



Development and testing of the DEAR 100 W flexible solar array blanket[☆]

Sven Holz^{a,*}, Patric Seefeldt^a, Kaname Sasaki^a, Tom Sprowitz^a, Yahya Motemani^b

^a *Mechanics and Thermal Systems, DLR Institute of Space Systems, Robert-Hooke-Straße 7, Bremen 28359, Germany*

^b *Saint-Gobain Research Germany, Glasstraße 1, Herzogenrath 52134, Germany*

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Abstract

The growing trend toward smaller, more compact satellites increases the demand for lightweight, deployable solar arrays. In response, the DEAR project - funded under ESA contract No. 4000133890/21/NL/MM/ra as part of the ARTES 4.0 programme - developed and tested a deployable 100 W solar array designed to fit within a single CubeSat segment. Beyond strict volume constraints, the most critical requirement was ensuring a minimum operational lifetime of 5 years, with an end-of-life (EOL) power output of 100 W.

To guide material selection and mechanical as well as electrical sizing, this paper defines multiple reference mission scenarios and analyzes key space environment factors - such as radiation, atomic oxygen erosion, and thermal loads. Based on this analysis, the design ensures 100 W power output at EOL, even under worst-case conditions.

The array was manufactured using a semi-automated welding process for electrical interconnections, with a coated FEP supporting film for atomic oxygen resistance and polyimide reinforcements for mechanical robustness. Triple-junction GaAs solar cells were used as photovoltaic devices.

As part of the DEAR system, the blanket underwent a qualification test campaign, including vibration and thermal-vacuum tests. This paper presents performance evaluations through multiple deployments, power output measurements (including IV-curve analysis), and electroluminescence inspections.

This work establishes a validated design and manufacturing process for a flexible solar array blanket, qualified through extensive testing for diverse space mission scenarios.

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1. Introduction

Solar arrays remain the primary power source for Small-Sats. Typical flown systems cluster around a specific power output of $\sim 30 \text{ W kg}^{-1}$ (Weston et al. (2025)). Volumetric

power density is equally important for nanosats, as the strictly limited bus volume of small satellite platforms constrains how much array can be stowed, regardless of its mass. Improving specific power requires either higher generated power or reduced system mass. High-efficiency cells address the former, while lightweight, deployable structures tackle the latter. Flexible-material panels, deployed via roll-out booms or similar mechanisms, offer extremely compact stowage and can achieve higher output than conventional rigid panels. Several initiatives pursue this approach. As summarized in Table 1, LISA-T achieves a

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* Corresponding author.

E-mail addresses: Sven.Holz@dlr.de (S. Holz), Patric.Seefeldt@dlr.de (P. Seefeldt), Kaname.Sasaki@dlr.de (K. Sasaki), Tom.Sproewitz@dlr.de (T. Sprowitz), Yahya.Motemani@saint-gobain.com (Y. Motemani).

Table 1
Comparison of deployable solar array systems.

System	Power (BOL/EOL)	Specific Power	Power Density	Mass/Volume
LISA-T (NASA)	230.7 W (planar mode)	378.5 W kg ⁻¹	461.8 kW m ⁻³	<1 U
UltraFlex 175 (ATK)	7 kW (5.5 m diameter)	175 W kg ⁻¹	—	—
ExoTerra FOSA	139 W per wing (BOL), scalable to 75–300 W	140 W kg ⁻¹	170 kW m ⁻³	1.07 kg (0.85 U stowed)
DCubed Origami	100 W at EOL (5 yr life)	—	100 kW m ⁻³ (EOL)	1 U
DEAR	100 W at EOL (10 yr life)	64.3 W kg ⁻¹ (EOL)	100 kW m ⁻³ (EOL)	1.555 kg (1 U stowed)

specific power of 378.5 W kg⁻¹ and a power density of 461.8 kW m⁻³ in a sub-1 U stowage volume (Carr et al. (2023)). ExoTerra FOSA reaches 140 W kg⁻¹ and 170 kW m⁻³ at 0.85 U stowed (ExoTerra Resource LLC (2020)), while UltraFlex ~175 delivers 175 W kg⁻¹ at 7 kW total output (ATK Space Systems (2008)). DCubed Origami provides 100 W at end of life over a five-year mission in a 1 U stowage volume, reaching 100 kW m⁻³ (DCubed GmbH (2026)). All four systems demonstrate that deployable architectures can substantially exceed the 30 W kg⁻¹ baseline of conventional rigid panels within the strict mass and volume constraints of small satellite buses.

Also at DLR research groups are working on deployable membrane structures in order to enable new applications including solar sails (Seefeldt et al. (2017)), drag sails (Sinn et al. (2017), Stelzl et al. (2023)) and deployable solar arrays. In the DLR project Gossamer, described in Seefeldt et al. (2017) and Sprowitz et al. (2019), technologies for the deployment of membrane structures were developed and tested. Within this activities, also the application of solar cells onto flexible membranes were investigated. The following project GoSolAr Sprowitz et al. (2019) shifted the focus more towards flexible solar arrays as the main application. Based on these predecessors, within the DEAR project a membrane-based solar array is developed that can be deployed out of a CubeSat segment, an artist's rendering is given in Fig. 1. A project overview is given in Sprowitz et al. (2023) and a description of the boom deployment system is presented in Tokarz et al. (2023). (see Fig. 2).

This paper is organized following the development logic of analysing the relevant space environment factors as a bases for the following design and sizing process, followed by a description of the manufacturing process and the final test campaign. In Section 2 the conceptual design is presented to give the reader a first impression of the developed array. This is followed by Section 3 describing the analysis of relevant space environment aspects for different reference missions. This includes the atomic oxygen (ATOX), radiation, and thermal analysis. Based on the analysis Section 4 presents more details about the design, including the material selection and mechanical as well as electrical sizing aspects. The manufacturing of the array is then explained in Section 5 covering the electrical contacting of the solar cells, the coating of the carrier polymer film, the assembly of all components to an array and finally the stowing of the

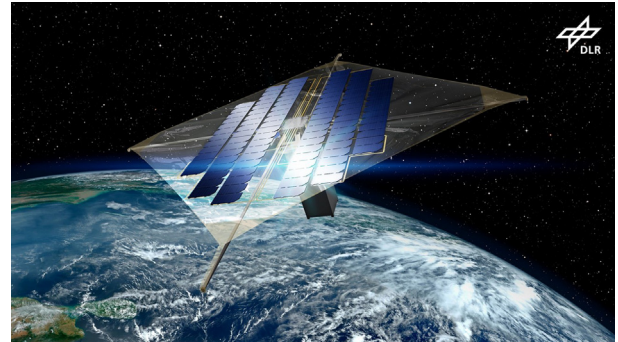


Fig. 1. The DEAR 100 W Array in space (artists' rendering).

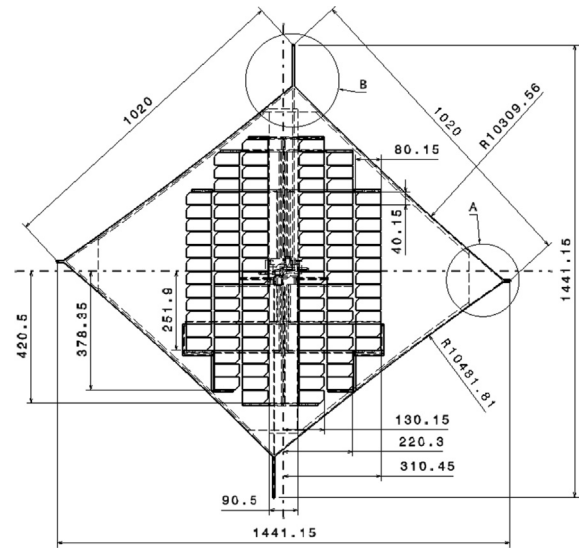


Fig. 2. General dimensions of the deployed array.

solar array blanket. The verification and testing is presented in Section 6 and the results are discussed in Section 7.

2. Conceptual design

The design of the array incorporates different mechanical and electrical aspects explained in the following sections. It has a size of about 910 mm × 910 mm and weighs 751g (including system margin). The power is generated by 100 triple junction GaAs solar cells. It should be noted that this design would allow for the mounting

of 40 additional solar cells, which was not necessary in order to be compliant to the requirement to generate 100 W at EOL.

All components are mounted on a FEP polymer carrier membrane. The busbars and grounding interconnections are made of Kovar stripes. Furthermore, mechanical interfaces and reinforcements with polyimide layers provide the required mechanical stability.

Fig. 3 gives an overview and visual impression of the design in deployed and stowed configuration.

3. Analysis of relevant space environment factors

For the analysis we consider both, satellites with 3kg, in order to consider CubeSats, and satellites with 200kg with a drag area of 1m^2 , which is the approximate size of the array. For the sake of simplicity, we consider that the area is pointing into ram-direction.

We make different assumptions for different parts of the analysis (orbit decay and atomic oxygen, radiation, thermal cycles). These assumptions are made such that different worst-case scenarios are considered while there is no mis-

sion in which all worst-case scenarios would appear at the same time.

All scenarios are deliberately placed at the upper bound of the expected environment: the orbit decay and atomic oxygen propagation use launch dates during a solar maximum, which maximises the atmospheric density and therefore the ATOX flux at a given altitude (see Fig. 4), and the radiation analysis assumes a mission start in 2003, likewise at solar maximum. The values given in Tables 2 and 3 are therefore upper bounds rather than nominal values. Fig. 4 shows the sun activity which has an impact on the atmospheric density at a certain altitude. In order to estimate atomic oxygen fluences we will analyse two different scenarios.

A low ISS like orbit, with about 500 km altitude, and a launch date in 2012 would constitute a mission during a moderate solar maximum (here during solar cycle 24). Such an orbit is shown in Fig. 5. The orbit lifetime of such a mission varies between a few months for a CubeSat and several years for a 200kg satellite. This orbit life will mostly be part of the operational lifetime.

For a mission that starts on a higher altitude like the SSOs the orbit lifetime can be many decades for a satellite with a weight of 200kg (without other means for de-orbiting). Such orbits are shown in Figs. 6a and 6b. The overall accumulated ATOX fluence is then higher but it is low during the first years, meaning during the operational lifetime. For the sake of completeness an extreme scenario is shown in which the satellite decays over the last two solar cycles (nearly 25 years). (see Fig. 7).

