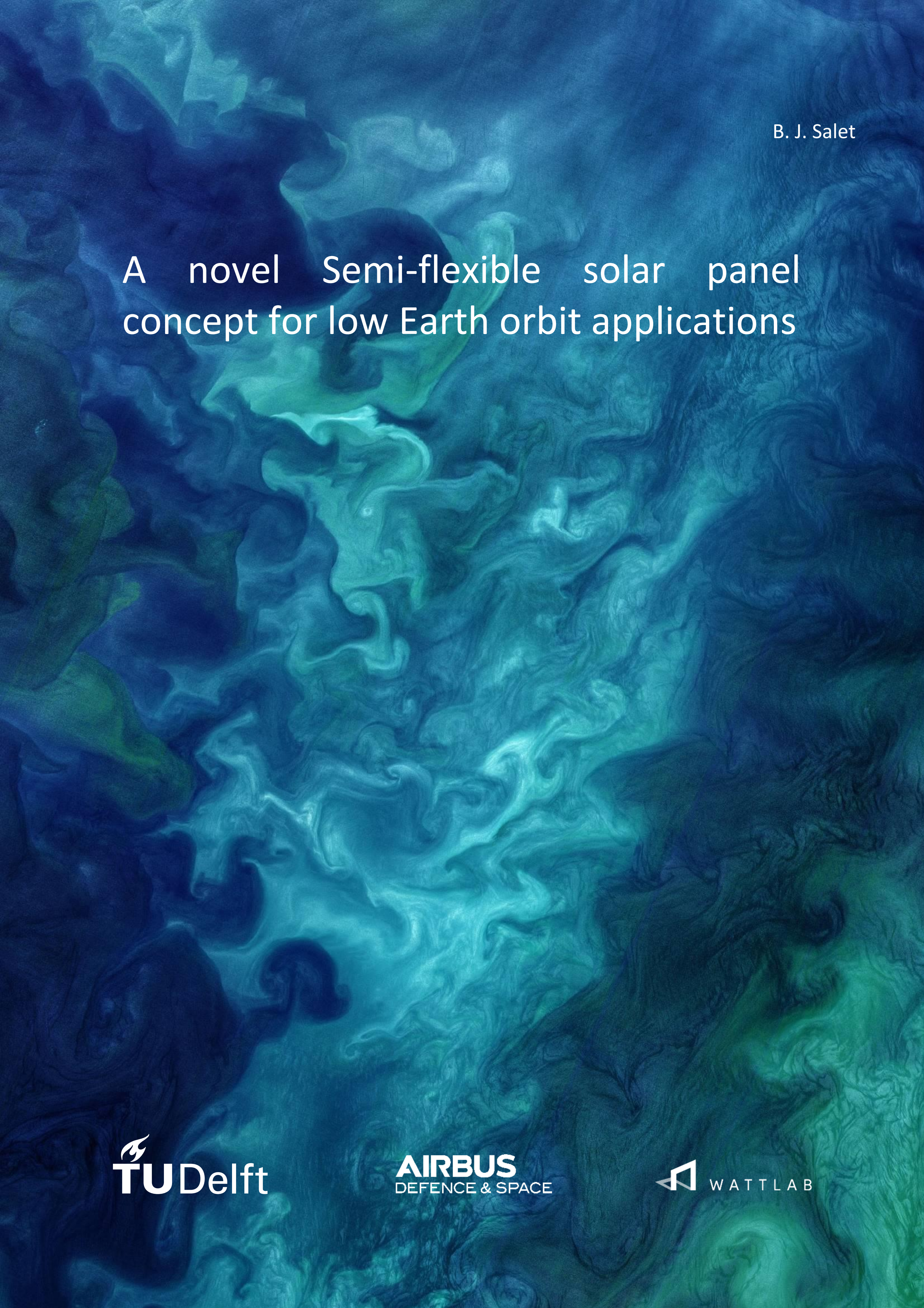

B. J. Salet

A novel Semi-flexible solar panel concept for low Earth orbit applications

TU Delft

AIRBUS DEFENCE & SPACE

WATTLAB

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A novel Semi-flexible solar panel concept for low Earth orbit applications

A collaboration between Delft University of Technology, Airbus Defence & Space and Wattlab

By

B. J. Salet
(4156021)

in partial fulfilment of the requirements for the degree of

Master of Science
in Space System Engineering

at the Delft University of Technology,
to be defended publicly on Thursday October 31st at 09:30 AM.

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Cover image credits to NASA [1]*: A satellite image showing the Alaskan coast in an almost artistic way.*
Title: Churning in the Chukchi Sea
Description: Regardless of the amount of winter ice cover, the waters of the Alaskan coast usually come alive each spring with blooms of phytoplankton.

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Preface

This report is the result of a master thesis project as part of the master Space System Engineering at Delft University of Technology. The project has been a collaboration between Delft University of Technology, Airbus DS NL and Wattlab. Airbus is aiming to enter a new market segment and is targeting that market under the name SparkWing, an Airbus company. As Wattlab was looking into the possibilities for the use of their solar panels for CubeSat missions, the connection between Wattlab and Airbus was established with this thesis as the result. This master thesis investigates the opportunities for space applications of a novel type of solar panel, which is used on solar cars. This way benefitting optimally from cross-pollination between different sectors. The goal of the project is to initiate a new direction in the space solar array sector. The result of this project is the beginning of new research opportunities and further development towards a commercially viable product.

B.J. Salet
Leiden, October 2019

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Acknowledgements

I want to start by thanking Jasper Bouwmeester and Chris Verhoeven for their guidance and knowledge from the academic perspective during the literature study and master thesis. I also want to thank Airbus DS NL for the graduation opportunity. Especially I would like to thank my daily supervisors at Airbus Jan Schutten and Marloes van Put for their valuable, consistent and constructive guidance throughout the project. Additionally, I would like to thank the rest of the SparkWing team and others at Airbus, Mark, Gerwin, Flip, Catarina, Remco, Gideon, Hans, Chris, Gemma, Henk and Dieter, for their creative and positive input and far-reaching expertise throughout the project. Furthermore, I want to thank my friends and colleagues at Wattlab, David, Siebe, Sarah, Kasper, Thijs, Flip, Frank, Joel and Ed. It was an honour to do my master thesis project as a collaboration project between three very different parties, Airbus, TU Delft and Wattlab. I had a great time and considered it a precious learning experience.

Finally, I would like to thank my family, girlfriend, friends and roommates for supporting me during this project. I hope I've not annoyed them too much with my endless tales about the project, they have been great sparring partners.

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Abstract

As the new space movement develops, so comes the request for cheaper space solar panels with a shorter delivery time. This master thesis investigates the possibilities of using a Semi-flexible terrestrial solar panel in a low Earth orbit space environment. A Semi-flexible solar panel is a sandwich of polymer films with the solar cells in between, these layers are laminated to create the end product. This, for space, new process is applicable to all major solar cell types. It enables a wide range of possible designs and could result in a drastically lower price, shorter production time and a lighter and smaller end product. By using system engineering methodology, the risks of Semi-flexible solar panels in space are identified after which the major risks have been tested. The risks investigated in this report are: outgassing, temperature cycling, vacuum UV radiation, charged particle radiation and the stiffness in deployed and stowed position. The results show no red flags for a seven-year 600 km Earth orbit. A transmission degradation of 10% is observed, resulting in lower power output for the selected material. When entering higher orbits, the amount of radiation leads to delamination and potentially no power output. When comparing the Semi-flexible solar panel to a conventional solar panel, a price difference of factor 4 and a power output difference of 33% is expected. The vast price difference shows the potential for the concept, but further investigation is needed to find out if the transmission degradation could be mitigated and whether designs comply with stiffness and vibration requirements.

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Contents

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List of abbreviations, acronyms and symbols

(A-Z)

AR	Anti-Reflective
b	Width
BSR	Back Surface Reflector
CAPEX	Capital Expenditure
CVCM	Collected Volatile Condensable Materials
d	Distance
E	Young's modulus
EB	Electron Beam
ECSS	European Cooperation for Space Standardization
EOL	End Of Life
EQE	External Quantum Efficiency
ESD	Estimated Space Days
ETFE	Ethylene tetrafluoroethylene
EVA	Ethylene-vinyl acetate
F	Force
FEP	Fluorinated ethylene propylene
FM	Flight Model
h	Height
I	Moment of inertia
KO	Kick-off
L	Length
LEO	Low Earth orbit
LNT	Lower Nominal Temperature
PVA	Photovoltaic Assembly
RML	Recovered Mass Loss
SA	Solar Array
TJ	Triple Junction
TML	Total Mass Loss
UNT	Upper Nominal Temperature
UV	Ultraviolet
UV/Vis	Ultraviolet/Visible
VUV	Vacuum Ultraviolet
WVR	Water Vapour Regained
XETFE	Crosslinked Ethylene tetrafluoroethylene

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1 Introduction

Fifty years ago, in 1969, the first human being set foot on the moon. Saturn V took the crew, orbiter and lander on a trip that will never be forgotten. Now, about 50 years later, the world observed Falcon 9 launch. Although with a goal not as spectacular as the Apollo 11 mission, it did change the space game. The launcher produced by SpaceX, showed the world that the high-tech space industry was ready for disruption. By lowering the price of launching a kilogram to low Earth orbit by about a factor 10, it did what nobody expected possible. The new space movement requires faster, cheaper and higher-quality access to space.

When comparing the space solar panel to the ones used on solar cars, it is observed that there is a price difference of a factor 1000 between the two. Additionally, there is a difference in thickness and mass of respectively 45 and 5 times in favour of the ones used on solar cars. On the other hand, space solar panels are more rigid and supply 1.5 times the power. The basic principle of protecting the cells is different, resulting in different properties.

Conventional space solar panels have been based on the same principle for a long time. A carbon aluminium sandwich is used as a base structure and expensive top-notch solar cells are placed on top protected by a cover glass. A variation of deployment methods has been used, but the basic principle stayed the same. This report dives into the opportunities for a new type of solar panel for space applications. The project is a collaboration between Delft University of Technology, Airbus Defence and Space and Wattlab. The expertise in research, space solar panels and Semi-flexible solar panels used on solar cars combined serve as the foundation for the project.

The goal of this master thesis project is to find out if the Semi-flexible solar panel concept has the potential to be used in space. Therefore the main research question is:

- *Is it possible to design a solar panel for small satellites in low Earth orbit based on the Semi-flexible solar panel concept?*

To find the answer to this question, it is subdivided into the following questions:

- *What are the requirements for small satellites in low Earth orbit?*
- *What is the difference between a conventional solar panel and a Semi-flexible solar panel?*
- *What solar array designs are possible with the Semi-flexible solar panel concept?*
- *What are the major risk factors of the Semi-flexible solar panel?*
- *Do those risk factors result in a "no-go" for the Semi-flexible solar panel concept?*

The project is based on the spiral model by Barry W. Boehm [2]. This model focusses on risk reduction and an iterative way of working; during this master thesis, one full iteration is performed. The cycle consists of four phases: determine the objectives, identify and resolve the risk, development and test and finally plan the next iteration.

In this report, this first phase consists of the mission and stakeholder analysis (chapter 0). To find the differences between the types of solar panels, a system functional analysis is performed (chapter 3), after which the requirements are determined (chapter 4).

The second phase is the risk identification, based on the requirements a brainstorm is performed where concepts are generated. Of those concepts, the most promising concept is chosen (chapter 0). Based on the concept, the risks are identified. To mitigate the risks, tests are selected and their procedure is described (chapter 0).

In the third phase, testing is done to evaluate the risks of which the results are presented (chapter 7). Finally, in fourth phase, an evaluation (chapter 8 and chapter 9) is done in the form of a comparison and conclusion. To end phase four, a plan is made for the next iteration as future work (chapter 10).

2 Mission & Stakeholder analysis

In order to get a feeling of the parties involved when making a space solar panel, a mission and stakeholder analysis is performed. This chapter starts by presenting the mission and need statement in section 2.1. Secondly, the stakeholders are identified and visualised in a stakeholder web in section 2.2. Finally the customer needs are listed in section 2.3.

2.1 *Mission and need statement*

Need statement

The space industry is developing into a more commercial market where the conventional space solar panel is no longer a solution for the demand. In order to be able to compete in this market lower cost and faster to customer solar arrays are needed.

Mission statement

The mission is to develop a product line to supply customers in the small satellite market with a cost effective & fast delivery solution for solar arrays subsystem to enter a new market segment for Airbus.

2.2 *Stakeholder identification*

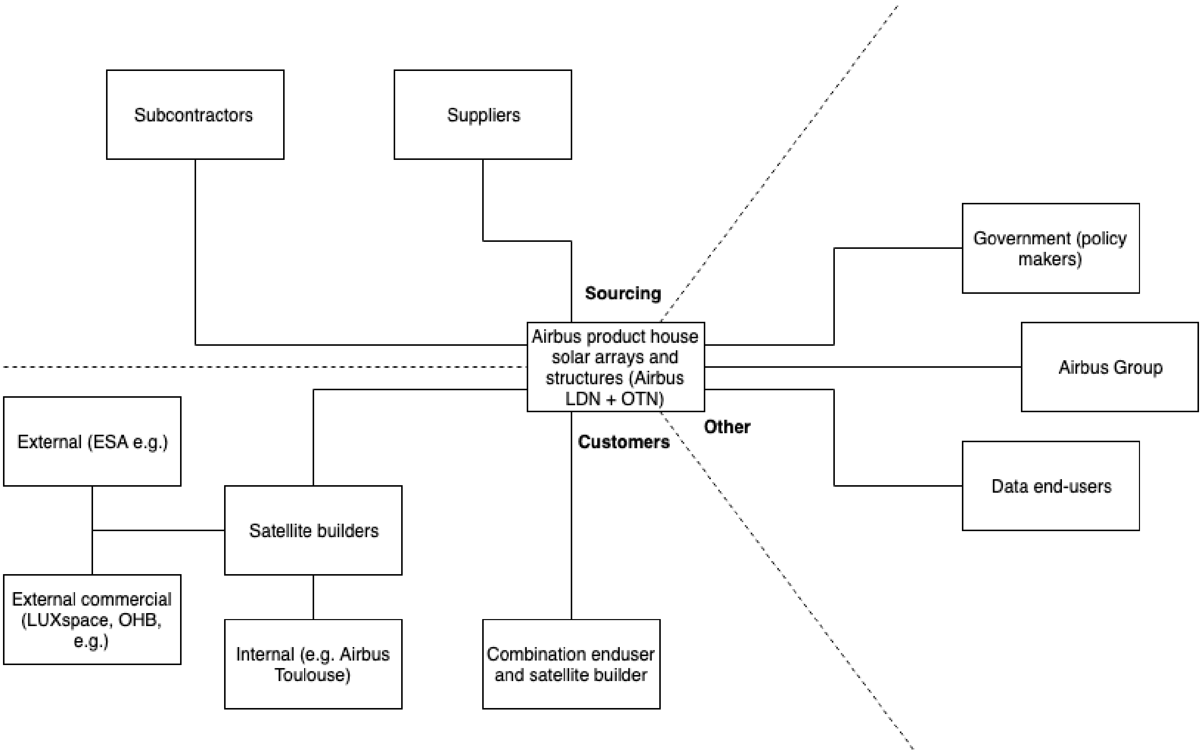

Figure 2.1: Stakeholder web

In order to give an overview of the involved stakeholders, a stakeholder web is shown in Figure 2.1. This web has Airbus in the middle and distinguishes three categories; sourcing, customers and other. The **sourcing** category is sub-divided into subcontractors and suppliers. Here the subcontractors are third parties that provide a service or are hired to perform a project specific task. Examples of these could be Delft University for radiation testing at the nuclear reactor, supply a custom designed substrate or assemble the photovoltaic assembly (PVA) on the substrate according to design. On the other hand there are the Suppliers. These are companies that deliver a product, which is then used by airbus in order to produce the solar panels.

The **customers** category is sub-divided into parties that build and sell satellites and parties that sell a service and use and build satellites in order to be able to provide this service. This distinction is made because the cost budgeting and risk acceptance could differ between the two types of customers. Finally the **other** category covers the rest of the stakeholders that directly or indirectly interact with the solar panels but are not involved as a customer or supplier in any form. These include but are not limited to: the policy making side of the government (the government could also function as a customer), the Airbus Group and data end-users.

2.3 *Customer needs*

Airbus DS NL customers are third parties that build satellites. The customer needs are listed in Table 2.1.

Table 2.1: Customer needs

NEED-ID	**Need**	**Rationale**
NED-CST-01	The price per watt for the complete solar array shall be lower than XXX[1] USD	Based on customer interviews performed by the Airbus SparkWing team
NED-PLN-01	The solar array shall be ready for installation within XXX months from the kick-off meeting	Based on customer interviews performed by the Airbus SparkWing team
NED-QLT-01	The solar array shall have a life cycle of at least 7 years in LEO	Based on customer interviews performed by the Airbus SparkWing team
NED-QLT-02	The solar array shall not harm the spacecraft	ECSS standards (ECSS-q-70-5.1.5)
NED-QLT-03	The solar array shall have a lower mass to power ratio than 20 g/W	Based on current state of the art Airbus space grade solar arrays
NED-QLT-04	The solar array shall have a lower volume to power ratio than 60 cm^3/W	Based on current state of the art Airbus space grade solar arrays
NED-QLT-05	The solar array shall supply the required power in the range of 200-1000 Watts	Based on customer interviews performed by the Airbus SparkWing team, majority of small satellites fit within this range

[1] Due to confidentiality reasons all requirements regarding cost and planning have been replaced by XXX

3 System functional analysis

This master thesis project focusses on demonstrating whether the Semi-flexible solar panel concept is feasible for use in low Earth orbit (LEO). To get a basic understanding about what this Semi-flexible concept is, and what the functions are a solar panel has to serve, a system functional analysis is done in this chapter. First a brief analysis of space, terrestrial and Semi-flexible solar panels is done in section 3.1. Then in section 3.2 the system functions are broken down into a tree to get a clear overview of the different functions of a space solar array. Finally in section 0 a function flow diagram is shown to illustrate how the different functions of the solar array interact with each other. The chapter will conclude with an overview of the Semi-flexible concept that will be used for concept generation, risk analysis and testing in section 3.4.

3.1 *Solar Panel System*

This section will cover an analysis of the current solar panels used both in space (section 3.1.1) and on Earth (section 3.1.2 and 3.1.3). This is done to show the differences between the three, which will be compared in section 3.1.4. Below a schematic visualization of the different types of solar panels is shown in Figure 3.1.

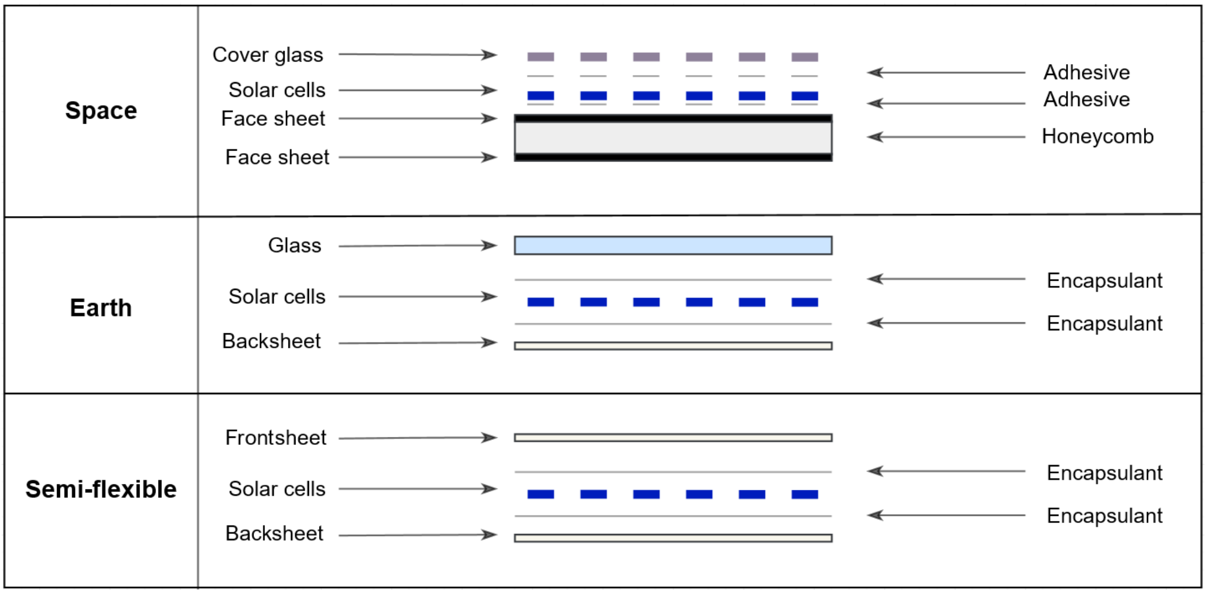

Figure 3.1: Schematic visualization of different types of solar panels

3.1.1 Space solar panel

Most of the solar panels used in space have consisted of the same composition for decades. Bottom up it consists of the following layers: carbon fiber face sheet, aluminium honeycomb structure (usually around 20 mm thick), carbon sheet, isolation sheet, the PVA and finally cover glasses on top of each solar cell. An example of such a panel is shown in Figure 3.2.

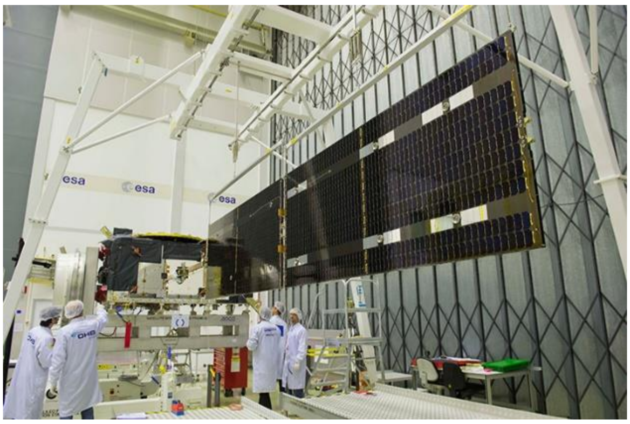

Figure 3.2: Space solar panel [30]

The substrate consists of a carbon sandwich with the aluminium structure in between that serves the purpose of creating surface area and generate the needed stiffness and strength for the panel. The Kapton sheet is used to electrically isolate the solar panels from the carbon sheet because the carbon sheet is conductive. The solar cells are used to generate the energy and are covered with cover glass to protect the cells from space radiation. Without the cover glass the solar cells would degrade as much in a few days as with a cover glass in 15 years [3].

3.1.2 Terrestrial solar panel

Terrestrial solar panels can differ a lot in appearance, but in general all terrestrial panels consist of the same sequence of layers. Changing the layers gives different properties to the panel of which examples are the thickness, mass, efficiency etcetera. The most common used combination of layers from bottom up is: plastic backsheet, encapsulant, the PVA, another layer of encapsulant and finally a glass layer on top.

Figure 3.3: Earth solar panel [29]

For the terrestrial solar panels, the solar cells are usually covered with glass to protect the PVA from weather conditions. As the backside of the solar panel is exposed to less harsh conditions since it is facing down, usually a plastic layer is used to seal it as it is cheaper and less heavy compared to glass. Finally in order to be able to mount the solar panels, an aluminium frame is constructed around the panel.

3.1.3 Semi-flexible panel

The Semi-flexible solar panel concept is a variation on the Earth based solar panel. The basic idea is the same as with a conventional solar panel, the solar cells are protected by encapsulating them in between sheet like materials. The big difference in this case is that the glass top layer is replaced by a flexible, thin sheet. Material examples for this different top layer are PET, ETFE and FEP. This results in a solar panel that is about 15 times less heavy (1 kg/m^2), 20 times as thick (1 mm) compared to a standard terrestrial solar panel and “Semi-flexible” with a radius of curvature of around 30 cm. The thickness of the adhesive and top/backsheet layers can be changed to adjust structural properties of the solar panel.

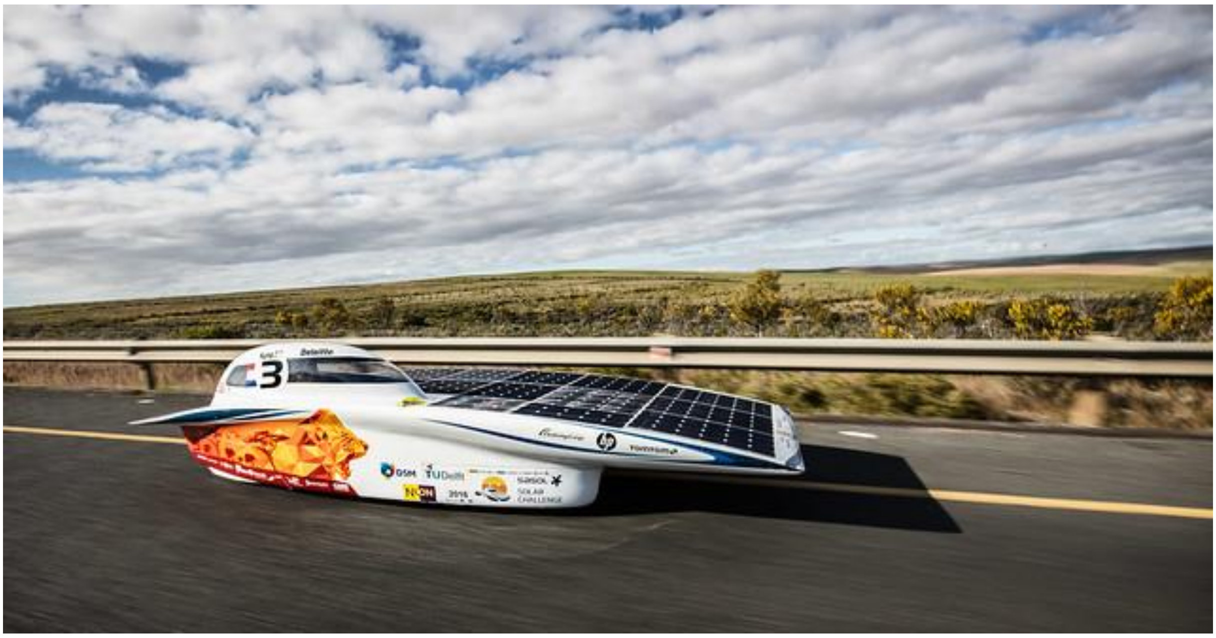

Figure 3.4: Picture of Nuna 8S, a solar powered car using Semi-flexible solar panels [4]

3.1.4 Solar panel functional comparison

The major difference between the solar panels is that the space solar panel is basically an “open” structure. Open in this case meaning that the fragile things are locally protected, but the whole PVA is not sealed from air or water. The PVA is placed on top of a carbon sandwich and finally topped off with the cover glass on each cell. On Earth this would not work as rain could easily short-circuit the cells, but since there is no atmosphere in space, this is not an issue. The result of this is that the PVA for solar panels on Earth is fully encapsulated in between a layer of plastic and a layer of glass and therefore sealed off from any moisture.

