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THE SOLAR ORBITER THERMAL DESIGN

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ABSTRACT

The Solar Orbiter mission is part of ESA's science program, Cosmic Vision 2020. It will explore the innermost regions of the heliosphere from high heliopsheric latitudes. From a distance of about 0.23AU and a max inclination of about 35 degrees with respect to the Sun's equator the Solar Orbiter will perform high resolution imagery of the sun and in-situ measurements of the heliosphere. At its closest distance to the Sun the spacecraft will experience a sun flux of approximately 28000W/m^2. To protect the spacecraft bus from this flux a sun shield is used. The shield requires innovative design and materials in order to keep both the radiated and conducted heat to a minimum. Additionally, all sun exposed elements such as the high gain antenna and the solar arrays need to be designed for surviving the intense sun flux. This paper will outline the work done on the Solar Orbiter thermal design during its assessment phase. A description of the technical challenges for the overall thermal control system will be given and some of the trade-offs will be discussed. Furthermore, a feasible heat shield design will be presented together with current solutions towards test and verification of the overall system.

INTRODUCTION

The Solar Orbiter mission was submitted to ESA in 2000 and then selected by ESA's Science Programme Committee (SPC) in October 2000 to be implemented as a flexi-mission, with a launch envisaged in the 2008-2013 timeframe (after the BepiColombo mission to Mercury) [1]. The mission was subsequently re-confirmed in May 2002 on the basis of implementation as a mission group together with BepiColombo. A re-assessment of BepiColombo was conducted in 2003, leading to an SPC decision in November 2003 to maintain Solar Orbiter in the Cosmic Vision programme, and to begin an assessment study [2]. In June 2004, ESA confirmed the place of Solar Orbiter in the Cosmic Vision programme, with the objective of a launch in October 2013 and no later than May 2015.

The Solar Orbiter mission has now completed the assessment phase where technical feasibility of the mission has been demonstrated. This paves the way for a start of the definition phase.

The Solar Orbiter mission will provide the next major step forward in the exploration of the Sun and the heliosphere to solve many of the fundamental problems remaining in solar and heliospheric science.

The mission will perform science at distances as close as 0.228 AU, providing first time in-situ measurements of regions so close to the sun and at the same time performing arcsec resolution imagery. Additionally, the Solar Orbiter will provide the first ever out-of-the-ecliptic imaging and spectroscopic observations of the Sun's poles and it will reach latitudes close to 35 degrees with respect to the Sun's equator.

MISSION

Obtaining an orbit with high sun latitudes at close distance requires high energy transfers and can currently not be done with conventional propulsion systems without taking substantial advantage of gravity assist manoeuvres. The Solar Orbiter will use Venus gravity assists to obtain the high inclinations reaching 35 degrees with respect to the Sun's equator at the end of the mission.

To reach the required inclination the Solar Orbiter will be in a 3:2 resonant science orbit with Venus, where at each Venus encounter the orbiter will use a gravity assist to raise the inclination sufficiently. When in science orbit, only orbit correction manoeuvres will be performed and hence only limited propellant is needed. 2 alternative options of reaching the science orbit have been studied during the assessment phase [3]; a mission using solar electric propulsion (SEP) and a mission using chemical propulsion, both launched on a Soyuz-Fregat launch vehicle from Kourou.

The electric propulsion scenario would utilize a Solar Electric Propulsion Module (SEPM). Using two Venus and one Earth Gravity assist manoeuvre would make the cruise duration about 1.8 year. To simplify the thermal design the SEPM would be jettisoned after the science orbit is reached, just before reaching the second Venus gravity assist manoeuvre where the spacecraft will be inserted into the 3:2 resonant orbit.

The chemical option would need two Earth and two Venus gravity assist manoeuvres before being inserted into the science orbit. Some impulsive manoeuvres would be required during cruise and the time to reach the second gravity assist manoeuvre at Venus would be increase to approximately 3.4 years.