The radiation analysis presented here is limited to photovoltaic End-of-Life (EOL) values estimation with SPENVIS. This is because the development and implementation of other electronics is not part of the development in the project at hand. For the radiation analysis we assume a maximum and constant altitude of SSO 1205 km. Assuming a constant altitude for the 5–10 years operational time for such a high orbit is justified because the orbit decay from this altitude is very slow. Furthermore, a longer time in a high altitude is the worst-case scenario for the radiation loads. In addition, we analyse a 700 km orbit for the radiation loads in order to show the already much lower impact of radiation loads in such orbits which are more fre-

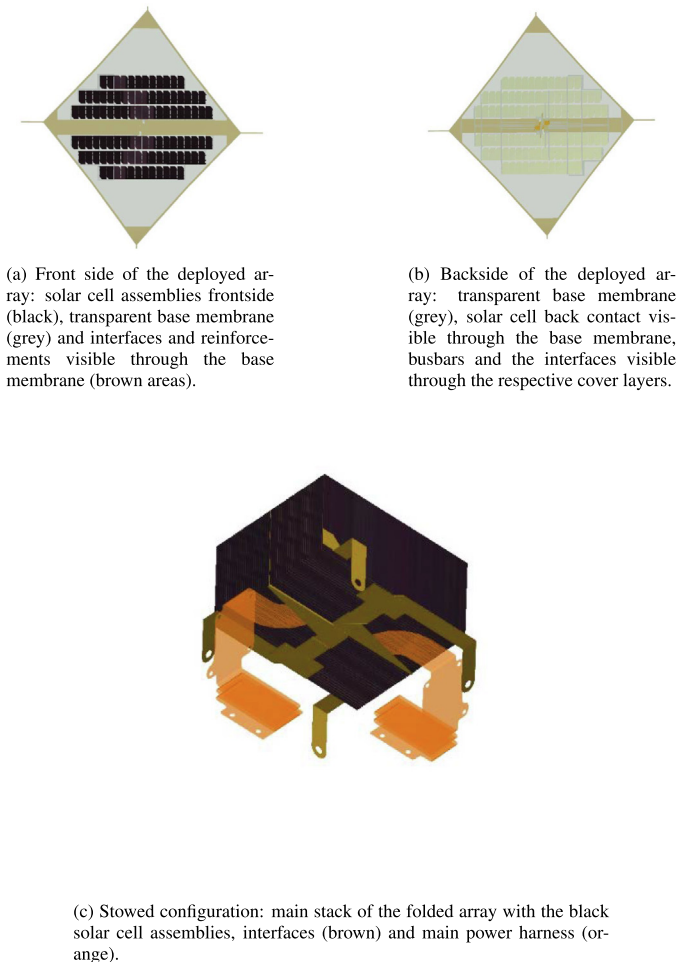


Fig. 3. Design Overview of the DEAR array.

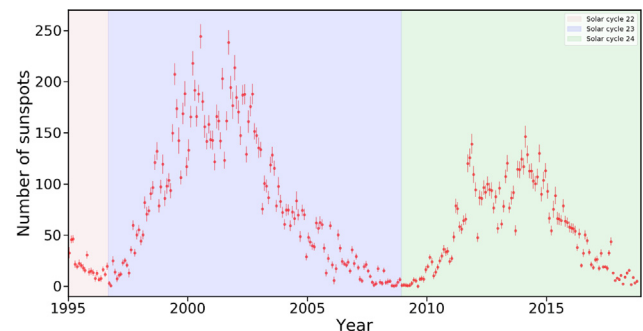


Fig. 4. Sun activity plot.

Table 2

Summary of ATOX Fluence for different cases.

Mission Scenario (Initial Altitude)	Satellite Weight [kg]	ATOX Fluence After 5 years [cm^{-2}]	Years at re-entry (200 km)	ATOX Fluence at re-entry (or after 25 years) [cm^{-2}]
ISS (500 km)	200	$7.88 \cdot 10^{20}$	13.6	$1.40 \cdot 10^{22}$
ISS (500 km)	3	-	0.16	$2.16 \cdot 10^{20}$
SSO (700 km)	200	$5.01 \cdot 10^{19}$	$\gg 25$	$3.83 \cdot 10^{20}$
SSO (700 km)	3	$(\approx 3 \cdot 10^{20})$	4.77	$2.91 \cdot 10^{20}$
SSO (550 km)	200	$5 \cdot 10^{20}$	23.5	$1.62 \cdot 10^{22}$

Table 3

Mission scenarios and occurring fluence, calculated with Spenvis.

Mission scenario	Fluence [$e \text{ cm}^{-2}$]		
	P_{mp}	V_{oc}	I_{sc}
700 km SSO, 5 year operation time	$1.08 \cdot 10^{13}$	$2.50 \cdot 10^{13}$	$3.20 \cdot 10^{13}$
700 km SSO, 10 year operation time	$2.16 \cdot 10^{13}$	$3.40 \cdot 10^{13}$	$4.50 \cdot 10^{13}$
1205 km SSO, 5 year operation time	$1.17 \cdot 10^{14}$	$1.50 \cdot 10^{14}$	$1.80 \cdot 10^{14}$
1205 km SSO, 10 year operation time	$2.34 \cdot 10^{14}$	$2.80 \cdot 10^{14}$	$3.10 \cdot 10^{14}$

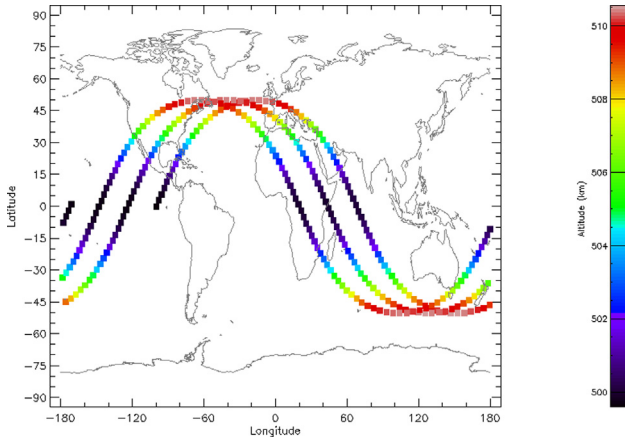
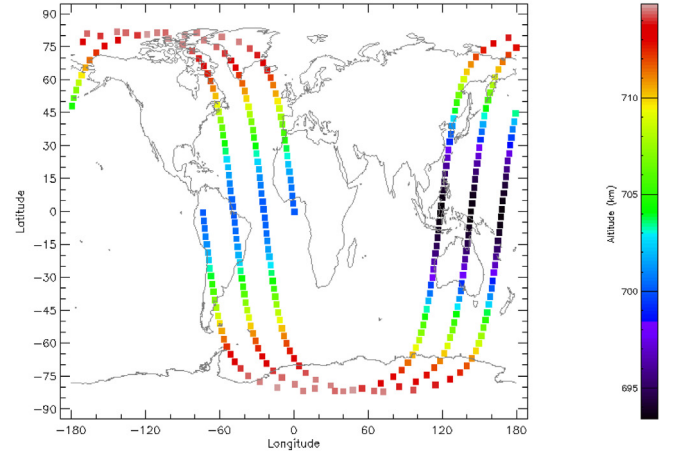
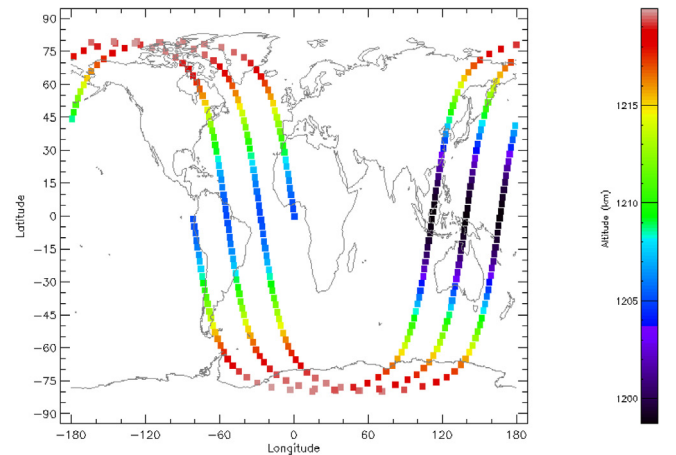


Fig. 5. ISS-like orbit, but with 500 km orbit altitude.



(a) SSO with 700 km altitude.



(b) SSO with 1205 km altitude.

Fig. 6. Higher SSO orbit plots for decay and atomic oxygen analysis.

quently used. For both orbits we look into the results for 5 years operational time (minimum required) and 10 years operational time.

Temperatures and thermal cycles are calculated for a 500 km orbit (hot case) for both a nadir and a sun pointing array. The hot case is the design driving condition, as the electrical sizing depends on the maximum cell temperature through the temperature coefficient of the maximum power point.

3.1. Orbit decay and atomic oxygen fluence

For the propagation of the orbit decay NASA's General Mission Analysis Tool (GMAT R2020a) using a SGP4 propagator was used for a fast generation of the Ephemeris and a MATLAB implementation of the NRLMSISE-00 atmosphere model which is part of the "High Precision

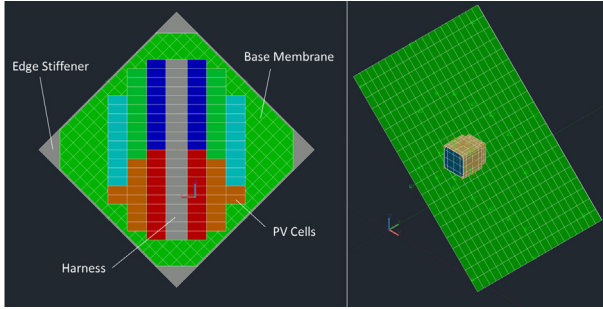


Fig. 7. Thermal model of the deployed DEAR membrane (left: front side, right: rear side).

Orbit Propagator” from Meysam Mahooti (version 2.2.1.1) was used to determine the atmosphere composition as recommended in ECSS-E-ST-10-04C. The NRLMSISE-00 provides the atmosphere density and composition for a certain altitude at a certain date, among them also the ATOX density.

The density is given in cm^{-3} , while the velocity v in cm s^{-1} , therefore the ATOX flux has a dimension of $\text{cm}^{-2} \text{s}^{-1}$. The ATOX values are given in Figs. 24–28 in the Appendix A.1 and in addition Table 2 gives an overview about the fluence after 5 years and at re-entry.

3.2. Radiation analysis

For a first sizing of the array ESA’s SSpace ENVironment Information System (SPENVIS) was used. In SPENVIS we used the Damage equivalent fluence model for solar cells (EQFLUX) that provides the 1 MeV damage equivalent

electron fluences for exposure to the fluences predicted by the trapped radiation and solar proton models, for a specified orbit and duration. Table 3 provides the 1 MeV equivalent electron fluences for the maximum Power P_{mp} , the open circuit Voltage V_{oc} and for the short circuit current I_{sc} considering a $100 \mu\text{m}$ thick cover glass. A mission during a solar maximum (start year 2003) with a duration of 5 years and 10 years is considered. The solar maximum is a conservative choice for the atomic oxygen environment, since the increased thermospheric density raises the oxygen number density and hence the ATOX flux at a given altitude.