Another large difference between the solar panels is that the space solar panels are produced in small numbers with extremely high precision and have a custom design. On the other hand, the common terrestrial solar panels are mass produced with the lowest cost price as a target. This also leads to the difference in the type of solar cells used. On Earth the most common type of solar cell used is the silicon solar cell whereas in space this is usually a triple junction (TJ) gallium arsenide based cell. The efficiency of the gallium cells could be almost 1.5 times as high as a common silicon cell, but the price could be up to 300 times higher per watt.

The major differences between the space, terrestrial and Semi-flexible solar panels are displayed in Table 3.1. Note that the comparison is made between a complete space solar array, including hinges and hold down systems and, on the other hand just a regular terrestrial panel. The purpose of this overview is to give an indication of the order of magnitude. The comparison is made based on the panel structures shown in Figure 3.1. Due to confidentiality reasons prices are relative and cannot be specified in more detail.

Table 3.1: Comparison of different types of solar panels

	Structure	Type of cells	Price [relative]	Mass [kg/m2]	Thickness [mm]	Goal
Space solar array	Open	Triple junction Gallium Arsenide	100[2]	4.5	20	Maximum power per size/mass
Terrestrial solar panel	Closed	Silicon	0.11[3]	15	40	Lowest cost per watt
Semi-flexible panel	Closed	Silicon	0.22[4]	1	1	Low mass, high efficiency, thin, flexible

3.2 *System functional breakdown*

This section breaks down the functions of a space solar array. Figure 3.5 shows the functional breakdown diagram. The solar array is first divided into the three main building blocks: the substrate, the PVA and the electrical and mechanical mechanisms. For each different building block, the functions are listed.

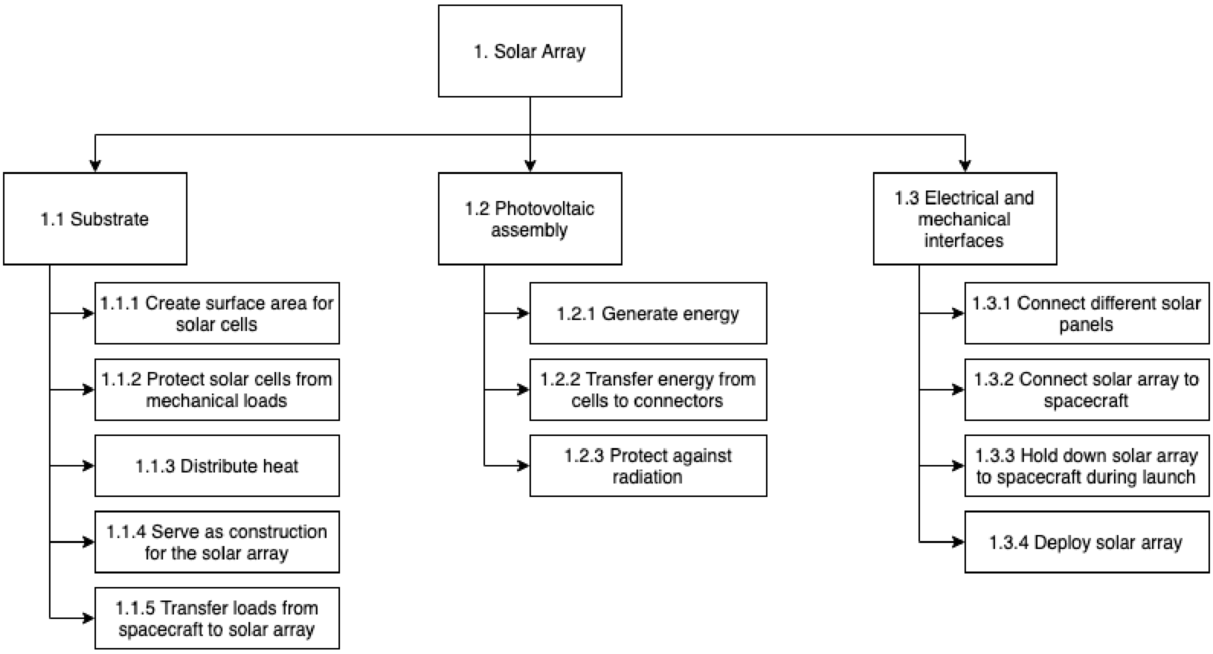

Figure 3.5: Solar array system functional breakdown diagram

[2] Price from recent Airbus projects
[3] Price from recent Wattlab projects
[4] Price from recent Wattlab projects

3.3 *System functional flow diagram*

In order to visualise how the different functions of the solar array interact with each other and with the environment a N2 is shown in Figure 3.6. In this case the environment it interacts with consists of Space, the Sun and the spacecraft. The arrows indicate the direction of interaction.

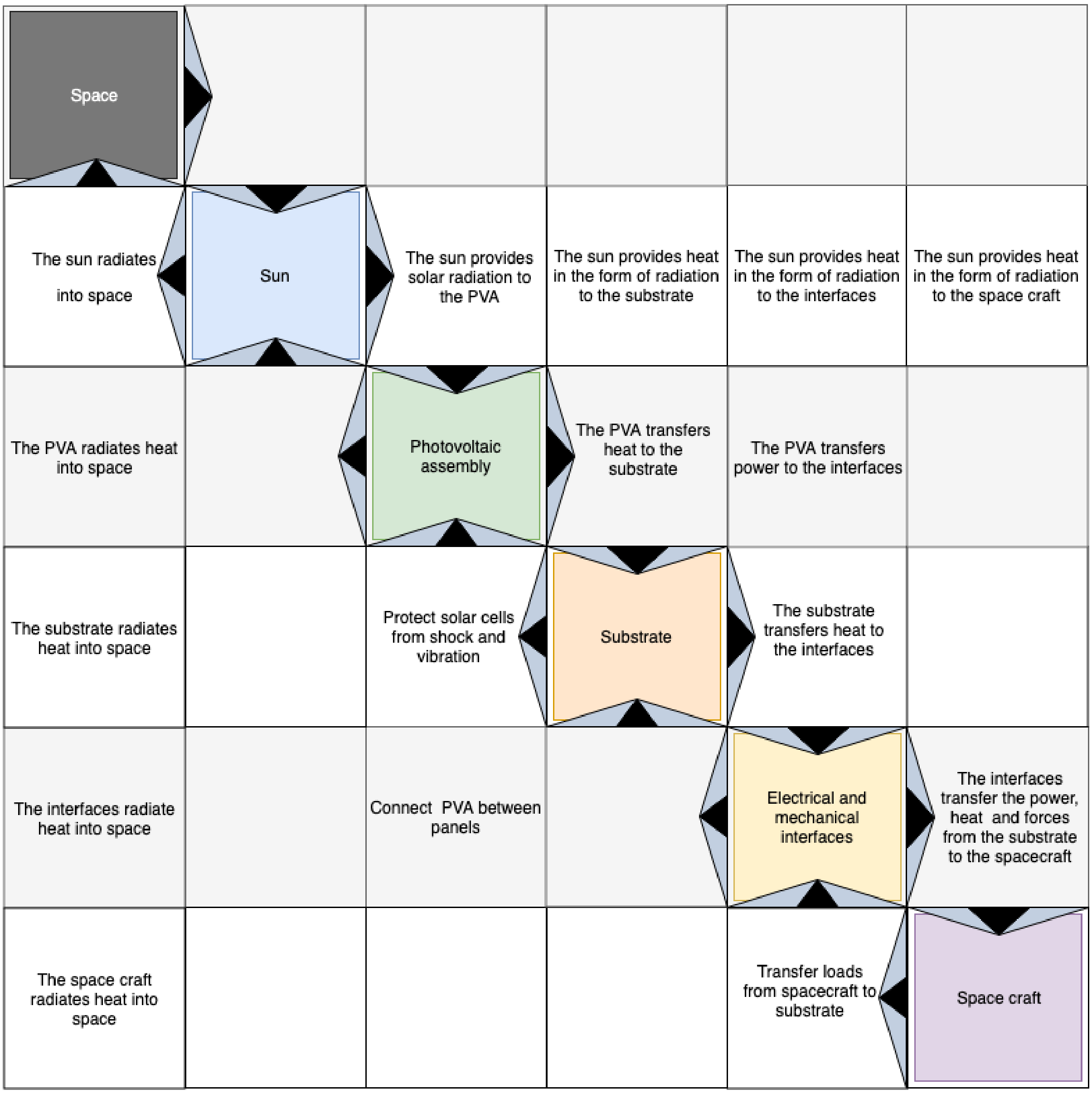

Figure 3.6: N2 diagram

3.4 *Conclusion*

This chapter starts by looking at the three main different types of solar panels and compares them in Table 3.1. This table does not tell the whole story as the Semi-flexible concept nor the terrestrial solar panel are ready for use in space, but it does show that the Semi-flexible concept has some significant advantages regarding the price, mass and thickness when compared to the space solar panel. Based on those advantages it is interesting to find out to what extend the Semi-flexible solar panel concept could be used in space. This master thesis focusses on finding the limitations and challenges of using this Semi-flexible concept for LEO applications. Since this concept is all about the encapsulation of the solar cells and can be used in different types of designs, a single type of solar cell is not chosen in this study yet. Focus will be on the total structure of the composition of materials and degradation of the layers in between the cell and the Sun (frontsheet and encapsulation layer, see Figure 3.1). The Semi-flexible concept can be equipped with all common types of solar cells including silicon based, gallium based triple junction and thin film cells. Because of this flexibility in cell usage the degradation of the cells is considered outside the scope of this research and will be listed in Chapter 10 as further research.

4 System requirement analysis

This chapter covers the system requirement analysis. First a requirement discovery tree is made after which the separate requirements are listed in section 4.1. Furthermore section 4.2 shows the house of quality which is used to show what design parameters have the most influence on the customer needs. In section 4.3 the design parameters that have most impact will be classified as the key drivers for the design of the solar array. The relationship between the customer needs from Table 2.1 and the requirement from section 4.1 can be found in Appendix A.

4.1 *Requirement discovery*

For the requirement discovery process a requirement discovery tree is made as shown in Appendix B. The requirement discovery tree is used to list the requirements. The list is separated into the program, industrialization, mechanical, electrical and environmental requirements. This list contains the relevant requirements and is therefore not complete.

4.1.1 Program requirements

Table 4.1: Program requirements

REQ-ID	TREE-ID	Requirement
SS-PRG-MKT-01	1.1.1.1	The program shall be able to deliver a first flight model (FM) for a commercial mission in XXX.
SS-PRG-MKT-02	1.1.1.2	The program shall be able to deliver 10 FMs for commercial missions in XXX.
SS-PRG-MKT-03	1.1.1.3	The program shall be able to deliver 40 FMs for commercial/institutional missions to be produced from XXX onwards.
SS-PRG-CST-01	1.1.1.4	The maximum cost price per flight model shall be lower than XXX $/W for order quantifies 1-10 FMs.
SS-PRG-CST-02	1.1.1.4	The maximum cost price per flight model shall be lower than XXX $/W for order quantifies >10 FMs.
SS-PRG-CST-03	1.1.1.5	Recurring cost targets shall be met by XXX.
SS-PRG-SBC-01	1.1.1.6	All flight hardware shall be free of export control regulations.
SS-PRG-BDG-01	1.1.1.7	The investment for research & development (design, development and qualification of both product technology and production line) shall not exceed XXX M€.

4.1.2 Industrialization requirements

Table 4.2: Industrialization requirements

REQ-ID	TREE-ID	Requirement
SS-IND-HLT-01	1.1.2.1	For every customer, the first (batch of) FM(s) shall be delivered within 6 months after project kick-off (KO), assuming these customer orders have a pre-defined, recurring SparkWing configuration. The 6 months' timeframe does not apply to custom designs.
SS-IND-BDG-02	1.1.2.2	The CAPEX investment for the set-up of a production line shall not exceed XXX M€.

4.1.3 Mechanical requirements

Table 4.3: Mechanical requirements

REQ-ID	TREE-ID	Requirement
SS-FNC-MCH-01	1.2.1.1	The system mass distribution shall be below 4.5 kg/m^2
SS-FNC-MCH-02	1.2.1.2	The first natural frequency shall be larger than 50Hz in stowed conditions
SS-SYS-MCH-03	1.2.1.3	The PVA shall have a power ratio of 20 g/W. This includes a suitable cover-glass or alternative radiation protective solution to fulfil life time requirements in the specified orbital environment
SS-SYS-MCH-04	1.2.1	The solar array shall not have a volume to power ratio higher than 60 cm^3/W.
SS-SYS-MCH-05	1.2.1	The first natural frequency shall be larger than 2 Hz in deployed conditions.

4.1.4 Electrical requirements

Table 4.4: Electrical requirements

REQ-ID	TREE-ID	Requirement
SS-FNC-ELE-01	1.2.2.1	The solar array (SA) shall provide the required power end of life (EOL) determined by the customer.

4.1.5 Environmental requirements

Table 4.5: Environmental requirements

REQ-ID	TREE-ID	Requirement
SS-FNC-ENV-01	1.2.3.1	The product life time shall be at least 7 years in LEO (300-1500 km) (operational life)
SS-FNC-ENV-02	1.2.3.2	The solar array hardware shall be able to perform under the following temperature range: Upper nominal temperature: • UNT = 115 ºC Lower nominal temperature: • LNT = -128 ºC
SS-FNC-ENV-03	1.2.3.1	The solar array shall be suitable for a minimum of 40000 thermal cycles.
SS-FNC-ENV-04	1.2.3.3	After an outgassing test is performed, the Recovered Mass Loss (RML) shall be <1% (percentage mass).
SS-SYS-ENV-05	1.2.3.4	After an outgassing test is performed, the collected volatile condensed material (CVCM) shall be < 0.1% (percentage mass)

4.2 *Design parameter analysis*

The design parameter analysis is performed to find what design parameters satisfy the customer needs the most. This gives insight into what to focus on when developing the concept in the next chapter. The House of Quality method is used to perform this analysis and is shown below in Figure 4.1. The weight factors of the customer needs have been determined by interviewing the Program manager, System Engineer and Project Controller of the SparkWing project team [5] [6] [7].

Relationships		**Weight**
Strong	●	9
Medium	○	3
Weak	▽	1

Direction of Improvement	
Maximize	/\
Minimize	\/

				Design parameters									
			Direction of Improvement	/\	\/	\/	/\	\/	\/	\/	\/	/\	/\
Trace	Relative Weight	Customer Importance	Customer Requirements	Solar cells performance	Cost	Mass	Expected life	Panel thickness	Surface area	Stowed volume	Production time	Power output	Mechanical performance
NED-CST-01	32%	35	Price per watt	●	●	▽	○	-	●	▽	▽	▽	○
NED-PLN-01	23%	25	Delivery time	▽	▽	-	▽	-	-	-	●	▽	▽
NED-QLT-02	18%	20	Launch and deployment reliability	○	●	●	-	●	○	○	-	-	●
NED-QLT-04	18%	20	Stowed volume	○	-	▽	-	●	●	●	-	○	○
NED-QLT-03	9%	10	Mass	▽	○	●	▽	○	●	▽	-	○	▽
			Importance Rating Sum (Importance x Relationship)	427	500	295	127	355	586	259	236	136	345
			Relative Weight	13%	15%	9%	4%	11%	18%	8%	7%	4%	11%

Figure 4.1: House of Quality for small satellite space solar panels

4.3 *Key drivers*

From the house of quality in section 4.3 the key drivers for the concept design process can be derived. The three design parameters with the highest influence on customer needs will be discussed below and will be considered the key drivers.

The **Surface Area (18%)** is the design parameter with the highest influence. This is the area that can be used to place the solar cells. This design parameter has a large influence on the cost of the solar panel since cheaper cells could be used if it is affordable to create a larger surface.

The **Cost (15%)** is the second most influential design driver. This is mainly the result of the price per watt being a decision driver for the customers. This cost is defined as the price per watt in space and therefore includes both the labour and the material costs as well as the launch cost due to the mass.

The **Solar cell performance (13%)** is named as the final key driver. The solar cell performance is determined by, but not limited to, the solar cell efficiency, mass, size, price and structure.

The key drivers will be used as a focus when deciding what concepts to pursue in the brainstorm phase.

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5 Conceptual Design

This chapter covers the process from the concept generation to the critical factors of the selected concepts. Based on the material properties of the Semi-flexible solar panel, section 5.1 will describe the process of concept generation. The pre-selected results of the first section will be discussed more thoroughly in section 5.2, after which a trade-off will be made using a Pugh matrix and a trade-off table in section 5.3.

5.1 *Concept generation and selection*

The concept generation process is performed by first doing a concept brainstorm (5.1.1), after which the output of the brainstorm will be displayed in a design option tree and visualised as strawman concepts in section 5.1.1, those concepts will finally be pre-selected in sub-section 5.1.2.

5.1.1 Concept generation brainstorm

A concept generation brainstorm was held at the Airbus DS facility in Leiden, a description of the brainstorm process can be found in Appendix C. The goal of this brainstorm was: "How can you, based on the Semi-flexible solar panel concept, make something go from as small as possible volume to an as large as possible surface?". The end product of the brainstorm was a concept to base the further research on. Figure 5.1 shows the design option tree for the Semi-flexible deployable solar panel concept. The concepts are visualised as strawman figures in Figure 5.2.

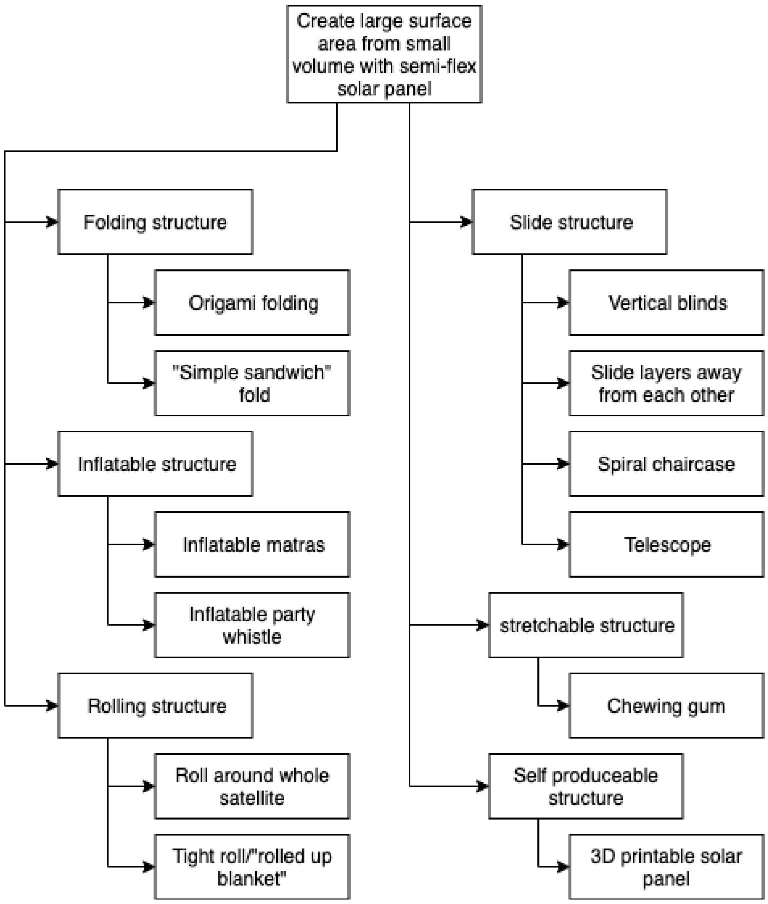

Figure 5.1: Design option tree

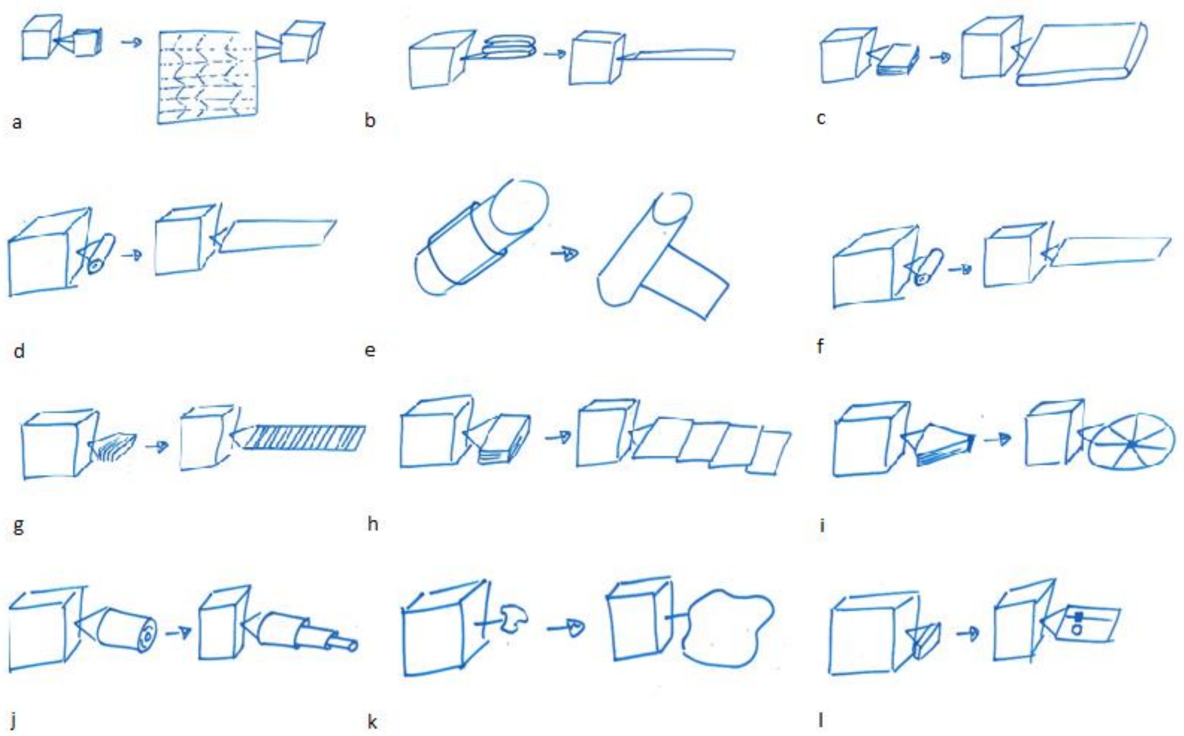

Figure 5.2: Strawman figures Semi-flexible concept

Names of drawings in Figure 5.2: a. Origami fold, b. Simple Sandwich fold, c. Inflatable matrass, d. Inflatable party whistle, e. Roll around whole satellite, f. Tight roll/Rollup blanket, g. Vertical blinds, h. Slide layers away from each other, i. Spiral chair case, j. Telescope k. Chewing gum, i. 3D printable solar panel

5.1.2 Pre-selection

The pre-selection is done in order to be able to perform a more thorough trade-off in section 5.3 on only four concepts. The pre-selection is based on the key drivers for the design discussed in section 4.3 which were:

- Cost
- Solar cell performance
- Available surface vs stowed volume

The reason of keeping in mind the key drivers while choosing the concepts to further develop is that these key drivers should have the largest effect on the selection criteria of the customer. Therefore focussing on those key drivers will have the largest benefit in the end. The selection process was executed by giving everyone present 3 points to score the different options with the key drivers in mind. Everyone could use a maximum of one point per concept and the concepts that scored the most points in this session where selected to move forward with.

Taking into account the key drivers, the following concepts are chosen after the brainstorm:

1. Pre-loaded leaf spring with vertical blind system
2. Slideable flower system
3. Wrap around spacecraft system
4. S-folding blanket

5.2 *Design options*

The four design options resulting from the previous section will be discussed in more detail below.

5.2.1 Vertical blinds system

This concept is a combination of two principles, it uses the pre-loaded leaf spring principle as a deployment mechanism for the vertical blind solar array system. In this case two leaf springs are used opposing each other which are pre-loaded before launch. When in stowed condition, the solar array is folded like a vertical blind system and when deploying the spring pulls on the blinds to span them over an area.

Table 5.1: Benefits and drawbacks Vertical blinds system concept

	Stowed position	**Deployment mechanism**	**Deployed position**
Benefit	Compact in volume	Leaf springs are a proven custom of the shelve option	Design could be stiff enough
Drawback	Bulky in shape (not flat)	The deployment process is uncontrolled	Bifacial solar cells is not an option in this case as the back is blocked by the leaf springs

5.2.2 Slideable flower system

This concept is based on five solar panels with the same dimensions stacked upon each other. Each solar panel has a hinge at one corner around which it can spin. When in orbit, the panels start to spin and lock into position after a 90 degree turn. In deployed position the array look like a four-leaf clover. This configuration can extend its deployed area up to five times when deployed. The benefit of this concept is that the array consists of separate solar panels of the same size which makes assembly line production more realistic.

Table 5.2: Benefits and drawbacks Slideable flower concept

	Stowed position	**Deployment mechanism**	**Deployed position**
Benefit	Compact	Could be passively deployed	Low moment of inertia because of 2 axis symmetrical design
Drawback	New type of hold down mechanism needed	Complex hinge and lock mechanism	Not possible to point solar array towards Sun

5.2.3 Wrap around spacecraft system

The key idea of the wrap around spacecraft system is to make a solar array that is flexible enough to be folded all around the spacecraft. This would work best with cylindrical spacecraft but the design could also be adopted for cubical spacecraft. In the latter case it must have a heterogeneous structure to be able to bend around tight corners. Which leads to a custom design per spacecraft and therefore to a longer delivery time. The deployment mechanism is based on centrifugal force, this way there is no active deployment mechanism needed. A down side of this concept is that when the solar array is installed the whole spacecraft will be covered and therefore unreachable for any adjustments.

