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The ARA Mark 3 Solar Array Design and Development

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Abstract

The ARA (Advanced Rigid Array) Mark3 solar array of Fokker Space BV is currently in its final stages of qualification (wing tests to be completed in March, 1996; unit/part tests in April, 1996). With regard to its predecessor, the ARA Mark2, the design has not only been improved in terms of mechanical and electrical performance, but also with regard to production cost and throughput time. This 'state of the art' array is designed to fit the needs of a wide variety of geostationary telecommunications satellites and is qualified for launch on the complete range of medium/large size commercial launchers (Ariane IV & V, Atlas, Delta, Proton, Long March, H2).

The first mission to fly the new ARA Mk3 array is Hot Bird 2 (customer: Eutelsat, prime contractor: Matra Marconi Space; launch: mid-1996). In this configuration, its end of life (EOL) power-to-mass ratio is 42 W/kg, with an operational life of more than 12 years.

The main mechanisms on a solar array are typically found in the deployment system and in the hold down and release system. During the design and development phase of these mechanisms, extensive engineering and qualification tests have been performed.

This paper presents the key design features of these mechanisms and the improvements that were made with regard to their predecessors. It also describes the qualification philosophy on unit/part and wing level. Finally, some of the development items that turned out to be critical, as well as the lessons learned from them, are discussed.

Introduction

At the start of the development program, an extensive study was carried out to identify the advantages and disadvantages of the previous ARA Mk2 design, based on the current and future solar array market demands. For this purpose, a range of possible applications was identified, such that the newly designed solar array family would be usable without any delta qualification being required on unit or wing level.

With this in mind, key requirements were defined at subsystem (solar array) level and were carefully translated and budgeted down to unit and part level. This resulted in the definition of critical design parameters that were to be monitored throughout the design process. This process is described in Figure 1, which shows the flow from detailed unit design to production and further on to unit qualification testing. Also, the connection with the activities on subsystem level, such as analytical modeling, qualification testing and correlation, is given.

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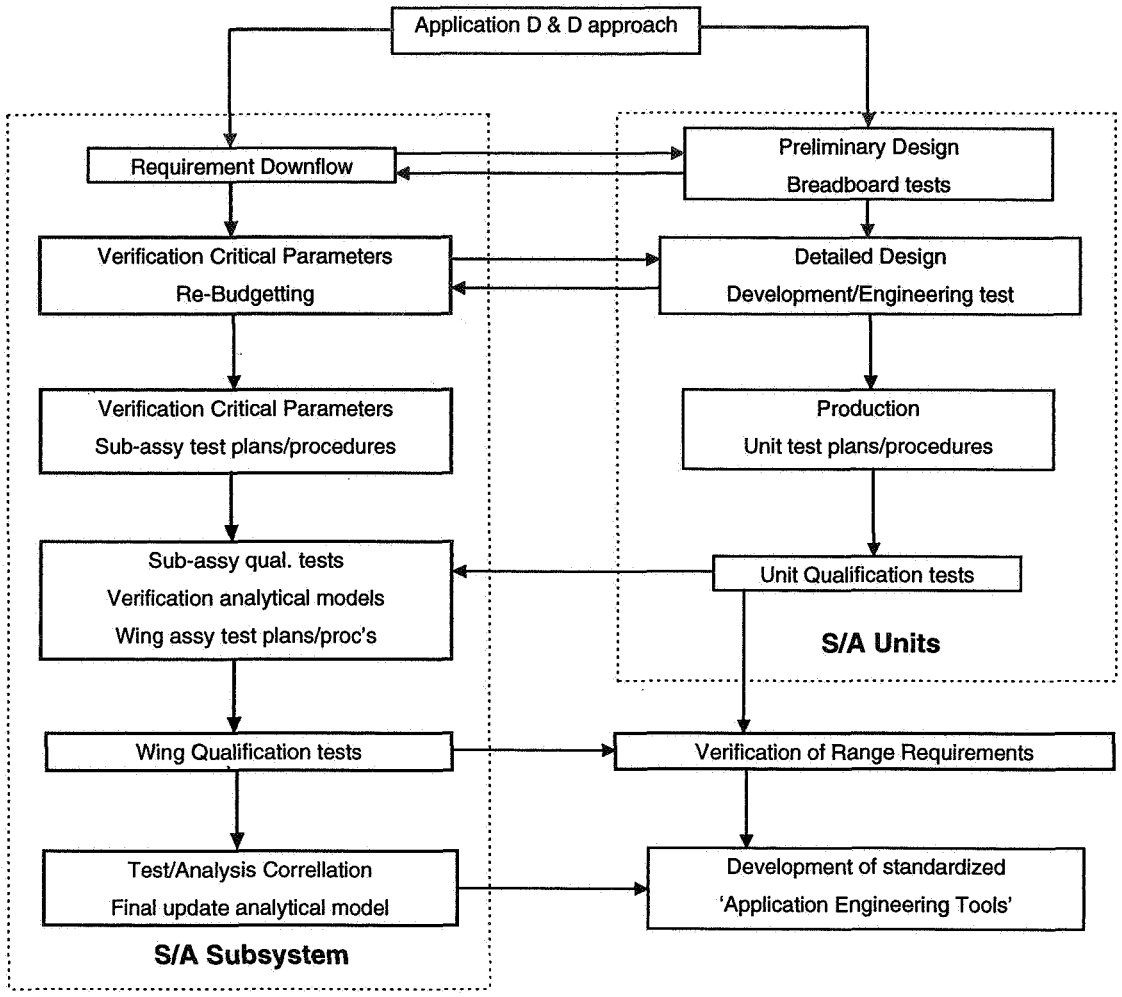

Figure 1 The ARA Mark3 requirement downflow process

Design Factors

Throughout the development program, we have tried to maximize the margins of the various units. This would allow us to use 'coarse' engineering and analysis tools, which would ensure that the non-recurring costs for a solar array application would be as low as possible.