Table 4 shows the AZUR Data Sheet values for different EOL radiation doses. The standard solar cell assemblies have a coverglass of $100 \mu\text{m}$ thickness. For all analysed cases with $100 \mu\text{m}$ coverglass the equivalent 1 MeV electron fluence is below $2.5 \cdot 10^{14} \text{e/cm}^2$ (first column in the data sheet). With that EOL values the 100W EOL can still be ensured even for extreme cases like 10 years in an altitude of 1200 km (also taking a higher temperature of $+91^\circ\text{C}$ into account).

In order to understand the limitations, a coverglass with $50 \mu\text{m}$ thickness was also investigated. Even with this limited shielding the EOL doses after 5 years operation (REQ-090 Lifetime) are always less than $2.5 \cdot 10^{14} \text{e/cm}^2$. Only for 10 years on an altitude of 1205 km (SSO) this value would be exceeded. In conclusion, between 5 and 10 years on that high altitude the EOL power of the array, operated at a temperature of 91°C , would drop below 100W. This is still in accordance to the minimum EOL performance of 100W after 5 years, as required in the DEAR project.

Table 4
AZUR SPACE 3G30C-Adv. solar cell performance parameter.

Electrical Data (SCA)

		BOL	2.5E14	5E14	1E15
Average Open Circuit V_{oc}	[mV]	2690	2606	2554	2512
Average Short Circuit I_{sc}	[mA]	519.6	517.9	513.4	501.3
Voltage at max. Power V_{mp}	[mV]	2409	2343	2288	2244
Current at max. Power I_{mp}	[mA]	502.9	501.7	499.1	485.1
Average Efficiency η_{ave} (1367 W/m^2)	[%]	29.3	28.4	27.6	26.3
Average Efficiency η_{ave} (1353 W/m^2)	[%]	29.6	28.7	27.9	26.6

Standard: CASOLBA 2005 (GS-20M/V1, etc), Spectrum: AMO WRC = 1367 W/m^2 , $T = 28^\circ\text{C}$

@fluence 1MeV [e/cm^2]

Acceptance Values (SCA)

Voltage V_{op}	2350 mV
Min. average current $I_{op, avg}$ @ V_{op}	500 mA
Min. individual current $I_{op, min}$ @ V_{op}	470 mA

Shadow protection

External Si protection diode	$V_{forward} (620 \text{ mA}) \leq 0.8 \text{ V}$ $I_{reverse} (4 \text{ V}) \leq 0.1 \mu\text{A}$
Operation Temperatures	-150°C to $+250^\circ\text{C}$

Temperature Gradients

		BOL	2.5E14	5E14	1E15
Open Circuit Voltage	$\Delta V_{oc} / \Delta T$ [mV/°C]	-6.2	-6.5	-6.6	-6.7
Short Circuit Current	$\Delta I_{sc} / \Delta T$ [mA/°C]	0.36	0.33	0.35	0.38
Voltage at max. Power	$\Delta V_{mp} / \Delta T$ [mV/°C]	-6.7	-6.8	-7.1	-7.2
Current at max. Power	$\Delta I_{mp} / \Delta T$ [mA/°C]	0.24	0.20	0.24	0.28

@fluence 1MeV [e/cm^2]

Table 5
Assumed orbit and attitude conditions for the hot case analysis.

Parameter	Hot case value
Altitude	500 km
Solar Constant	1419 W m ⁻² *
Inclination	90 °
Earth IR	Latitude dependent model**
Albedo	Latitude dependent model**
Beta angle	0°
RAAN	90°
Attitude	Front side Sun pointing

* see Anderson et al. (2001)

** see European Cooperation for Space Standardization (2008)

3.3. Thermal analysis

The thermal analysis focuses on the membrane temperature in the hot case, when directed to the sun, is of major interest. With increasing temperature the efficiency of the solar cells drops, this needs to be considered when evaluating the power output of the array. Therefore, the evaluation presented in this section focuses on the maximum temperatures of the solar cells.

In the thermal model, the DEAR membrane is constructed as the transparent base membrane with different mechanical and electrical components: solar cells, harness and edge stiffener.

The assumed orbit and attitude conditions for the study are summarized in Table 5. The orbit is assumed to be a 500 km altitude polar orbit with a beta angle of 0°. With this orbit assumption, the satellite experiences the highest thermal radiation from the Earth around the sub-solar point. The membrane constantly receives maximal solar flux, based on the Sun pointing assumption, which is partially converted to electrical energy. In the analysis, it is assumed that the solar cells convert 25% of the solar flux to electrical energy and the rest is treated as heat input for the thermal model.

Fig. 8 shows the temperatures over the membrane under the hottest time-frame in the assumed orbit. The maximum temperature calculated for the solar cells is approx. 70 °C, which is the hottest part of the membrane. The other components of the membrane, such as the harness and edge stiffeners are expected to reach lower temperature than the solar cell temperatures, because these components have lower ratios absorptivity to emissivity.

4. Design and sizing

Based on the presented analysis polymer materials for the carrier film and coatings are selected and an appropriate thicknesses are selected. Furthermore, the electrical sizing, specifically the amount of overall cells and the amount of cells per string is selected based on the radiation and thermal analysis.

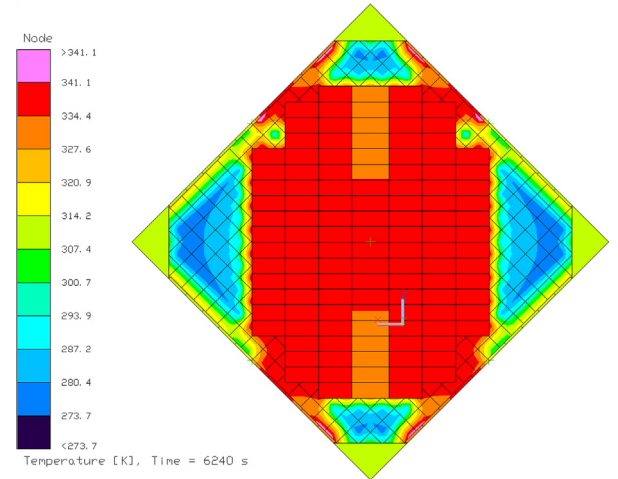


Fig. 8. Temperature distribution on the DEAR membrane at the hottest time-frame.

4.1. Material and coating selection and sizing

The most driving aspect in the material selection is the atomic oxygen resistance of the membrane. In order to safely achieve the 5 years life time, the base membrane should withstand an atomic oxygen fluence of about of 10²¹cm⁻² (compare with Table 2). Nevertheless, additional coatings and reinforcements will significantly increase this life-time, ensuring structural integrity over the complete orbit lifetime.

The material that is used as base membrane and for protective cover layers is an FEP thin-film that is coated with silicon oxide (SiO₂) on the frontside and on the backside with indium-tin oxide (ITO). FEP was selected due to its superior atomic oxygen and thermal resistance.

From atomic oxygen erosion tests at ESTEC, we know the erosion yield (*Y*) of FEP to be 6.3·10⁻²⁵ cm³/AO. The relation between fluence (*f*) and eroded thickness (*t*) is given by

$$t = Y \cdot f. \quad (1)$$

The eroded thickness due to the fluence of 10²¹cm⁻² is therefore 6.3 μm and for the maximum fluence of 1.6·10²²cm⁻² (see Table 2) it would be about 100 μm. This specifies the range of our focus in the design. Available are three different thicknesses: 25 μm, 50 μm and 100 μm. When evaluating the applicability of those thin-films, it was found that a material of 100 μm thickness is already difficult to fold and unfold. Due to the volume restrictions it was decided to use a 25 μm FEP film because for a thicker film the resistance against folding is much higher which ultimately results in a much larger folded blanket.

The SiO₂ provides a shielding against UV-radiation and provides an adhesion interface. The coating thickness of the SiO₂ can be sized according to the Beer-Lambert law (e.g. for shielding 99% of the intensity):

$$I_{\lambda} = I_{\lambda,0} \cdot e^{-\frac{4\pi k l}{\lambda}} = (1 - 0.99) \cdot I_{\lambda,0} \quad (2)$$

$$t = -\frac{\ln(0.01)}{\frac{4\pi k}{\lambda}} \quad (3)$$

The absorbance in a material along the penetration depth (or thickness, t) varies with the wavelength and is specified by the extinction coefficient. The data for SiO_2 is taken from the website www.refractiveindex.info (see Fig. 9). Evaluation of the Beer–Lambert law for SiO_2 is shown in Fig. 10. It reveals that 50 nm are sufficient to completely block UV radiation up to 120 nm wavelength from the substrate.

Considering furthermore an erosion of SiO_2 by AO with an erosion yield of about $10^{-27} \text{ cm}^3/\text{AO}$ the thickness is increased by the amount that would be eroded by a fluence of 10^{21} cm^{-2} :

$$t_{\text{SiO}_2} = 50 \text{ nm} + 10^{21} \text{ cm}^{-2} \cdot 10^{-27} \text{ cm}^3/\text{AO} = 60 \text{ nm} \quad (4)$$

The resulting minimum thickness is therefore 60 nm. For the engineering qualification model a SiO_2 thickness of 100 nm was used, providing an additional margin of a factor of five on the eroded fraction. This accounts for the fact that the degradation of dense oxide coatings in LEO is driven by undercutting at coating defects rather than by uniform recession, which is not covered by the simple erosion model applied here.

The ITO provides electrical conductance in order to implement a grounding strategy. Without an analysis on system level grounding requirements are difficult to define. However, the target surface resistance for SLI and MLI is often between $2 \text{ k}\Omega/\square$ and $10 \text{ k}\Omega/\square$ and it can be adjusted by the coating. This can be achieved by only applying a coating of less than 10 nm thickness. In consequence the main driver for the thickness is again the AO erosion. Considering an erosion yield of about $10^{-26} \text{ cm}^3/\text{AO}$ the required thickness is calculated as

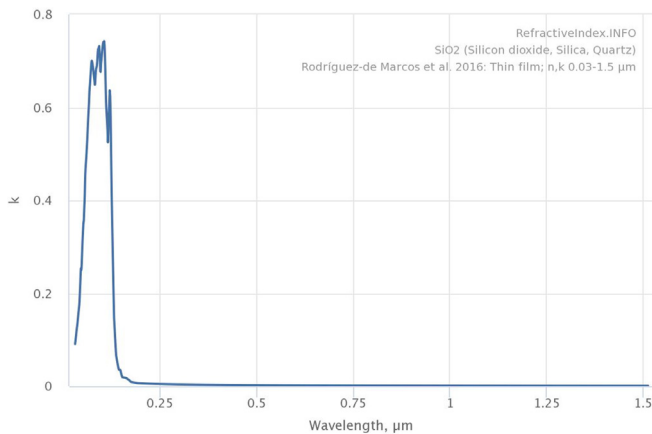


Fig. 9. Extinction coefficient index for SiO_2 (measure for the absorbance of radiation in a material), taken from Polyanskiy (2008).

