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Title High Efficiency Solar Arrays – Executive Summary Report

Summary To be able to provide future satellite missions using electrical propulsion with sufficient power, innovations in the field of solar arrays are needed. This study aims to find the best concept able to provide a Spacecraft with 150 kW to 300 kW of power. The most important driving requirements are a power specific density of at least 60 kW/m^3 and a power specific mass of at least 200 W/kg.
A number of different concepts were generated with different architectures and deployment techniques. A trade-off was performed to find the best concept. Not only the specific volume and mass driving requirements were considered, but also Testability, Scalability, Modularity, Retractability and Cost were taken into account. With good performance scores across the board and excellent scores for Testability, Modularity and Cost, the I-beam concept was the selected winner.
Finally, a detailed mechanical evaluation of the I-beam concept is presented, including a thorough examination of foldable membranes, harness design and their implication at S/C level.

Keywords: Flexible Blanket; ESA HESA study; High Efficiency Solar Arrays

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1	Dec 2023	9	R. Torenbeek	all	First issue

Acronyms and Abbreviations

ADSO	Airbus Defence & Space Ottobrunn
ADST	Airbus Defence & Space Toulouse
AN	Airbus Netherlands B.V, Airbus NL
AOCS	Attitude, Orbit and Control System
AIT	Assembly Integration & Testing
CFRP	Carbon Fiber Re-enforced Plastic
CTM	Collapsible Tubular Mast
DLR	Deutsches Zentrum für Luft- und Raumfahrt
ESA	European Space Agency
FE	Finite Element
GEO	Geostationary Earth Orbit
GSE	Ground Support Equipment
HESA	High Efficiency Solar Array
HDRM	Hold Down Release Mechanism
MGSE	Mechanical Ground Support Equipment
PDM	Primary Deployment Mechanism
PSV	Power Specific Volume
PV	Photo Voltaic
PVA	Photo Voltaic Assembly
S/A	Solar Array
S/C	Spacecraft
SCA	Solar Cell Assembly
TTM	Tip Tensioning Mechanism

Executive Summary

Introduction

The Global Exploration Roadmap, published by the International Space Exploration Coordination Group which comprises of ESA and other agencies, has identifies mid and high class solar arrays as a key technology. Current advancements for mainly GEO applications take a step in the right direction, however, these are very much focused on just one environment and on powers levels lower than those required for future planned lunar Martian missions, requiring electrical propulsion. In addition to such exploration missions, Space Based Solar Power applications have also recieved attention recently (ref. ESA Solaris concept). These missions have even greater power needs and should not be overlooked, as these may make future use of the technology developed for exploration missions.

In order to obtain insight into the concepts that might enable such new applications, with their specific sets of requirements and limitations and to focus on areas for future development, ESA awarded a study contract named “High Efficiency Solar Array for high power solar electric propulsion missions” to a team comprising Deutsches Zentrum für Luft- und Raumfahrt (DLR), Airbus Defence & Space Toulouse, Airbus Defence & Space Ottobrunn and Airbus Netherlands to generate various concepts, suitable for solar arrays in the 150 to 300kW power range. A second follow-on objective of the study has been to identify the main performance, limitations and points of attention for further studies and development.

Problem Definition and Requirements

To make practical use of electrical propulsion, one would not only need a larger solar array area, but this array should also be able to sustain the loads resulting from re-boosts, aero-braking and docking. The High Efficiency Solar Array (HESA) study aims to generate and develop viable concepts that would achieve these goals. In this study, the focus will be on solar arrays producing at least 150kW, with a possibility of reaching 300kW.

These larger solar arrays will face many different challenges comparison to existing arrays, not only in terms of deployed load cases, but also in stowage volume and mass. It is not feasible to launch arrays with the intended power capability with the current power to mass and power to volume ratios. New ways of stowing and producing solar arrays have been investigated to make this possible. Within the study, a set of requirements were provided and grouped in terms of performance goals, constrains and ‘nice-to-haves’. The most important driving requirements are a specific power/volume of at least 60kW/m^3 and a specific power/mass of at least 200W/kg.