However, the solar array market currently sees a trend towards increased system requirements in combination with reduced mass budgets. Therefore, frequent interaction between the unit and subsystem requirement definitions was mandatory in order to balance the criticality and margin of key parameters.

At wing level, the definition of applicable environments was rather straightforward: they should simply envelop the requirements of the desired range of spacecraft and launchers, including qualification margins and a safety factor against yield.

On unit level, however, this definition was more complex. Environments and boundary conditions had to be defined in such a way that they would ensure qualification of the unit for the full application range.

For the thermal environment definition, a qualification range that would envelop all desired applications was chosen. This was done by careful study of all thermal analyses of previous ARA Mark2 solar array designs and by increasing margins to allow for uncertainties.

For the mechanical loads, a set of factors was defined, as found from subsystem analyses. The definition of these factors is given below. For each individual unit, these factors were quantified by analyzing the criticality of the design, the sensitivity to parameter changes, and the uncertainties in the design.

Growth Factor *[GF]*:

- The design of the solar array shall allow for an approximate 50-mm shift of the hold down locations without any effect on the overall qualification status. This is based on our experience with spacecraft layouts changes at a late stage in the program. The consequences of such a shift are normally a change of the natural frequencies and mode shapes and a change of the load pattern in the panels.

- Furthermore, the design shall allow for power growth of an existing solar array geometry by changing from silicon to gallium arsenide solar cells. The latter have a considerably higher mass, thus causing higher stresses in the substrate and higher loads on the hold downs.

Parameter Inaccuracy Factor *[PIF]*:

- Although all design parameters are well known after completion of the development and qualification program, inaccuracies must be taken into account for variations in both stiffness and mass properties (panels, hold down stacks, hinges, yoke, etc.) that were yet unknown.

Analysis Inaccuracy Factor *[AIF]*:

- The analytical models that we used for load predictions, of course, do not have a complete 1:1 relationship with the hardware.

General Reserve Factor *[GRF]*:

- Boundary conditions (Ground Support Equipment, test setups, supports etc.) for unit tests will very often not be identical to those on wing level.

Thus, the overall safety factor for a unit (on top of the regular factor against yield) is:

Overall Unit Factor = *[GF]* x *[PIF]* x *[AIF]* x *[GRF]*

Key Design Features

The solar array design features that were chosen to meet the desired application range are described below.