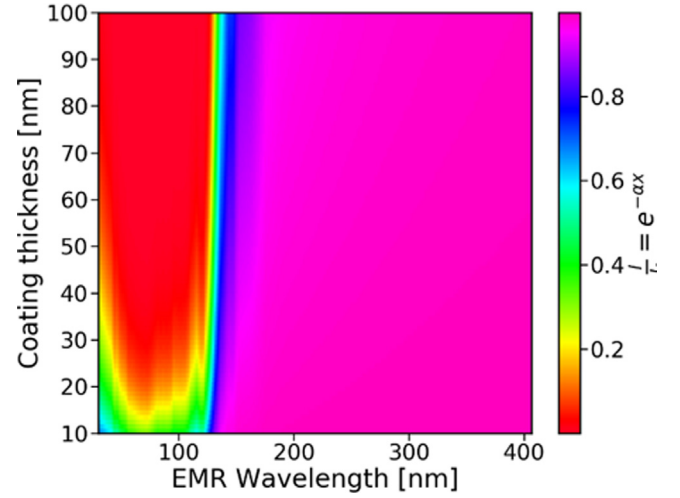


Fig. 10. Absorbance of SiO_2 . Red areas - no penetration of radiation, purple areas complete penetration.

$$t_{\text{ITO}} = 10 \text{ nm} + 10^{21} \text{ cm}^{-2} \cdot 10^{-26} \text{ cm}^3/\text{AO} = 110 \text{ nm} \quad (5)$$

Because ITO might get too brittle when the coating thickness is too large, the fluence of a five years mission is considered instead. According to Table 2 this fluence is $7.88 \cdot 10^{20} \text{ cm}^{-2}$ and therefore the minimum ITO thickness is

$$t_{\text{ITO}} = 10 \text{ nm} + 7.88 \cdot 10^{20} \text{ cm}^{-2} \cdot 10^{-26} \text{ cm}^3/\text{AO} = 89 \text{ nm} \quad (6)$$

For the engineering qualification model 80 nm of ITO were deposited as a compromise between erosion allowance and coating flexibility. This corresponds to an erosion allowance of 70 nm, i.e. a fluence of $7.0 \cdot 10^{20} \text{ cm}^{-2}$. The five year fluences of all SSO reference missions (see Table 2) are covered by more than one order of magnitude; for the 500 km ISS-like worst case the allowance corresponds to 89% of the five year fluence.

Samples of such a material were manufactured with a surface resistance of 60Ω (sometimes also referred to as $\Omega_{\square}$ or Ω_{sq} .) This value would increase over mission time while the coating is eroded.

For mechanical reinforcement, layers of $12.5 \mu\text{m}$ Upilex-S Polyimide are foreseen which are always protected by FEP cover layers. The reason for that is the low tensile strength of the FEP. The tensile modulus of FEP is 0.48 GPa and the tensile strength is 24 MPa, while the tensile modulus of the Polyimide is 9.7 GPa and the tensile strength is 578 MPa.

For the bonding of the material the acrylic adhesive 3 M 9460 is used. It is comparable to the 3 M 966 that is widely used in the MLI field, but is optimized for higher temperatures. It is also used in the ESA ADEO (see [3]) projects and proved its applicability in different environmental tests.

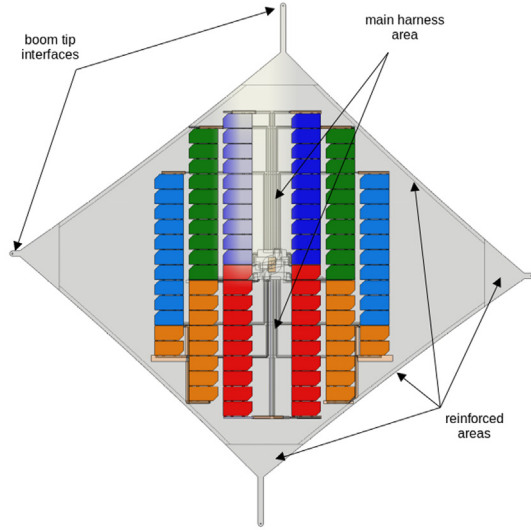


Fig. 11. String accommodation for 10 strings with 10 cells that minimizes the magnetic dipole. The location of the main harness, reinforcements and the boom tip interfaces can be seen.

It should be noted that the differences between the two adhesives are minor and one could easily be replaced by the other. For the electrical contacting of the grounding paths the conductive acrylic transfer adhesive 3 M 9703 is used.

4.2. Electrical sizing

The AZUR SPACE solar cells are characterized for different EOL values as shown in Table 4. According to the analysis presented in Section 3.2, we used the values for $2.5 \cdot 10^{14} \text{ e cm}^{-2}$. Furthermore, the maximum temperature of 70°C (see thermal analysis in Section 3.3) were used in order to evaluate the hot-case performance. Using the performance parameter as shown in Table 4, for 70°C ($\Delta T = 42^\circ\text{C}$) the following can be found:

- $V_{mp} = 2057.4 \text{ mV}$
- $I_{mp} = 493.3 \text{ mA}$
- $P_{mp} = V_{mp} \times I_{mp} = 1.01 \text{ W}$

Consequently, 100 solar cells are needed to achieve 101W in the defined worst-case scenario at 1205 km altitude with a 10 year operation time. Considering further that a SmallSat with a 100W solar array would likely use one of the more sophisticated power systems, it is logical to assume that the battery would consist out of four Lithium-ion cells each providing 3.7 V (14.8 V total). This allows for a 12 V regulated system. For this system the strings of the array always have to deliver a voltage above the 14.8 V of the battery. With some margin this can be achieved with strings made of 10 cells, which allows a safe down-conversion by the power system. By these considerations the array would consist out of 10 strings each with 10

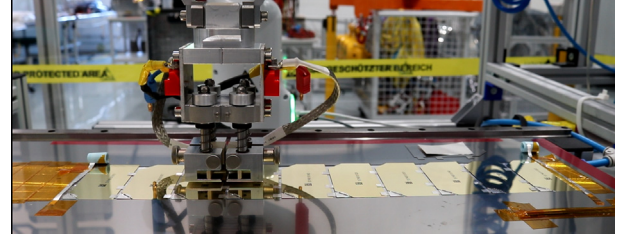


Fig. 12. Automated cell welding with a welding head attached to a Franka Emika Panda Robot. Ten cells are connected to one string with bus bars for the electrical routing. The cells are placed on a temperature controlled vacuum table.

cells providing at the end-of-life still 20.57 V and 4.93 A (101W).

4.3. Magnetic compensation

When configuring the strings and the electrical routing one aspect is the minimization of the magnetic dipole in order to reduce disturbance torques for the satellite. This requires a mirrored arrangement of the strings so that the magnetic fields, that arise due to the electrical current, cancel each other out. Fig. 11 shows a possible accommodation. Note that the strings of the same color generate magnetic dipoles in opposite directions, so that the arrangement is intended to reduce the net magnetic dipole of the array. This was implemented as a design guideline during the electrical routing; a numerical simulation of the residual dipole of the as-built harness was not performed within the project. (see Fig. 12).

5. Manufacturing of the engineering qualification model

This chapter describes the manufacturing of the engineering qualification model of the blanket. The purpose of this engineering qualification model is to test the design's manufacturability and the functionality under representative environmental conditions during the test campaign. The manufacturing workflow is divided into four parts. The first part describes the cell welding, followed by the coating of the carrier polymer films, the blanket assembly and it is finalised by the folding and stowing of the blanket.

5.1. Cell welding

The manufacturing of the DEAR-100 W solar-array blanket begins with the welding of the photovoltaic cells to ten strings. Each cell is first inspected with an electroluminescence (EL)-image to verify cell health and to record any defects before welding. The strings are then placed on a temperature-controlled vacuum table ($25 \pm 1^\circ\text{C}$). Kovar bus-bars are laid out along the back side of the first cell and the front contacts of the last cell of each string. They are temporarily fixed with Kapton tape. Spot-welding is performed automatically by an iSpot welding

head mounted on a Franka Emika Panda robot. In a first step all the cell-to-cell joints are welded. In two separate next steps the busbars are welded to the string. After welding the pre-cut transfer adhesives are glued to the backside of each cell, as a preparation for the later blanket assembly. The whole string is put into a storage box using a vacuum gripper, for safe storage until further assembly.

5.2. Coated carrier polymer film

The coatings were synthesized at pilot coater lines at Saint-Gobain Research Germany. Both SiO_2 (100 nm) and ITO (80 nm) were deposited at ambient temperature by reactive magnetron sputtering in a vertical sputter system from Von Ardenne (Fig. 13). The vertical coater was able to handle substrate sizes of up to 1.6m x 1.8m which is suitable for small batch of thin polymer film sheets that needed for the solar array. The thicknesses of the coatings were measured by ellipsometry using coated samples on Si wafer. The base vacuum and working pressure were approximately 10^{-5} mbar and 10^{-3} mbar respectively. SiO_2 deposition was performed using a double rotary Silicon target attached to a cathode equipped with MF power supply. The sputter power of 15 k W and Ar/O_2 ratio of 250 sccm/ 150 sccm were applied for this deposition. The FEP substrate was moving back and forth in front of the cathode with speed of 1.10 m min^{-1} to deposit approx. 100 nm of SiO_2 . As for ITO thin films, the deposition was performed using a double planar ITO target attached to a DC power supply. Each ITO target received 3.5 k W power during deposition. The Ar/O_2 ratio was 500 sccm/14 sccm. The substrate, coated FEP with 100 nm SiO_2 on one side, was moving back and forth in front of ITO cathode by speed of approx. 1.10 m min^{-1} to deposit approx. 80 nm ITO on the other side.

5.3. Blanket assembly

In a first step, Polyimide reinforcement patches (central harness, edges and mechanical interfaces) and all the FEP cover sheets are cut from stock material using CAD-derived templates. The transfer adhesive is applied to the patches before cutting. For the FEP cover sheets, 2mm of FEP is left to overlap on each side to prevent excess adhesive from escaping at the edges after bonding. The previously welded strings, which have been equipped with transfer adhesive on each cell, are placed into a mold to ensure they are correctly aligned with each other. In the next step, the uncut FEP main membrane was rolled onto the aligned strings and bonded to them using a long polyurethane roll, which is guided on a sled. This is shown in Fig. 14. The roll's own weight ensures that the membrane is pressed flat onto the back of the cells without wrinkles. To achieve a defined bonding, additionally a smaller roll is rolled over the back of each cell. Since the pressure is solely determined by the roll's own weight, it is ensured that the same pressure is applied to each cell.