The main performances to be reached are:

- Area: 640 – 1280m^2 per spacecraft
  (State of the art Flexible Compact Array: 100m^2, assuming 240W/m^2)
- Power/Mass Ratio: > 200W/kg
  (State of the art Flexible Compact Array: 120W/kg)
- Power/Volume Ratio: > 60kW/m^3
  (State of the art Flexible Compact Array: 32kW/m^3)

The main constraints to fulfill are:

- Re-boost load: 0.21m/s^2
- Docking shock: 0.2g
- Testable on-ground

The ‘nice-to-have’ functions are:

- Aero-breaking: 0.3-0.5N/m^2
- In-flight (partial) retractability

Literature Survey Flexible blanket and S/A Characteristics

A set of 17 flexible solar arrays, either previously developed or currently in development, have been evaluated. Peculiarities and performance parameters have been identified, in order to illustrate the strong and weak aspects of a given concept. Only flexible blanket type solar arrays (and thus no rigid panel concepts) have been retained, due to the challenging requirements of mass and stowed compactness. From the review and evaluation, one could derive the following specific topics, which are needed to ensure power growth to this high value of 300kW:

- The blanket shall be very flexible, shall sustain loads during vibration and shall be capable of fold-out or roll out. Blanket harness shall be assumed to be embedded in the blanket. Section voltage should be as high as possible: 150Vdc as a starting point, with growth to 300Vdc.
- A modular concept is needed in case of linear deployments. This is to ensure cost reduction as only one module will be qualified to extreme loads. Tests are always in horizontal orientation, due to the complexity of ground support equipment (GSE) and to prevent unwanted blanket loadings when hanging vertically.
- Serviceability requires full retraction and re-stowage. For autonomous fold-out and retraction, a fold by fold retraction is envisaged. Full wing retraction also provides robustness against re-boost and docking loads. These loads may also be considered in partially retracted configuration, if needed.
- The concepts should allow scalability, namely that higher power outputs can be attained by varying dimensions. This would lead to 2-dimensional deployment concepts, as in the case of the Flex Array and Solar Sail (DLR). However, this might necessitate full requalification, since such concepts do not lend themselves to modularity. Building blocks have to been examined to investigate their perimeters, which in turn show growth potential limitations, based on the selected strawman mission concept (ESA).
- Testing shall be performed at unit, equipment and at subsystem level. This will necessitate a full size deployment rig for array modules.

Concept generation and Performance

Four design teams have evaluated the HESA problem definition and came up with a series of concepts, based on:

- 1D and/or multi 1D deployment via booms or guy wires
- Adaptation of 2D Z- or Origami-folded solar sails, deployed via booms
- Roll-out solar arrays deployed via booms

Each team has evaluated the performance of the concepts via a common set of trade parameters [weighting factor], incl.; Specific stowed volume [kW/m^3] [5], Specific mass [kW/kg] [4], Testability and AIT effort [5], Scalability ease [3], Modularity / common designs [2], Retractable in orbit [2], and Cost / Complexity [PM]. The first two criteria are evaluated quantitatively, whereas the remaining are only assessed qualitatively. The I-Beam roll-off array solution appears to be the overall best performing concept.

In the I-Beam roll-out design, the central structure (mast) consists of I-beams that in stowed configuration house the rolled up solar blankets. The solar blankets are attached to the I-beam and deployed using Collapsible Tubular Masts (CTMs) that are rolled onto the drum alongside the solar array. The nominal configuration has four I-beam segments with two blankets of 4x20m each, so 640m^2 in total, producing 150kW. Each blanket is tensioned via a tip tensioning mechanism (TTM). The solar array is stowed by stacking the I-beams one-by-one as they rotate simultaneously in an alternative plane. After deployment, the wing comprises a "staggered" central I-beam structure. See Figure 1 for dimensions in stowed and deployed configuration, and Figure 2 for PDM details.