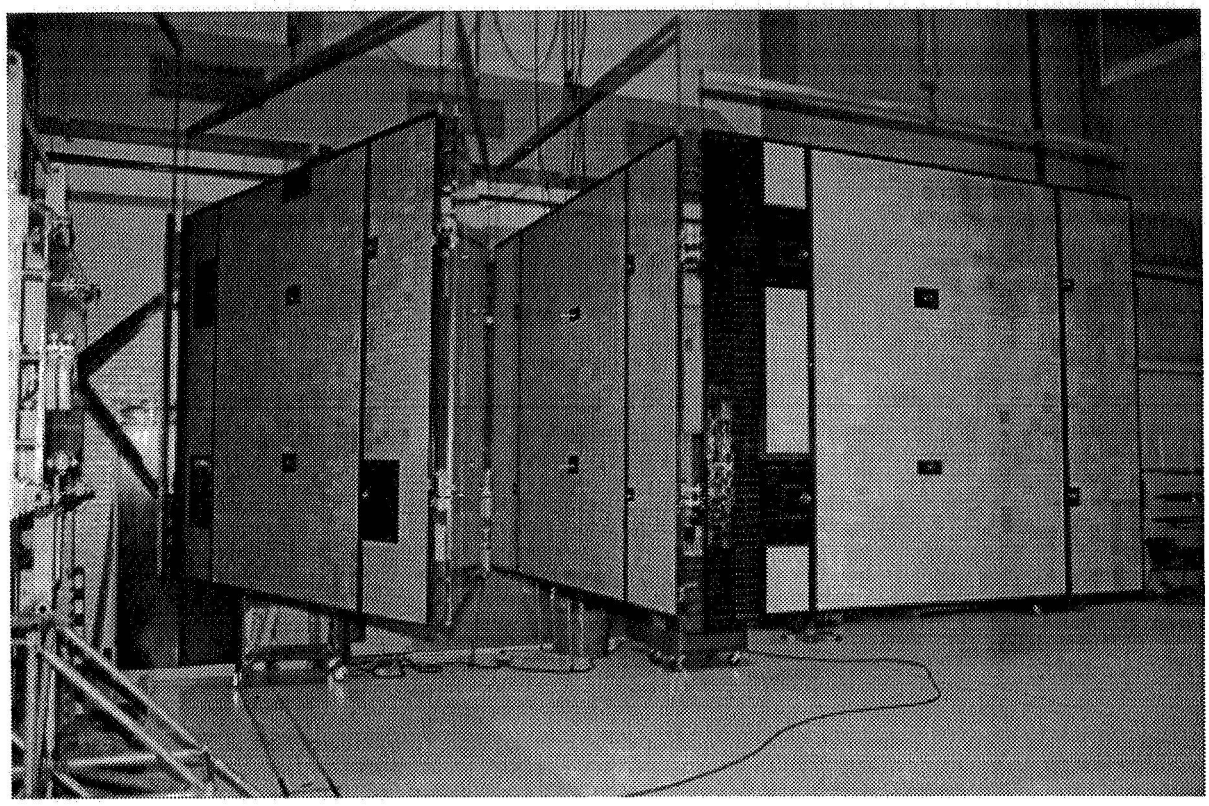

Figure 2 The ARA Mark3 solar array qualification model

- An array typically consists of two identical wings, mounted on the north and south side walls of the spacecraft. Figure 2 shows an overview of the ARA Mark3 Qualification Model (QM) wing during one of its deployment tests (4 panels of more than 6 m^2 each; total wing span of more than 13 m).

- Each wing is composed of a maximum of 5 sandwich panels with aluminum honeycomb core and CFRP face sheets, covered by solar cells (cell type is project specific).

- During launch, the wings are stowed in folded positions against the side walls of the satellite by means of 4 to 6 (inboard) hold downs per wing.

- The deployment system is equipped with an eddy current damper. Thus, the deployment shocks on the Solar Array Drive Mechanism (SADM) hinge and the interpanel hinges can be limited.

- In transfer orbit, two configurations are possible: either the satellite is spinning and the wings are kept folded, or it is 3-axis stabilized, such that the outer panel of each wing is deployed over 90° to a partially deployed position. The remaining stowed panels are kept folded to the side walls by means of 1 or 2 hold downs. After arrival in its final orbit, the wing is deployed to its fully deployed position.

- The solar array wings are released for partial and full deployment by cutting aramid restraint cables by means of thermal knives (identical principle to ARA Mk2).

- The adverse effects of shadowing are countered by:
  - the yoke length; maximum yoke length without extra hold down provisions is 1.80 m
  - shunt diodes

- Each wing can be equipped with:
  - thermal fins for thermal shielding of the spacecraft radiator panels and to prevent sun trapping between the solar array wings and the satellite side wall
  - plume shields to avoid thruster plume impingement on the wing
  - an AOCS flap (solar sail) to save fuel by assisting satellite attitude control
  - solar array-mounted sun sensors

Design Description

In the following paragraphs, the design of the various components is described in detail.

Solar Panel Substrates

The panel design is based on "standardized" components (skins, honeycomb, subassembly panels for hold down areas, edge members), such that it fits the stiffness and strength requirements, while keeping the mass to a minimum.

For each application, the panel loading, which depends on the spacecraft/launcher combination, must be checked to ensure that it is within the qualification spectrum. The nominal substrate covers the application range of solar arrays from 3 minimum sized panels (~ 1.2 x 1.7 m^2) up to 5 panels with maximum sizes (~ 2.25 x 2.75 m^2) per wing. Outside this range, it is of course still possible and justifiable to apply the ARA Mark3 design components. However, one must realize that the performance parameters (such as power-to-mass ratio) then start to deteriorate.

The skins of all panels of each solar array are kept identical and consist of M55 CFRP prepreg in closed lay-up. Reinforcement patches are located between each pair of opposite hinges to provide extra stiffness and skin strength. Since mass budgets are invariably tight, the skin layout is mass-optimized.

The panel core consists of low density aluminum honeycomb (height of 22 mm). Edge members ('C-sections') of CFRP weave are applied in the hingeline edges along the full length of the panel edge. Figure 3 shows a cross-section of a substrate at the panel edge.

Local reinforcement plugs ('subassembly panels') are used to give extra strength and stiffness to the hold down areas. These are circular and consist of relatively dense aluminum honeycomb and several layers M55 CFRP. They provide margin against yield and fatigue of the nominal honeycomb and reinforce the skin at the hold down locations.

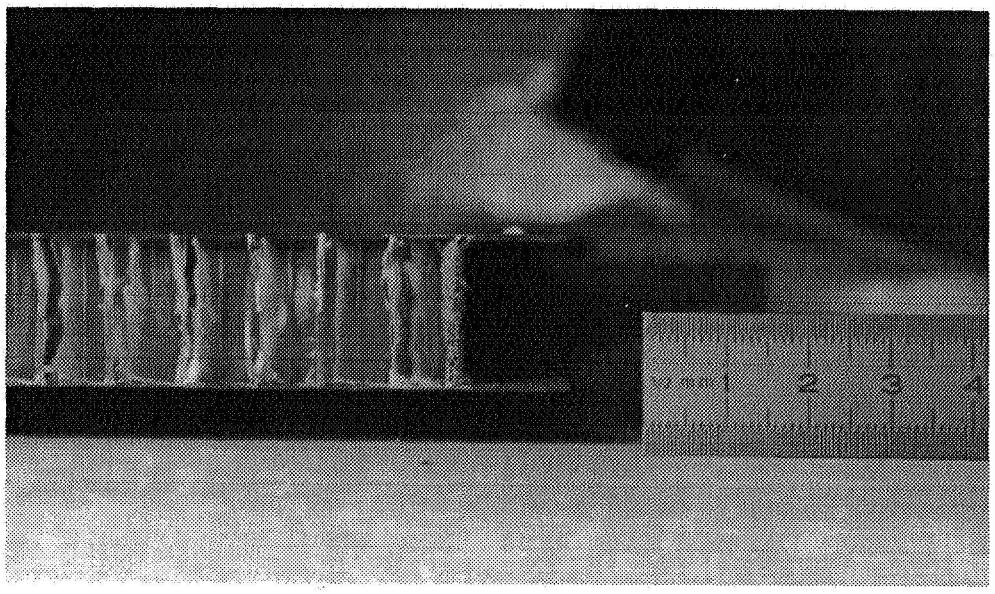

Figure 3 Detail of panel substrate with edge member

Photovoltaic Assembly

The solar cells and their cover glasses are typically project specific. The solar cell isolation is provided by a layer of Kapton on the front CFRP skin. Isolation with regard to the power subsystem, if required, can be obtained with flat blocking diodes mounted on the panel rear side.