Fig. 13. Photo of Vertical Sputter Coater at Saint-Gobain Research Germany.

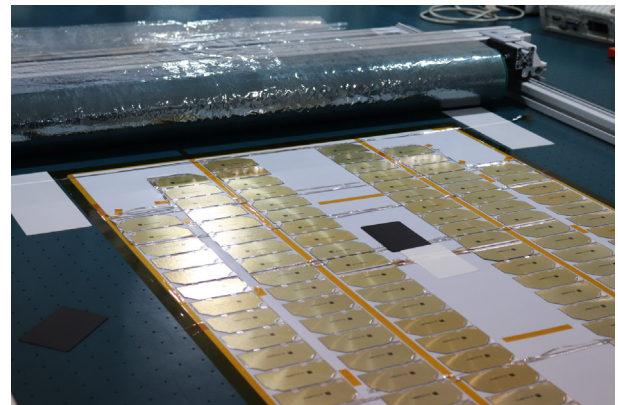


Fig. 14. Rolling of the FEP main membrane onto the welded strings. Transfer adhesive is applied to all cells directly after welding. All strings are placed into a 3D-printed alignment pattern (white parts). The busbars are placed below the cells for the later routing.

Next, all polyimide reinforcement patches are applied (edges, interfaces, main harness). For alignment, paper templates are used, which in turn are aligned with the cells. In the next step, the contour of the entire blanket is cut out. A 2 mm distance is left to the polyimide parts to ensure that excess adhesive is fully encapsulated and does not lead to bonding of the blanket when folded. Subsequently, holes are made to guide the busbars from the front side (sun-exposed side) to the back of the blanket. Then, the busbars are routed from the strings to the main harness and further to the center of the blanket. At the outer mechanical interfaces, washers are applied, which will serve as screw points for the booms later on. All polyimide layers and busbars are then covered with FEP. Again, 2 mm of overlap is left to ensure that the adhesive is fully covered. In the final step, the flexible PCBs, which serve as electrical interfaces, are

applied in the center of the blanket. The busbars, which have been routed to the center, are then soldered to the FlexPCBs.

5.4. Folding and stowing

It was decided to use a double folding pattern. Analysis showed that 100W EOL can already be achieved by 10 strings with 10 cells each. Further upscaling by integrating additional strings appears feasible without increasing the deployed area, but the stowed volume then becomes the limiting factor: a first estimate indicates that the 1U stowage envelope would be exceeded beyond approximately 12 strings. A detailed stowage-volume verification for such configurations was not carried out within this work and is left for the follow-up activities.

The stowing and deployment strategy process is shown in Fig. 15. The verification of the strategy was started by employing the membrane bread board. This pattern requires a sequential deployment along the two dimensions. The membrane is stowed in both dimensions with a zig-zag folding. First, a zig-zag folding is applied along the diagonal on which the main harness is located. The resulting folded membrane strip is then folded a second time (which results in a configuration that may look like a snake, therefore this pattern is internally referred to as snake folding). The deployment is carried out in the opposite order. Note that the first deployment direction is now unfolding in the horizontal plane rather than in the vertical plane. This is an adaptation of the previously investigated GoSolAr design (see [4]). This is the main advancement that allows to stow the membrane in one half the available 1U-volume and the deployer with the boom in the other half.

6. Test campaign

This chapter describes the test campaign conducted for the engineering qualification model. The test flow for the whole system consist of mechanical testing, venting, and



Fig. 15. The stowed blanket with the electrical interfaces on the right (FlexPCBs). One boom tip interface can be seen in the lower left corner.

thermal-vacuum tests. The objective of the test campaign was to demonstrate that the system could survive the launch loads and the space environment before and during operations, while still delivering the required power output at EOL. In this paper we limit ourselves to the blanket itself, verifying its structural integrity and performance under the applied loads. Functional verification was performed through visual inspection, electroluminescence (EL)-inspection, and power-output measurements taken under a solar simulator both before the test campaign and after each deployment. The complete test flow is illustrated in Fig. 16.

6.1. Visual- and EL-Inspection

During the qualification campaign the DEAR-100 W blanket was subjected to regular visual checks and to EL-imaging of the photovoltaic strings.

The complete assembly was examined for macroscopic damage such as cracks, broken or detached components, tears in the blanket membrane and any signs of delamination (e.g. peeled-off sections).

EL-imaging was employed solely as a non-destructive diagnostic tool to reveal potential cell damage and to verify the integrity of the series interconnects. For each measurement the fully assembled strings were forward-biased with a constant current of 0.250A. Because the harness contains blocking diodes, they had to be bypassed; this was realised with external clamps. All EL-pictures were taken inside a closed vacuum chamber to eliminate ambient light.

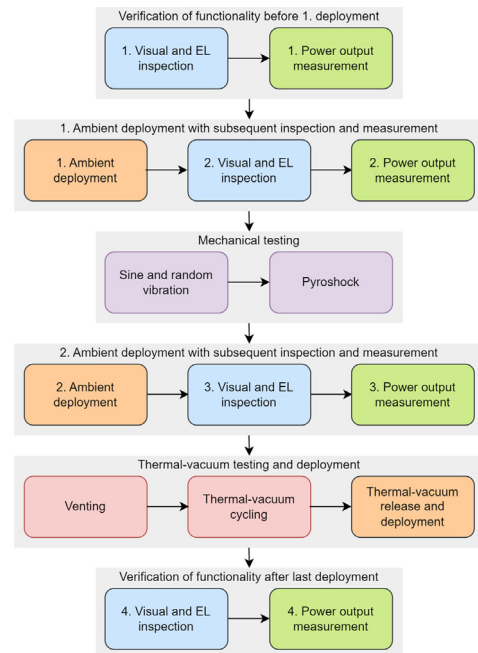


Fig. 16. Test flow of the test campaign for the DEAR engineering qualification model.

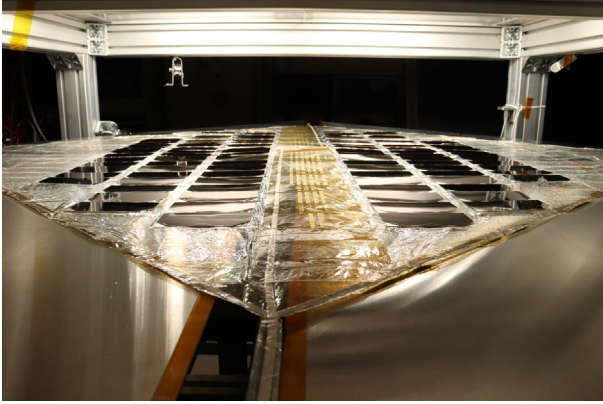


Fig. 17. Test setup for the power-output measurements.

6.2. Power output measurements

To verify the predicted performance, power-output measurements with a self-made solar simulator were carried out several times during the test campaign. These measurements also probe the robustness of all electrical connections, because any degradation appears as a reduction of the measured power. In addition to the plain power values of each string, the characteristic I–V curves were recorded. All measurements were performed with a source measurement unit (SMU). To suppress any stray ambient light, the tests were conducted inside the closed vacuum chamber. The test set-up with the unfolded solar array is shown in Fig. 17.

During the campaign it became clear that the inhomogeneous illumination of the solar simulator induced step-like features in the measured I–V curves, as partially shaded cell strings trigger the forward conduction of the corresponding bypass diodes, leading to discrete voltage drops. An example of a measured curve with these steps is shown in Fig. 18 (blue curve). For comparison an idealised I–V curve with the same open-circuit voltage V_{oc} and short-circuit current I_{sc} but without steps is plotted in green. The measured curve shows a considerably lower MPP than the theoretical one. The steps are generated by the bypass diodes that are integrated in each cell. When a cell receives less illumination it cannot sustain the string current, so the diode conducts only after its forward-bias threshold is reached, resulting in a step wise voltage drop (Rauschenbach, 1976). Consequently, the absolute MPP value is not reliable and was used only as a relative indicator to track changes between successive measurements.

6.2.1. Performance evaluation

To assess the overall performance of the array, the open-circuit voltages V_{oc} and short-circuit currents I_{sc} of each string were measured. These parameters are only weakly affected by the non-uniform illumination and therefore can be used to estimate the in-orbit power output (Rauschenbach, 1976). For a realistic comparison the local irradiance, the cell temperature and the resistances in the

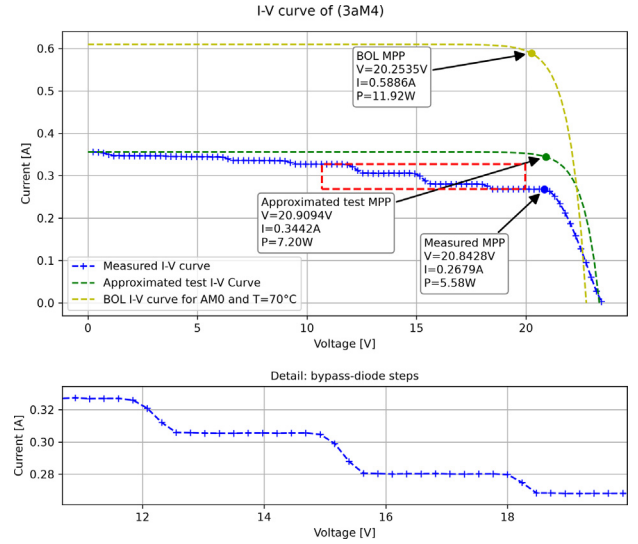


Fig. 18. I–V curve of string 3a from the final power-output measurement (blue), the approximated I–V curve for the same I_{sc} and V_{oc} under test conditions (green), and the reconstructed BOL I–V curve for AM0 and $T = 70^\circ\text{C}$ (yellow), each with its maximum power point. The lower panel shows a magnified view of the voltage region marked in red, in which the bypass-diode-induced current steps are clearly visible.

measurement harness were measured and taken into account. The test conditions together with the values from the cell data sheet were used to calculate the expected V_{oc} and I_{sc} , which were then compared to the measured values. The comparison process is shown in Fig. 19.

6.2.2. Environmental and test-setup related measurements

The following subsections describe the environmental quantities that influence the measured V_{oc} and I_{sc} .