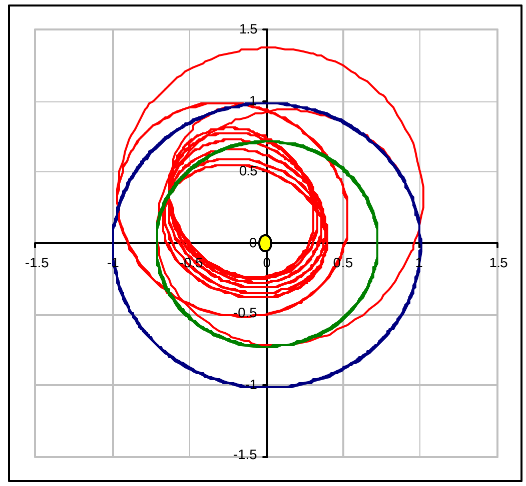

Figure 1 Ecliptic View of the trajectory for a chemical mission, launched in 2013. Red marks the trajectory, blue marks the Earth orbit and green marks Venus orbit

ENVIRONMENT

After insertion to the science orbit the Solar Orbiter will have three orbits with perihelions of 0.228 AU from the Sun. Compared to BepiColombo that is going to Mercury the Solar Orbiter will receive almost twice the sun flux. This implies a sun flux of about 20 solar constants or approximately 28 000 W/m^2. Figure 2 shows how the sun flux varies over the mission timeline.

At close proximity to the Sun also the plasma environment is expected to be severe. This is mainly due to the solar wind that is increasing in density when approaching the Sun.

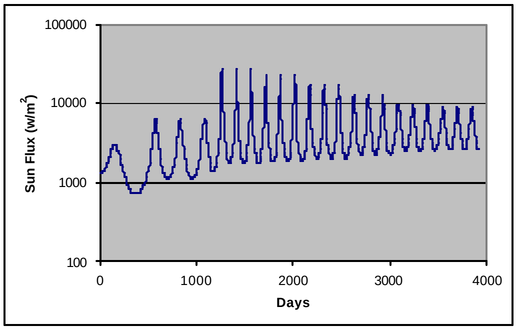

Figure 2 Sun flux as a function of the mission timeline for the 2013 chemical propulsion scenario

The high concentration of solar wind particles together with the high UV-flux is likely to cause increased degradation of materials. This can cause a change in the thermo-optical properties of the materials, leading to a degraded thermal performance.

In order to minimize the effect of this environment the use of organic materials will be avoided. Furthermore, when doing the thermal design, conservative values of the properties will be used such as for the absorption (a) and emissivity (ε). A specific technology development and test plan is also baselined.

THERMAL DESIGN

The thermal design of the orbiter is driven by the sun flux at 0.228 AU, which will cause extreme temperatures and is likely to cause a large heat flux into the spacecraft. The spacecraft bus, payloads and all appendages would need to be designed in order to survive this large amount of heat.

The spacecraft would at times also be at large distances to the sun. This will impose a further challenge as the heat input would be very low and the spacecraft is likely to require substantial amounts of heating power. The required heating power would be very dependent whether a SEP or chemical scenario would be selected. As the furthest distances from the sun are achieved while in cruise the SEPM could provide the heating power necessary during cruise. For the chemical scenario the spacecraft would need to provide the necessary heating power at up to 1.5 AU. This would require heating power in excess of 300 W largely driving the solar array sizing. The use of louvers have been investigated to reduce the heating, however it has been decided to keep the system simple, thus exclude mechanical systems such as louvers.

At close distances to the Sun the spacecraft will have to be shielded. The thermal design would be very similar for both the SEP and the chemical option due to the jettisoning of the SEPM. After jettisoning the two spacecraft will be very similar in overall configuration the main difference being the size, the propulsion system and the solar array sizing. The two configurations are shown in Figure 3 and Figure 4.

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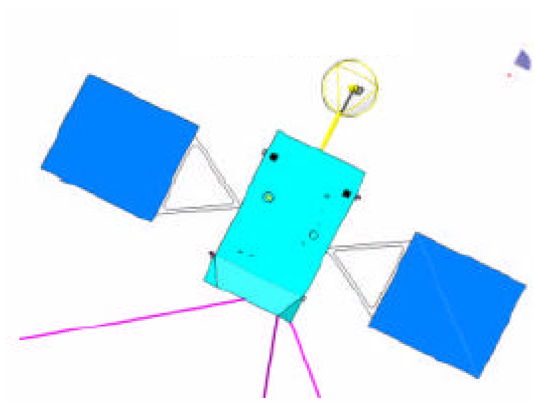

Figure 3 A potential configuration of the orbiter showing the shield and the instrument apertures located in the shield

Most components inside the spacecraft bus require room temperature for operation. To achieve this temperature the spacecraft will utilize a sun shield. This sun shield will cover the spacecraft bus and some of the external components and it will contain aperture openings providing the required field of view for the remote sensing instruments. Some of the appendages require being sun-exposed thus requiring specific technology development.