Shadow protection in the fully deployed configuration is provided by shunt diodes (flat, mounted on the panel front and/or rear side).

The panel wiring and solar cell string design are positioned, such that the string ends and feedthrough holes are all located at the same panel edge for each panel.

Transfer Harness

The transfer harness is lightweight and modular. The panel wiring from the solar cell string ends, as well as the harness coming from the more outer panels, is routed to dedicated female 'flexprint connector' parts. These connector parts are attached to the

edge members along the panel hingeline edges. Therefore, the harness can be installed on the panel rear side completely by the cell supplier, thus resulting in a decrease of assembly cost and throughput time.

The hingelines are then crossed by flexible flat wiring, so called 'flexprints' (Figure 4), with male connector parts attached to them and are inserted into the female connector parts on the panel edges. The flexprints are made of two Kapton front and rear sheets, with 12 etched cupper tracks sandwiched between them. They are designed, such that they ensure low electrical resistance and minimum retarding torque under extreme thermal conditions.

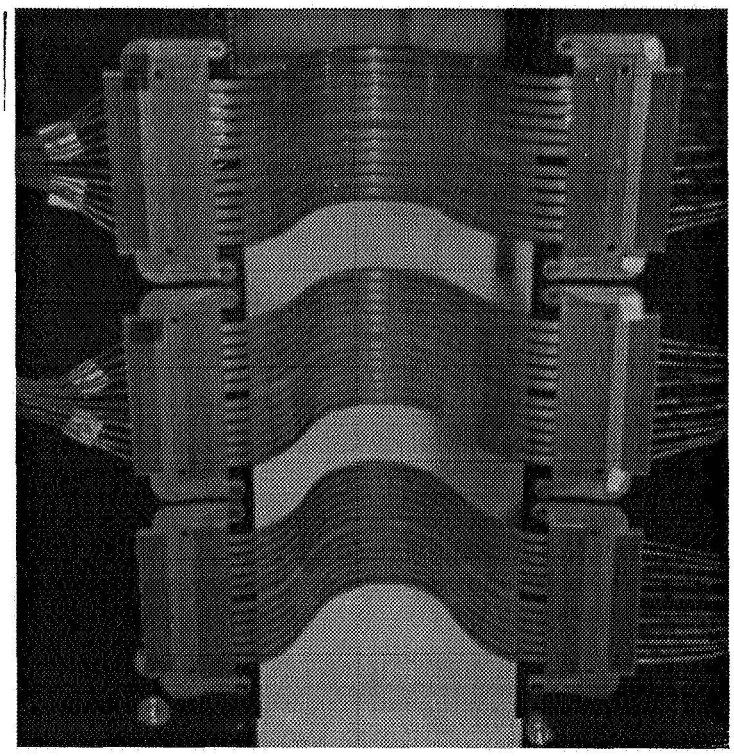

Figure 4 Flexprint/Connector assembly on panel edge

Deployment Mechanism

The deployment mechanism consists of the following items:

- Yoke
- SADM ('root') hinge
- Deployment damper
- Interpanel hinges
- Synchronization system
- Partial Deployment Mechanism (PDM)

Yoke
The V-shaped yoke ranges in length from 1.0 to 1.8 m. The length of the yoke pipes may vary, as well as the angle between them. As a consequence, the interfacing parts of the root hinge and the hinge attachment brackets will then also have a different angle.

The arms of the V-shaped yoke consist of CFRP filament wound tubes with a circular cross-section. The thermal expansion coefficient is kept as close to zero as possible in order to be able to meet the alignment requirements for the complete range.

Flexprints can be attached to the dedicated yoke flexprint panel (standard M55 CFRP skin with edge members for flexprint and hinge attachment and nominal honeycomb), running parallel to the panel hingeline edge.

SADM Hinge
The SADM hinge interfaces with the satellite mechanically, thermally, and electrically and consists of two hinged brackets providing attachment points for the SADM hinge torsion spring, the deployment damper, the yoke tubes, the SADM interface plate, and the I/F connector bracket. One hinge axis serves as the input shaft for the damper gearbox, while the other is used as feedthrough for the yoke harness.

Deployment Damper
A deployment damper is included in the ARA Mark3 design baseline in order to reduce deployment shocks, while maintaining ample margins with respect to retarding torques. The damper is of the eddy current type, in combination with a gearbox mechanism, manufactured by Honeywell. Its performance can serve the full ARA Mark3 range.

The damper/gearbox combination is mounted on one of the two axes of the SADM hinge. In order to enable this configuration, the gap between the inner panel and the satellite side wall was increased with regard to the ARA Mk2 design.

The use of a damper allows for relaxation of the strength requirements from deployment shock of the SADM and the panel hinges after secondary, full deployment. The load case of the deployment shock at the end of the primary, partial deployment of the outer panel still requires a higher allowable bending moment of the hinges because this deployment is not damped.