Local irradiance. The irradiance distribution over the test area was obtained with a grating spectrometer at 28 probing points. Cubic radial basis function (RBF) interpolation was used to generate a radiance map as shown in Fig. 20. The positions of the PV-strings are indicated as red rectangles. Over the area covered by the array the irradiance ranges from 739 W m^{-2} to 961 W m^{-2} with a mean

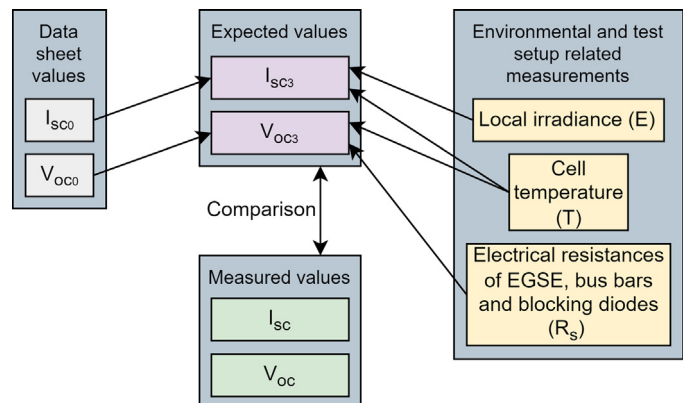


Fig. 19. Power output comparison process.

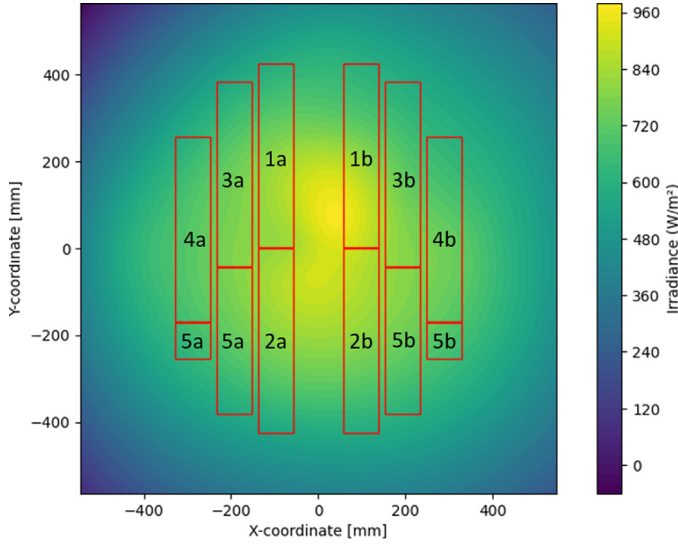


Fig. 20. Irradiance distribution with the approximate locations of the PV-strings (red rectangles). The number/letter combinations represent the corresponding string on the blanket.

of 840 W m^{-2} . This corresponds to a non-uniformity of 13.1% according to IEC 60904-9 (International Electrotechnical Commission, 2020) and to 61.5% of the AM0 solar constant. The self-made simulator is therefore below Class C and is explicitly not a calibrated performance measurement facility. It is used to verify functionality and to track relative changes between deployments, while the absolute performance assessment relies on the normalisation described in the following.

Varying irradiance shifts the I-V curve along the current axis but does not change its shape, provided that all cells are illuminated equally (Rauschenbach, 1976). The voltage variation is negligible and therefore omitted:

$$V_{oc1} \approx V_{oc0}$$

The short-circuit current scales linearly with the local irradiance E relative to the solar constant S_0 :

$$I_{sc1}(E) = I_{sc0} \frac{E}{S_0}. \quad (8)$$

Here the subscript “0” denotes the data-sheet value (AM0 spectrum) and “1” the value under the measured intensity E .

Temperature. The AZUR cells are specified for a nominal temperature of 28°C , as seen in Table 4. With increasing temperature the short-circuit current rises slightly while the open-circuit voltage drops noticeably (Ley et al., 2019; Kitai, 2018). A PT-100 sensor placed under one cell of string 4b recorded the cell temperature during each measurement. Assuming a uniform temperature across the blanket, the temperature correction is applied as

$$V_{oc2}(\Delta T) = V_{oc1} + \frac{\Delta V_{oc}}{\Delta T} \Delta T \quad (9)$$

$$I_{sc2}(\Delta T) = I_{sc1} + \frac{\Delta I_{sc}}{\Delta T} \Delta T \quad (10)$$

The coefficients $\Delta V_{oc}/\Delta T$ and $\Delta I_{sc}/\Delta T$ are taken from the cell data sheet; ΔT is the temperature difference between the measurement and the nominal 28°C .

Electrical resistance. The series resistance R_s in the measurement harness reduces the open-circuit voltage:

$$V_{oc3}(R_s) = V_{oc2} - I_{sc2} R_s. \quad (11)$$

The total series resistance consists of three contributions:

$$R_s = R_{bb} + R_{EGSE} + \frac{V_{diode}}{I_{diode}}. \quad (12)$$

The short-circuit current is unaffected by R_s :

$$I_{sc3} = I_{sc2}. \quad (13)$$

For the resistance of the Kovar bus bars R_{bb} a four-wire measurements of two Kovar samples (lengths 100mm and 200mm) were conducted. The linear relation yielded

$$R_{\text{sample}}(x) = \rho A_{CS,\text{sample}} x + R_{\text{contact}}. \quad (14)$$

From the two measured resistances $R_{\text{sample},1}$ and $R_{\text{sample},2}$ the product $\rho A_{CS,\text{sample}}$ is obtained:

$$\rho A_{CS,\text{sample}} = \frac{R_{\text{sample},2} - R_{\text{sample},1}}{x_2 - x_1}. \quad (15)$$

Multiplying this value by the actual conductor length of each string gives the bus-bar resistance:

$$R_{bb,\text{string}} = \rho A_{CS,\text{sample}} \sum l_{\text{conductor}}. \quad (16)$$

The resistance of the blocking diodes is provided by the I-V curve in the diodes data sheet (see Fig. 21). Using the measured string current, the corresponding voltage drop V_{diode} is read from that curve. At the MPP current and a diode temperature of 75°C the drop is 0.71 V.

The electrical ground support equipment (EGSE) connects the flexible PCB to the breakout box where the SMU is plugged in. A four-wire measurement of the complete cable-assembly provided R_{EGSE} .

7. Results

7.1. Visual and EL-Inspection Results

During the first EL-inspection, which was performed after the membrane integration, a crack was found on the upper side of one photovoltaic cell in string 2a (see the comparison in Fig. 22). In Fig. 22b the bright area in the upper-left corner marks the damage. Because the defect was detected prior to the membrane and deployer being assembled, it must have been introduced during membrane integration or handling, but before the first membrane folding. The cell remains electrically functional, and the expected loss in power output is likely small since most of its surface is still illuminated. Nevertheless, this assumption must be confirmed by the subsequent power-output measurements.

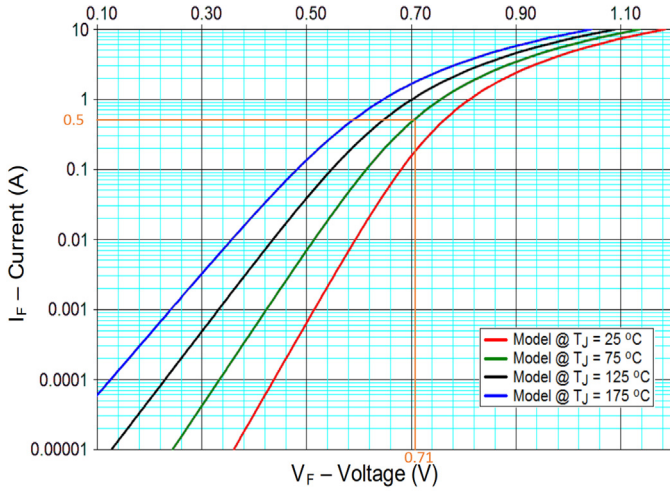
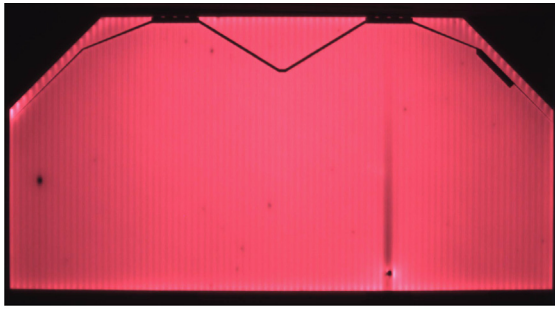
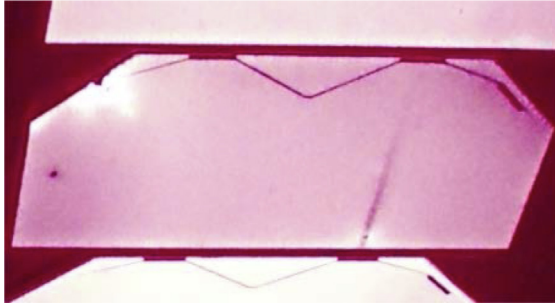


Fig. 21. I-V curve of the blocking diodes with the MPP current marked; voltage loss at 75 °C is 0.71 V.



(a) Undamaged cell directly after contacting in string 2a.



(b) Corresponding cell after integration in string 2a.

Fig. 22. EL-images of damaged cell in string 2a after contacting and after integration.

After the first deployment test, damaged cells were identified in strings 1a and 4b. Fig. 23 shows the EL-images of these cells. The cell in string 1a (Fig. 23b) exhibits a darker region in the lower-right corner compared with the same cell inspected immediately after contact (Fig. 23a). The damage in the cell of string 4b (Fig. 23c) is smaller, yet also visible in the lower-right corner. Both defects most likely arose during system integration or during the first deployment.

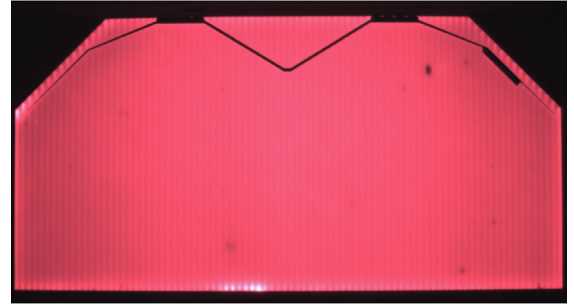
The impact of these defects on the array performance can be bounded in two ways. Empirically, the total array



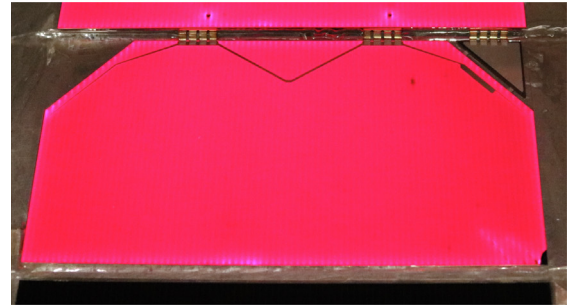
(a) Undamaged cell after contacting in string 1a.



(b) Corresponding cell after first deployment in string 1a.



(c) Undamaged cell after contacting in string 4b.



(d) Corresponding cell after first deployment in string 4b.