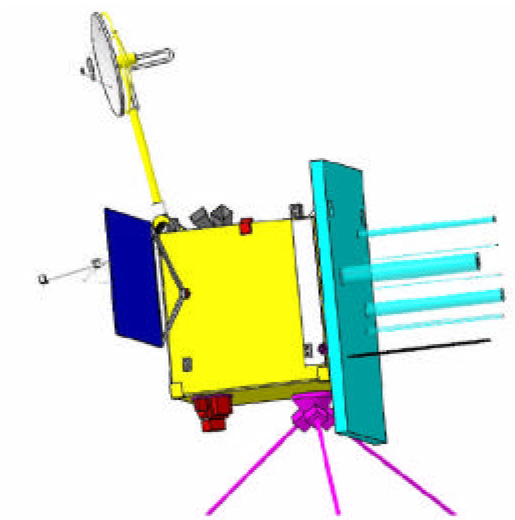

Figure 4 Orbiter configuration after jettisoning the SEPM

The spacecraft will need to be sun pointing at all times to keep the lateral faces from being exposed to the Sun. A robust attitude control system is therefore needed ensuring that even in case of failure the spacecraft will quickly re-point towards the sun. Nevertheless, for short periods of time, off pointing from the sun could happen for instance if the actuators are responding incorrectly. To ensure that the spacecraft can sustain off pointing for shorter periods, the radiators on the lateral faces would be covered with optical solar reflectors (OSR).

<u>Sun shield design</u>

The sun shield will provide a more benign thermal environment for the spacecraft and it will be one of the mission critical elements of the Solar Orbiter. Therefore a large proportion of the work during the assessment study went into finding feasible solutions for the sun shield. A number of trade-offs were conducted on the shape of the shield, the size, front layer materials etc. As the launch margins (in particular for the SEP option) are rather tight all these trade-offs where conducted on criteria such as mass, complexity and interface to the payload.

The size of the shield is given by the size of the orbiter. The shield needs at least to be large enough to cover the orbiter and the external components in a nominal sun pointing mode. As the closest distance the sun will cover 2.4 degrees of the sky and hence the shield would have to extend at least 1.2 degrees off from the most protruding element. This is clearly shown in Figure 4. In addition the spacecraft should be designed to sustain some nominal off-pointing. The shield will therefore extend at least 2.5 degrees of the most protruding external component.

The shape of the shield could be chosen as to minimize the absorbed sun flux. For instance, through having a conical shield or a v-groove design the temperature of the first layer could be greatly reduced, thereby potentially making the shield more effective. However, the drawbacks of having such a design would be the mechanical complexity. Furthermore, it would complicate the interface to the payload, which requires a clear view through the shield. This interface would be greatly simplified if a flat surface would be used. Hence this simpler geometry is the baseline for the current shield alternatives.

The thermo-optical properties of the front-shield are dictating the temperature of the shield and the sun flux absorbed. Figure 5 shows how changing the a/ε affects the temperature at 0.23 AU. Having a low temperature of the front layer would simplify the thermal design and limit the heat flux into the spacecraft. Several alternatives have been investigated such as coatings, ceramics, metals, optical solar reflectors (OSR) etc.

Only non-organic coatings would be possible to use due to the UV-flux. These coatings could potentially have a rather low a/ε value, although the End of Life (EOL) value would be uncertain. With a typical white coating it is assumed that the EOL a/ε could be as about 0.6. Nevertheless it would require extensive testing in a representative environment to make sure that this value is not exceeded, as the coating is likely to not sustain very high temperatures. Using coating on a metal will also have the problem that if some part of the metal gets exposed to direct sun light the metal could heat up drastically and thus the shield could fail.

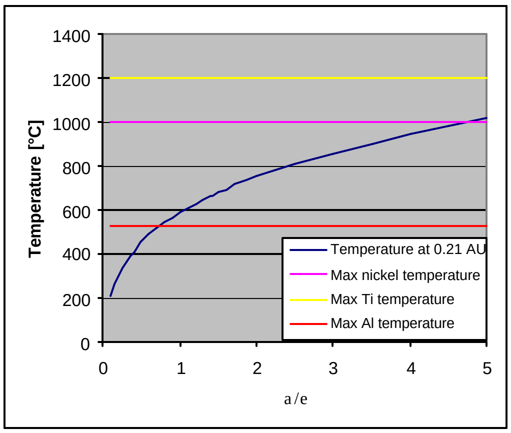

Figure 5 Temperatures for different alpha/epsilon values at 20 solar constants. The graph also shows typical maximum values for some candidate materials

Metals have typically high a/ε values, which would cause very high temperatures on the front shield. Based on this a metallic front layer was discarded.