The damper can cover the range of applications up to 5 maximum size panels per wing, enveloping the complete range of thermal environments. For each application, the damping coefficient must be determined, while considering the applicable thermal environments and the deployment safety factors.

Interpanel Hinge
Each interpanel hingeline consists of with two co-axial hinges (Figure 5). These are equipped with dedicated 'clock' springs to counter the retarding torque of the Power Transfer Harness, the friction of the synchronization system, and the internal friction of the hinge bearing itself.

The required torque is strongly dependent on the number and size of panels and will vary per hingeline. Therefore, the actuating torque that has to be produced by the hinges will also deviate over a large range. From both a mass and a strength point of view, one type of spring to cover the complete torque range is not acceptable, so the hinge design accommodates several different spring types.

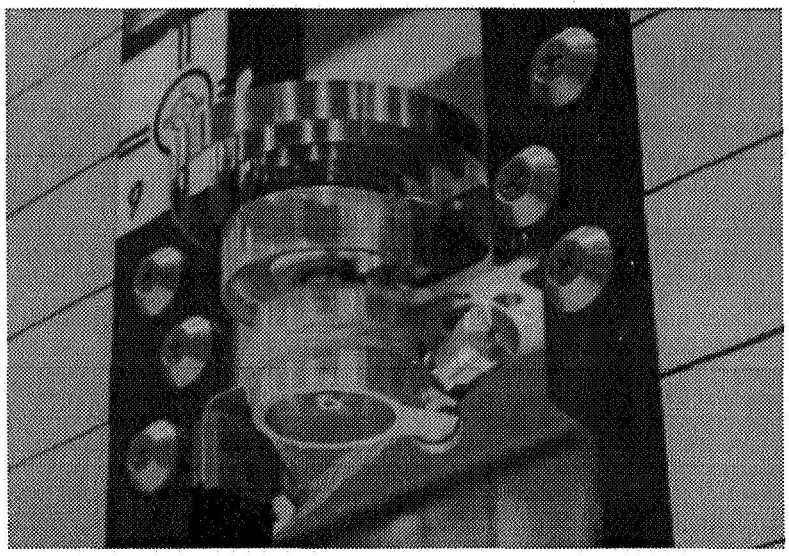

Figure 5 Interpanel hinge

The interface with the panel substrate for each hinge bracket is provided by three titanium pins. The I/F holes are not in line in order to reduce hysteresis effects under bending loads. The brackets are mounted internally in the edge member, and shear webs in the brackets provide the necessary stiffness.

Synchronization Mechanism

The synchronization of the deploying panels is ensured by braided aramid cables, running from hingeline to hingeline. To accommodate this, each hinge is equipped with a dedicated pulley.

Because of the deployment damper, the stiffness of these synchro cables is very high. Without this, the combination of high hinge actuation torques (driven by the high deployment safety factor) and a high damping coefficient (for cold temperatures) would cause an asynchronous deployment, thus resulting in unpredictable high deployment latch-up shocks at the interpanel hinges.

A slack compensator spring is attached internally in the braided cables in order to maintain a minimum cable tension and to prevent the cables from running off their pulleys under dynamic loading during launch.

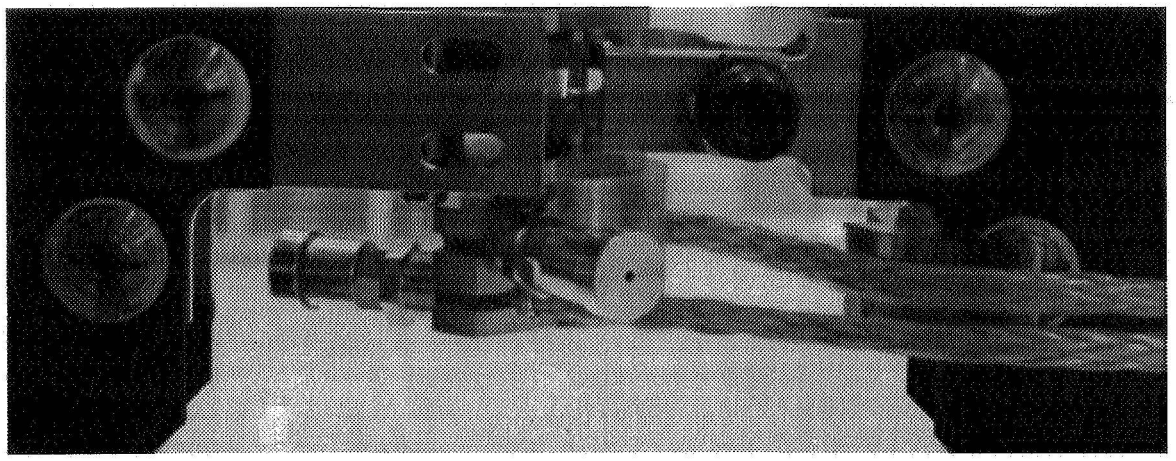

Figure 6 Synchro cable/pulley system

Partial Deployment Mechanism

The bracketry for the PDM is shown in Figure 7. The various brackets are in 'closed form' configuration, as long as the inner panels are kept stowed, thus providing the required stiffness and load path. The rotation axes are provided by the interpanel hinges between panels 3 and 4. A bracket with a titanium wheel is attached to the outer panel. In a partially deployed position, the wheel is pressed against the flange of the bracket on panel 2.