Fig. 23. EL-images of damaged cells in string 1a and 4b after first deployment.

power changed by 0.92 W (1.7%) between the first and the last power-output measurement (Table 6), a figure that already includes the combined effect of all cell damage detected by EL together with the observed misalignment and the temperature increase over the campaign. This constitutes an upper bound on the loss associated with the cell cracks under the applied test conditions. Analytically, a margin analysis of the EOL power output against the fraction of damaged cells shows that the worst-case mission scenario (SSO, 1205 km, 10 years) tolerates more than

Table 6
Compilation of the power-output results.

String	Measurements							
	Measurement 1		Measurement 2		Measurement 3		Measurement 4	
	T [°C]	Power [W]	T [°C]	Power [W]	T [°C]	Power [W]	T [°C]	Power [W]
1a	50.1	5.072	55.4	5.301	56.2	5.187	56.6	5.364
1b	50.8	4.769	56.6	5.390	56.9	5.333	56.7	5.550
2a	51.5	5.361	57.3	4.864	57.3	4.655	57.0	4.729
2b	52.1	5.420	57.8	4.973	57.6	4.618	57.4	4.842
3a	52.5	4.997	58.2	5.639	58.1	5.600	57.7	5.584
3b	53.2	5.026	58.9	5.055	58.4	4.969	57.8	4.962
4a	53.8	5.672	59.3	5.619	58.6	5.500	58.0	5.437
4b	54.2	5.889	59.8	5.443	59.4	4.749	58.5	5.352
5a	54.6	4.967	60.1	5.293	60.1	5.071	58.9	4.953
5b	55.0	5.577	60.3	5.011	60.2	4.830	59.0	5.057
Total Power [W]		52.750		52.587		50.511		51.829

4% of damaged cell area before the 100W requirement is violated, while the less demanding scenarios tolerate the damage of six to seven full cells. As only three partially damaged cells were detected across the whole array, the requirement is met with margin in all scenarios.

7.2. Power output measurement results

The compiled results are shown in Table 6. Measurement 1 was performed at the beginning of the test campaign, before the first folding of the array. Measurements 2–4 correspond to the three deployments. Each string was measured individually and the total array power is the sum of all strings. The inhomogeneous illumination of the solar simulator can be seen by the uneven power outputs of the strings. Note that the array was rotated by 180° for the first measurement, which explains the different power distribution.

The fields at the bottom of Table 6 indicate the calculated total power for each measurement. Between measurement 1 and 2 the total power drops only slightly. Measurement 3 shows a larger loss of about 2.2W, which is attributed mainly to a lower irradiance level (supported by the higher power observed in measurement 4). Overall, the difference between the first and the last measurement is 0.92W, which may be linked to the visual- and EL-inspection findings (Section 7.1) as well as to a slight misalignment between the array and the simulator and a modest temperature increase.

7.2.1. Comparison of measured and expected values

Only the last measurement (representative of the expected in-orbit condition) was compared to the calculated values. Tables 7 and 8 list the short-circuit current and open-circuit voltage, respectively, for each string.

The calculation starts with the data-sheet I_{sc0} , scales it with the local irradiance (Section 6.2.2), and then applies the temperature correction (Section 6.2.2). For most strings

the calculated and measured values agree well. Strings 5a and 5b are lower, while strings 3a and 4a are higher. Two plausible reasons are:

1. The irradiance interpolation deviates from the true illumination, especially for the outer strings where fewer probing points are available.
2. Only the temperature beneath string 4b was measured; other strings may have experienced slightly different temperatures.

Because the voltage is only weakly dependent on irradiance, the temperature correction (Section 6.2.2) was applied first, followed by the voltage drop due to series resistance (Section 6.2.2). Measured voltages are consistently a few percent lower than the expected ones. The discrepancy is mainly attributed to higher actual series resistances than estimated, and in the case of string 1a also to the cell damage observed during visual inspection. For string 2b the deviation is still small and can be explained by the same temperature/irradiance effects discussed above.

Because the measured V_{oc} and I_{sc} are only weakly affected by the non-uniform illumination, they can be used to reconstruct the in-orbit performance. The maximum power point is recovered from the measured V_{oc} and I_{sc} using the V_{mp}/V_{oc} and I_{mp}/I_{sc} ratios of the data sheet and a constant I–V curve shape (Rauschenbach (1976)), and the resulting values are then corrected to in-orbit intensity, cell temperature and end-of-life radiation damage. For the worst-case mission scenario (SSO, 1205 km, 10 years) this yields a predicted EOL array power of 104.8W, and higher values for all other scenarios, so that the 100W requirement is met with margin.

The absolute MPP read directly from the measured I–V curves is therefore not used as the performance figure, as it is reduced by the bypass-diode steps under non-uniform illumination. A verification under a homogeneous light source, or in orbit, would allow a direct MPP measurement and remains the main open point of the performance

Table 7
Comparison of calculated and measured short-circuit current.

String	E	$I_{sc1}(E)$	T	$dI_{sc}(\Delta T)$	I_{sc3}	Meas. I_{sc}	Match
	[W/m ²]	[A]	[°C]	[A]	[A]	[A]	[%]
1a	878.997	0.334	56.6	0.010	0.344	0.360	104.38%
1b	961.315	0.365	56.7	0.010	0.376	0.377	100.23%
2a	879.130	0.334	57.0	0.010	0.345	0.347	100.61%
2b	912.999	0.347	57.4	0.011	0.358	0.375	104.78%
3a	803.437	0.305	57.7	0.011	0.316	0.356	112.50%
3b	832.936	0.317	57.8	0.011	0.327	0.328	100.08%
4a	739.412	0.281	58.0	0.011	0.292	0.326	111.63%
4b	770.044	0.293	58.5	0.011	0.304	0.313	103.00%
5a	804.767	0.306	58.9	0.011	0.317	0.306	96.37%
5b	818.634	0.311	59.0	0.011	0.322	0.315	97.57%

Table 8
Comparison of calculated and measured open-circuit voltage.

String	T	$dV_{oc}(\Delta T)$	R_s	I_{sc3}	$dV_{oc}(R_s, I_{sc2})$	V_{oc3}	Meas. V_{oc}	Match
	[°C]	[V]	[Ohm]	[A]	[V]	[V]	[V]	[%]
1a	56.6	−1.773	3.8	0.36	−1.381	23.746	23.448	98.74%
1b	56.7	−1.779	3.9	0.38	−1.451	23.670	23.448	99.06%
2a	57.0	−1.798	3.9	0.35	−1.340	23.762	23.448	98.68%
2b	57.4	−1.823	3.9	0.37	−1.444	23.633	22.974	97.21%
3a	57.7	−1.841	3.9	0.36	−1.397	23.661	23.448	99.10%
3b	57.8	−1.848	3.9	0.33	−1.291	23.761	23.448	98.68%
4a	58.0	−1.860	4.0	0.33	−1.314	23.726	23.448	98.83%
4b	58.5	−1.891	4.0	0.31	−1.265	23.744	23.449	98.76%
5a	58.9	−1.916	4.2	0.31	−1.285	23.699	23.448	98.94%
5b	59.0	−1.922	4.2	0.31	−1.321	23.657	23.211	98.12%

assessment. Such a measurement is foreseen within the DLR in-orbit demonstration mission PLUTO (Aksteiner et al. (2025)).

With a system mass of 1.555 kg and the 100 W EOL output, the DEAR blanket reaches a specific power of 64.3 W kg^{−1} and a volumetric power density of 100 kW m^{−3} in a 1 U stowage volume, more than twice the 30 W kg^{−1} baseline of conventional rigid panels mentioned in Section 1 (Table 1).

8. Conclusion and outlook

This work presents a validated design and manufacturing process for the DEAR 100 W flexible solar array blanket, which was qualified through extensive testing for a wide range of space mission applications.

The radiation and atomic-oxygen analyses show that the selected polymer and cell materials comfortably exceed the expected fluence limits for the targeted mission lifetimes, while thermal studies confirm that the membrane temperature remains within the operating envelope, preventing efficiency loss.

Mechanical vibration, venting, and thermal-vacuum tests confirmed the structural integrity of the flexible poly-

mer carrier, the Kovar bus-bars, and the electrical harness, while meeting the required mass and volume constraints.

Electrical performance measurements using metal halide lamps confirmed the functionality of all ten strings and, after normalisation of the measured V_{oc} and I_{sc} to AM0 conditions, are consistent with the predicted EOL output of 100 W after five years in a 1205 km circular orbit under worst-case temperature and radiation conditions. The absolute MPP measured under the available light source is not a calibrated performance figure: the non-uniform illumination introduced shading artefacts and step-like distortions in the I–V curves. A homogeneous light source would be required for a calibrated end-of-life performance assessment. With a resulting power density of 100 kW m^{−3}, the DEAR array is competitive with the deployable solar array systems presented in Section 1.

The series-string configuration (10 × 10 cells) satisfies a possible spacecraft's 12 V power-system requirement.

Electroluminescence inspection revealed no critical defects, and the measured series resistance and diode voltage drops remain within the predicted range, confirming the robustness of the electrical design. Even though the cell damages do not critically effect the array performance, it proved that handling of the solar array blanket with its

triple-junction GaAs cells is difficult during integration and testing, as this damage appeared only after the membrane was mounted and folded. As the damage was not introduced by the vibration or thermal-vacuum tests, manual handling during integration is identified as the dominant risk. The definition and validation of dedicated handling and folding procedures to mitigate this risk for flight models requires a further manufacturing campaign and is therefore addressed within the follow-up activities.

The folding concept allows for stowing within a single 1U CubeSat segment and deploys reliably after being exposed to the simulated launch loads and space thermal-vacuum environment during the test Campaign.

In summary, the DEAR 100W blanket satisfies the performance, reliability, and integration criteria required for small-satellite missions, providing a lightweight, deployable, and highly efficient power system that can be readily integrated into future CubeSats or similar platforms.

Building on these results, several activities are planned to further advance and validate the DEAR solar array technology in a real space environment. The array is scheduled to fly aboard the upcoming DLR in-orbit demonstration mission PLUTO, which will demonstrate the deployment mechanism under actual orbital conditions and provide direct measurements of the power output in space (Aksteiner et al. (2025)).

In parallel, DLR is participating in the European Material Ageing (EMA) experiment, in which material samples relevant to the DEAR array are included in a broader investigation of material behaviour in the space environment (Perraud et al. (2025)). The results are expected to give valuable insights into the long-term durability and degradation characteristics of the array materials in low Earth orbit.

Declaration of competing interest

The authors declare that they have no known competing financial interests or personal relationships that could have appeared to influence the work reported in this paper.