Another option as a front layer is the use of ceramics. A ceramic material should degrade less than coatings, and an interesting alternative could be Alumina-Boria-Silica (ABS). However, as with any white material the ceramics would also have an uncertain degradation in a Solar Orbiter environment. Hence, degradation tests in a representative environment would be needed.

Materials with a/ε close to 1 could also be used. Typically this could be a Carbon-Carbon based shield. This material is also the baseline for NASA's solar probe [4]. This alternative would have a much higher front layer temperature than some of the more white alternatives. However, the use of such materials could simplify the AIV/AIT concept as there would be less need for testing degradation and the testing of the shield could be done with infrared lamps instead of a representative sun-flux.

From the front layer the heat transmitted to the spacecraft must be significantly reduced. This can be done by introducing several highly infrared reflecting layers or by using a high temperature MLI similar to the ongoing development in the BepiColombo programme. By having sufficient number of layers and by utilizing gaps with view factor to space between layers, the heat input to the spacecraft can be greatly reduced.

After performing thermal analysis of several options it is clear that several feasible alternatives exist for the sun shield, both with black or white (grey) first layer. As the shield is such a critical element for the mission it was decided to keep alternative designs in order to minimize development risk. The sun shield mass will vary greatly dependent on the front layer and using a black sun shield will be heavier than a white sun shield due to the need for more shielding due to the higher temperatures.

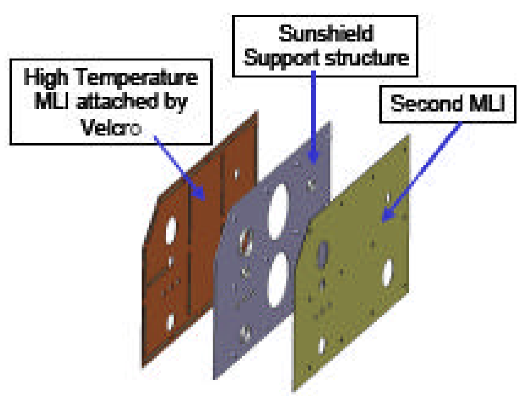

Figure 6 One sun shield design alternative

A white front cover solution is shown in Figure 6 [2]. This solution uses a High Temperature MLI similar to what is expected for the BepiColombo mission. This MLI would use ABS as the front layer, which would have an a/ε of about 0.45 and high temperature resistant metallic foils such as titanium. The MLI would be supported by a sun shield support structure consisting of aluminium honeycomb. On the backside of this support structure a second and more traditional MLI could be used. Using this type of shield the heat into the spacecraft would be about 30 W and the temperature of the front layer would be 420 degrees. Using foils with gaps instead of the High Temperature MLI would potentially reduce the input heat even further. However, struts would then be required to keep the foils at the required distance from each other and the shield would grow in thickness.

Payload thermal design

The solar orbiter payload will be provided by institutes after an Announcement of Opportunity (AO). In preparation for this future AO a reference payload has been used [5]. The reference payload has been defined in close cooperation with the Solar Orbiter Payload Working Group (PLWG) that is organized by the scientific community.

The payload consists of two groups of instruments. The remote sensing and the in-situ instruments, where most of the in-situ instruments can be located behind the sun shield while the remote sensing instruments will be observing the Sun and thus would need openings in the shield.

The remote sensing instruments have a range of aperture sizes and Table 1 shows the current aperture sizes for the different instruments and the assumed absorbed sun flux.

In order to have an effective thermal control, each instrument is being controlled individually. There are two instruments that have substantially larger apertures than the others Visible-Light Imager and Magnetograph (VIM) and the Coronograph (COR). The COR has an external occulter that will run at very hot temperatures. The philosophy is to have this radiatively cooled and limit the conductivity of the occulter to the rest of the structure. Behind the occulter there will be a sun rejection mirror reflecting most of the sun flux back through the instrument. By utilizing this technique, only a minor portion of the incoming light is absorbed in the instrument.

Instrument	**Aperture diameter (mm)**	**Heat load (W)**
VIM[1]	125 - 180	19 - 40
EUS	70	91
EUI (HRI)	20	3 × 10
EUI (FSI)	20	10
COR	180	62
STIX	40	~0
Total	-	**212 - 235**

Table 1 Heat load for the different remote sensing instruments. [1] The VIM aperture size is still not fully defined but will be within the sizes in the table.