Table 5.3: Benefits and drawbacks Wrap around spacecraft system concept

	Stowed position	Deployment mechanism	Deployed position
Benefit	Little added volume	Only release mechanism but no further actuators needed	Stable due to centrifugal forces (in case of spinning spacecraft)
Drawback	Design of solar array is extremely dependant on spacecraft design	Not sure if the stresses in stowed position will harm the solar cells	Stiffness when deployed could be too low

5.2.4 S-folding blanket

The S-folding blanket concept is based on the conventional “simple sandwich” concept which has been used a lot in space. It consists of multiple areas with constant dimensions connected to each other and when in stowed position zig-zag folded together. The large difference between the conventional “simple sandwich” principle and the S-folding blanket is that the S-folding blanked is made as one part with an inconsistent combination of layers. At the places where the array has to be folded there will be less material to make it more flexible and a tape spring could be laminated into the layers to function as deployment mechanism and to bring stiffness on the bended areas. During launch the multiple layers of the solar array will be pulled against the spacecraft which will be released when in orbit so it can deploy itself.

Table 5.4: Benefits and drawbacks S-folded blanket system concept

	Stowed position	Deployment mechanism	Deployed position
Benefit	Compact and flexible design	Only release mechanism but no further actuators needed	Simple concept so little risk when deployed
Drawback	New type of hold down mechanism needed	Uncontrolled self-regulating deployment process	Stiffness when deployed could be too low

5.3 *Selection*

This section will cover the selection process from the four concepts discussed in section 5.2 to finally one concept to proceed with. This final concept is later used to do first calculations on concept level and perform comparison with the state of the art conventional solar arrays. To perform the selection first the selection criteria are listed in section 5.3.1. Those criteria will be used to perform the trade-off in section 5.3.2. Two separate methods where used, the Pugh matrix and the graphical trade-off table.

5.3.1 Selection criteria

The selection criteria are based upon the customer needs listed in section 2.3 and are discussed below. As not every criterion is equally important, every criterion will be given a weight factor in a range from 1-5 and is in line with the weight factors of the House of quality from section 4.2.

The **Cost (weight factor: 5)** is the purchase price of the solar array. As the volumes of satellites go up in the "new space" and the budgets decrease, the cost gets more and more relevant. Additionally the amount of private (non-governmental) stakeholders is increasing in the space market and those companies need to make a profit which will result in a tighter budget. Since the budget in new space is a lot lower and decisions between suppliers are driven by cost this criterion receives a weight factor of 5.

The **Delivery time (weight factor: 4)** is the time it takes between project kick-off and delivery. As mentioned above, the space market is getting filled with private companies who sometimes compete the service for which they need satellites. If the delivery time is too long, a competitor could be there first and claim market share. Therefore the delivery time is also a strong criterion and receives a weight factor of 4.

The **Reliability (weight factor: 3)** is an indicator of how sure the customer can be that the solar array will work as requested. Since in new space it is more common to launch constellations of multiple satellites it is less reliant on a single satellite than with conventional research missions based only on one satellite. In case one solar array wouldn't function out of a 1000 satellite constellation, this could be an acceptable loss if the price is right. Therefore this criterion receives a weight factor of 3.

The **Mass (weight factor: 2)** is the mass of the solar array. As the price of the launch depends on the total mass of the solar array this is of influence in the design choice. Since it is less critical than the reliability and has rather low impact on the cost it receives a weight factor of 2.

The **Volume (weight factor: 3)** is determined by the volume of the solar array in stowed position. Together with the mass of the solar array this determines what influence it will have on the launch costs of the satellite. Different than the mass there are restrictions on the volume to be able to fit in the launcher. It receives a weight factor of 3.

5.3.2 Trade-off process

Two trade-off methods are used to find which of the four options is will score the best on the criteria listed in 5.3.1. First a Pugh matrix, shown in **Error! Reference source not found.** nd second a Graphical trade-off table, shown in **Error! Reference source not found.**. Both clearly show that the S-folding blanket has the most potential. From here on the S-folding blanket will be used as the main concept idea for the Semi-flexible solar panel concept.

Table 5.5: Pugh matrix

Criteria	**Weight factor**	**Vertical blind**	**Slideable flower**	**Wrap around spacecraft**	**S-folding blanket**
Cost	5	-1	-1	1	1
Delivery time	4	0	1	-1	0
Reliability	3	0	-1	-1	0
Mass	2	0	0	1	1
Volume	3	0	1	1	1
	∑(+)	0	2	3	3
	∑(0)	4	1	0	2
	∑(-)	1	2	2	0
	Results	7	1	3	17

Legend		**Weight factor**	
1	Positive impact on system	1	Least important
0	no impact on system	5	most important
-1	negative impact on system		

Table 5.6: Graphical trade-off table

Criteria	Vertical blind	Slideable flower	Wrap around spacecraft	S-folding blanket
Cost	**(R)** Complex design and assembly, therefore higher costs	**(R)** Complex design and assembly, therefore higher costs	**(G)** Simple design, little additional interfaces needed, low cost	**(G)** Simple design, little additional interfaces needed, low cost
Delivery time	**(Y)** Medium delivery time, because of assembly of different parts	**(G)** Low delivery time, because of standard sizes possible	**(R)** Long delivery time because of strong dependency on spacecraft	**(G)** Low delivery time, because of standard sizes possible
Reliability	**(Y)** Medium reliability, many moving parts but proven concept	**(R)** Low reliability, many moving parts	**(R)** Low reliability, depends strongly on spacecraft	**(Y)** Medium reliability, simple design but not proven yet
Mass	**(Y)** Medium mass because of extra mechanism	**(B)** Low mass because of little extra interfaces	**(G)** Extremely low mass because of no mechanisms	**(G)** Extremely low mass because of no mechanisms
Volume	**(B)** Good because of bulky shape but low volume	**(G)** Excellent because of low volume	**(G)** Excellent because of low volume	**(G)** Excellent because of low volume

Colour	Judgement
(green)	Excellent, exceeds requirements
(blue)	Good, meets requirements
(yellow)	Correctable deficiencies
(red)	Unacceptable

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6 Test selection and procedure

This chapter first investigates the risks for the Semi-flexible solar panel based on the s-folded blanked concept in section 6.1. Next, based on the most critical factors, experiments are explained by stating a hypothesis, elaborate on the experiment setup and samples used in section 6.2.

6.1 *Test selection*

To prioritise the test selection, a risk assessment is performed. This way the most critical factors can be determined and a prioritized test plan is set up. This starts by analysing what risks are present and ranking them in terms of likelihood and impact as done in section 6.1.1. The critical factors for this concept will be listed in section 6.1.2 and are used to determine what tests will be performed to get a step closer to a feasible design.

6.1.1 Risk Analysis

Table 6.1 shows the risks concerning the s-folded blanket concept based on the Semi-flexible concept. Each risk is given an ID which is later referred to in the risk map. For each risk the effect is described and the risks are rated with a Likelihood and an Impact factor. Both on a scale from 10-50 where 10 means low and 50 means high. The risks are partially (R1-R11) based on the ECSS-Q-ST-70C document [8], the remainder is concept related (R12-R14). The likelihood and impact factors have been determined both on literature research as well as by interviews with experts at Airbus. Experts with the following functions have been interviewed: Functional Specialist in the Materials & Processes department [9], Functional Specialist in the Product engineering department [10], Design Leader [6] and Project Manager [5] from the SparkWing team and Team Leader Testing [11] at Airbus DS.

Table 6.1: Risk listing for S-folded blanket concept

ID	**Risk**	**Effect**	**L**	**I**
R1	**Temperature** in orbit gets too high or too low	Structural integrity is effected by temperature so construction fails or electrical properties of solar array don't function as a result of temperature.	40	50
R2	**Temperature cycles** wear down the solar array	Construction will fail because of fatigue as a result of thermal cycling according to requirements.	40	50
R3	Solar array is not resistant against **vacuum** environment	Material properties change and don't function anymore and/or because of outgassing instruments could be contaminated.	40	50
R4	Solar array catches fire **(flammability)**	Because of high temperatures and/or electrical properties the solar array could catch fire (while still on Earth).	10	50
R5	Solar array degrades because of UV **radiation**	Radiation degrades the material in between the Sun and the solar cell and solar array will produce zero power. Crosslinked ETFE has shown to degrade in space [12], this is a similar material as used in the Semi-flexible solar panel.	50	50

R6	Solar array degrades because of **particle radiation**	Radiation degrades the material in between the Sun and the solar cell, as a result the solar array will not produce power. Possible delamination could occur.	50	50
R7	**Electrical charge and discharge** appears on surface	Local malfunctions of the solar array.	30	10
R8	Solar array will malfunction due to **Corrosion** degradation	Solar array construction will degrade.	10	40
R9	Solar array will break due to **Stress corrosion**	Solar array construction will fail.	10	40
R10	**Atomic oxygen** breaks down solar array	Structure is consistently broken down until solar array stops functioning at all. A study by NASA has shown that both FEP and ETFE are well resistant against atomic oxygen [13].	10	50
R11	**Micrometeorites** and debris crash into solar array	Local or complete malfunction of solar array due to high velocity impact of millimetre to centimetre particles [14]. Likelihood and impact might increase as a result of higher amounts of debris in space.	50	30
R12	**Stiffness** of solar array is too low	The spacecraft is not able to position itself according to requirements. Dynamic interference with Attitude and orbit control system could result from low stiffness.	50	40
R13	**Vibrations** during launch break solar array	Solar cells fracture before it enters orbit.	50	50
R14	**Deployment system** not working	Solar array will only provide little to no amount of energy.	40	50

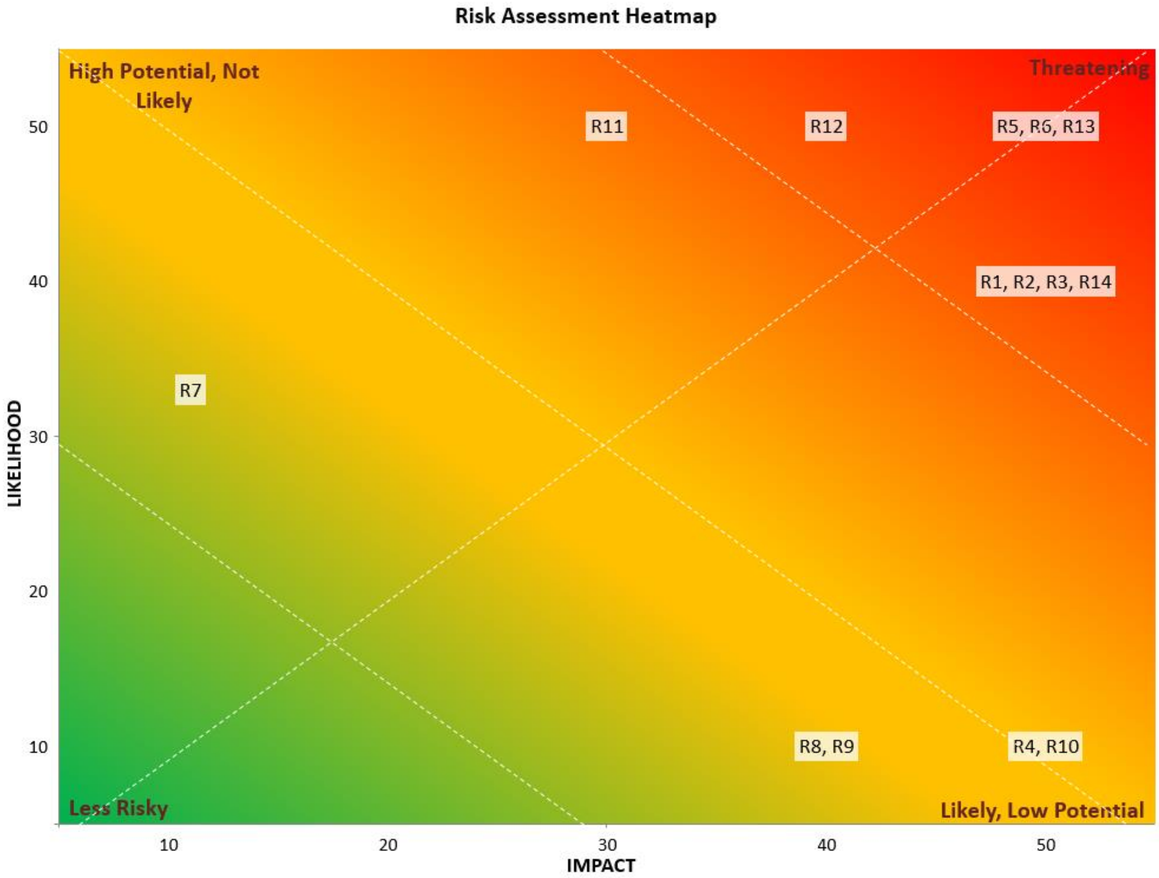


Figure 6.1: Risk map for S-folded blanked concept

6.1.2 Critical factors and test selection

From Figure 6.1 in section 6.1.1, it can be observed that the most threatening risks are:

1. UV Radiation (R5)
2. Particle radiation (R6)
3. Vibration (R13)
4. Stiffness (R12)
5. Temperature (R1)
6. Temperature cycle (R2)
7. Vacuum (R3)
8. The deployment system (R14)
9. Micrometeorites and space debris (R11)

Resulting from the list of critical factors four tests and a structural analysis are selected to investigate if the risks form a treat to the solar array. Risks 11 and 14 are considered outside of the scope of this project. The four tests that are selected are:

- Electron beam test (R6)
- Vacuum UV transmission degradation test (R5)
- Thermal cycling test (R1, R2)
- Outgassing test (R3)

Additionally a first structural analysis (R12, R13) will be performed to find out if the stiffness and vibration requirements will be met.

6.2 *Test procedures*

This chapter describes the experiments chosen in section 6.1.2. First the test sample production process and materials are described and a complete list of the different sample compositions is given in section 6.2.1. Than in section 6.2.2 to section 6.2.6, each of the experiments is elaborated upon. This is each time done by first stating the hypothesis, than describing the test and elaborate on the experimental setup and finally listing the samples for each specific test.

6.2.1 Test samples

The samples used in the experiments performed are all produced using a lamination process at the Wattlab production facility. Although the workshop is thoroughly cleaned it is not a certified cleanroom. The materials used are taken from rolls and the solar cells are supplied both by Airbus (the space grade cells) and Wattlab (terrestrial cells). This section will give an overview of the production process, the materials used in the samples and finally a full overview of all the samples used in the experiments.

6.2.1.1 Production process

The production of the samples starts by cutting the sheets to the desired sizes. The sheets and solar cells are stacked in an up-side-down order on an aluminium plate. The samples follow a curing process using a 3S hybrid system solar panel laminator. The curing process consist of combination of temperature and pressure changes**.** After the curing process, the solar modules are trimmed using a box cutter. An image made during the stacking process is shown in Figure 6.2.

Figure 6.2: Sample production process

6.2.1.2 Sample materials

As illustrated in Figure 3.1, the Semi-flexible solar panel consists of a frontsheet, backsheet, solar cells and adhesive layers. For the frontsheet two different materials are used, Ethylene tetrafluoroethylene (ETFE) and Fluorinated ethylene propylene (FEP). The ETFE material used is transluX EC 100 from Feron [15], this 100 micron thin film is a high endurance, high transmission film widely used for terrestrial purposes. The FEP material is Type-C gauge 500 from Chemours [16], this 125 micron film has a slightly higher transmission than ETFE but is about double the price and therefore not commonly used for terrestrial purposes. It has been tested by NASA and proved its transmission not to degrade in LEO space environment after four years [13]. Both ETFE and FEP are Teflon products and are therefore treated on one side to make it suitable for the lamination process.

As an adhesive, PHOTOCAP 15580P from STRE is used, this is a 200 micron ethylene vinyl acetate copolymer (EVA) film [17]. The backsheet used is a 100 micron CPX1000 film from Feron [18].

Finally three different types of solar cells are used in the test samples. The main reason is to find out if the different types of solar cells will influence the adhesion of the layers. The solar cells used in the process are Sunpower Gen 3 terrestrial silicon cells [19], old ASEC BSR space grade silicon cells and Azure space triple junction gallium arsenide based cells which where declined from the Galileo mission. These cells are chosen because together they represent almost all different types of rigid solar cells and include the most common ones used in space.

6.2.1.3 Sample list

Table 6.2 shows an overview of the sample labels and the corresponding composition of all of the samples used in the experiments in this chapter. The table indicates the layers of material used where the left side of the table corresponds with the Sunny side of the sample and the right side with the dark side. For example, the test sample “SMP-CYC-TRE” is a sample with the following layers from Sunny side to dark side: ETFE, EVA, Triple junction space cell, EVA, EVA, backsheet. The sample labels all correspond to the ones used throughout the rest of this document.

Table 6.2: Test sample list

Label	FEP	ETFE	EVA	Terrestrial silicon cell	Space silicon cell	junction cell	EVA	Terrestrial silicon cell	EVA	ETFE	Backsheet
SMP-VUV-ETF											
SMP-VUV-FEP											
SMP-EBT-TJF											
SMP-EBT-SIF											
SMP-EBT-TRE											
SMP-EBT-TRF											
SMP-EBT-RFE											
SMP-EBT-RFF											
SMP-CYC-TRE											
SMP-CYC-SIE											
SMP-CYC-SUE											
SMP-CYC-BFE											
SMP-OGT-ETF											
SMP-OGT-FEP											
SMP-OGT-BFM											
SMP-STR-SUN											

6.2.2 Electron beam test

Hypothesis 1: If the Semi-flexible solar panel is exposed to an equivalent dose of particle radiation for an orbit of 1000 km and lifetime of 7 years, it will not delaminate.

Hypothesis 2: If the Semi-flexible solar panel is exposed to an equivalent dose of particle radiation for an orbit of 1000 km and lifetime of 7 years, the layers between the solar cell and the Sun will not lose more than 10% transmission.

To find out what the effect is of charged particle radiation on the solar panel an electron beam (EB) test will be performed. The total test doses are based on the SparkWing project requirements. The doses applied to the samples are calculated using Spenvis [20] and are listed in Table 6.3. The overview can be found in Appendix D.

Table 6.3: Experimental doses Electron Beam test

Orbit altitude [km]	Lifetime [year]	Dose [e-/cm^2/s]	Dose [Rad]
400	3.5	3.50E+14	5.10E+3
400	7	7.25E+14	10.57E+3
600	7	1.68E+15	24.49E+3
1000	7	1.03E+16	150.17E+3

6.2.2.1 Experimental setup

The test will be performed at the nuclear reactor of Delft University of Technology using the van de Graaff electron accelerator shown in Figure 6.3 and Figure 6.4 [21]. This type of accelerator is commonly used for space testing purposes and the one in Delft specifically is special for its large exposure area. A limitation of the setup is that the electrons pass all the way through the material. As a result one should calculate the dose of a specific layer and test each layer separately because one layer could block a part of the radiation for the next. In this test case the whole assembly was exposed to the dose calculated for the outer layer, therefore the layers behind the top layer have been overexposed. This has been done since this was the most conservative way of testing delamination of the assembly. Expected is that the results for delamination would therefore be less in space for the given orbits and duration than during this test.

The transmission measurements will be done on a PerkinElmer LAMBDA 950 UV/Vis Spectrophotometer. For the transmission measurements a sample will also be taken which will not be exposed to radiation but functions as a reference sample.

Figure 6.3: Test bed of the van de Graaff electron accelerator

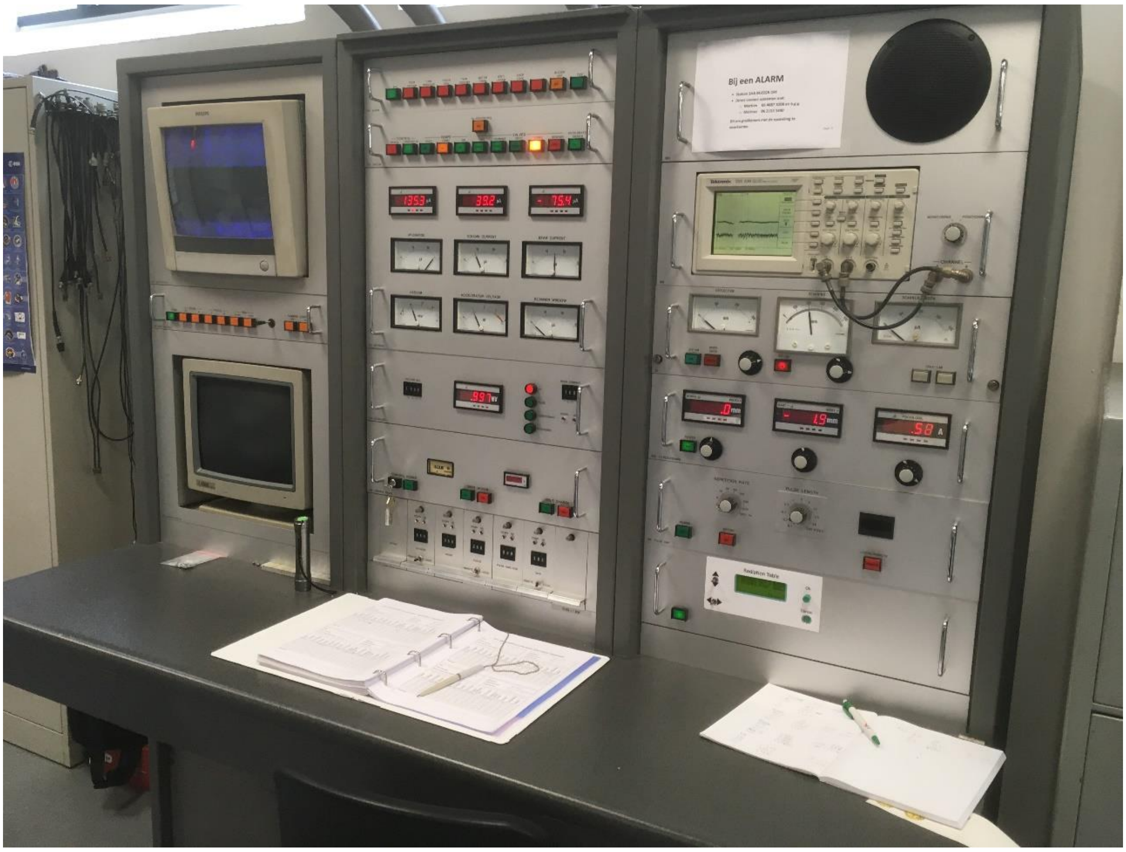


Figure 6.4: Monitoring and setup system for the van de Graaff electron accelerator

6.2.2.2 Test samples

The samples for the EB test are listed in table Table 6.4

Table 6.4: Sample list electron beam test

Sample label	Size [mm]	Dose [e-/cm^2/s]	Dose [Rad]
SMP-EBT-TJE-00	100x50	0	0
SMP-EBT-TJE-01	100x50	7.25E+14	10.57E+3
SMP-EBT-TJE-02	100x50	1.68E+15	24.49E+3
SMP-EBT-TJE-03	100x50	1.03E+16	150.17E+3
SMP-EBT-SIF-00	100x50	0	0
SMP-EBT-SIF-01	100x50	7.25E+14	10.57E+3
SMP-EBT-SIF-02	100x50	1.68E+15	24.49E+3
SMP-EBT-SIF-03	100x50	1.03E+16	150.17E+3
SMP-EBT-TRE-00	40x40	0	0
SMP-EBT-TRE-01	40x40	7.25E+14	10.57E+3
SMP-EBT-TRE-02	40x40	1.68E+15	24.49E+3
SMP-EBT-TRE-03	40x40	1.03E+16	150.17E+3
SMP-EBT-TRF-00	40x40	0	0
SMP-EBT-TRF-01	40x40	7.25E+14	10.57E+3
SMP-EBT-TRF-02	40x40	1.68E+15	24.49E+3
SMP-EBT-TRF-03	40x40	1.03E+16	150.17E+3
SMP-EBT-TRF-04	40x40	3.50E+14	5.10E+3

6.2.3 Vacuum UV transmission degradation test

Hypothesis: If the Semi-flexible solar panel is exposed to a space equivalent dose of vacuum UV radiation the transmission of the layers between the solar cell and the Sun will stabilize after 150 equivalent space days.

The atmosphere functions as a filter of a part of the light the Sun emits. A relevant part of the light which is filtered out is called Vacuum UV (VUV). This is light with a wavelength of 200 to 280 nm. As can be seen in Figure 6.5 the AM0 (space) spectrum contains these wavelengths but the AM1.5 (terrestrial) spectrum does not. This light is known for its darkening or blackening effect on polymers [12].

As this darkening effect can result in a lower transmission of the layers in between the solar cell and the Sun, this will eventually decrease the power output of the solar panel. The degradation of XETFE is found to be stabilizing after 150 equivalent space days (ESD) of exposure to VUV [12]. This VUV degradation test will have the goal of finding out how much transmission is lost as the result of VUV and will simulate 255 ESD. The test will be performed at a constant temperature of 65 degrees Celsius. This temperature is based on the maximum temperatures of a cold orbit when using triple junction cells as shown in Appendix E. As further research the effect of changing this temperature on the transmission degradation is listed in chapter 10.