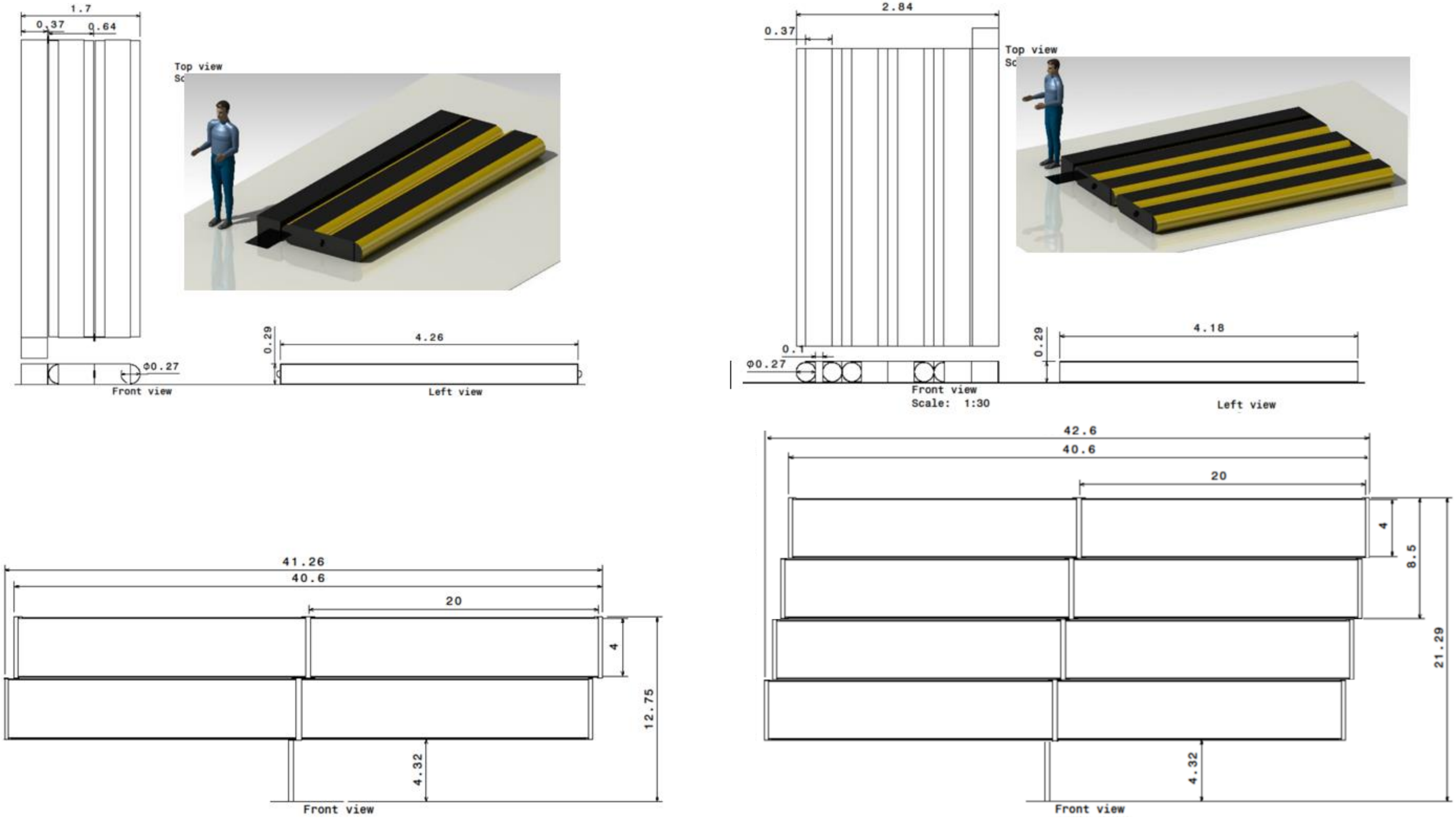

Figure 1: I-Beam roll-out solution dimensions 75kW (left) and 150kW (right) Solar Array wing, in stowed and deployed configuration

After concept selection, the I-beam solution was further detailed by performing re-sizing activities for the CTM boom, I-Beam support structure, drum and blanket tension, based on mechanical analysis, such as running loads and I/F load, quasi-static (10g) and stowed and deployed modal analysis.