In order to limit the sun flux into the VIM instrument an external filter is baselined, limiting the transmitted flux to about 5 %. The absorbed flux would be about 10%. This specific filter will be subject to a technology development activity and the specific characteristics and material is therefore not yet selected.

As the Spectrometer/Telescope for Imaging X-rays (STIX) instrument is measuring in X-Ray an opaque surface can be placed in front of the aperture. Depending on the sun shield material the STIX could therefore be located behind the front cover of the sun shield. To further limit the heat input infrared reflective screens could be placed in the baffle.

Extreme Ultra Violet Imager (EUI) consists of several small apertures, which have a long baffle in front of a filter. This filter would be able to reflect most of the incoming heat flux back into the baffle thus limiting the heat into the instrument.

The Extreme Ultra Violet Spectrometer (EUS) would use a heat stop to reject the majority of the heat in the instrument. The heat stop will be placed after the primary mirror.

Each instrument will have a thermal baffle through the heat shield. Doors and related closing mechanisms will be provided for each instrument in order to avoid contamination of optical surfaces during thrusting and in periods when the instruments are not operating. These doors will have to be designed to sustain the full sun flux at 0.23 AU and it is therefore likely that they will be made of the same materials as the sun shield. These doors would then also have a purpose for thermal protection in case of severe off-pointing or in case of filter failure such as for VIM and the EUI.

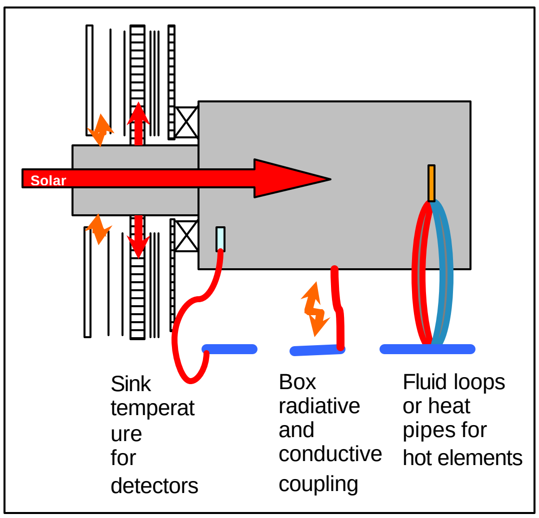

Figure 7 General thermal interface concept of the remote sensing instruments.

All instruments have a similar thermal interface to the spacecraft. This is shown in Figure 7. In each instrument there will be hot elements such as heat stop, primary mirrors etc. These would have a direct interface to a dedicated hot radiator through either use of heat pipes, fluid loops or heat straps. Some of the instrument will have detectors requiring very low operating temperatures. For instance the EUS detector is an Active Pixel Sensor that requires -80 °C. These low temperature detectors need dedicated cold radiators in order to avoid complex active cooling. The remaining components such as secondary mirrors and optical bench can be thermally coupled to the instrument box and commonly cooled.

<u>Appendages</u>

The Radio Plasma Wave experiments (RPW) antennas, the High Gain Antenna (HGA) and the Solar Arrays are appendages that will be exposed to high solar flux and hence would require dedicated technology developments effort.

The RPW antennas need to be at an angle of 90 degrees in respect to each other and located in a position where they will have a uniform sun illumination. This implies that they will be exposed to the sun at the closest distances and thus would reach high temperatures. The Strawman design of the RPW antennas is using stacers, similarly to what is used for the STEREO Waves Experiment [6]. Although, the antennas will reach hot temperatures, the conducted heat into the spacecraft will be rather limited and the challenge is more to find a suitable material for the antennas themselves.

A high temperature High Gain Antenna (HGA) is under development in frame of the BepiColombo project. A potential material for such an antenna is based on SiC, which can sustain high temperatures. For the Solar Orbiter this HGA antenna would need to withstand much higher flux if used at the close perihelion passes. Alternatively the BepiColombo antenna could be used down to 0.3 AU and stowed at distances closer than this. This would however, increase the required telemetry data rate and would slightly decrease the scientific return as no science data would be available during the closest perihelion passes and the scientist could therefore not be in a loop commanding the spacecraft to observe a specific position on the sun based on the data received. One of the main challenges for qualifying the antenna at the closest distance is to develop a feeder for these temperatures. At certain angles the temperature of the feeder could exceed 300 degrees. Hence a dedicated high temperature feeder development is on its way where alternative materials such as titanium, SiC and CFRP is being investigated.