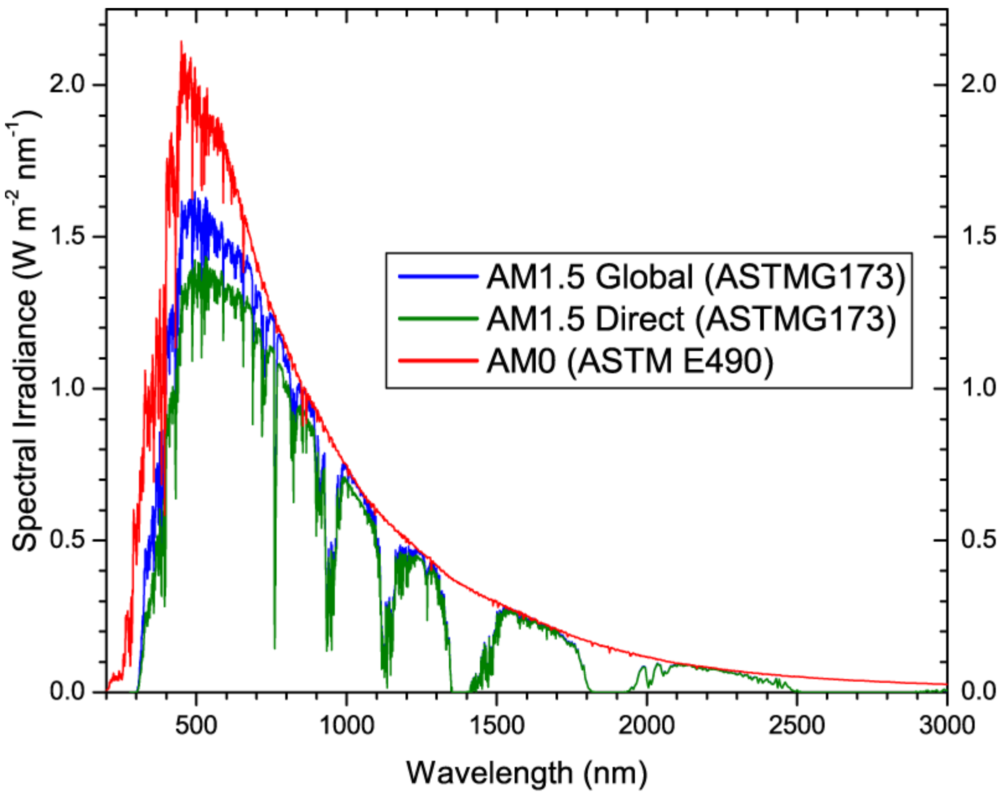


Figure 6.5: Space and terrestrial solar spectra [22]

6.2.3.1 Experimental set-up

The setup of this experiment is shown in Figure 6.6 and schematically visualized in Figure 6.7. The cooling is done by adjusting a fan. Cooling is needed because the radiation source heats up the test samples more than desired. For the temperature monitoring, a FLIR T650sc thermal camera is used. In order to control the amount of light that reaches the sample, a spectrometer is installed at the same distance as the samples from the radiation source. The amount of light reaching the sample each day is equivalent to 10 ESD based on the spectrum analysis. The light is provided using a Dr. Hönle UV 400 H/2 lamp and its spectrum is shown in Appendix F. This lamp has a higher energy in the lower end of the 200-280 nm range compared to the AM0 spectrum. The spectrum is measured before and after the test. For the transmission measurements a sample will also be taken which will not be exposed to VUV but functions as a reference sample. The transmission measurements will be done on a LAMBDA 950 UV/Vis Spectrophotometer.

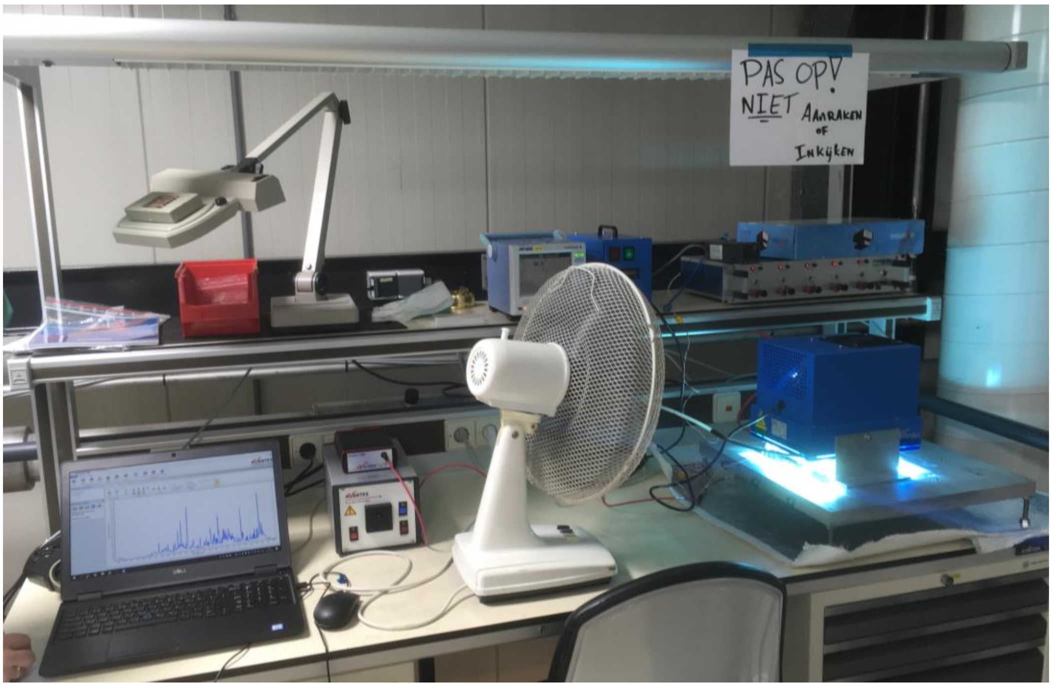


Figure 6.6: VUV degradation experimental setup

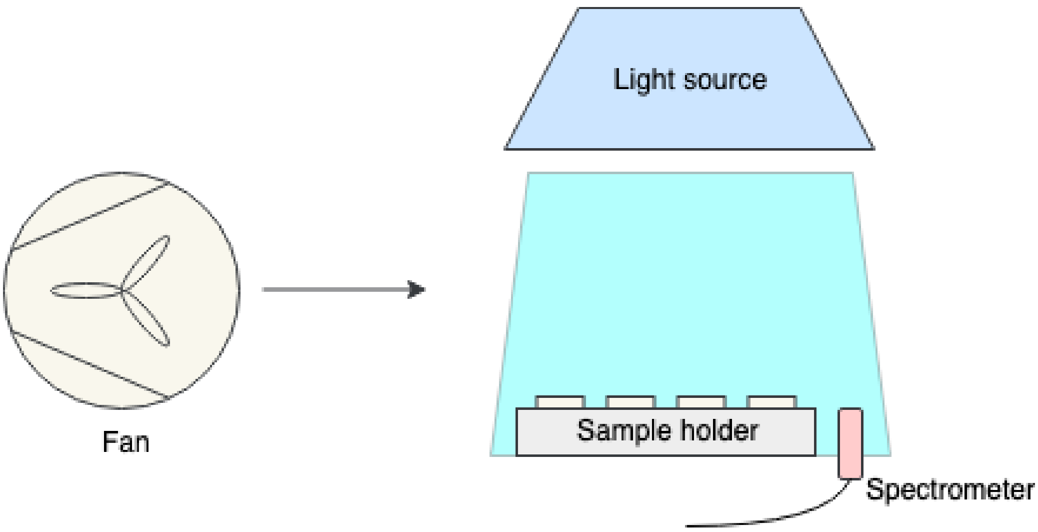


Figure 6.7: Schematic representation of VUV degradation experimental setup

6.2.3.2 Test samples

Two types of samples are tested in two separate sessions of little less than a month. The first sample consists of a 200 micron layer of ETFE and a 200 micron layer of EVA, the second consists of a 200 micron layer of FEP and a 200 micron layer of EVA. 16 test samples of 40 by 40 mm are placed in a matrix. Of those samples, two at a time are taken out and placed in sealed labelled bags with date and time. Each time two samples are taken, samples from opposite sides of the sample matrix are taken. This way inconsistencies in the radiation source can be noted if present. Of each of the sample types one sample will not be exposed to the light and will function as the reference sample.

Table 6.5: UV test samples

Sample label	Size [mm]	Number of samples
SMP-VUV-ETF-00 to SMP-VUV-ETF-16	40x40	17
SMP-VUV-FEP-00 to SMP-VUV-FEP-16	40x40	17

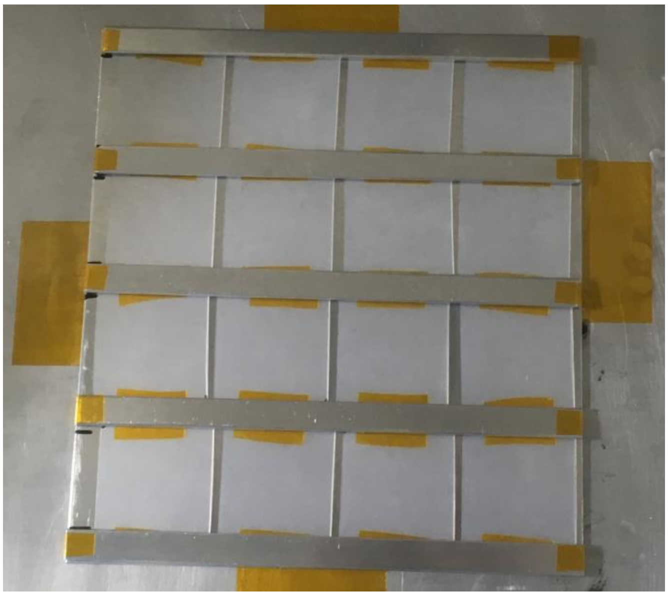


Figure 6.8: VUV degradation test samples in sample holder

6.2.4 Thermal cycling test

Hypothesis: if the Semi-flexible solar panel is exposed to 40000 thermal cycles (7 years lifetime) from -70 to +80 degrees Celsius the solar panel will not delaminate.

To find out if the samples are able to withstand the required temperatures a thermal cycling test is performed. The goal of this test is to find out if the samples construction will not fail due to differences in thermal expansion coefficients or material properties. Failure in this case is defined by delamination or dislocation of the layers. The temperature range chosen for the test is from -70 to +80 degrees Celsius. This range is chosen based on thermal calculations performed for the SparkWing team shown in Appendix E. A temperature range of 150 degrees is common looking at this data so the chosen temperature range gives a good indication for most orbits. In order to find out if other orbits with more extreme temperature reaches are possible, an additional test was performed. During this test 50 cycles with a temperature range of -120 to +120 degrees Celsius was chosen to cover a wider range of orbits from Appendix E**.**

6.2.4.1 Experimental setup

For the test the samples will be exposed to thermal cycles in a Hielkema Testequipment liquid nitrogen thermal cycling facility gas environment as shown in Figure 6.9: Samples in the thermal cycling chamber before cycling. Two thermocouples will be applied to the samples in the cycling facility to check the temperature during the test. To check if the structure has changed during the temperature cycling, the samples will be reviewed by inspection with a microscope.

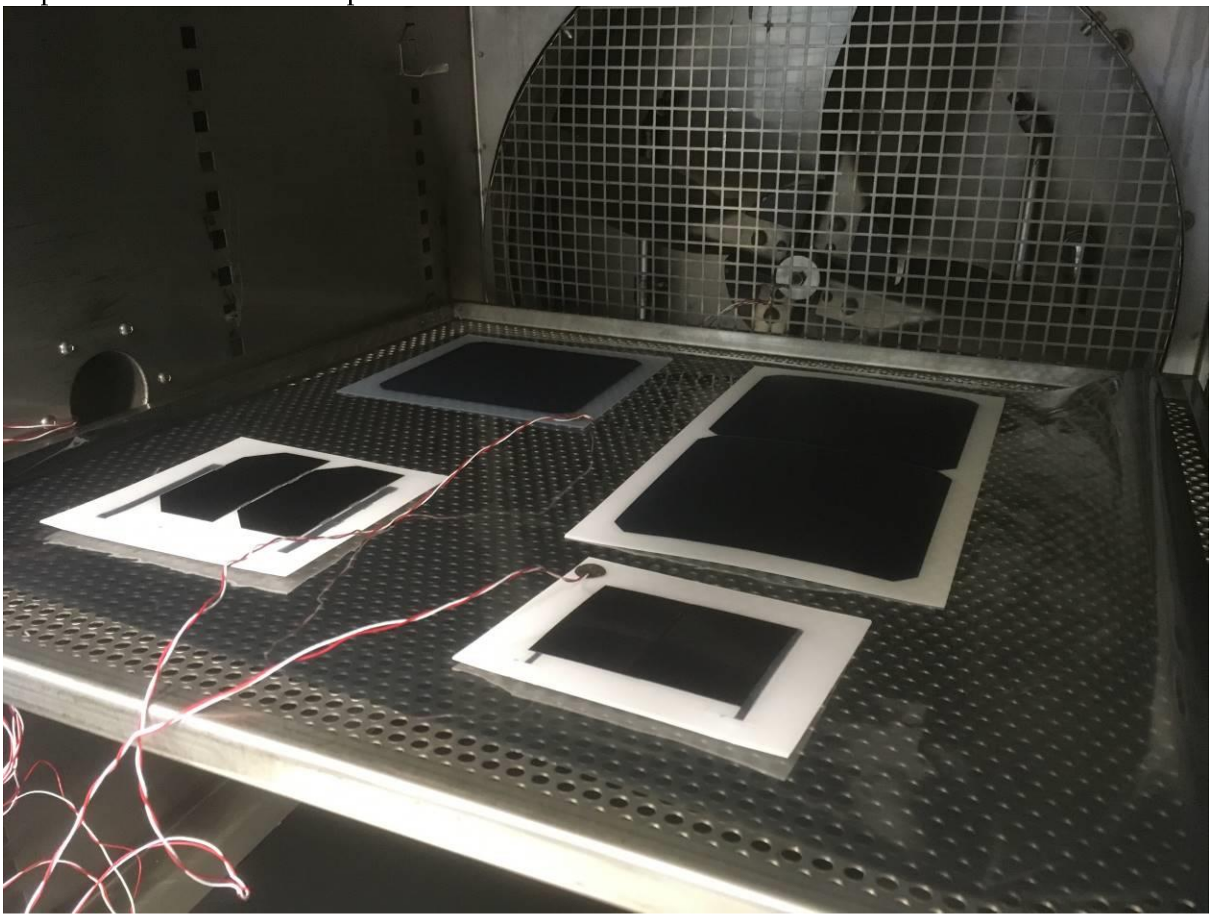


Figure 6.9: Samples in the thermal cycling chamber before cycling

6.2.4.2 Test samples

The samples for the thermal cycling test are listed in Table 6.6.

Table 6.6: Thermal cycling samples

Sample label	Size	Solar cell type	Construction
SMP-CYC-TRE-01	50x60 mm	Triple junction GaAs	Semi-flexible, backsheet material different from frontsheet
SMP-CYC-SIE-01	50x60 mm	Space grade silicon	Semi-flexible, backsheet material different from frontsheet
SMP-CYC-SUE-01	280x180 mm	Terrestrial Integrated back-contact silicon (Sunpower)	Semi-flexible, backsheet material different from frontsheet
SMP-CYC-BFE-01	150x150 mm	Terrestrial Integrated back-contact silicon (Sunpower)	Semi-flexible Bifacial, backsheet and frontsheet are same material (ETFE)

6.2.5 Outgassing test

Hypothesis: If the Semi flexible solar panel is exposed to a vacuum environment of <1 mbar and 125 degrees Celsius for 24 hours, the recovered mass loss (RML) will not be higher than 1%.

To find out if the solar panel will function in a vacuum environment an outgassing test is performed. The requirements state that the RML should not be higher than 1% and the collected volatile condensable material (CVCM) should not be higher than 0.1%. The RML can be calculated with formula 1.

$$RML = TML - WVR \tag{1}$$

Here the total mass loss (TML) is subtracted by the water vapour regained (WVR) to get the RML. The TML is the mass loss directly after outgassing and the WVR is the mass regained after exposure to atmospheric conditions at 22(+- 3) degrees Celsius and 65% relative humidity [23]. The mass change course is shown in figure Figure 6.10.

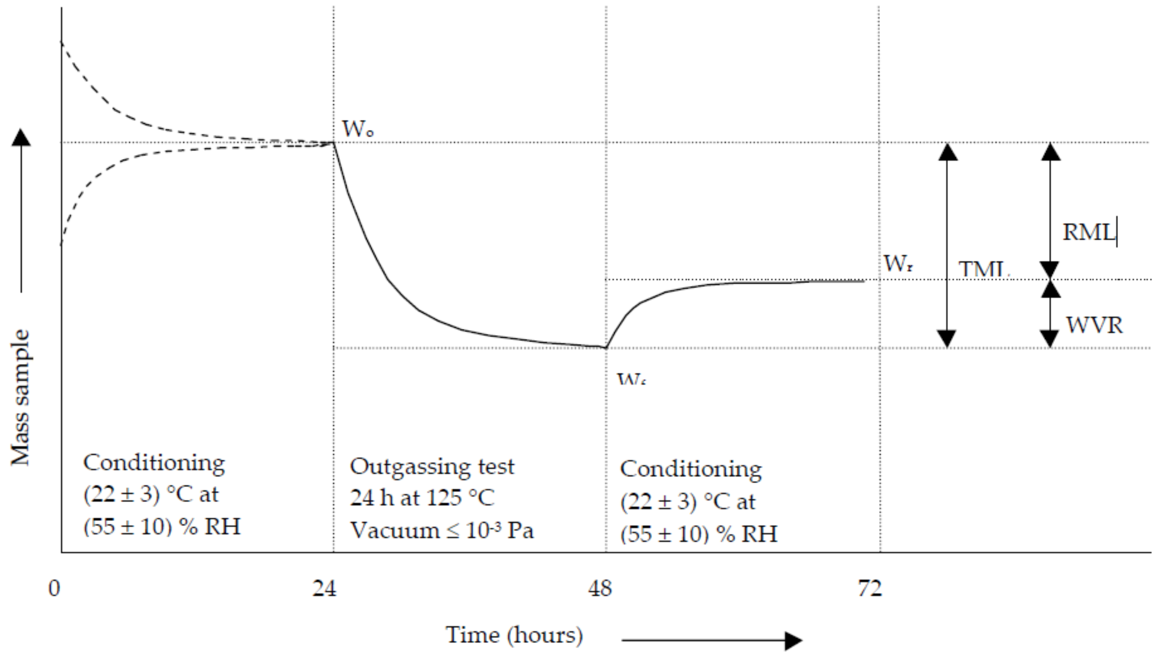


Figure 6.10: Mass change course outgassing test [14]

6.2.5.1 Experimental setup

Before starting the test the samples will be exposed in a cleanroom environment for 24 hours at a temperature of 22(+- 3) degrees Celsius and 55(+- 10) % relative humidity. The samples will be placed in a vacuum oven at a temperature of 125 degrees Celsius and a pressure of 0.37 mbar for 24 hours. When the samples are taken out of the oven they will again be placed in a cleanroom environment as described above. Before placing the samples in the oven and 24 hours after taking the samples out of the oven, the samples will be weighed on a Sartorius Research Balance RC 210P shown in Figure 6.11. In the oven the samples are spread out as much as possible to minimize the effect of outgassing contamination of the samples among each other shown in Figure 6.12.

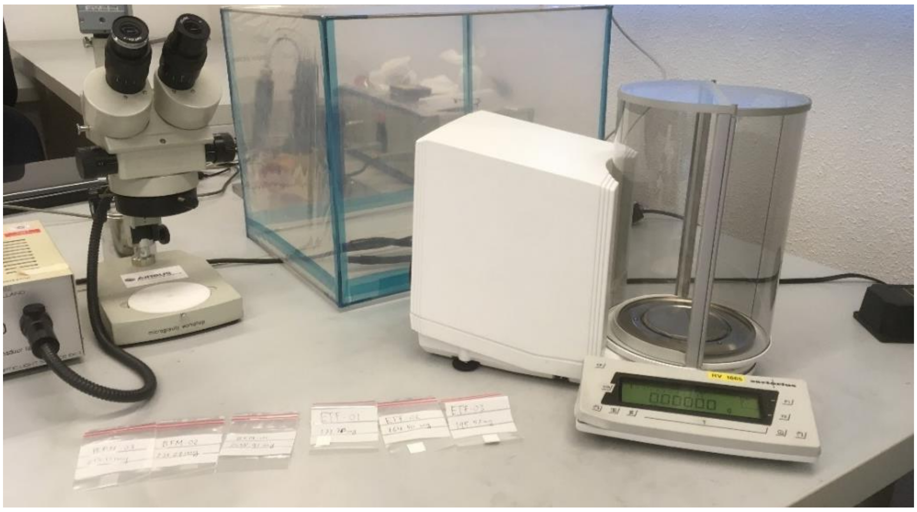


Figure 6.11: Outgassing samples and balance

6.2.5.2 Test samples

The samples will have a mass of 100 – 300 mg per sample and are about 1 cm2. A total of 3 samples will used to validate the results. Cleaning of the samples is done using a microfiber cloth and dry pressure air. The samples are handled while wearing gloves. Along with the samples that will be exposed to vacuum, reference samples are used to weigh at the same moments as the other samples but will not be put in the vacuum oven. This will make sure inconsistencies with the scale should be presented if any. The samples for the outgassing test are listed in Table 6.7.

Table 6.7: Sample list outgassing test

Sample label	Mass [mg]
SMP-OGT-ETF-00	193.76
SMP-OGT-ETF-01	173.73
SMP-OGT-ETF-02	164.50
SMP-OGT-ETF-03	195.57
SMP-OGT-FEP-00	192.19
SMP-OGT-FEP-01	224.46
SMP-OGT-FEP-02	199.57
SMP-OGT-FEP-03	197.04
SMP-OGT-BFM-00	155.00
SMP-OGT-BFM-01	208.91
SMP-OGT-BFM-02	231.29
SMP-OGT-BFM-03	213.11

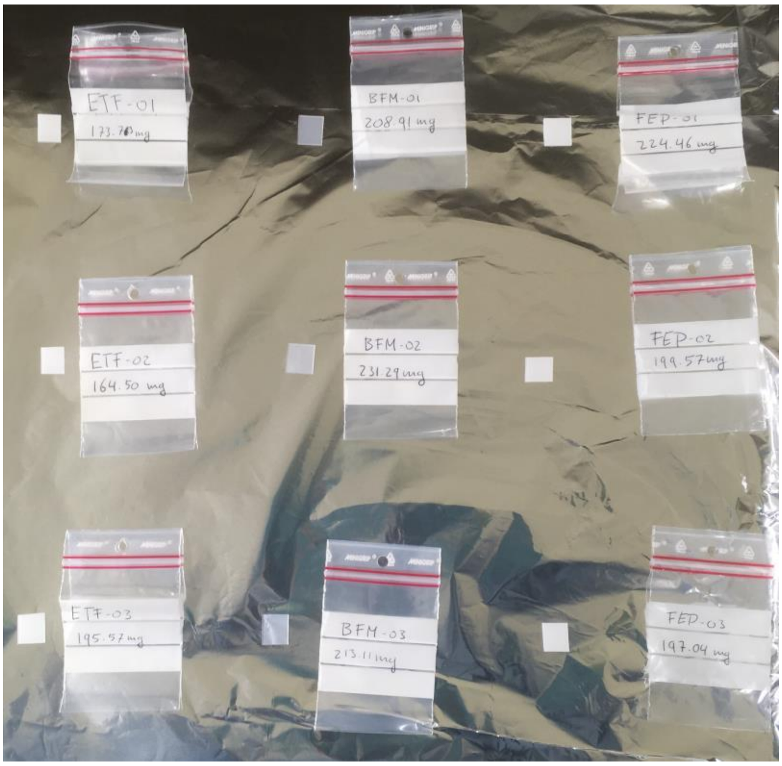


Figure 6.12: Outgassing samples before entering the vacuum oven

6.2.6 Structural analysis test

Hypothesis 1: If a solar panel (1-3 m^2) is made using the Semi-flexible concept it is stiff enough to comply with the deployed stiffness requirement of having a higher first natural frequency of 2 Hz.

Hypothesis 2: If a solar panel ($1m^2$) is made using the Semi-flexible concept it is stiff enough to comply with the stowed stiffness requirement of having a higher first natural frequency of 50 Hz.

To find out if there will be any problems regarding the stiffness and vibration during flight and launch a structural analysis test is performed. This is done by measuring the stiffness of a Semi-flexible module laminated with a 3 Sunpower cell string. Based on this stiffness, a structural analysis is performed using Abaqus.

6.2.6.1 Experimental setup

The stiffness is measured by restraining a sample on one side and applying a force on the far end of the sample, the setup is visualized in Figure 6.13. The sample has a length of 340 mm (L). By applying different forces (F) and registering the corresponding replacements (d) the equivalent Young's modulus (E) can be found using formula 2 & 3. The force was applied using a spring suspension and the displacement was measured using a tape spring. The initial displacement as a result of gravity was set as a baseline.

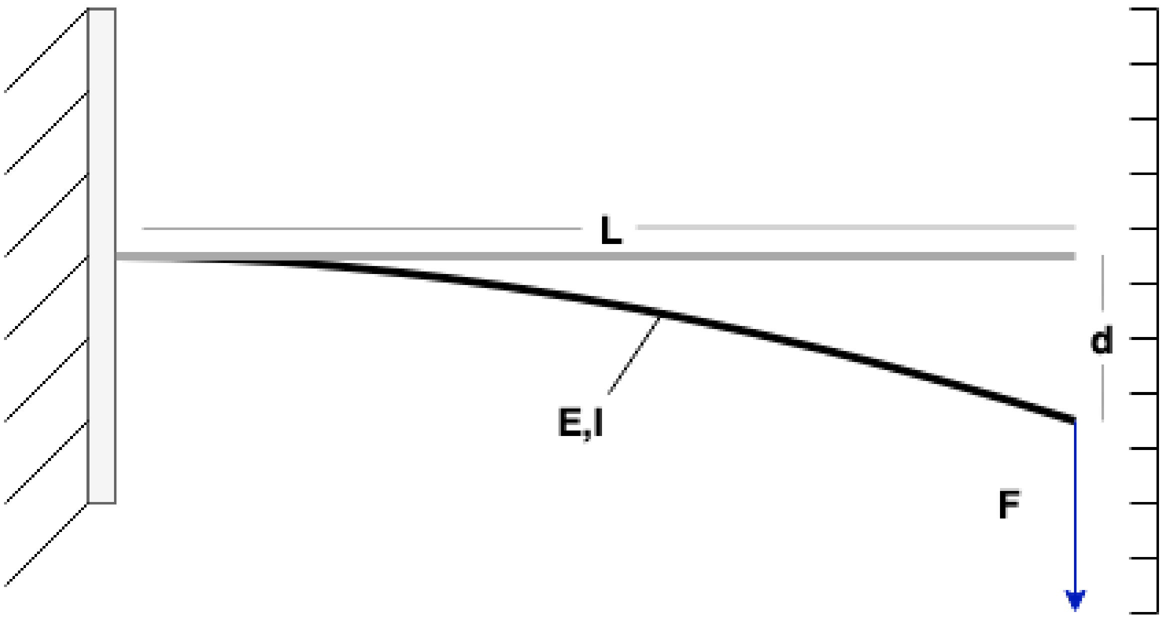


Figure 6.13: Schematic representation of stiffness measurement setup

$$d = \frac{FL^2}{3EI} \tag{2}$$

$$I = \frac{bh^3}{12} \tag{3}$$

6.2.6.2 Test sample

Only one test sample was used for this test, shown in Table 6.8

Table 6.8: Sample list structural analysis test

Sample label	Length [mm]	Width [mm]	Height [mm]
SMP-STR-SUN-01	340	150	1.3

7 Test Results & Analysis

This chapter covers the results and analysis of the experiments described in section 6.2. Each experiment is separately discussed, the results are then later used in chapter 8 as a foundation for the comparison and in chapter 9 and 10 as a base for the conclusion and recommendations.