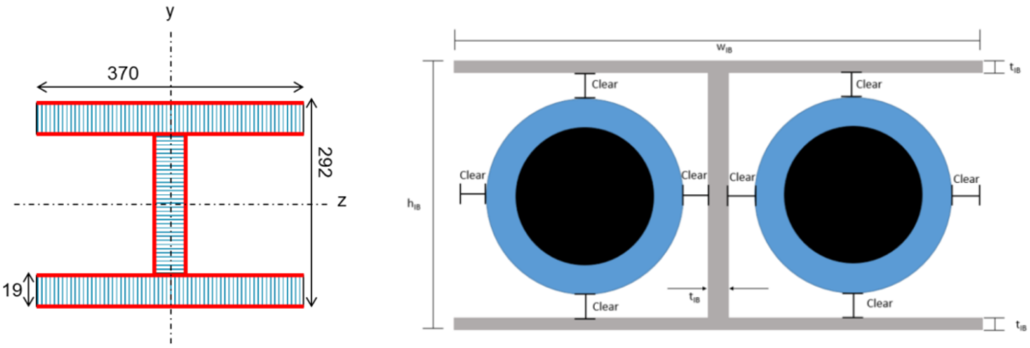

Figure 2: I-beam (PDM) with rolled-up blanket on drum positioned on each side of center plate

The updated design revealed a volume specific power of 86kW/m^3, therefore the requirement is well achieved. The mass specific power requirement of 200W/kg is not met, but varies dependent on the installed power on the S/C. For instance, for the 150kW spacecraft, the power specific mass is 138W/kg, while for 300kW spacecraft, this figure becomes 129W/kg. The used solar blanket with 1.1kg/m^2, consist of 0.8kg/m^2 PVA and wiring (non-redundant) and a 0.3kg/m^2 backing substrate (incl. cushioning). This is in line with currently available solar blankets.
Note that, to achieve the 200W/kg requirement, the actual number needs to be lower than that. It is estimated that to achieve 200W/kg, with the current I-beam roll-out solution, the blanket density should be reduced to 0.85kg/m^2. This can be achieved by introducing flexible cell, higher operating voltage (potential on-blanket power conversion) and improved backing substrate technology.
Impact of blanket wire redundancy should not be overlooked, as current feasibilities studies tent to show. Taking the current I-beam roll-out solution and adding harness redundancy to the design, the blanket density becomes 1.3kg/m^2 for the 150kW spacecraft, and its power specific mass would then become 122W/kg, while for 300kW spacecraft, this figure becomes 115W/kg. In order to obtain a realistic reliability for the flexible solar array, the blanket density of 1.3kg/m^2 is taken into account in the mechanical I-Beam roll-out evaluation.

The stowed I-Beam roll-out concept FE-model predicts the lowest in-plane mode at a frequency of 33.7Hz and the lowest out-of-plane mode at 38.2Hz.
The deployed out-of-plane frequency of the I-Beam solar array solution, is fully dictated by the flexible blanket, namely 0.0172Hz. This is because, at very low frequencies, the vibration of the mast is decoupled from the blankets, due to the greater stiffness of the I-Beams.

In general, it is noted that the docking load requirement, definition as a quasi-static 0.2 g acceleration, is not well defined. Consequently, it is an unrealistic design driving. A docking load can best be represented by a step function, acting within a short time interval. As the I-Beam solar array has a very low first natural frequency, there is a decoupling of events; the docking event and the resulting solar array response. In a follow-up HESA study, this load case should be defined in a more realistic manner, with the knowledge of the dynamic behaviour of the I-Beam concept.