The Solar Arrays are very challenging components of the Solar Orbiter Design. The Solar Arrays would not only need to sustain close distances to the Sun, but would also need to be able to produce the necessary amounts of power at distances far from the Sun. This is in particular important for the chemical option where the orbiter might be as far as 1.5 AU from the sun. The solar arrays can limit the temperatures by tilting them so that the incident angle of the sun is larger. However, using very large incident angles would potentially cause major difficulties such as edge effects, internal reflection in solar array and uncertainties in the degradation of the cells. Furthermore, at large incidence angles slight change in attitudes could cause large changes in temperatures. The decision is therefore to use a solar array that is operating at an incidence angle of less than 70 °. Using triple junction GaAs cells a fully populated solar array with only cell would be at a typical temperature approaching 300 °C if the angle is about 70 degrees. This is clearly an unacceptable operating temperature and hence using Optical Solar Reflectors (OSR) substituting the solar cells is envisioned. The substrate for the solar arrays will be made of carbon reinforced carbon to sustain higher temperatures. Even so, the maximum tolerable temperature is expected to be lower than 230 °C and the Cell to OSR ratio would then need to be between 40% - 50 %. Figure 8 shows how the temperature of the solar array changes with the cell to OSR ratio.

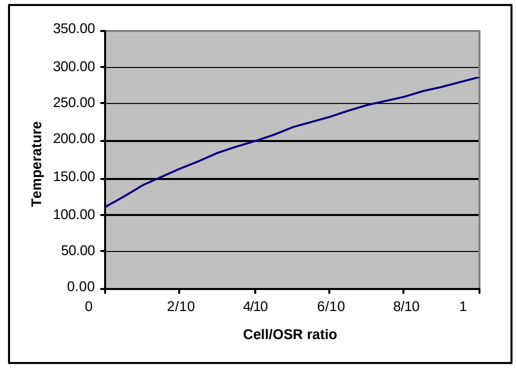

Figure 8 Temperature for different cell to OSR ratios

AIV/AIT

The use of sun shield causes room temperature environment inside the spacecraft. The thermal testing of the components inside the spacecraft that are not subject to sun flux can therefore be limited to thermal cycling and thermal vacuum in standard facilities. For the sun exposed components this is not sufficient. However, testing the entire sunshield at 20 solar constants will also be prohibitively expensive as no facility exists able to provide this large beam width (larger than 2 by 2 m) at those flux levels. The tests would therefore need to be made on representative samples.

All materials that will be sun exposed at 0.228 AU would have to undergo degradation testing in relevant environment. This would imply a test under vacuum with presence of a representative plasma environment. Existing facilities could be modified to provide this capability although at a more limited sample size. Having a smaller size for degradation testing should be acceptable and it is envisioned that such tests will be performed for all materials that would be exposed to the sun.

Test on components would usually be required if the shape is complicated and shadowing and reflection effects are important. The sun shield is typically such a shape with thermal baffles and doors making the shape complicated. The largest aperture of the sun shield is in the order of 20 cm in diameter. It might therefore be that this would be sufficient size to perform the necessary testing of the shield. An alternative solution would be to build a representative smaller model of the shield (or parts of the shield) and use this in order to perform the testing. It is evident that testing is more complicated for a reflective sun shield such as the white cover compared to an absorptive shield. Additionally, with a black sun shield the testing could be performed largely by using infrared laps instead of a full sun spectrum.

Both the HGA and the solar array will go through testing with BepiColombo. However, as the flux is much higher it will still be necessary to re-qualify these components after potential design changes. The reflector used in HGA antenna would be larger than 1 m. However for the HGA it is envisioned that testing could be done on a smaller model. For the solar arrays the critical areas of the arrays could be tested, however this would not require a large beam.

CONCLUSION

The Solar Orbiter will face extremely challenging thermal environment during its mission lifetime. The finalized assessment study has shown that there exist technically feasible alternatives for the thermal control system. This thermal control will employ a sun shield to limit the heat input to the spacecraft. Both using a black and a grey front layer have been shown as viable solutions for the heat shield, which could be implemented while keeping complexity to a minimum. The remaining sun exposed components would require some specific technology development, but can be largely based on the heritage from BepiColombo.

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