7.1 *Electron beam test results*

Figure 7.2 shows the transmission of the different samples, AM0 spectrum and the External Quantum Efficiency (EQE) curve of a triple junction GaAs solar cell (same graph but with silicon EQE curve can be found in Appendix G). The largest difference can be found below 500 nm as can be observed in Figure 7.3. Since the rest of the transmission stayed almost equal the drop in transmission is not more than 3.2% for the ETFE based sample and only 1.9% for the FEP based sample.

The effect of the degradation on the expected power output per cell type relative to its original state can be found in Table 7.1. It can be observed that the power output for the 400 and 600 km orbits increased. This is due to the fact that the radiation degrades the material in such a way that the transmission first increases. For the 1000 km orbit a power loss can be observed.

The degradation of the solar cells is not measured in this test. Since to a first-order approximation the shielding effectiveness of a cover is proportional to the mass per area of the cover [24]. The mass per area of the frontsheet and encapsulant layer combined can be compared to the one of a conventional cover glass. A commonly used 0.15 mm cover glass has a mass per area of 0.0335 g/cm^2, the combination of frontsheet and adhesive combined has a mass per area of 0.035 g/cm^2 (ETFE based). Because of this marginal difference in mass per area it will be assumed that no extra cell degradation is caused due to a different type of shielding material. The color degradation can be observed in Figure 7.1. Delamination can also be observed for samples SMP-EBT-TRE-02, SMP-EBT-TRE-03 and SMP-EBT-TRF-03.

Table 7.1: Power output by solar panel relative to BOL in different orbits and lifetimes due to particle radiation

Orbital altitude	**Lifetime**	**Relative power output**			
-	-	ETFE		FEP	
-	-	TJ	Silicon	TJ	Silicon
0	**0**	100	100	100	100
400	**3.5**	-	-	101	101
400	**7**	101	102	102	103
600	**7**	101	102	102	103
1000	**7**	92	90	96	95

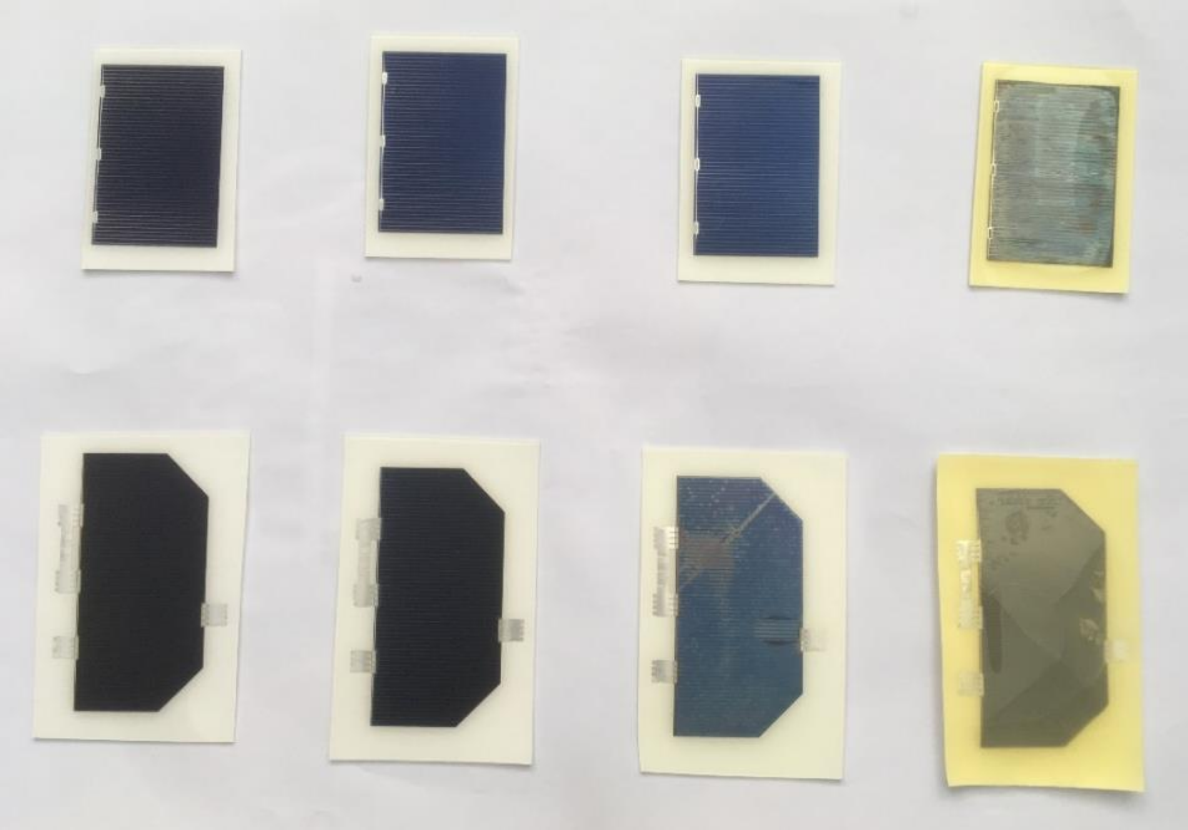


Figure 7.1: EB test samples after exposure

(Top, left to right: SMP-EBT-SIF-00 to SMP-EBT-SIF-03, Bottom, left to right: SMP-EBT-TJE-00 to SMP-EBT-TJE-03)

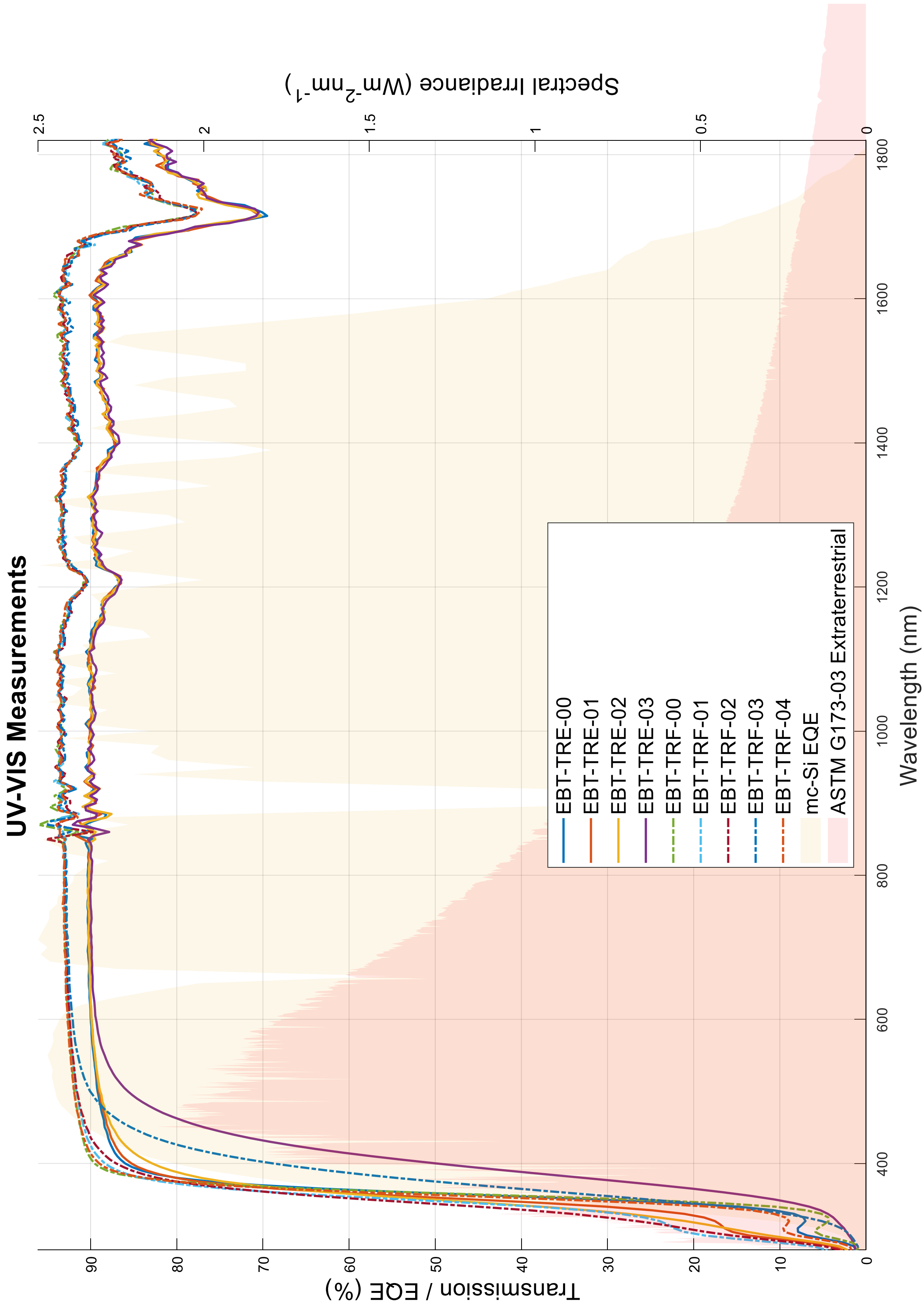


Figure 7.2: Transmission measurements EB Test

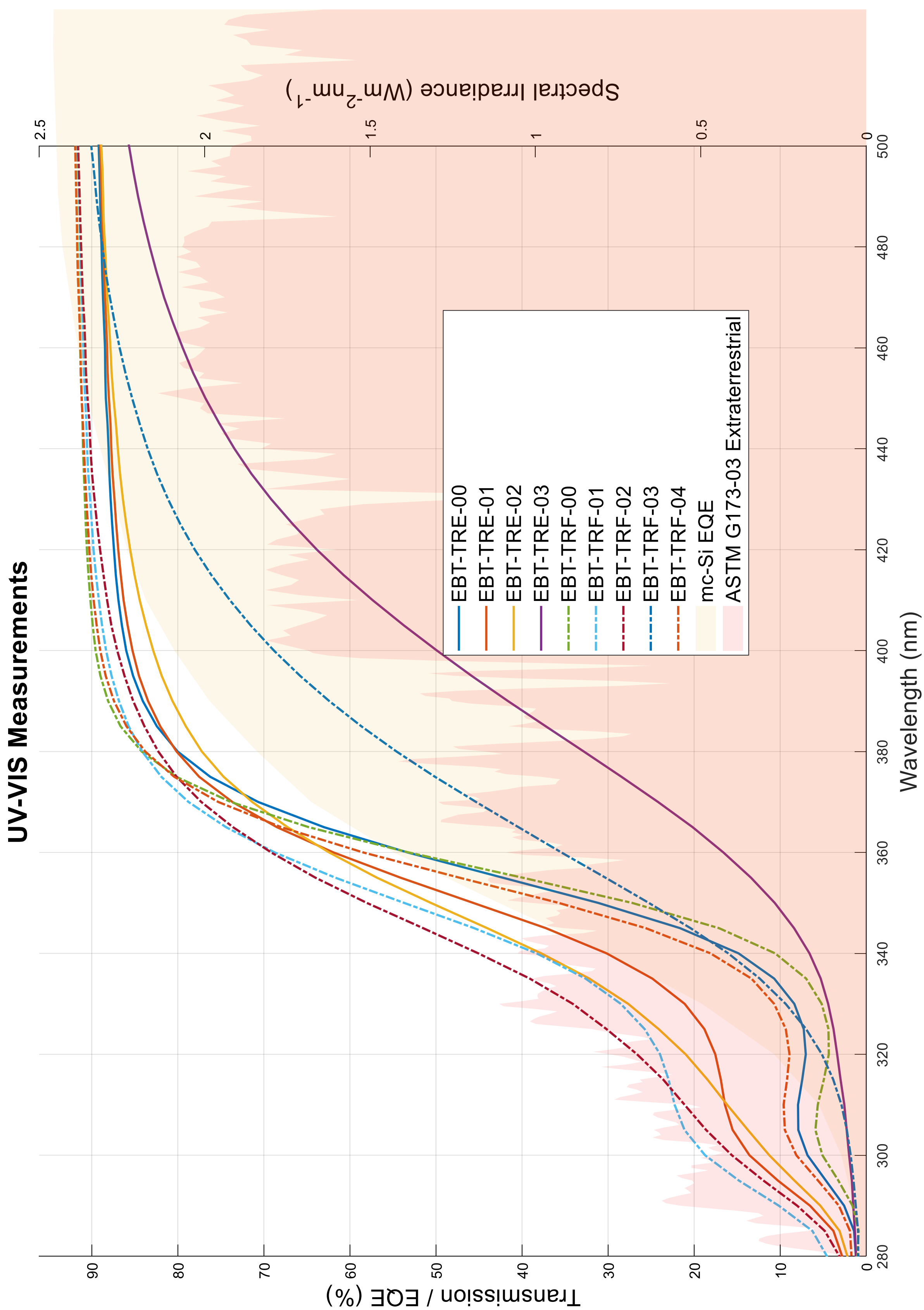


Figure 7.3: Transmission measurements EB Test zoomed in

7.2 *Electron beam test conclusion*

Missions with a lifetime of 7 years an orbit of 600 km showed to be feasible, but the dose of a 1000 km orbit resulted in delamination which could lead to a complete malfunction of the whole panel. This rejects the first hypothesis, but this does not mean that the panel is unusable for lower orbits. The ETFE sample showed cracks in the top layer whereas the FEP samples did not. The adhesive layer loses its function due to the radiation but the FEP doesn't seem to degrade. Since the mass per area of the adhesive layer and frontsheet are comparable to that of a cover glass no extra cell degradation will be expected. The transmission results show an unexpected power increase for orbits until 600 with a lifetime of 7 years. Only for the 1000 km orbit a power degradation was observed but as it was not more than 10% the second hypothesis is confirmed.

7.3 *VUV Test results*

Table 7.2 shows the relative expected power output after 255 ESD compared to begin of life (BOL) for the different materials and two cell types. The difference in transmission before and after the test is visualised in Figure 7.4, in this plot the AM0 spectrum and EQE curve are also shown. It can be observed that the degradation is strongest in the lower end of the spectrum, from 700 nm downwards. The transmission measurements for the ETFE and FEP based samples are listed in Table 7.3 and Table 7.4. The degradation is visualized in Figure 7.5 and Figure 7.6 respectively and show that the degradation did not reach a steady state. The test simulated a total of 255 ESD.

Table 7.2: Power output by solar panel relative to BOL after 255 ESD of UV exposure in LEO

	Relative power output			
	ETFE		FEP	
	TJ	Silicon	TJ	Silicon
P_{out} BOL	100	100	100	100
P_{out} 255 ESD	80.9	78.0	91.1	88.6

7.4 *VUV Test conclusion*

The results show a transmission degradation of 15% and 5.5% for the ETFE and FEP based samples respectively. This will result in a power output loss of about 20% and 10% for ETFE and FEP respectively after 255 estimated space days as shown in Table 7.2. Since the degradation did not stabilize after 255 ESD the hypothesis is not confirmed. The degradation stabilization time was taken from a test on white opaque XETFE [12]. There the change in absorption coefficient stabilized after 150 days. The difference in stabilization time could be due to the fact that the materials in this test where transparent instead of opaque.

It is not possible to say anything yet about further degradation if the test was extended. The test was performed at a temperature of 65 degrees Celsius. At this point it is not possible to say anything about the temperature influence on this degradation. Based on the difference in transmission loss between the two samples and the NASA MISSE 2 [13] research it can be concluded that the total transmission loss of the FEP based sample is due to the transmission degradation in the EVA layer. It can be concluded that FEP is the better option as a frontsheet and that further research is needed to find adhesives that don't lose transmission due to VUV.

The lamp used for the test had a 200 by 200 mm radiation area. The intensity difference between the edges of the square and the centre showed to be around 5-10%. As the samples that where last taken out where in the centre of the sample board, they will have been exposed to the most radiation. The spectral measurement used to make the translation to estimated space days has also been done in the centre. The spectrum of the lamp used is not completely the same as the AM0 spectrum. The main difference is in the fact that the lamp has more energy when in the lower end of the 200-280 nm range, whereas the AM0 spectrum keeps lowering towards 0 in this same range. Therefore it can be concluded that the degradation data in this test could be worse than it would have been in space.

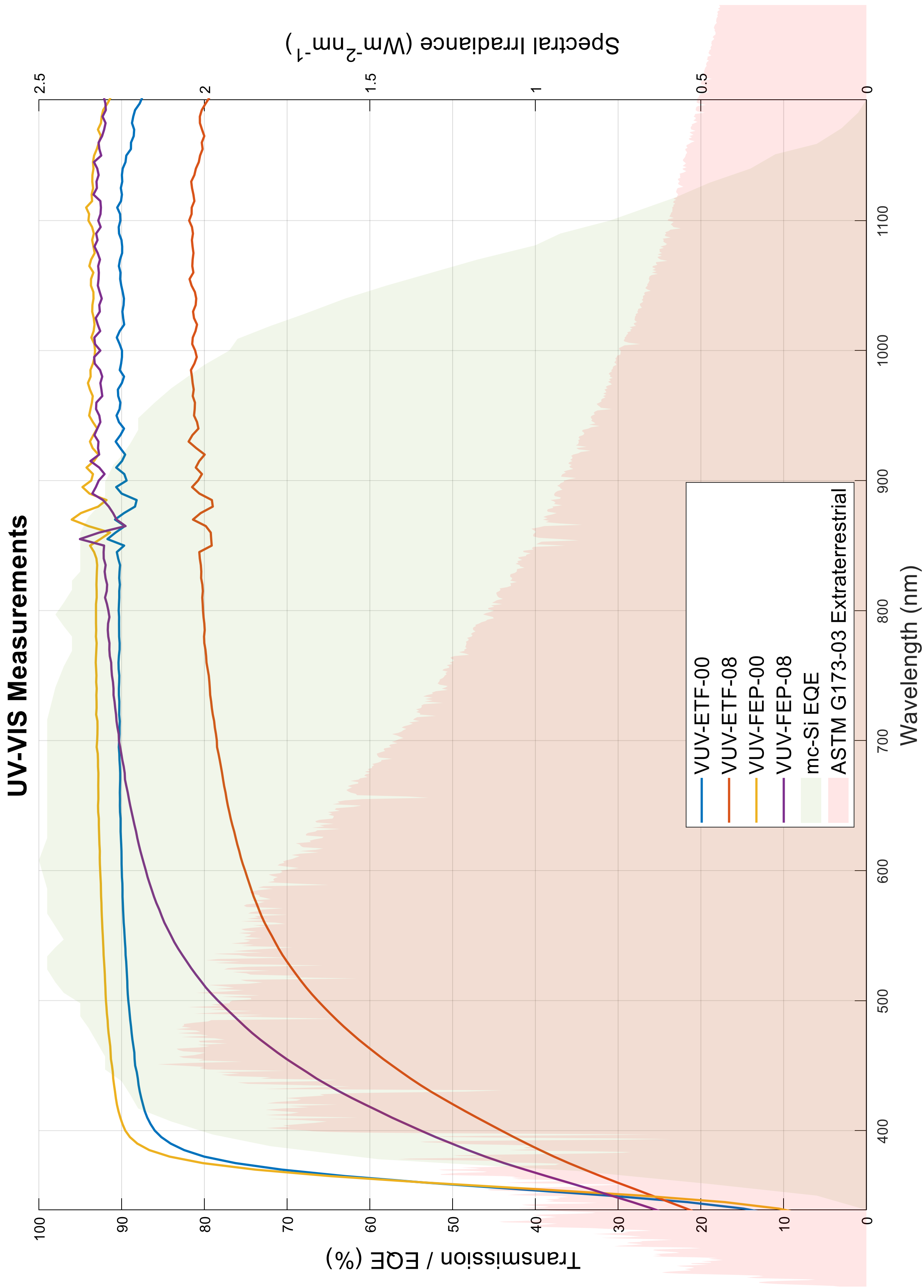


Figure 7.4: Transmission measurement UV degradation test

Table 7.3: Transmission measurement results of ETFE based sample for a TJ and a Silicon cell as a result of VUV degradation

Sample label	ESD	Transmission for a TJ cell (%)[5]	Transmission for a terrestrial silicon cell (%)[6]
SMP-VUV-ETF-00	0	88.5	89.8
SMP-VUV-ETF-01	15	88.7	89.6
SMP-VUV-ETF-16	15	88.6	89.5
SMP-VUV-ETF-02	25	88.8	89.5
SMP-VUV-ETF-15	25	88.9	89.6
SMP-VUV-ETF-03	35	88.5	89.2
SMP-VUV-ETF-14	35	88.6	89.4
SMP-VUV-ETF-04	55	87.9	88.7
SMP-VUV-ETF-13	55	87.9	88.8
SMP-VUV-ETF-05	85	87.0	87.8
SMP-VUV-ETF-12	85	87.0	87.9
SMP-VUV-ETF-06	125	85.5	86.2
SMP-VUV-ETF-11	125	85.8	86.5
SMP-VUV-ETF-07	185	81.3	81.2
SMP-VUV-ETF-10	185	81.1	81.1
SMP-VUV-ETF-08	255	75.9	75.6
SMP-VUV-ETF-09	255	75.3	74.8

[5] Transmission on the 350-1600 nm range based on the effective bandwidth of a triple junction Azure Space solar cell [20]

[6] Transmission on the 400-1150 nm range based on the effective bandwidth of a terrestrial p-type silicon solar cell [19]

Table 7.4: Transmission measurement results of FEP based sample for a TJ and a Silicon cell as a result of VUV degradation

Sample label	ESD	Transmission for a TJ cell (%)[7]	Transmission for a terrestrial silicon cell (%)[8]
SMP-VUV-FEP-00	0	91.9	92.8
SMP-VUV-FEP-01	15	92.1	92.3
SMP-VUV-FEP-16	15	92.4	92.7
SMP-VUV-FEP-02	25	92.0	92.4
SMP-VUV-FEP-15	25	92.0	92.3
SMP-VUV-FEP-03	35	91.8	92.3
SMP-VUV-FEP-14	35	91.7	92.2
SMP-VUV-FEP-04	55	91.2	91.8
SMP-VUV-FEP-13	55	91.1	91.7
SMP-VUV-FEP-05	85	90.0	90.6
SMP-VUV-FEP-12	85	90.1	90.8
SMP-VUV-FEP-06	125	89.1	89.5
SMP-VUV-FEP-11	125	88.9	89.4
SMP-VUV-FEP-07	185	88.2	88.3
SMP-VUV-FEP-10	185	88.2	88.5
SMP-VUV-FEP-08	255	87.3	87.3
SMP-VUV-FEP-09	255	87.1	87.0

[7] Transmission on the 350-1600 nm range based on the effective bandwidth of a triple junction Azure Space solar cell [20]

[8] Transmission on the 400-1150 nm range based on the effective bandwidth of a terrestrial p-type silicon solar cell [19]

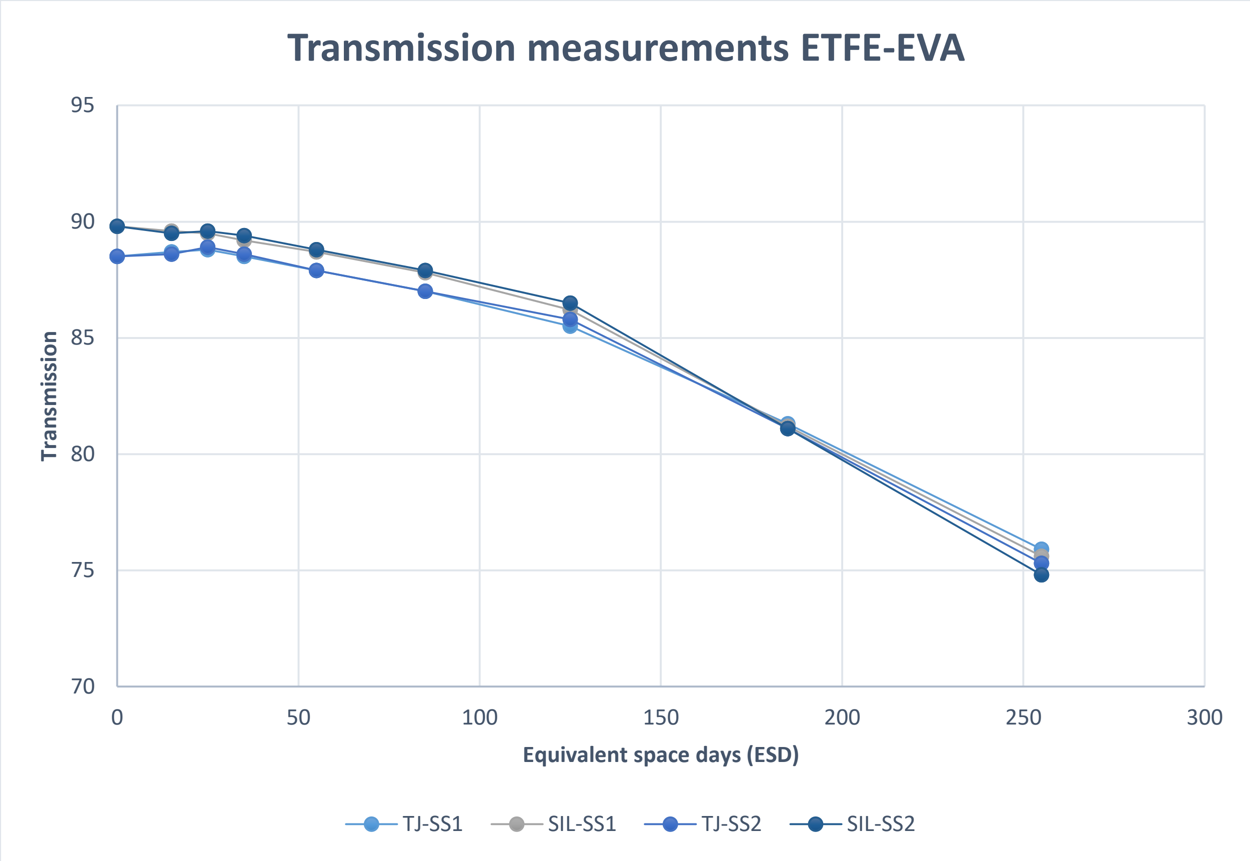


Figure 7.5: Transmission of ETFE sample for a TJ and a Silicon cell as a result of VUV degradation

