 32 cells. To allow for high peak power demands, a battery voltage higher than the bus voltage has been chosen, and battery reconditioning will be performed before each eclipse season.

The MRU
The Main Regulator Unit provides a 28 V regulated main bus voltage, with protection to

Figure 2. One of XMM's two solar-array wings (flight model), each of which has three folding rigid panels

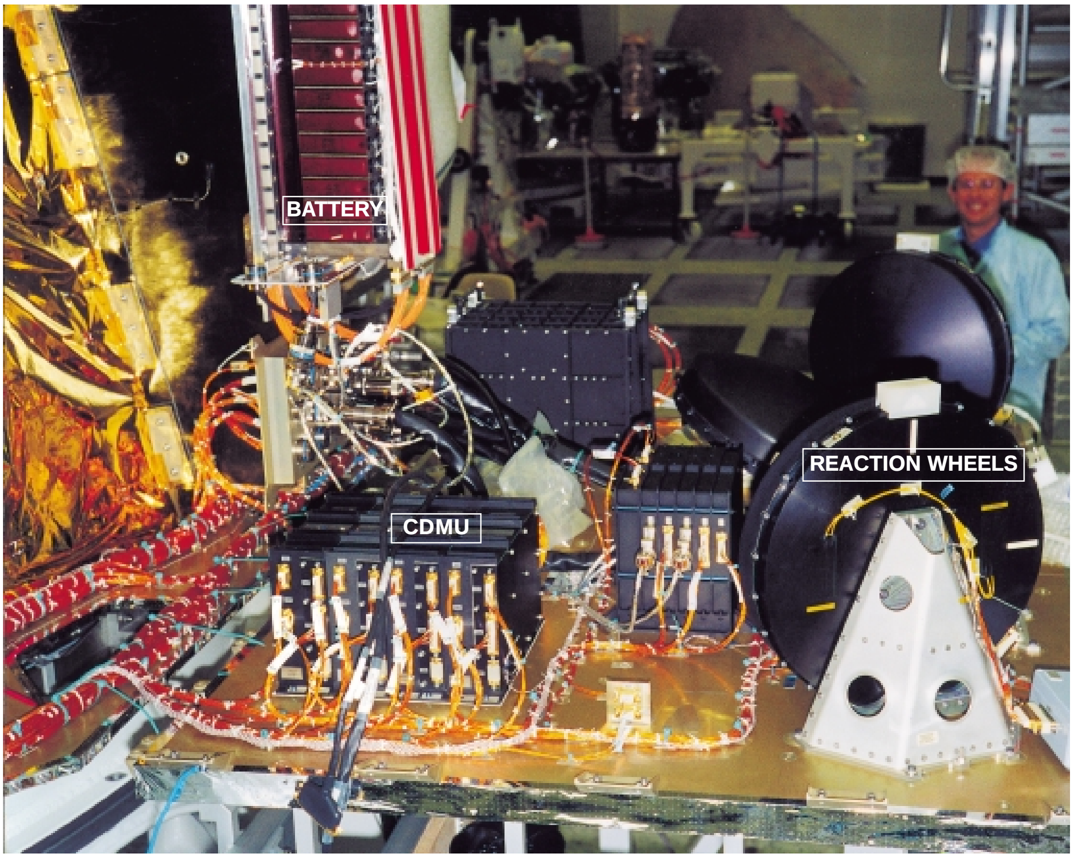

Figure 3. One of XMM's four equipment panels, carrying items powered by the Electrical Power Subsystem: front left, the Command and Data Management Unit (CDMU) and, far right, the AOCS Reaction Wheels; in the upper left corner, one of XMM's two onboard batteries is visible

ensure uninterrupted operation even in the event of a single-point failure. During sunlit periods, the MRU provides power via Sequential Switching Shunt Regulators (S3Rs), as well as managing battery charging. In eclipse mode, the MRU controls the discharging of the two batteries to ensure correct current sharing. To reduce the power demand on the batteries and ensure that all non-essential loads are switched off during eclipse, an eclipse signal ECL is generated by the MRU and sent to the PDUs and MTCU.

To provide protection against a battery over-discharge that could lead to a cell voltage inversion, a Disconnect Non Essential Loads (DNEL) signal is generated by the MRU and can be sent to the PDUs and MTCU. A DNEL signal will be issued as soon as the voltage from one group of four cells of either battery falls below 3.9 V.

Table 1. Key MRU features

Dimensions	540 x 520 x 180 mm^3
Mass	24 kg
Output voltage	28 V +1%-2%
Max bus load demand in sunlight	2100 W
Maximum output power in eclipse	1680 W

Table 2. Key PDU features

	SVM PDU	FPA PDU
Dimensions	455 x 270 x 200 mm^3	312 x 270 x 200 mm^3
Mass	16 kg	11kg
LCL/FCL solid-state switches	2 x 24	2 x 12
TSW heater switches	2 x 24	2 x 24
Keep-alive lines	2 x 2	2 x 2
Maximum continuous load current	25 A	20 A
Maximum peak load current	50 A	30 A

The MRU provides 2 x 2 power lines for the SVM-PDU and the FPA-PDU, and 1 x 2 switched lines for the MTCU. For ground testing, it provides interfaces with solar-array and battery simulators.

The MRU is composed of the following elements:
– bus capacitor and Main Error Amplifier (MEA)
– 24 non-redundant S3R sections (sunlit regulation)
– 2 x 2 hot-redundant Battery Discharge Regulator (BDR) modules for eclipse regulation
– 2 hot-redundant Battery Charge Regulator (BCR) modules to control battery charging
– 2 x 2 power relays to switch the MTCU and PRU on and off
– redundant telemetry/telecommand interfaces
– hot-redundant DC/DC converters for MRU auxiliary supplies.