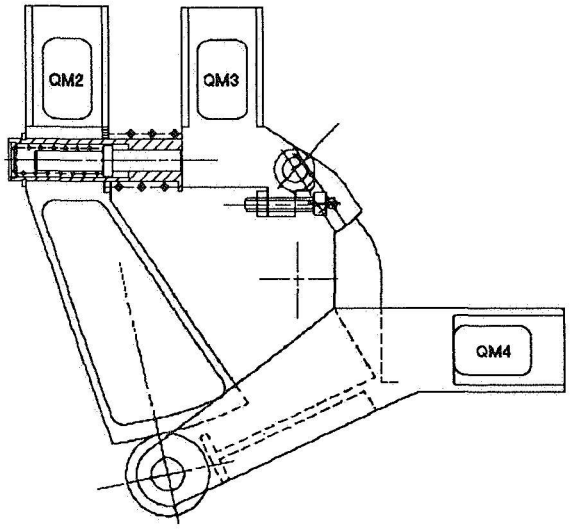

Figure 7 Partial Deployment Mechanism (PDM)

Positive locking of the PDM assembly is provided by the latch and cam combination on the brackets of panels 3 and 4. As soon as the secondary, full deployment starts, the wheel and flange, as well as the latch and cam, will start to move away from each other. All attachments to the panels are identical to the panel/hinge interface.

Hold Down and Release System

Functionally, the same basic concept as for ARA Mk2, using a flexible tie-down element (aramid cable) under high preload, is applied. The stack height can vary between 2 panels (in case of a partially deployed 3-panel wing) and 5 panels (completely stowed 5-panel wing). The hold down points are located inboard on the panels, thus allowing optimization of the mechanical interface location with the satellite side wall. Figure 8 shows a typical hold down stack.

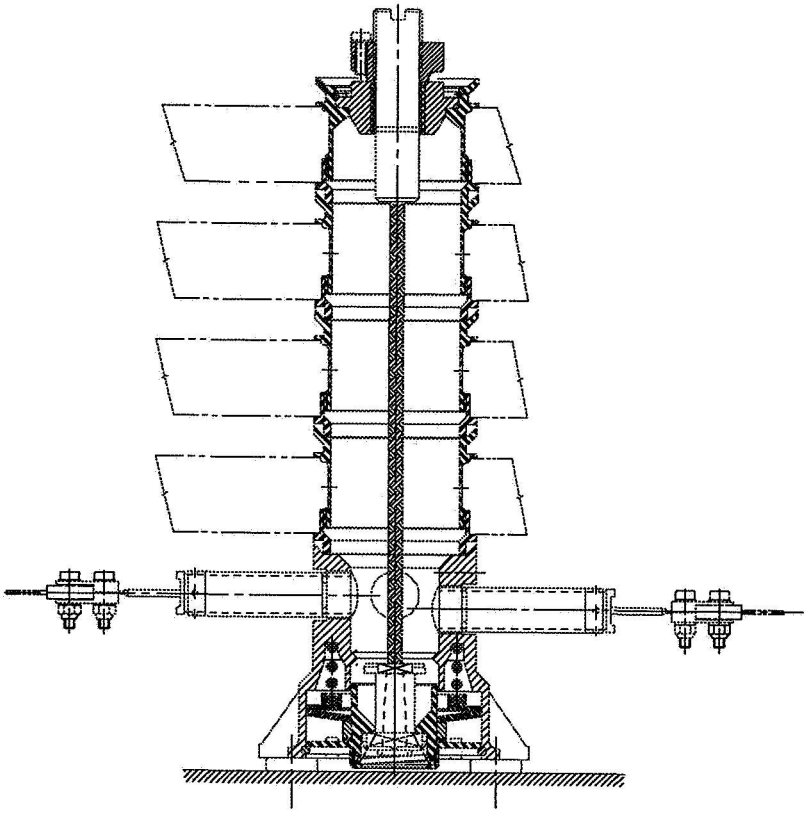

Figure 8 Typical hold down and release stack

The first row of honeycomb cells around the cylinder are filled with a potting compound in order to create a good shear connection between the honeycomb and the cylinder. The cups and cones are also bonded to the skin.

Cup/Cone

The hold down points in the panel consist of a cup and a cone part. Both parts consist of a threaded titanium (good compatibility with the thermal expansion of the aramid restraint cable) cylinder. The conical shape is chosen to ensure that they do not start to cant or to gap relative to each other under the severe launch loads. In order to prevent fretting and/or cold welding of the cups and cones under those same loads, the contact surfaces on the cone side are covered by a surface coating.

Thermal Knife
The aramid restraint cable is cut by means of a thermal knife. The knife consists of a ceramic plate that is heated to approximately 1000°C, while being pressed against the cable. At that temperature, the cable is gradually degraded at the location of the plate/cable interface until it is completely cut (Figure 9).

The thermal knife has proven to be very reliable in numerous applications. Furthermore, it avoids shocks at the moment of wing release.