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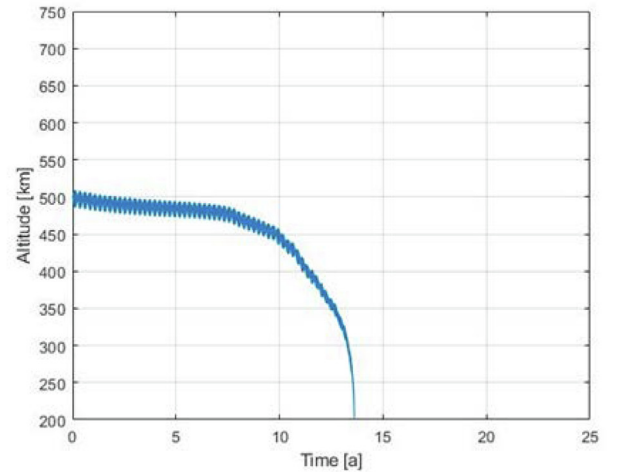
Further we thank the German ARTES delegation, in particular David Futterer, for the support of this activity.

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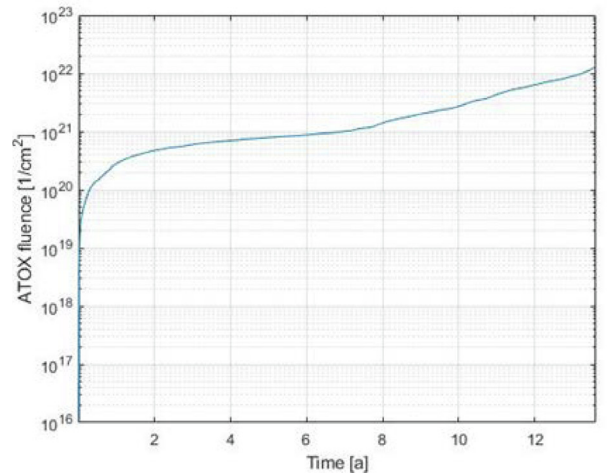
Appendix A. Appendix

A.1. Decay and atomic oxygen fluence plots

(see Figs. 24–28).

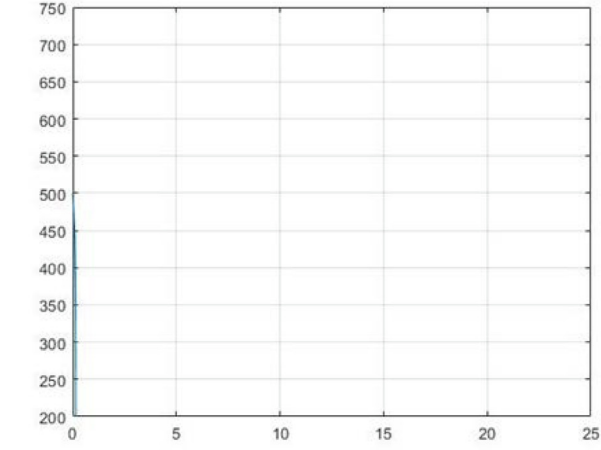


(a) Altitude over time.

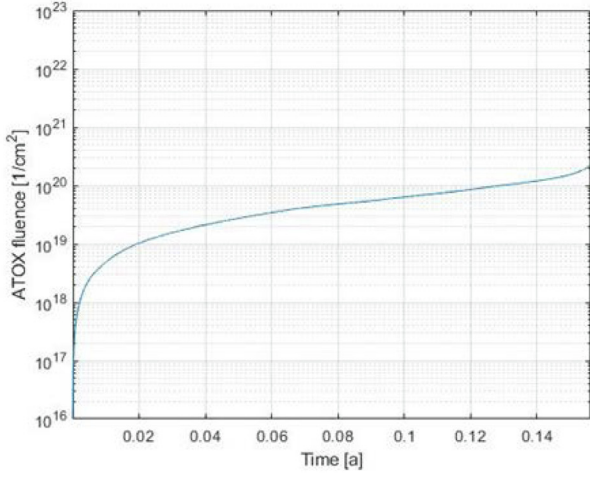


(b) ATOX fluence over time.

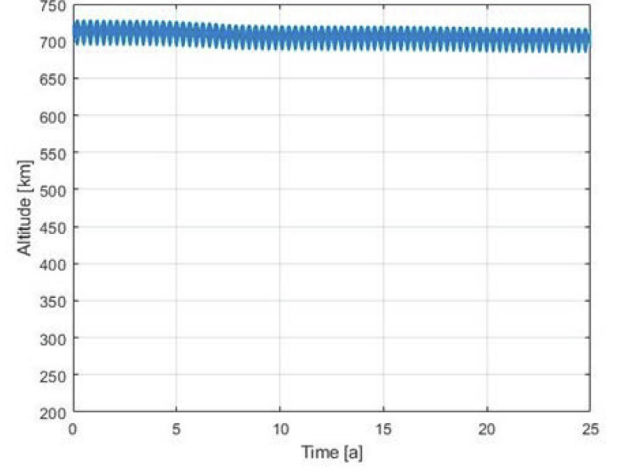
Fig. 24. Atomic oxygen fluence for an ISS like orbit with about 500 km initial altitude, for a 200kg satellite launched 01/01/2004, orbit life is about 13 years with a final fluence of $1.40 \cdot 10^{22} \text{ cm}^{-2}$.



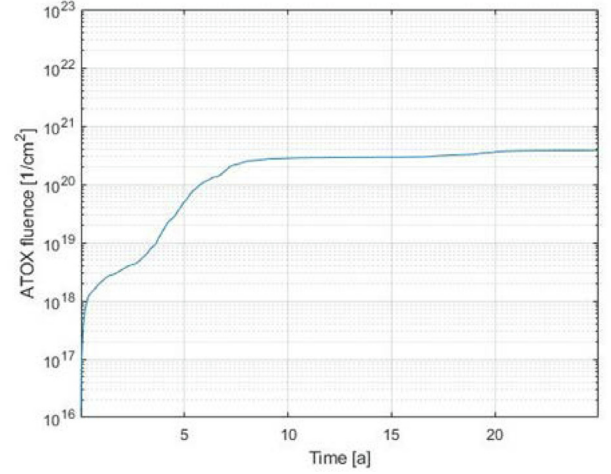
(a) Altitude over time.



(b) ATOX fluence over time.



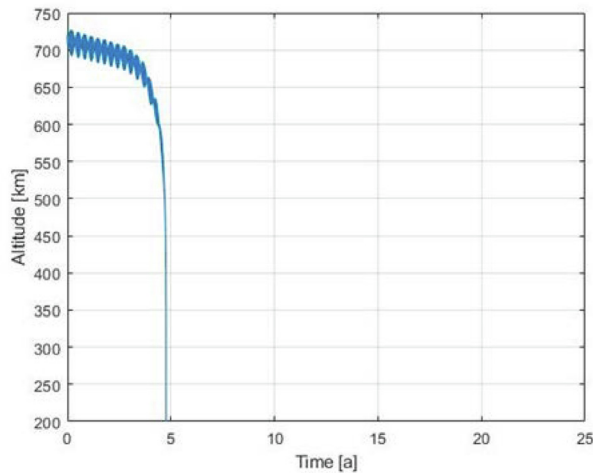
(a) Altitude over time.



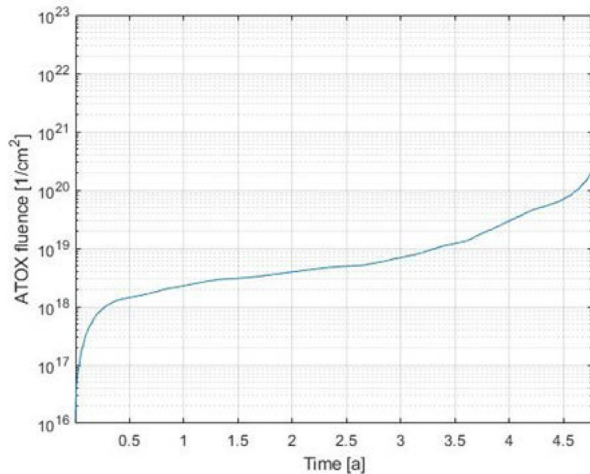
(b) ATOX fluence over time.

Fig. 25. Atomic oxygen fluence for an ISS like orbit with about 500km initial altitude, for a satellite with 3kg launched 01/01/2004, orbit life is less than 2 months with a final fluence of $2.16 \cdot 10^{20} \text{ cm}^{-2}$.

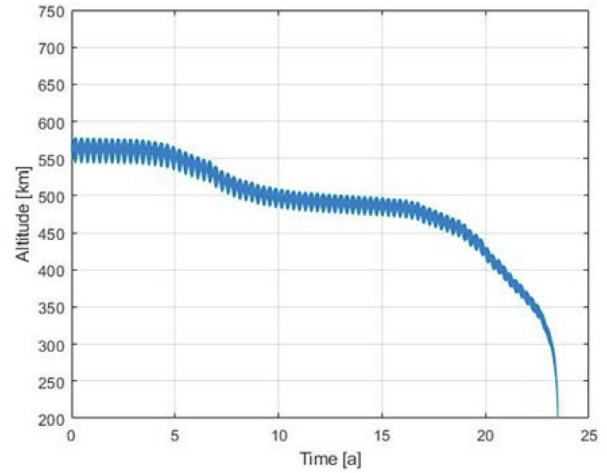
Fig. 26. Atomic oxygen fluence for an SSO orbit with an initial altitude about 700km for a 200kg satellite launched 01/01/1995. After 25 years a fluence of $3.83 \cdot 10^{20} \text{ cm}^{-2}$.



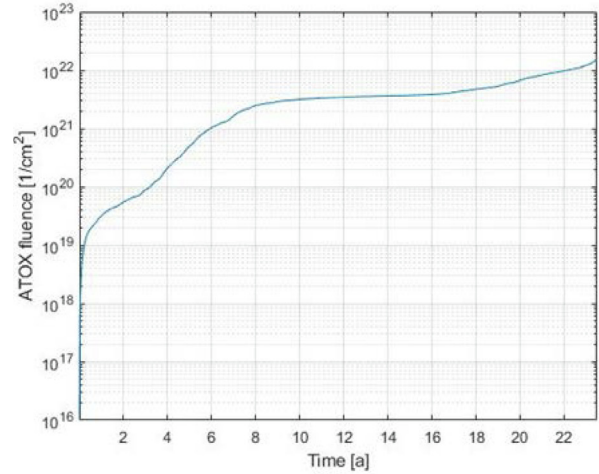
(a) Altitude over time.



(b) ATOX fluence over time.



(a) Altitude over time.



(b) ATOX fluence over time.

Fig. 27. Atomic oxygen fluence for an SSO orbit with an initial altitude about 700km, for a 3kg satellite launched in 01/01/1995. The orbit life is a little less than 5 years with a fluence of $2.91 \cdot 10^{20} \text{ cm}^{-2}$.

Fig. 28. Atomic oxygen fluence for an SSO orbit with an initial altitude about 560km, for a 200kg satellite launched in 1995. The orbit life is about 23 years with a fluence of $1.62 \cdot 10^{22} \text{ cm}^{-2}$.

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