S/C consideration

The integration of a High Efficiency Solar Array, providing power from 150kW to 300kW per wing, represents a major design driver for the satellite or in-orbit station, to which such a large system will be attached.

The different operation phases of such huge composite structures also represent significant challenges for the satellite design in terms of:

- Accommodation and storage phase within the fairing: the high specific power/volume ratio allows an easy accommodation of the stowed structure. The stowed I-Beam rolled-out solution nearly fits on a classical geostationary telecommunication satellite of 6.0m (height) x 2.8m (lateral) x 2.8m (lateral).
- Deployment phase: The deployment sequence of such a high inertia appendage will require a non-standard attitude control strategy. This topic represents a major challenge for the overall AOCS deployment.
- Electrical Orbit Raising & AOCS control phases: The Electrical Orbit Raising phase consist of using the full electrical power generated by the solar array to produce a continuous propulsion force (Xenon, Krypton or Iodine based propulsion) in order to reach geostationary orbit, or to accelerate the spacecraft during a transfer orbit from Earth to Moon, from Moon/Earth to Mars or to other planets. The I-Beam rolled-out concept shall withstand the accelerations provided by the propulsion system and by the manoeuvre loads induced by the AOCS. The AOCS strategy will need to adapt the control loop system to the high inertia and low frequency of the composite structure of 0.017Hz (out-of-plane) which represents a major challenge.
- In orbit assembly phase: The I-Beam rolled-out concept can be used to provide electrical energy to a single spacecraft but also to a more large and complex structure assembled in orbit. In this case, the I-Beam rolled-out concept should be launched into space and attached in orbit to larger composite structure (ISS for example, space station around Moon or Mars, …). In such a case, design shall accommodate all the constraints resulting from the in orbit assembly process (full robotic assembly or human extra vehicular assembly)
- In orbit maintenance phase: The large photovoltaic surface exposed to space environment will have to endure micrometeoroid and debris impacts. This could affect the in-life performance of the solar array. Even if this natural degradation is taken into account in term of safety and reliability, in orbit maintenance will be a major challenge, especially for such a large and complex structure with a long life-time.
- Thermal aspects: The onboard available power will be a major design driver for the satellite thermal control system. The electrical propulsion system generates a great quantity of energy that needs to be thermally radiated to space (Joule Effect with PPE electronics). This energy dissipation will have a major consequence: the need of large radiating surfaces. Additional deployable radiators will probably be needed to supplement classical radiating surfaces. This represents a thermal design challenge for the satellite.

Observation and Conclusion

- Specific Mass (W/kg) of roll-out looks favourable, especially in the case of double roll-out, development of thin flexible PVA technology is driving stowed volume.
- Modular sub wing design of roll-out allows simple acceptance testing.

- Environmental tests (i.e. vibration, TV etc.) on individual sub wings (and …partially on full wing).
- PDM / central mast must always be tested separately (with dummy stacks; TBC; Ref. Envisat S/A).
- Many separate deployment steps are not favorable, as MGSE may become very complex / $ infrastructure.
- Simple (sub) wing retraction allows fast testing of modular wings, as well as being beneficial for ground handling and/or on-orbit loads.
- Partially retraction is only feasible for roll-out solar array (single / double roll-out), not for fold-out blanket (TBC).
- The integration of a High Efficiency Solar Array, providing from 150kW to 300kW electrical power per wing, represents a major design driver for the satellite or for the in-orbit station onto which such a large system will be attached.
- As observed during the mechanical evaluation, it is concluded that the deployed frequency of these large structures is very low. I-Beam concept has a lowest frequency of 0.017Hz in Out-of-Plane direction. Impact on the AOCS system has to be further evaluated.
- Aspect ratio has large impact on harness mass, next to the redundancy and/or in-orbit reliability. The harness mass determines to a large extent the overall specific power/mass performance ratio.