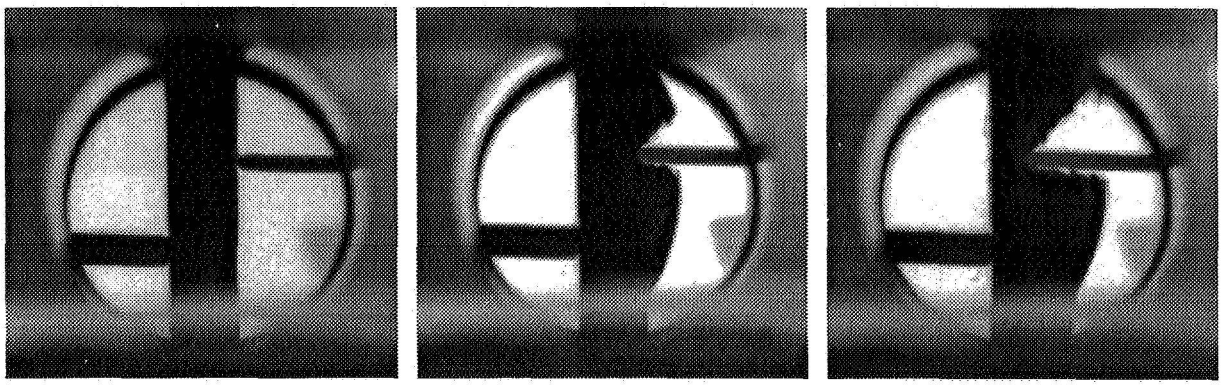

Figure 9 Thermal knife, cutting through restraint cable

Restraint Cable
As already mentioned, the cable is made of aramid. It is braided and equipped with end fittings. The bottom end fitting of the restraint cable is bayonet-shaped to ensure simple installation and traction of the cable in the stack in the stowed wing configuration. The upper end fitting is threaded to enable tensioning of the restraint cable. Both end fittings are attached to the cable by means of a conical wedge, or 'spike'.

Hold Down Bracket
The hold down bracket interfaces with the satellite side wall. In the hold down bracket, a bayonet bus is located, constraining a package of belville springs. The bayonet bus contains the lower end fitting of the restraint cables. During the cutting process, the tension in the restraint cable decreases. The belville washers ensure a minimum pretension level in the hold down cable.

The Wing Qualification Test Program

The qualification tests for ARA Mark3 are performed both on wing level and on unit/part level. Since the wing and unit/part level qual tests have been scheduled in parallel in order to enable delivery for the first flight program, both test flows still have tests in progress or not yet performed.

Wing Qualification Tests

Wing level dynamic tests (acoustic and sine vibration) have been performed successfully at the test facility of IABG, Germany. The output of these tests was helpful in ensuring that the mechanical load levels for the unit/part tests indeed envelop the wing levels. Table 10 shows the total wing qualification test program.

Table 10 Overall QM wing qualification test program

Configuration / Item	Feature / Requirement to be verified	Verification method
Solar Array	Mass	Mass analysis
QM wing	Mass	Weighing of hardware
Stowed QM wing	Stiffness, Strength, fatigue	NASTRAN analysis
		Vibration test
		Acoustic test
Partially deployed array	Deployment shock strength	Deployment analysis
		Shock test
	Stiffness	NASTRAN analysis
		Stiffness test
	Alignment	Alignment test
Fully deployed array	Deployment shock strength	Deployment analysis
		Shock test
	Stiffness	NASTRAN analysis
		Stiffness test
	Alignment	Alignment test
QM wing	Deployment torque margin	Deployment analysis
		Energy surplus test
QM wing	Functional partial deployment	Deployments, including thermal knife firing
QM wing	Functional full deployment	Deployments, including thermal knife firing
QM wing	Electrical performance	Electrical health checks
		Flasher tests

Key Performance Parameters

Based on the 'Hot Bird 2' solar array configuration (2 x 4 panels of 2.736 x 2.250 m; 6 hold downs per wing; yoke length of 1.8 m; 10-Wcm Si cells), the following key performance parameters have been established:

Mechanical Subsystem

- 133-kg total mass
- 52-Hz first stowed global natural frequency
- 0.6-Hz partially deployed frequency
- 0.05-Hz deployed frequency
- minimum out-of-plane input at natural frequencies $\geq$ 2.0 g
- acoustic input 148.7 dB overall sound pressure level

Electrical Subsystem
- 44-V I/F voltage
- at autumn equinox, EOL power of 5.5 kW

Critical Design Items And Lessons Learned

Deployment Hinge Development Issues
Although several parts for the hinges had already been manufactured, assembled, and tested before the Critical Design Review of the development program, there was one design feature that was not tested on the engineering level simply because it was not considered critical: the stiffness and backlash properties of the deployed hinge.

However, after technical discussions with ESA experts, we had decided to increase the angular play in the hinges to allow for a larger build misalignment budget in the hinge assemblies. This would then result in very low friction during the deployment which would lead to a favorable reduction of the total required deployment energy budget.

At that time, it was not realized that this also introduced a considerable amount of backlash in the hinges in deployed configuration. Thus, the deployed stiffness was almost nil for low disturbance loads (Figure 11). For large solar arrays, this feature is unacceptable, and it was only found during hardware acceptance testing, just before installation in the qualification wing.

The solution that we found was as elegant as it was simple: The male hinge bracket rotates around a bearing. This bearing was originally cylindrical, but the design was updated to a conical one. Now it allowed for relatively large angular rotations, while keeping lateral play to the minimum. Figure 11 shows both stiffness curves.

Hinge Lessons Learned
- A change of one design parameter does not necessarily affect only one aspect of the performance. Track down all the possible effects, both positive and negative, of the design change.
- ARA Mark 3 has succeeded in developing a low friction hinge with positive locking and no backlash in deployed position.