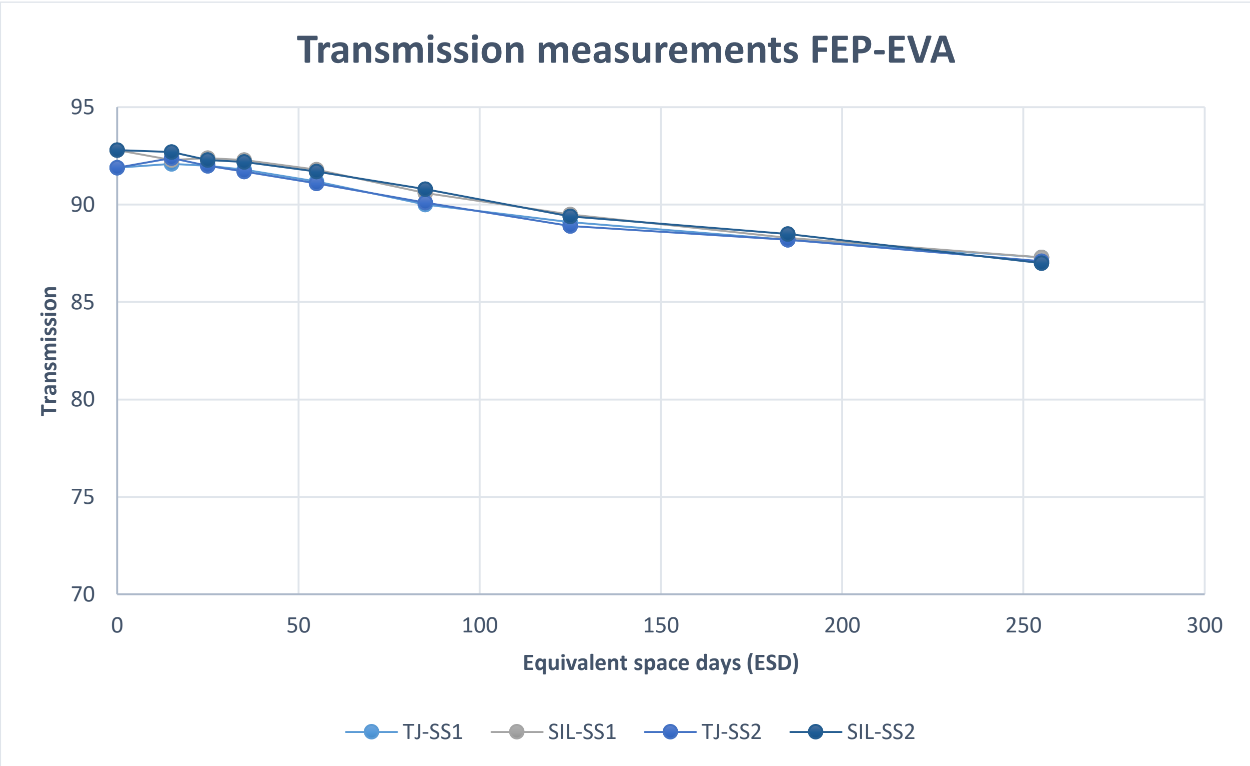


Figure 7.6: Transmission of FEP sample for a TJ and a Silicon cell as a result of VUV degradation

7.5 *Thermal cycling test results*

Figure 7.7 shows the samples in the thermal cycling chamber after 100 cycles from -70 to 80 degrees Celsius. It can be observed that the samples are all still intact. Some marginal deformation can be observed, this mostly happened at the place where the tabs exited the modules. Furthermore it was observed that the bifacial module was the only module that stayed flat. This is due to the use of a symmetrical layup. Using the same material on both sides resulted in having no difference in thermal expansion coefficient. The same results were observed after 50 cycles from -120 to 120 degrees Celsius. These tests cannot confirm the hypothesis, but they do show that there are no major infant mortality issues.

A thorough inspection has been performed on the samples after the -70 to 80 cycles test by checking the samples with a microscope. The locations on the samples are labelled in Appendix H. The pictures show minor differences, a selection of the before and after microscope images can be seen in Table 7.5. The three major differences are the following:

- The distance between the cells of sample SMP-CYC-SUE-01 seems to have decreased by almost a factor 2 (location 1 in Table 7.5).
- Something seems to have happened to a part of the adhesive (location 2 Table 7.5).
- The tabs of sample SMP-CYC-TRE-01 are not as flat anymore (location 3 Table 7.5).

This could result in stress on the connectors. Note there is a slight difference in the colour of the images, this is due to different lighting settings and not the result of the samples changing colour.

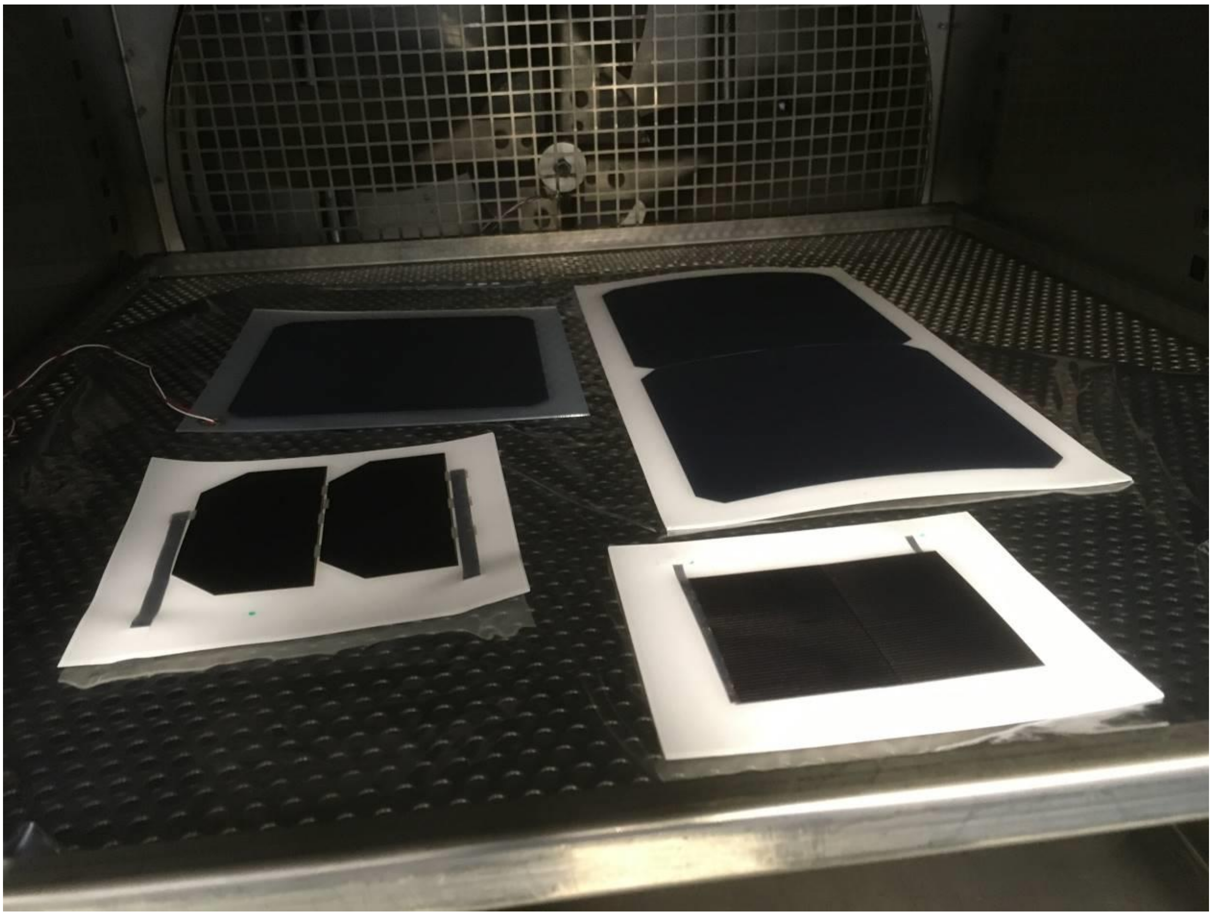
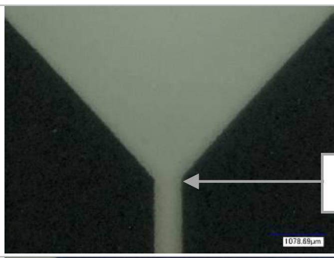
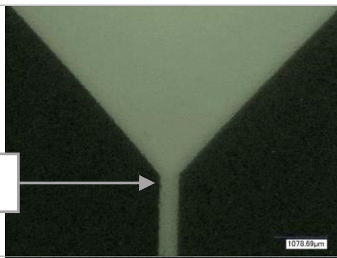
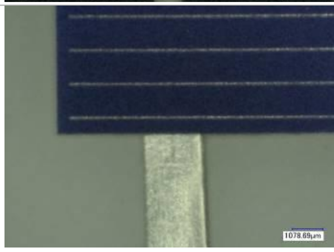
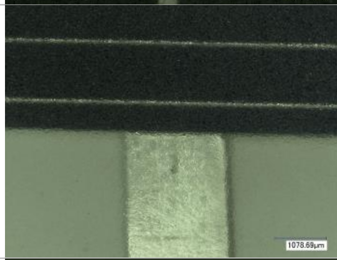
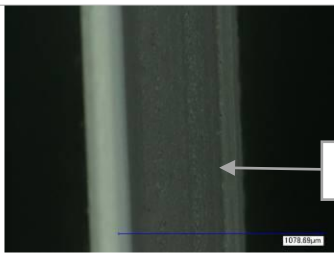
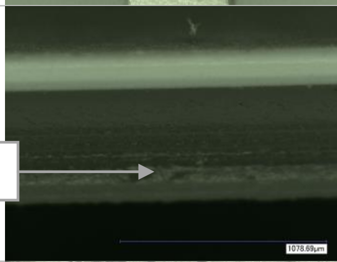
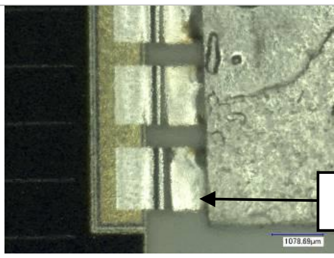
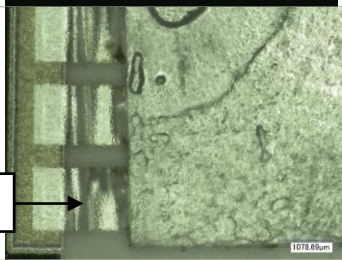
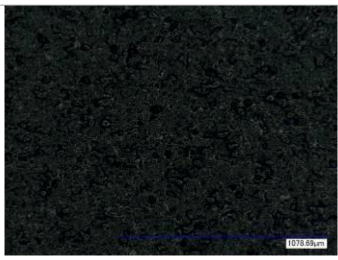


Figure 7.7: Samples in thermal cycling chamber after cycling

Table 7.5: Microscope images before and after thermal cycling test

Sample	Before	After
SMP-CYC-SUE-01 50x zoom Location 2		
SMP-CYC-SIE-01 50x zoom, 100x zoom Location 3		
SMP-CYC-SIE-01 100x zoom Side view		
SMP-CYC-TRE-01 50x zoom Location 3		
SMP-CYC-BFE-01 200x zoom Location 2		

7.6 *Thermal cycling test conclusion*

The samples where all structurally intact after going through 100 temperature cycles from -70 to +80 degrees Celsius. It has been observed that there where minor changes in the flatness of the samples when using a backsheet. These changes where not observed in the bifacial sample. The samples did not have the same size and are therefore hard to compare, but this could be the result of using a similar film for both sides since there is no difference in the thermal expansion coefficient in that case. Thus it could be interesting in using the bifacial composition of layers even for non-bifacial purposes as it results in less deformation. The amount of cycles is not enough to conclude that the compositions tested will survive for a lifetime of 7 years, but it can be concluded that the major infant mortalities are excluded. It still has to be investigated what effect the thermal cycling will have on the contacts between the solar cells. When choosing two frontsheets instead of a backsheet frontsheet combination it has to be investigated what effect this will have on the thermal properties.

7.7 *Outgassing test results*

Table 7.6 shows the results of the outgassing test. It can be observed that the RML is not higher than the required limit of 1%. The reference samples show to have a maximal difference of 0.03% and therefore confirm that the accuracy of the scale is accurate enough to determine whether the samples comply with the RML requirement.

Table 7.6: Results outgassing test

Sample label	Mass before (mg)	Mass after (mg)	RML%
SMP-OGT-ETF-00	193.76	193.70	0.03
SMP-OGT-ETF-01	173.73	172.89	0.48
SMP-OGT-ETF-02	164.50	163.66	0.51
SMP-OGT-ETF-03	195.57	194.60	0.50
SMP-OGT-FEP-00	192.19	192.20	-0.01
SMP-OGT-FEP-01	224.46	223.62	0.37
SMP-OGT-FEP-02	199.57	198.95	0.31
SMP-OGT-FEP-03	197.04	196.35	0.35
SMP-OGT-BFM-00	155.00	155.04	-0.03
SMP-OGT-BFM-01	208.91	207.87	0.50
SMP-OGT-BFM-02	231.29	230.26	0.45
SMP-OGT-BFM-03	213.11	212.27	0.39

7.8 *Outgassing test conclusion*

The outgassing results have shown to fulfil the requirement of a RML lower than 1%, therefore confirming the hypothesis. In the setup used it was not possible to measure the CVCM nor the TML. As the RML was between 0.31 and 0.51% it could not yet be said if the CVCM requirement of lower than 0.1% will be met. According to the ECSS standards the vacuum should be 1e-5 mbar, during the test it only got to 3.7e-1 mbar. It is hard to say what effect the difference in pressure would have for an effect on the RML.

7.9 *Structural analysis test results*

The force-displacement measurements resulted in a Young's modulus between 1.3E+9 and 2.1E+9 N/m2. The different replacements, forces and corresponding Young's moduli are listed in Table 7.7.

Table 7.7: Young's modulus Semi-flexible concept

Force [N]	**Displacement [m]**	**Young's modulus [N/m^2]**
0.147	0.055	1.3E+09
0.392	0.105	1.8E+09
0.491	0.120	2.0E+09
0.638	0.145	2.1E+09

Based on the most conservative Young's modulus (1.3E+09 N/m^2) and the dimensions of the sample an analysis could be performed in Abaqus. This was done for the following four cases (visualised in Figure 7.8 to Figure 7.11):

1. A Semi-flexible array without reinforcements of 1x1 meter constrained in all directions on two corners.
2. A Semi-flexible array with 2 reinforcement bars with a 1x1 meter size constrained in all directions on two corners.
3. A Semi-flexible array with 2 reinforcement bars with a 1x3 meter size constrained in all directions on two corners.
4. A Semi-flexible array with 4 reinforcement bars with a 1x1 meter size constrained in all directions on all sides.

Case 1-3 serve as an analysis for a deployed state, whereas case 4 serves as an analysis in stowed position. The reinforcement bars used in this case are 20x20x2 mm box profile aluminium bars. Case 1-3 are constrained in on the left and bottom corners and case 4 is constrained along all four sides. The required first natural frequency while deployed should be at least 2 Hz and in stowed position 50 Hz is required. No of the deployed cases meet the deployed stiffness requirement (natural frequencies between 0.3 and 1.9 Hz). Case 4 showed that the stowed requirement is not met in this configuration (3.1 Hz). Figure 7.8 to Figure 7.11 show the structural analysis of those cases, Table 7.8 summarizes the results.

Table 7.8: Summary of calculated first natural frequencies

	First natural frequency (Hz)	**Required first natural frequency (Hz)**
Case 1	0.3	2
Case 2	1.9	2
Case 3	1.3	2
Case 4	3.1	50

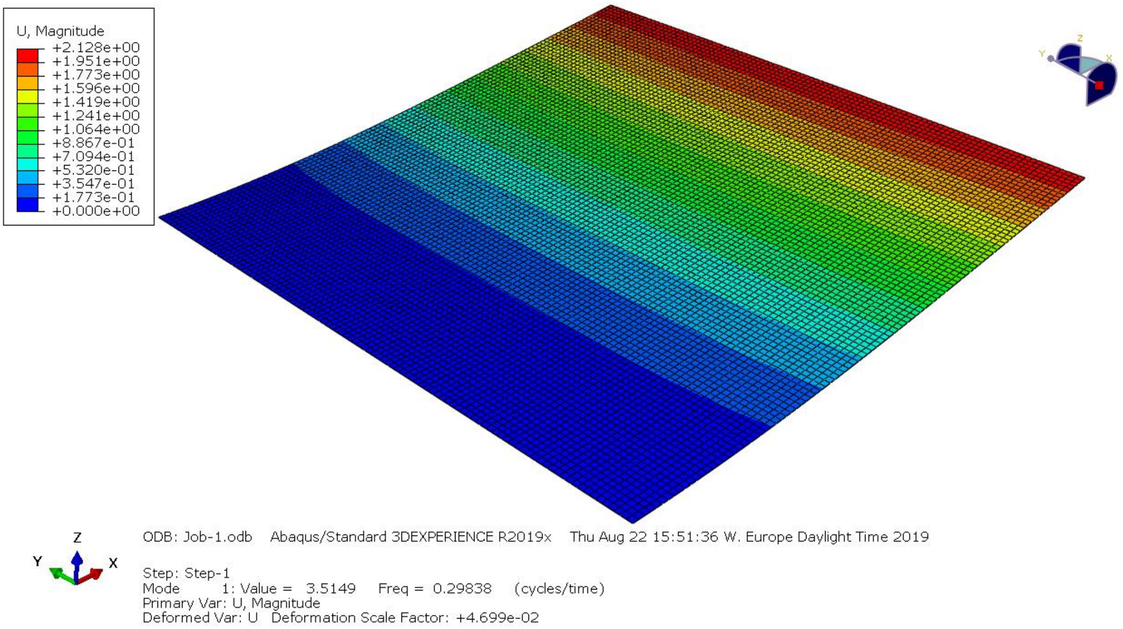


Figure 7.8: Structural analysis of case 1

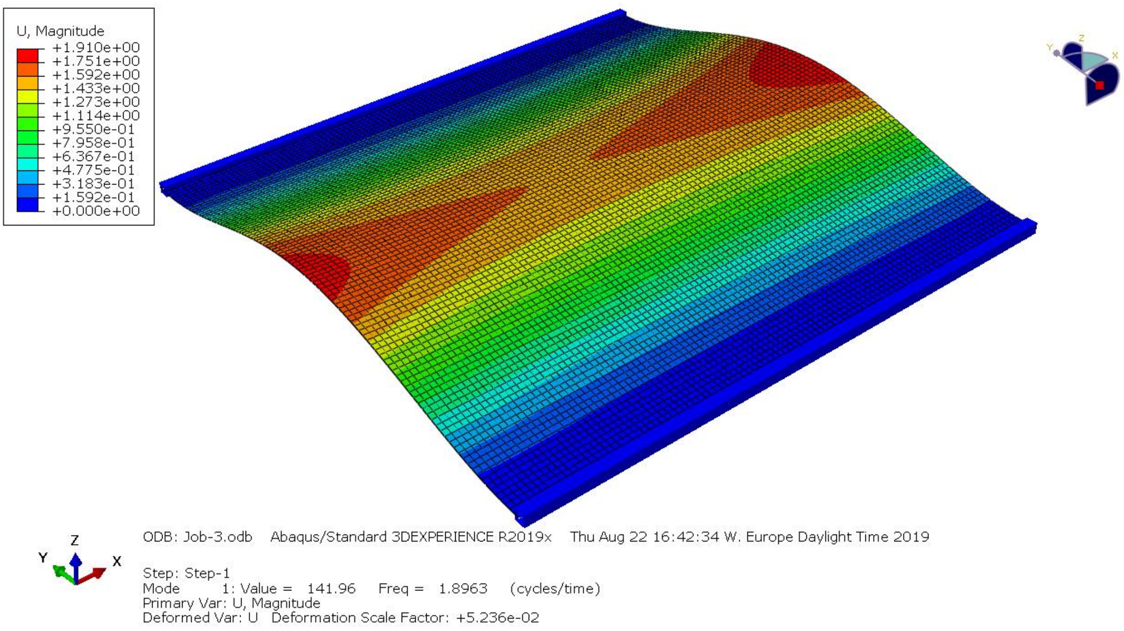


Figure 7.9: Structural analysis of case 2

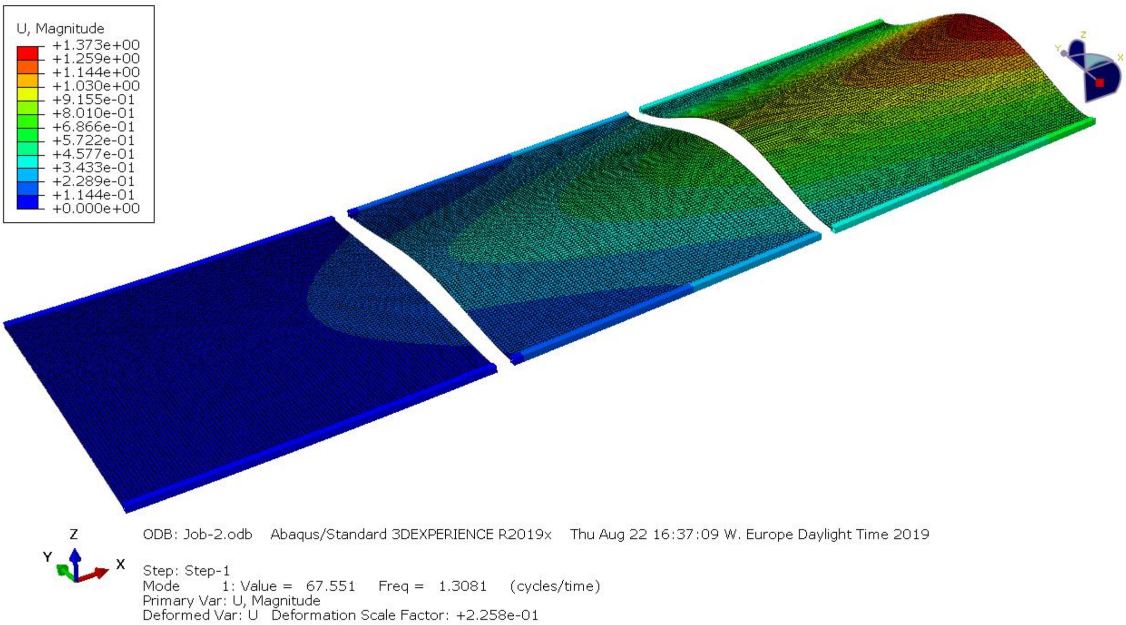


Figure 7.10: Structural analysis of case 3

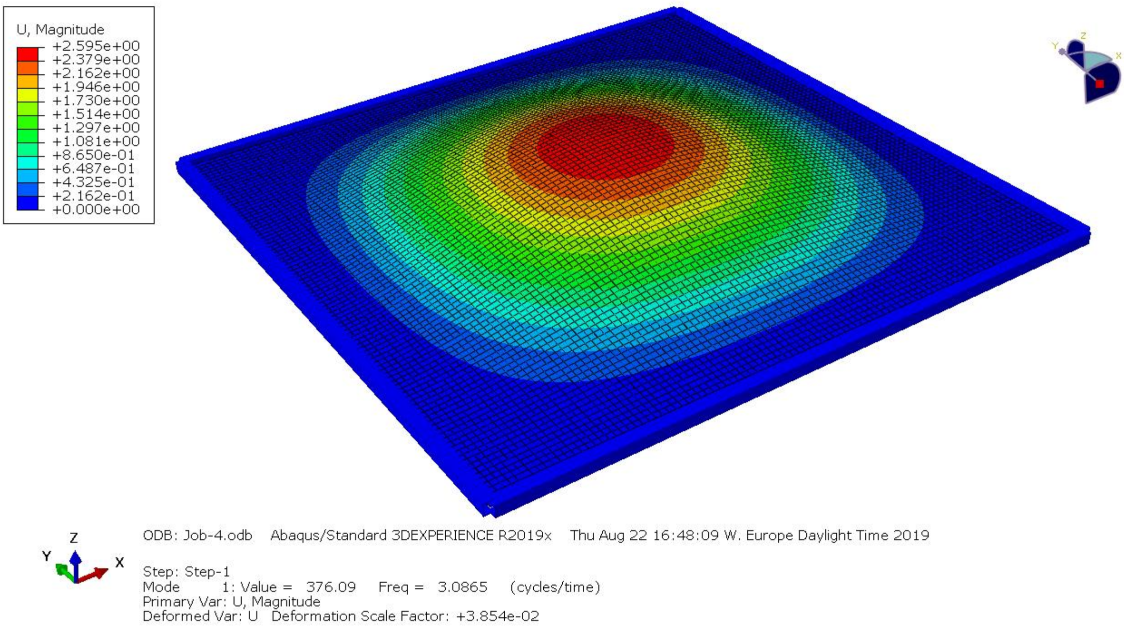


Figure 7.11: Structural analysis of case 4

7.10 *Structural analysis test conclusion*

The results show that the stiffness requirement when in deployed state is not met, this rejects the first hypothesis. It does show that the order of magnitude is right and it should therefore be possible to adjust the design in such a way that it will be met without major changes. To achieve the 1x3m array size a reinforcement is needed but as the mass of the reinforcement is 1.5 kg/m2 the total array mass still is about 2.5 kg. The stowed requirement is not met, therefore rejecting the second hypothesis. To find out if this can be met, different designs to attach the Semi-flexible array to the spacecraft during launch have to be tested. To find the equivalent Young's modulus only one sample has been tested and therefore this value is more an indication than an accurate value. Additionally the stiffness largely depends on the type of solar cell encapsulated in the materials and the thickness of the total encapsulation material. Depending on the cell type the thickness of the laminate could be used as a design parameter to get to the right stiffness.

8 Comparison with conventional solar array

Based on the results a comparison can be made between a conventional solar array and a solar array based on the current state of the Semi-flexible concept. This comparison will be done by looking at a one square meter solar array for a 7 year mission in an orbit of 600 km. The main specifications of the three solar arrays will be listed below in Table 8.1. The numbers are indicative and based on interviews [5] [6] [7]. Due to confidentiality reasons more detailed pricing and lead time specifications cannot be given and all price related indications will be relative instead of absolute.