The PDUs

XMM has two independent and fully redundant PDUs, one for the Focal-Plane Assembly (FPA-PDU) and one for the Service Module (SVM-PDU). Both are supplied from the spacecraft 28 V bus. The outputs can be configured by ground telecommanding, or using a read-only memory (ROM). The PDUs are able to reconfigure their outputs to a predefined configuration, for example in the case of an eclipse or a low battery voltage (ECL, DNEL).

The PDUs provide power distribution via two types of solid-state current limiters:
– Fold-Back Current Limiters (FCLs): for powering receivers, decoders, and essential AOCS units that are always powered on. In the event of an overload, these current limiters enter a fold-back mode.
– Latching Current Limiters (LCL): which operate as telecommanded switches with current limitation protection (thresholds 1A, 2A, 3A, 4.5A or 7A). The switch automatically trips off if the current limitation period exceeds a maximum time limit. The PDUs also provide groups of heater outlets, each outlet being controlled by a transistor switch (TSW) and protected by an LCL.

In addition, the PDUs provide keep-alive lines to maintain instrument memories during eclipse.

The MTCU

The Mirror Thermal Control Unit is responsible for controling and monitoring XMM's Mirror Assemblies (MA) and their support platform (MSP) in order to maintain thermal stability. It operates from the MRU regulated bus and is protected against output line short-circuits.

Latching Current Limiters (LCLs), each of which drives three transistor switches, perform the MSP and MA heater control. An 8 bit telecommand controlling the pulse-width modulation of the TSWs controls the power fed to the individual heaters. In addition, for failure management, the MTCU monitors four critical temperatures and provides automatic switch-over to the redundant side in the event of a temperature-limit failure.

At power on, eclipse and DNEL transitions, the MTCU will automatically configure the heaters into a default status.

The PRU

The Pyrotechnic Release Unit provides sequential activation of the spacecraft subsystems and release mechanisms. The PRU is internally redundant, and it will be 'off' until the spacecraft's separation from the

Table 3. Key MTCU features

Dimensions	402 x 274 x 220 mm^3
Mass	15.8 kg
LCL (3A) solid-state switches	2 x 8
TSW heater switches	2 x 24
Maximum continuous load current	24 A

Table 4. Key PRU features

Dimensions	250 x 250 x 200 mm^3
Mass	5.5 kg
LCL (3A) solid-state switches	2 x 8
Thermal-knife outputs	8 + 8
EED outputs	8 + 8

launcher. It will then be connected to the unregulated battery supplies and will autonomously trigger a timer to power-up basic spacecraft subsystems (AOCS and transmitter) and activate the thermal knives for deploying the solar arrays. Via arming relays and arm/fire switches operated by telecommand, the PRU will fire the Electro-Explosive Devices (EEDs) needed to activate the spacecraft subsystems.

Lessons learnt from SOHO's recovery

Following the successful recovery of the SOHO satellite, the XMM design was re-examined to see how, even in the case of a catastrophic failure, including the loss of the spacecraft main power bus, the probability of recovering the spacecraft could be improved. As a starting point it was assumed that the spacecraft would be in a worst-case configuration, that it would have totally-discharged batteries, would have lost attitude, and that the solar arrays would no longer be Sun-pointing, and with most sub-systems cold. In this case, main-bus recovery could only occur when the solar arrays came into sunlight and provided sufficient power to support all connected spacecraft loads. At such a time, it can be assumed that:

– the spacecraft is spinning and not stable and Sun-pointing
– at least some parts of the reaction control system are frozen
– survival thermostats are closed, connecting main (and redundant!) heaters to the spacecraft bus.

Significant lessons learnt from SOHO were:

– to switch on the transmitters as soon as possible (to provide a 'beacon') to show the ground control centre that the spacecraft is powered, and its location
– to minimise power consumption at switch-on and to power up spacecraft units sequentially, allowing time for ground intervention; it is even desirable not to instantly power on the AOCS for example, in case the RCS is frozen. On XMM, the bus load at power up is now limited to 'essential units' only (MRU, receivers, transmitter, Command Data Unit) to maximise the probability that the spacecraft main bus will recover
– to ensure that if sufficient power is available, the spacecraft will recover even without ground intervention
– to ensure that the batteries can be charged as quickly as possible; e.g. to recover a spinning spacecraft, the batteries would be required to power the spacecraft whenever the solar arrays were shadowed and for long enough to process commands and ultimately for the AOCS to stabilise the spacecraft.

Throughout all of the investigations into possible improvements, great care was taken not to create new failure modes that might jeopardise the mission. The changes were to be implemented at a late stage in the programme, when most flight units had already been built and were being integrated. However, the inherent flexibility of the design and the timely work by the spacecraft contractors involved allowed it to be done with minimum impact on the overall schedule.

Ultimately, the following modifications were made to XMM:

– New timer units were added, which generate pulses to the PDU control inputs.
– The ROMs controlling the PDU output configurations were modified.
– The harness was modified so that direct commands (commands from the CDU which are available directly at switch-on without the need to load software or power-on RTUs) were routed to allow the ground to power-off the MTCU and Timer units.
– In addition, the MRU was modified to ensure that the batteries went into full charge mode at power-on (using excess power only!).

The modifications were successfully tested on the integrated flight-model spacecraft and showed that, for example, initial spacecraft power-on now occurs at less than 300 W, compared to a power-on figure prior to modification of more than 1500 W. This provides increased confidence that, even in the unlikely event of a total power loss, the probability of recovering the spacecraft is significantly increased and that it would now begin to recover for Solar Aspect Angles (SAAs) of up to 80 deg.