Partial Deployment Mechanism Development Issues
At the start of the program, in-house development and qualification tests on a PDM design for another project had just finished. It was decided, given the success of these tests, to keep the design 'as is' for ARA Mark3 and concentrate on more urgent matters.

At the end of the QM wing program, at the start of the deployment shock qualification tests, a flaw in the design was found. Under low loads, the mechanisms behaved perfectly. However, under increasing load levels (e.g., the undamped shocks at latch-up after deployment), the stiffness dropped sharply, while the adjacent brackets showed large relative displacements.

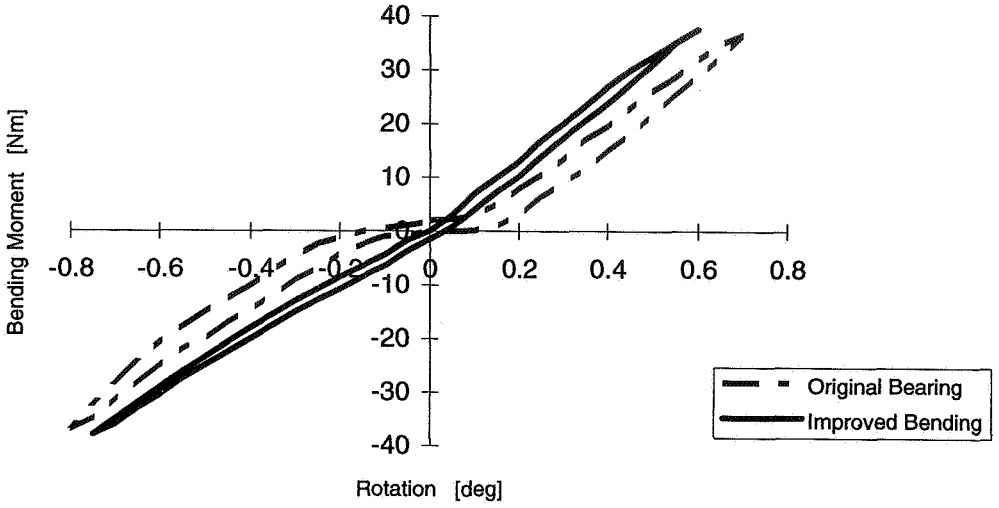

Figure 11 Deployed stiffness of the 'old' & 'new' bearing design

Detailed investigation of the stresses and displacements, as found by test and by analysis in the partially deployed wing, revealed the following causes. At the end of partial deployment, outboard panel QM4 tends to rotate further due to its inertia. Furthermore, the out-of-plane flexibility of panel QM3 (outermost stowed panel) allows a displacement in the outboard direction of the hinges that connect panel QM3 and QM4 at the same time. This resulted in the combination of large relative displacements and low stiffness.

The re-design had to be, as usual, a compromise between stiffness (required to meet the needs of the Attitude & Orbit Control System of the Hot Bird 2 spacecraft) and strength (as a consequence of increasing the stiffness, the shock loads after deployment would increase).

The new brackets (Figure 7 shows the unloaded situation, and Figure 12 shows the deformed configuration) have been installed, and the tests for alignment, stiffness, and deployment shock are currently under way, with the bracketry behaving flawlessly. After this, the test program is to be resumed in order to conclude the ARA Mark3 qualification program at wing level.

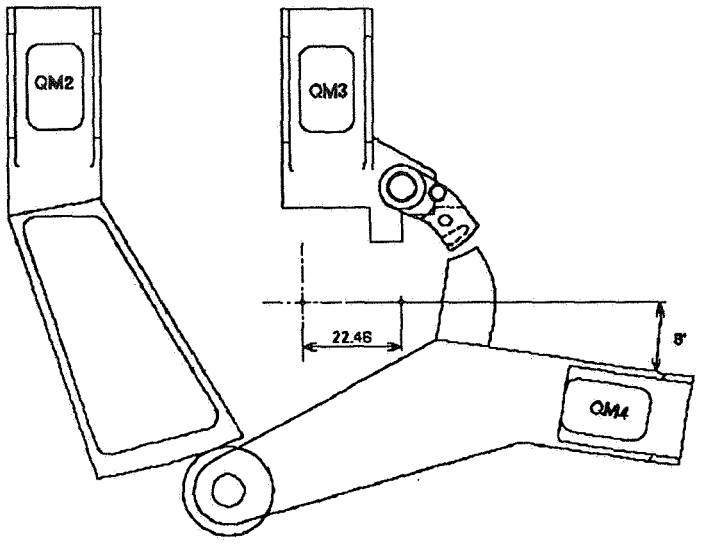

Figure 12 Deformed Partial Deployment Mechanism (e.g., shock load)

PDM Lessons Learned

- A unit/part that functions correctly in a system is designed using a certain set of boundary conditions. Check if these conditions also apply to your system. Try to find killer requirements/conditions before they kill you!
- ARA Mark 3 has succeeded in developing a Partial Deployment Mechanism that is insensitive to panel size and to location of hold down points.

Conclusions

- The ARA Mark3 solar array combines high performance with low mass, low cost, and short schedule time.
- The development philosophy successfully concentrated the efforts on two fields:
  - requirement derivation from system level down to unit/part level to cover the complete application range.
  - extensive breadboard/engineering tests to correct design flaws before the actual qualification tests (the PDM being the exception that proves the rule).

Acknowledgment

The author would like to thank the members of the ARAFOM solar array project team for their comments and their never-wavering enthusiasm and devotion throughout the project.