Table 8.1: Comparison conventional and Semi-flexible solar array

	SparkWing solar array	**Semi-flexible array**	**Semi-flexible array**
Cell type	Triple junction gallium	Triple junction gallium	P-type silicon
Power BOL [W]	250	250	167[9]
Price[10] [relative]	100	70	25
Power EOL[11] [W]	225	203[12]	150[13]
Price per watt EOL [relative]	100	77	39
Mass [kg]	4.5	2.5[14]	2.5[15]
Mass per watt EOL [g/W]	20	12	17
Thickness [mm]	14	1.3	1.3
Natural frequency deployed	2.8	1.9	1.9
Lead time[16] [months]	6	4	4

There is a large difference in price and in lead time. The difference in lead time is due to the independency from the substrate supplier. The price difference based on the following:

- No carbon sandwich substrate and cover glasses needed for Semi-flexible array, replacement materials very cheap
- Type of solar cells used
- Labour intensity decrease due to the replacement of single cover glasses with one frontsheet

Additionally there is a mass and thickness difference which will result in a lower price for launch costs for the customer. This is not taken into account in this overview.

[9] Efficiency of 2/3 is taken for silicon compared to triple junction gallium solar cells
[10] Price for semi-flexible concept based on Wattlab production experience and environment.
[11] A power degradation of 10% over the lifetime for the solar cell is used for all cases.
[12] Power loss of 10% due to transmission degradation.
[13] Power loss of 10% due to transmission degradation.
[14] With reinforcements on the sides
[15] With reinforcements on the sides
[16] Time from design freeze (critical design review) till ready to send based on Airbus and Wattlab experiences.

Based on Table 8.1 it can be concluded that for the decrease in price the customer should be willing to give in on the stiffness and power EOL. Although the power EOL is lower, since the price, thickness and mass are significantly lower, this could be covered by using multiple arrays on space craft level.

9 Conclusion & Discussion

The main research question was to find out if it was possible to design a solar panel for small satellites based on the Semi-flexible concept for LEO applications. To find out what the customer needs are for this domain, the mission and stakeholder analysis have been performed. Second, a system functional analysis has been performed to find the similarities and differences between the conventional space solar panels and the Semi-flexible concept. This showed that the Semi-flexible concept has a lot of potential due to a lower cost, lower mass and lower volume to power ratio. After the requirements were determined, a brainstorm was done and concept selection was performed. This way, the risks could be identified and a better, fair comparison could be made later on. The winning concept was based on the conventional sandwich principle and was named the S-folding blanket.

To find out if the S-folding blanket is applicable in low Earth orbit, tests have been performed to find out if the most critical factors from the risk assessment would cause a problem. The tests have not resulted in any show stoppers. The different tests did show that there are limitations to the missions for which the concept could be applied. The conclusion is that making a solar panel based on the Semi-flexible concept is feasible for missions with an Earth orbit at an altitude of 600 km with a lifetime of 7 years. For higher altitudes with the same lifetime, the increased amount of radiation will lead to delamination and malfunctioning of the solar panel. A series of different tests have been performed, the main conclusion of each of the tests are given below:

- **EB test –** Exposure to radiation did not show to be a problem for orbits with an altitude up to 600 km and a lifetime of 7 years. The counter intuitive finding was, that the degradation of the material actually resulted in a higher power output due to an increase of transmission in a part of the spectrum. For an orbit with a height of 1000 km and 7 years lifetime the dose of radiation showed to result in delamination of the samples.
- **VUV test –** The exposure to VUV resulted in a degradation of the transmission and therefore a decrease in potential power output. This degradation did not show to stabilize after 255 estimated space days. It can be concluded that for the FEP-EVA samples the degradation was only the result of the EVA degradation.
- **Thermal cycling test –** Two different temperature ranges have been cycled, 100 cycles from -70 to +120 degrees Celsius and 50 cycles from -120 to +120 degrees Celsius. Both did not show delamination. It was observed that there were some locational changes to the PVA.
- **Outgassing test –** The test showed that the samples had a maximum recovered mass loss of 0.51% which complies with the requirements. The required vacuum was not reached and the collected volatile condensed material could not be measured in this setup.
- **Structural analysis –** The analysis has shown that the stiffness of this concept is in the order of magnitude required for deployed state of the SparkWing products. Further design iterations are needed to adjust the array design to meet the mission specific requirements. The first natural frequency for the stowed position was an order of magnitude to small, design iterations will have to show if it is possible to adapt the design in such a way that it will be able to meet its stowed requirement.

The Semi-flexible solar panel concept has some advantages when compared with the current state of the art space solar arrays. Designs based on the Semi-flexible concept can result in a lower cost, mass and volume solar array compared to conventional solar arrays as the result of different material and process use. This research has proven the potential of the concept, to better specify the limitations, increase the performance and increase the technology readiness level additional research is needed. The next steps are discussed in chapter 10.

10 Recommendations & Outlook

This chapter gives an overview of the recommendations and future outlook that result from this master thesis project. It is divided into three main parts. Starting with the recommendations on the main experiments performed (10.1) after which the possible improvements to the Semi-flexible concept are discussed (10.2). Finally the recommendations and future outlook at an array concept level will be listed (10.3).

10.1 *Experimental tests*

1. **Perform more thermal cycles on the Semi-flexible solar panel composition to determine the mechanical wear out failure lifetime.** In the thermal cycle test performed 100 cycles have been done, this is not enough to say anything about the wear out failures of the concept. First concept designs should determine the final composition of the layers for the solar panel. When this is determined, more samples and a larger amount of thermal cycles could give more insight into the effects of mechanical wear on the concept. This test could alternatively be replaced by a mechanical cycling test to reduce costs. It should also be investigated what effect the thermal cycling process has on the internal connections of the PVA

2. **Perform an official outgassing test to find the CVCM value and find out if the higher vacuum will result in a higher RML.** The test showed that the RML requirement was met, but the ECSS requires a higher vacuum than has been reached during the test [23]. Additionally the test setup was not capable to measure the CVCM and could therefore not conclude on that requirement.

3. **Repeat the VUV test at higher temperatures to find out if this has an effect on the total transmission degradation and/or functional properties of the adhesive.** This study was limited to testing at one temperature. It would be interesting to find the relation between the temperature and the total transmission degradation. This test should be performed only after an adhesive film trade-off has been done.

4. **Validate if the combination of environmental effects tested will comply with the separate tests by doing an in orbit demonstration.** This is the ultimate test. Some partial results will show fast, among these are the degradation due to VUV and the effect of the temperatures in combination with high vacuum. The ones that will only show over the full mission time are the effect of radiation and mechanical wear due to thermal cycling.

10.2 *Semi-flexible concept*

1. **Test the transmission degradation of different types of adhesive films due to VUV and Charged particle radiation.** Both the VUV and the EB test showed that when using FEP as a top layer the limiting factor in transmission degradation is the adhesive film. Finding an adhesive film that keeps its function and does not have transmission loss due to VUV and EB could result in a power output increase of 10%.

2. **Research if a rejection filter which cuts off wavelengths below 400 nm can be applied on the FEP frontsheet and find out if this is space environment resistant.** If this is possible the degradation due to VUV will not happen and the only transmission loss is due to charged particle radiation. This could be an alternative to finding an adhesive that does not lose transmission due to VUV.

3. **Research if an anti-reflective (AR) coating can be applied on the frontsheet (FEP) and find out if this is space environment resistant.** FEP has a transmission of 93-96% depending on the supplier, applying an AR coating could increase this transmission with as much as 4% [25].It has to be tested whether the coating is resistant to the space environment since it is located on the outside of the solar panel.

4. **Test what the effect is of using a carbon face sheet as a backsheet on the stiffness.** This could be a way to increase stiffness without increasing the mass and volume too much. The application of the face sheet could be done during the lamination process or applied afterwards. It could affect the thermal properties of the solar panel, this has to be investigated. A first bonding test of Kapton to carbon has been tested already, the results after a vacuum oven test (24 hr, 120 degrees Celsius) and thermal cycle test (-120, +120 degrees Celsius, 50 cycles) can be found in Appendix I.

10.3 *Solar array concept level*

1. **Find out if designs based on the Semi-flexible concept can meet the launch and deployed stiffness and vibration requirements.** Since the major risks to the composition of the Semi-flexible concept do not show a no-go for the use in space it has to be investigated what new design opportunities this results in. The main challenge now is to come up with a concept that complies with the stiffness and vibration requirements. As concluded in section 7.10, the main challenge will be to meet the stowed requirements.

2. **Perform a solar cell trade-off based on the concepts generated with the Semi-flexible concept.** With conventional solar arrays a trade-off performed by Airbus determined that going with more expensive solar cells will win over creating a larger surface and use cheaper cells [10]. This is because the extra costs of creating the additional surface area to generate the same power with cheaper cells is higher than the savings on the solar cells. This trade-off is strongly dependent on the composition of the panel and the assembly method which are both completely different with the Semi-flexible concept. A new trade-off has to be done to find out if possibly cheaper solar cells stand a chance and could decrease the price of the solar array drastically.

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APPENDIX

I.1 Appendix A

Table A.1: Requirement traceablity matrix

	NED-CST-01	NED-PLN-01	NED-QLT-01	NED-QLT-02	NED-QLT-03	NED-QLT-04	NED-QLT-05
SS-PRG-MKT-01							
SS-PRG-MKT-02							
SS-PRG-MKT-03							
SS-PRG-CST-01							
SS-PRG-CST-02							
SS-PRG-CST-03							
SS-PRG-SBC-01							
SS-PRG-BDG-01							
SS-IND-HLT-01							
SS-IND-BDG-02							
SS-FNC-MCH-01							
SS-FNC-MCH-02							
SS-SYS-MCH-03							
SS-SYS-MCH-04							
SS-SYS-MCH-05							
SS-SYS-MCH-06							
SS-SYS-MCH-06							
SS-FNC-ELE-01							
SS-FNC-ENV-01							
SS-FNC-ENV-02							
SS-FNC-ENV-03							
SS-FNC-ENV-04							
SS-SYS-ENV-05							

I.2 Appendix B

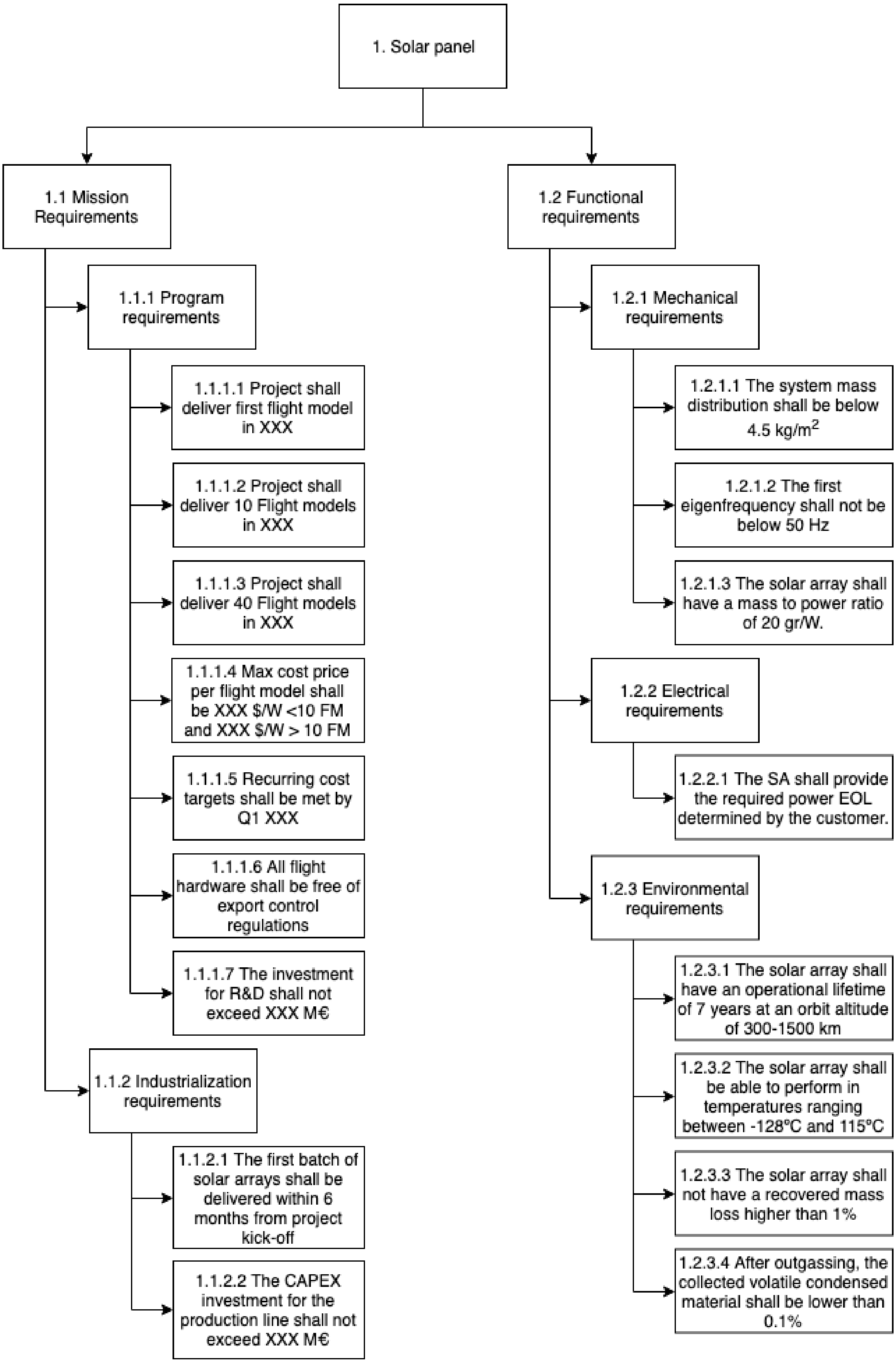

Figure A.1: Requirement discovery tree

I.3 Appendix C

Brainstorm setting

To achieve the goal of the brainstorm, a group of people was gathered. The group was chosen in such a way that a variety of knowledge was available about both space solar arrays as well as the Semi-flexible solar panel. The group consisted of four Airbus employees with different backgrounds: design leader, design engineer, innovation engineer and a system engineer**.** Additionally there were four Wattlab employees, also with different backgrounds: R&D manager, production manager, financial manager and service designer. Sarah Arntz, service designer at Wattlab facilitated the brainstorm process which is described in the next section. The brainstorm was held at the Airbus DS facility in Leiden, the setting is shown in Figure A.2.

Figure A.2 Concept brainstorm in progress

Process

The brainstorm started off with an introduction of everyone present, after which the goal of the brainstorm was presented and put into context. Before starting the idea generation, a brief explanation of both state of the art solar panels, space and Semi-flexible, where presented. When all on the same page, the next step was to write down the answers to two separate questions: “How can you get the volume of something as low as possible?” and “How can you create a large surface area?” Everyone had an A3 sized paper in front of them and got 30 seconds to write down as many answers to those questions as possible after which the papers circulated and the process was repeated. This was done 9 times until everyone had the paper on which they started back in front of them. This way a lot of ideas to realize the two questions above where created and ideas of one person could inspire the next person.

The second stage in the brainstorm process was to combine the answers of the two questions above to come up with design concepts for possible solar arrays. Those combinations where then grouped and doubles were eliminated. The result of this first brainstorm part are shown as strawman concepts in section 5.1.

Next, the customer needs where explained after which four concepts where chosen by groups of two in order to create a more in depth concept. Those four concepts will be discussed in section 5.2. After 30 minutes every group presented its concept based on the following aspects:

- Stowed configuration setup
- Deployment mechanism
- Deployed setup
- Positive and negative aspects
- Price per watt
- Mass per watt
- Volume stowed

An example of the format in which the concepts where presented is shown in Figure A.3. These concepts are used to perform the final selection and come to one design to proceed with.

Figure A.3: Concept presentation during brainstorm

I.4 Appendix D

Calculations performed using Spenvis by Remco van der Heijden [26].

Table A.2: Radiation calculations for different orbits and lifetimes using Spenvis

400km

Coverglass thickness			Total (annual)		
g cm^{-2}	mils	micron	P_{max}	V_{oc}	I_{sc}
0,0E+00	0,0E+00	0,0E+00	1,0E+14	1,0E+14	1,7E+13
5,6E-03	1,0E+00	2,5E+01	1,8E+13	1,8E+13	7,7E+12
1,7E-02	3,0E+00	7,6E+01	9,5E+12	9,5E+12	4,5E+12
3,4E-02	6,0E+00	1,5E+02	5,6E+12	5,6E+12	2,9E+12
6,7E-02	1,2E+01	3,0E+02	3,0E+12	3,0E+12	1,8E+12
1,1E-01	2,0E+01	5,1E+02	1,7E+12	1,7E+12	1,1E+12
1,7E-01	3,0E+01	7,6E+02	1,2E+12	1,2E+12	8,4E+11
3,4E-01	6,0E+01	1,5E+03	5,9E+11	5,9E+11	4,8E+11

500Km

Coverglass thickness			Total (annual)		
g cm^{-2}	mils	micron	P_{max}	V_{oc}	I_{sc}
0,0E+00	0,0E+00	0,0E+00	1,6E+14	1,6E+14	2,2E+13
5,6E-03	1,0E+00	2,5E+01	2,1E+13	2,1E+13	9,0E+12
1,7E-02	3,0E+00	7,6E+01	1,1E+13	1,1E+13	5,3E+12
3,4E-02	6,0E+00	1,5E+02	6,5E+12	6,5E+12	3,5E+12
6,7E-02	1,2E+01	3,0E+02	3,6E+12	3,6E+12	2,2E+12
1,1E-01	2,0E+01	5,1E+02	2,2E+12	2,2E+12	1,5E+12
1,7E-01	3,0E+01	7,6E+02	1,5E+12	1,5E+12	1,2E+12
3,4E-01	6,0E+01	1,5E+03	8,6E+11	8,6E+11	7,1E+11

600km

Coverglass thickness			Total (annual)		
g cm^{-2}	mils	micron	P_{max}	V_{oc}	I_{sc}
0,0E+00	0,0E+00	0,0E+00	2,4E+14	2,4E+14	2,8E+13
5,6E-03	1,0E+00	2,5E+01	2,6E+13	2,6E+13	1,1E+13
1,7E-02	3,0E+00	7,6E+01	1,3E+13	1,3E+13	6,6E+12
3,4E-02	6,0E+00	1,5E+02	8,0E+12	8,0E+12	4,5E+12
6,7E-02	1,2E+01	3,0E+02	4,6E+12	4,6E+12	2,9E+12
1,1E-01	2,0E+01	5,1E+02	2,9E+12	2,9E+12	2,1E+12
1,7E-01	3,0E+01	7,6E+02	2,1E+12	2,1E+12	1,7E+12
3,4E-01	6,0E+01	1,5E+03	1,3E+12	1,3E+12	1,1E+12

800km

Coverglass thickness			Total (annual)		
g cm^{-2}	mils	micron	P_{max}	V_{oc}	I_{sc}
0,0E+00	0,0E+00	0,0E+00	6,8E+14	6,8E+14	6,2E+13
5,6E-03	1,0E+00	2,5E+01	4,9E+13	4,9E+13	2,1E+13
1,7E-02	3,0E+00	7,6E+01	2,4E+13	2,4E+13	1,3E+13
3,4E-02	6,0E+00	1,5E+02	1,5E+13	1,5E+13	8,8E+12
6,7E-02	1,2E+01	3,0E+02	9,0E+12	9,0E+12	6,1E+12
1,1E-01	2,0E+01	5,1E+02	6,1E+12	6,1E+12	4,6E+12
1,7E-01	3,0E+01	7,6E+02	4,7E+12	4,7E+12	3,8E+12
3,4E-01	6,0E+01	1,5E+03	3,2E+12	3,2E+12	2,8E+12

1000km

Coverglass thickness			Total (annual)		
g cm^{-2}	mils	micron	P_{max}	V_{oc}	I_{sc}
0,0E+00	0,0E+00	0,0E+00	1,5E+15	1,5E+15	1,3E+14
5,6E-03	1,0E+00	2,5E+01	9,5E+13	9,5E+13	4,2E+13
1,7E-02	3,0E+00	7,6E+01	4,8E+13	4,8E+13	2,5E+13
3,4E-02	6,0E+00	1,5E+02	3,0E+13	3,0E+13	1,8E+13
6,7E-02	1,2E+01	3,0E+02	1,9E+13	1,9E+13	1,3E+13
1,1E-01	2,0E+01	5,1E+02	1,3E+13	1,3E+13	9,6E+12
1,7E-01	3,0E+01	7,6E+02	9,9E+12	9,9E+12	8,0E+12
3,4E-01	6,0E+01	1,5E+03	6,7E+12	6,7E+12	5,9E+12

I.5 Appendix E

From an email from Niels van der Pas to Jan Schutten [27]:

Cases:
A hot orbit, panels are facing Sunward, PVA is off. 500 km altitude.
A cold orbit, panel are pointing 30° away from the Sun, PVA is on. 1400 km altitude

Results
LDT: Lower design temperature (incl. -15°C uncertainty)
LNT: Lower nominal temperature
UNT: upper nominal temperature
UDT: Upper Design Temperature (incl. +15°C uncertainty)

Table A.3: Resulting temperatures excluding eclipse case

Concept	LDT [°C]	LNT [°C]	UNT [°C]	UDT [°C]
ALTA 12mm	-143	-128	86	101
ALTA 18mm	-142	-127	86	101
ALTA 22mm	-141	-126	86	101
GA3J 12mm	-124	-109	113	128
GA3J 18mm	-123	-108	113	128
GA3J 22mm	-122	-107	114	129
Yoke panel 12mm	-127	-112	115	130

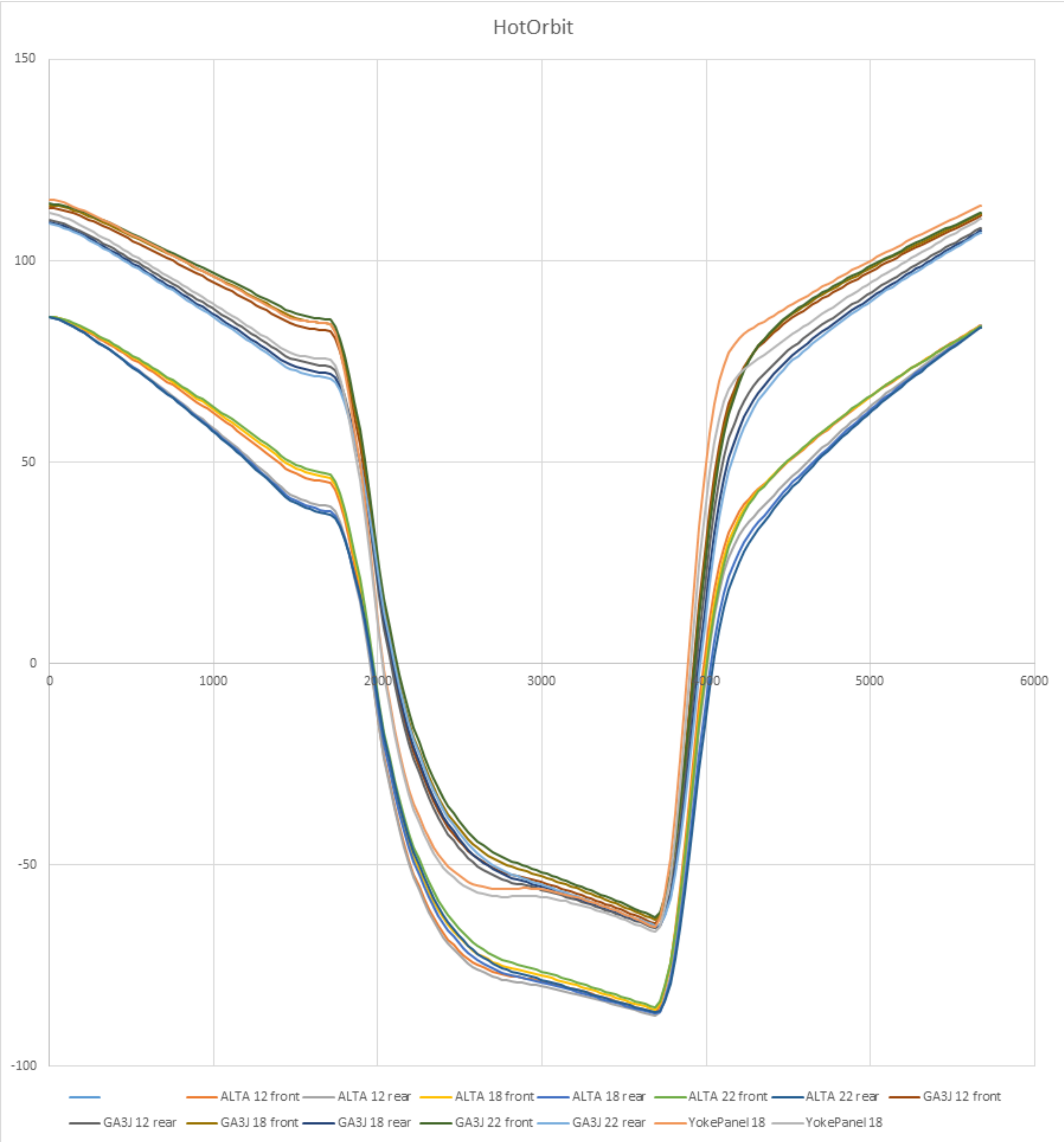

Figure A.4: Hot orbit graph

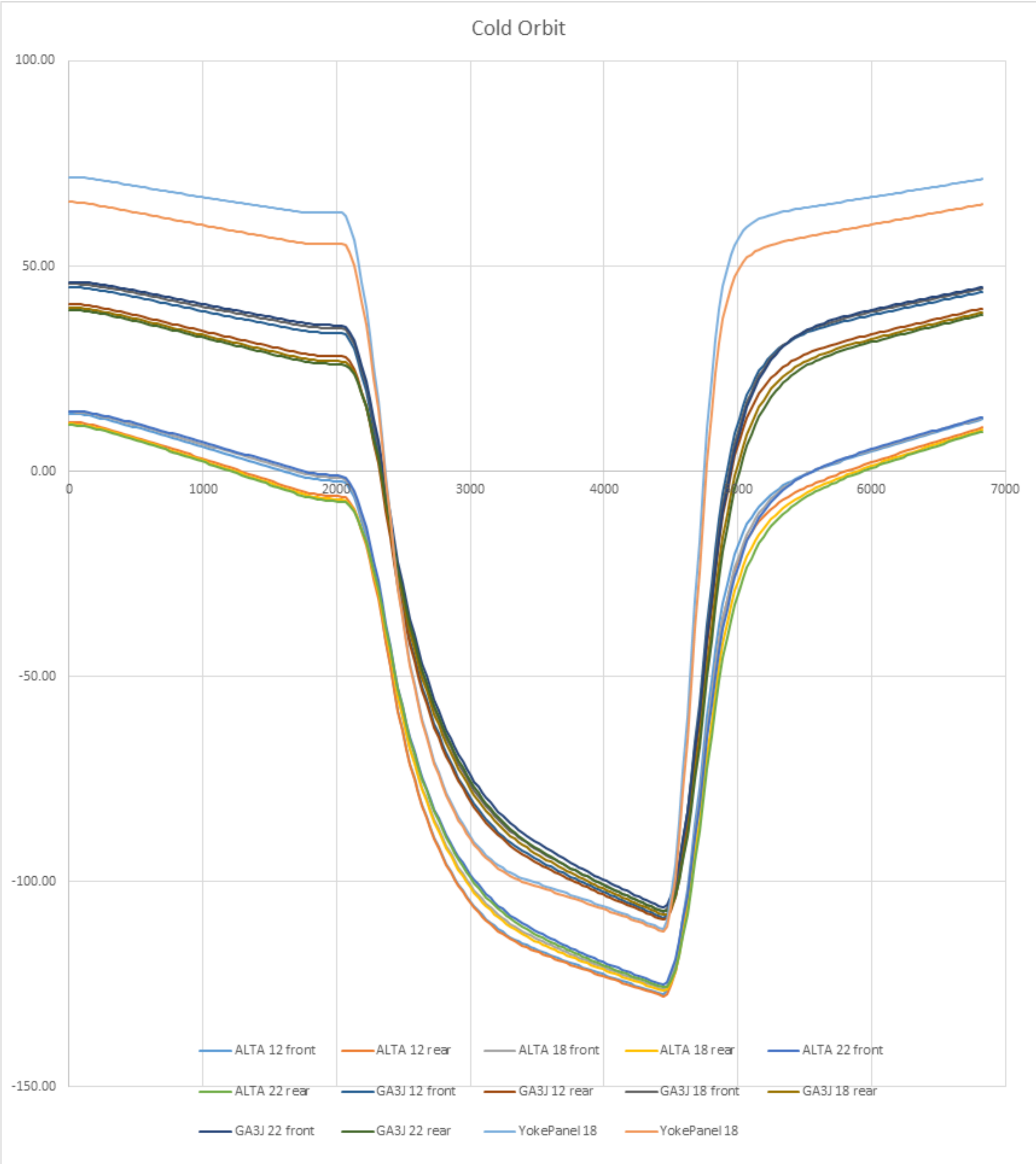

Figure A.5: Cold orbit graph

I.6 Appendix F

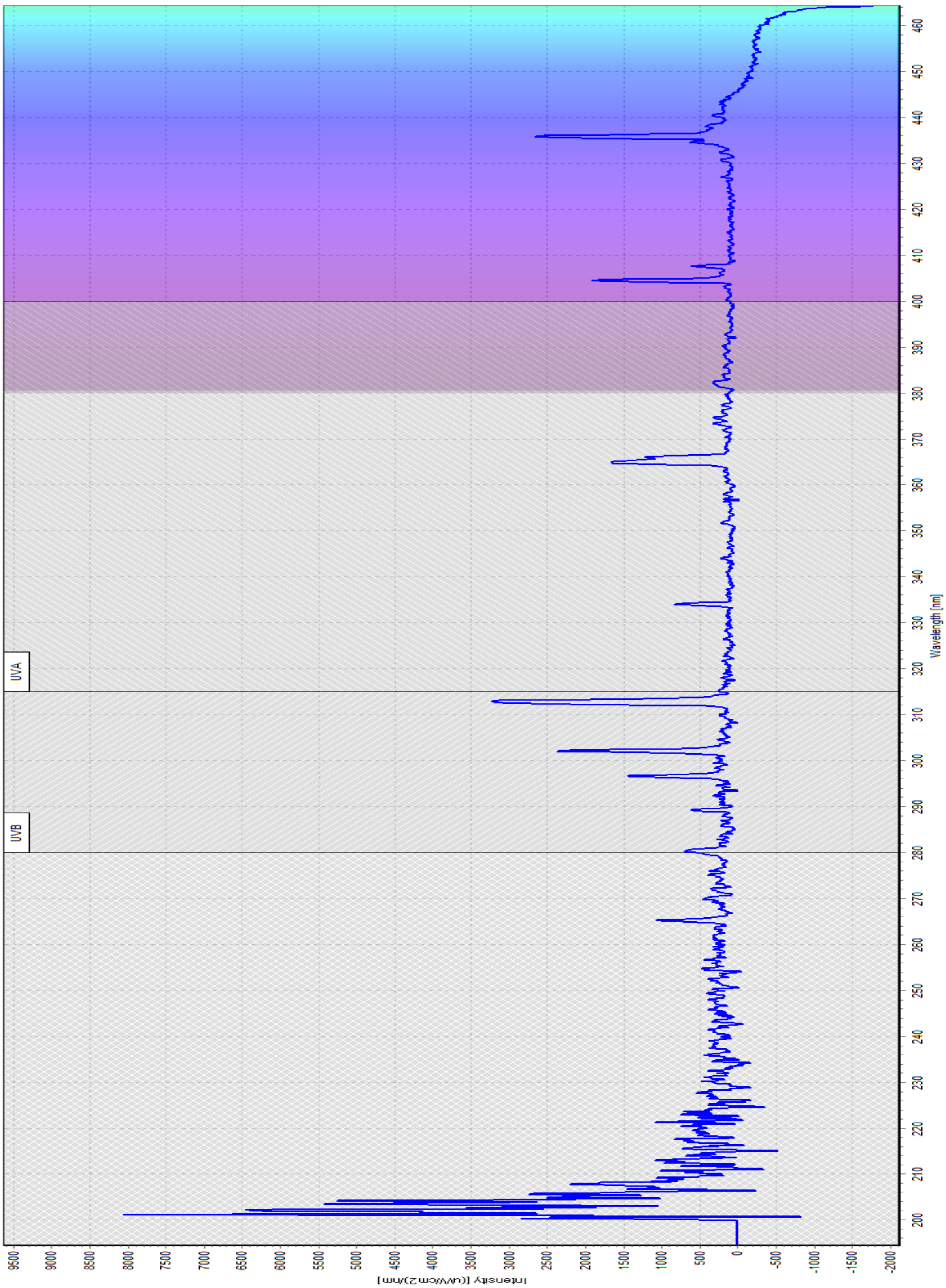

Figure A.6: Screenshot of spectral analysis of VUV lamp used for VUV degradation test

I.7 Appendix G

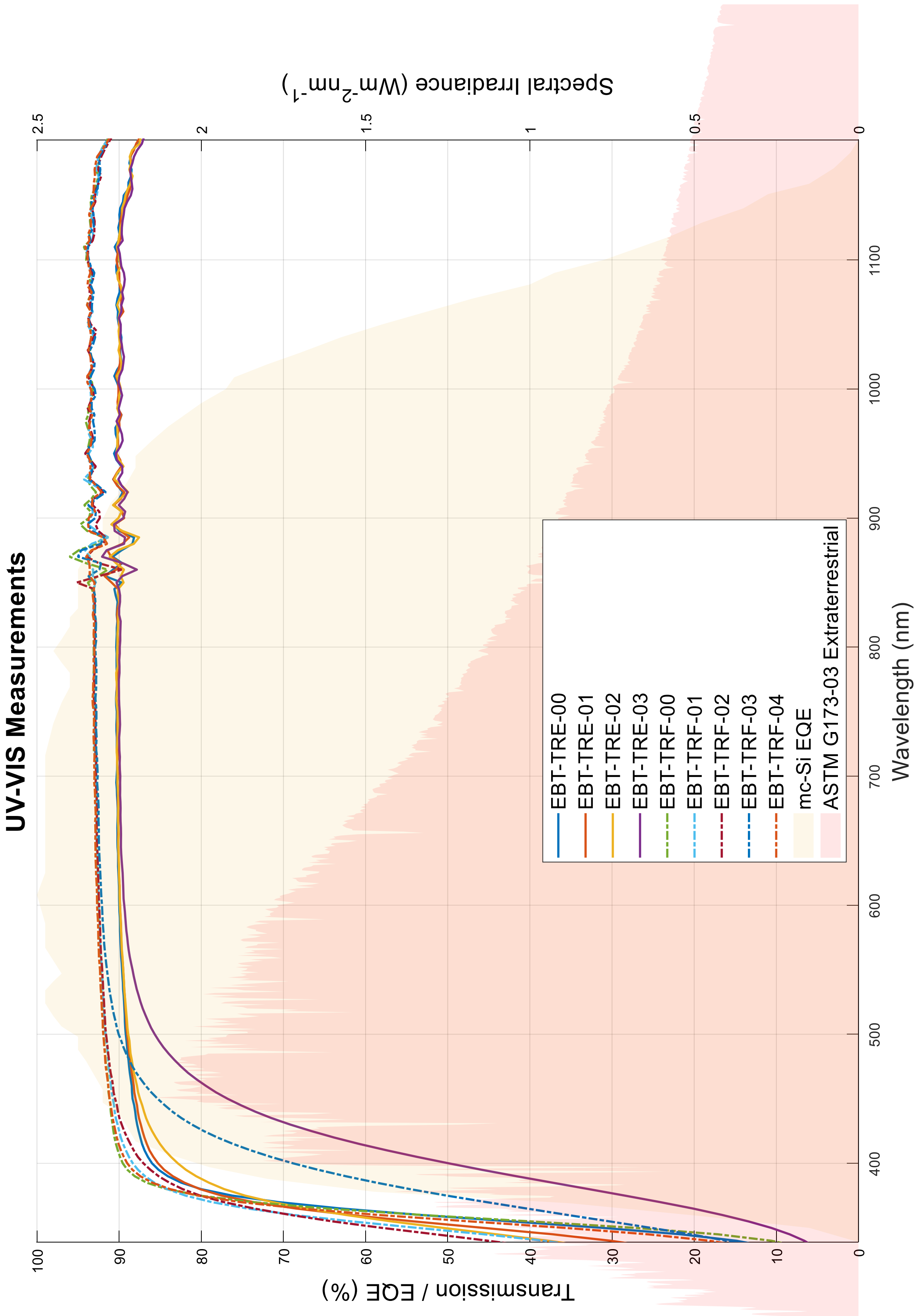

Figure A.7: Electron Beam test results plot with silicon solar cell EQE curve

I.8 Appendix H

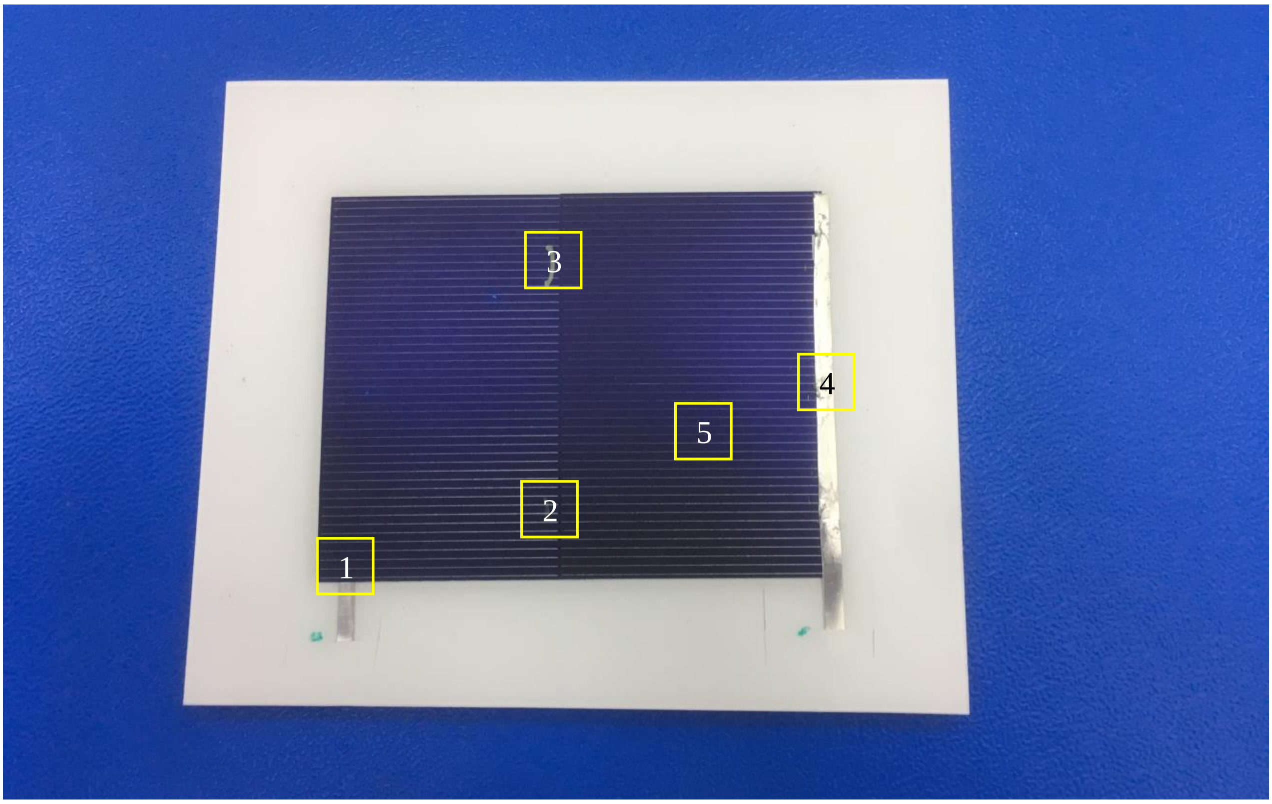

Figure A.8: Sample SMP-CYC-TRE-01

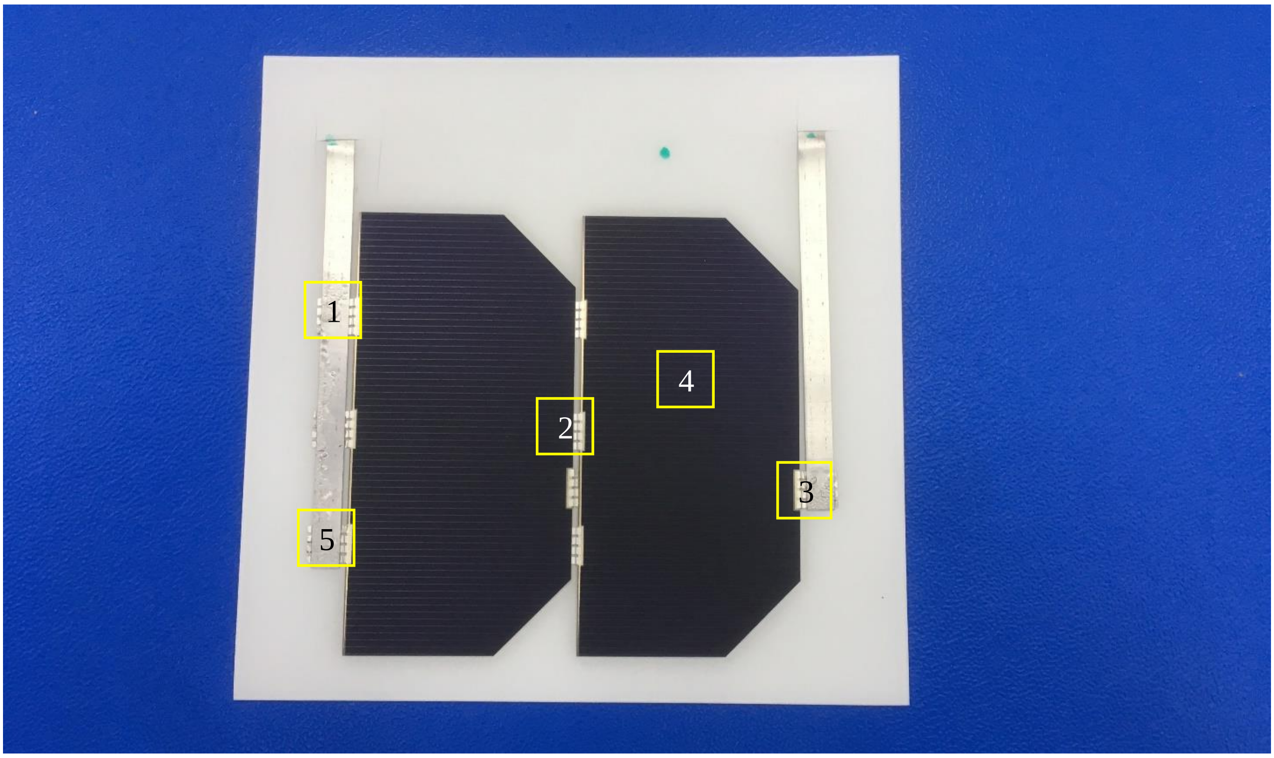

Figure A.9: sample SMP-CYC-SIE-01

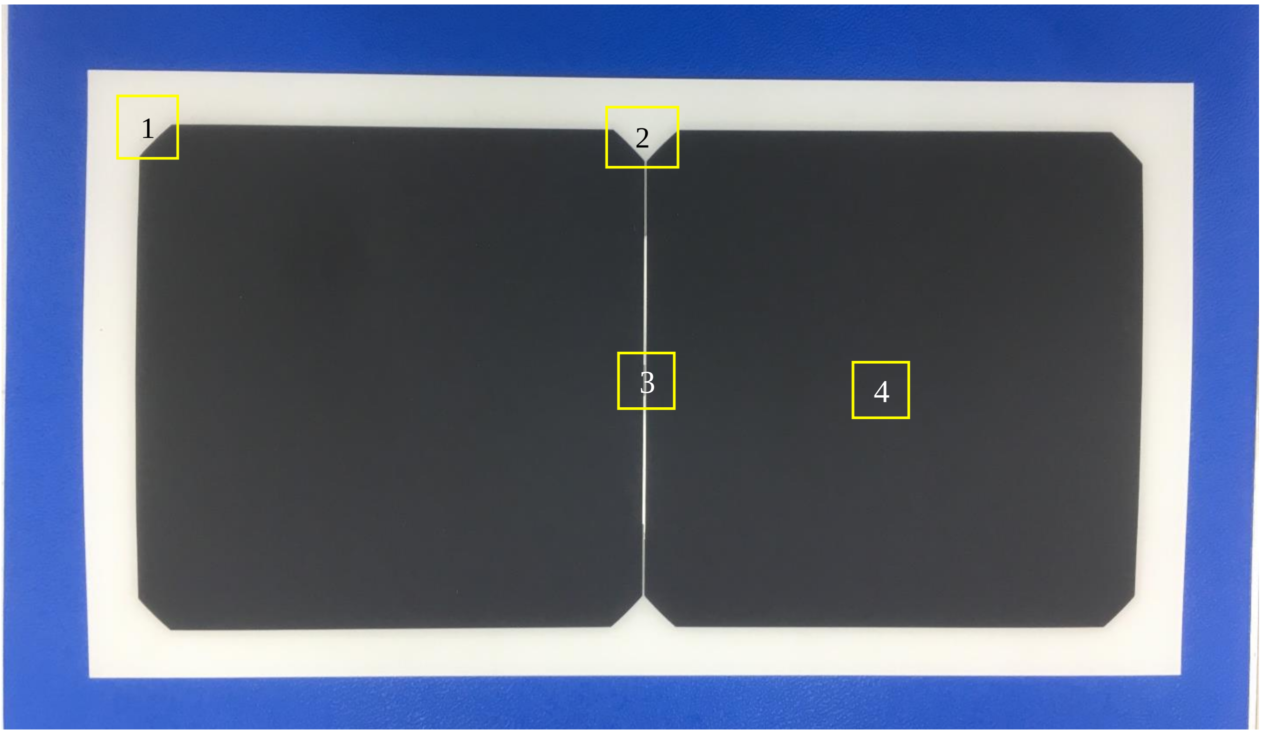

Figure A.10: SMP-CYC-SUE-01

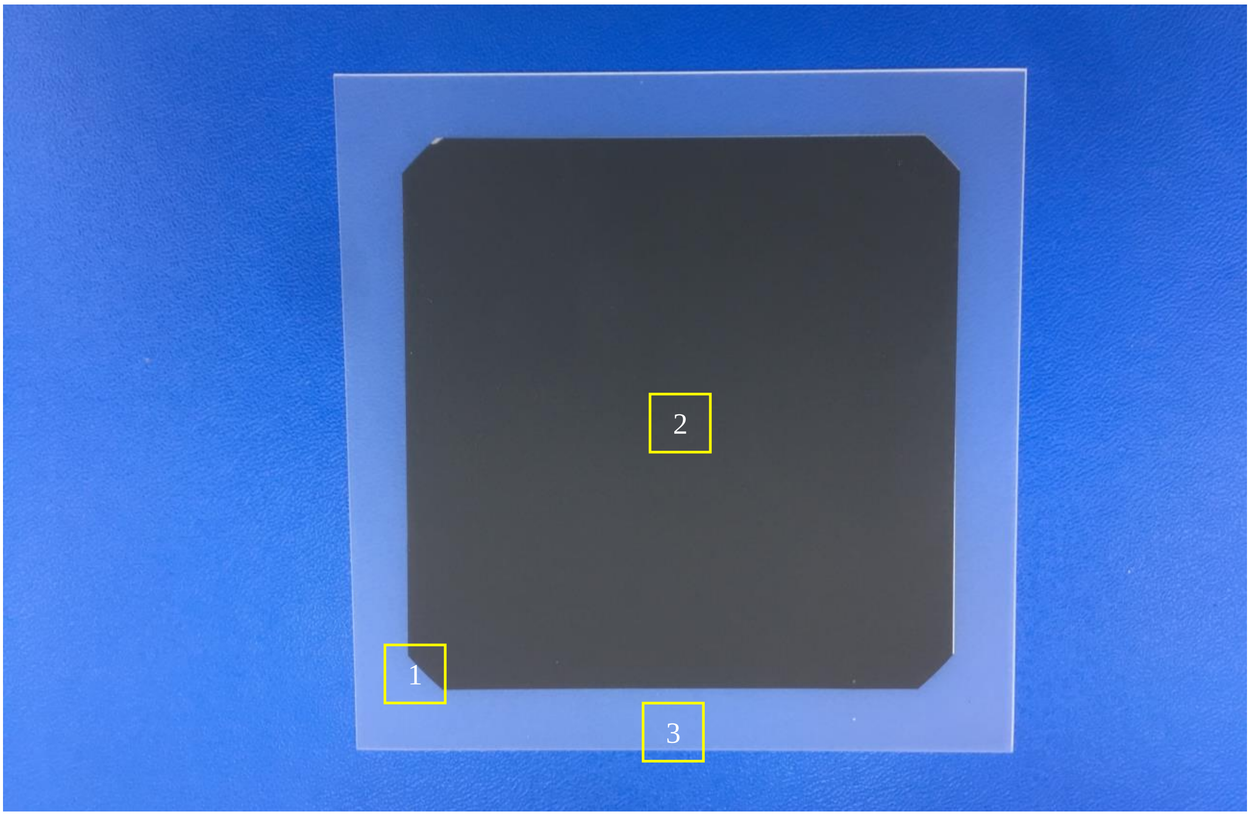

Figure A.11: Sample SMP-CYC-BFE-01

I.9 Appendix I

The bonding of Kapton to the carbon face sheet rough side was intact after cycling (50 cycles, -120 to 120 degrees Celsius) and vacuum oven (120 degrees Celsius, 24h). The bonding on the smooth side did show some bubbles.

Figure A.12: Bonding of kapton on carbon face sheet rough side

EVA (top 4) and Akaflex (bottom four)

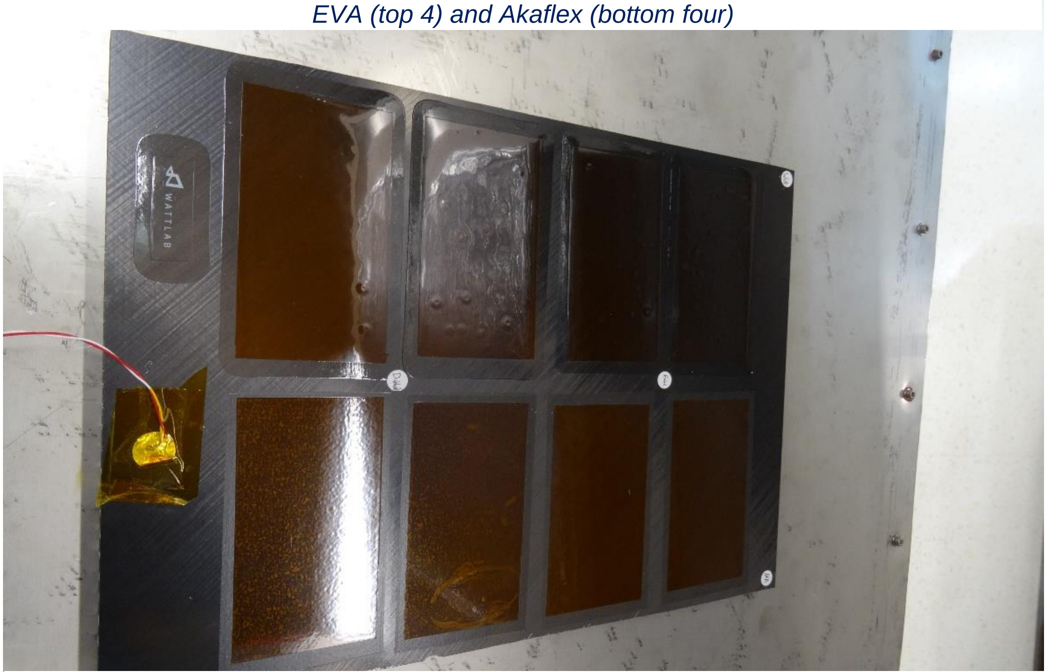

Figure A.13: Bonding of kapton on carbon face sheet smooth side

EVA (top 4) and Akaflex (bottom four)