The Solar Orbiter Mission and Design Recommendations

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A short overview is given of the Solar Orbiter mission. First, the key scientific aims of the mission are briefly described. As the mission profile has consequences on the design of the payload instruments and their calibration, the mission design is described. Possible implications and problems for the cleanliness and the calibration stability of the instruments are outlined. Some solutions are discussed.

28.1 Introduction

As one of ESA's future solar missions, the Solar Orbiter was selected in October 2000. It is planned to be launched around 2011. Similar to SOHO, its payload will consist of a suite of instruments to investigate the Sun and the heliosphere. It is thus appropriate to consider here if any lessons learned from SOHO could be usefully transferred to Solar Orbiter. We take an excerpt from the "Solar Orbiter Assessment Study Report" [ESA, 2000] in order to give an overview of the mission perspectives, the prospective payload and the mission orbit all of which have specific implications for the cleanliness and stability of calibration of the instruments. We review some of the cleanliness procedures adopted for the SOHO project which may be applicable to the Solar Orbiter.

28.2 Solar Orbiter Scientific Aims

The Sun’s atmosphere and the heliosphere represent uniquely accessible domains of space, where fundamental physical processes common to solar, astrophysical and laboratory plasmas can be studied in detail and under conditions impossible to reproduce on Earth or to study from astronomical distances. With the results from missions such as Helios, Ulysses, Yohkoh, SOHO, ACE, and TRACE we have advanced enormously our
understanding of the Sun, its corona, the associated solar wind and the three-dimensional heliosphere. After these missions, we have reached the point where further in-situ measurements, now much closer to the Sun, together with high-resolution imaging and spectroscopy from a near-Sun and out-of-ecliptic perspective, promise to bring about new insights and perhaps major breakthroughs in our understanding of solar and heliospheric physics. The Solar Orbiter will, through a novel orbital design and innovative instrumentation, provide the required measurements and observations, and will, for the first time, explore the uncharted, innermost regions of our solar system. It will study the Sun from close-up (45 solar radii, or 0.21 AU), fly by the Sun and examine the solar surface and the space above from a co-rotating vantage point. It will provide images of the Sun’s polar regions from heliographic latitudes as high as 38°. The near-Sun interplanetary measurements, together with simultaneous remote-sensing observations of the Sun, will permit us to disentangle spatial and temporal variations at the solar surface, the corona and inner heliosphere. They will allow us to understand the characteristics of the solar wind and energetic particles in close linkage with the plasma conditions in their source regions on the Sun. By approaching as close as 45 solar radii, the Solar Orbiter will view the solar atmosphere with unprecedented spatial resolution (35 km pixel size, equivalent to 0.05’’ from Earth). Over extended periods of time the Solar Orbiter will deliver images and data of the polar regions and the side of the Sun not visible from the Earth at that time. The scientific goals of the Solar Orbiter mission can be summarized as follows. It will determine, in-situ, the properties and dynamics of plasma, fields and particles in the near-Sun heliosphere. It will investigate the fine-scale structure and dynamics of the Sun’s magnetized atmosphere, using close-up, high-resolution remote sensing. Using the solar co-rotation passes, it will identify the links between activity on the Sun’s surface and the resulting evolution of the corona and inner heliosphere. Finally, it will observe and fully characterize the Sun’s polar regions and equatorial corona from high latitudes. The Solar Orbiter will achieve its wide-ranging aims from a specially-designed orbital trajectory and using a suite of sophisticated instruments.

28.3 Mission Profile and Spacecraft Design

The Solar Orbiter scientific requirements define a mission profile in terms of orbital parameters, launch windows, payload mass, etc., that is the driver for the spacecraft design. The required final orbit is characterized by a perihelion distance as close as possible to the Sun with a high orbital inclination with respect to the solar equator. To achieve such a trajectory, a long transfer-period is required and low-thrust solar electric propulsion (SEP) will be used. In addition, several swing-bys at Venus are necessary to make use of planetary gravity assists. The projection of the orbit on the equatorial plane is shown in Figure 28.1. The mission profile is composed of three phases:

1. The cruise phase, which starts at spacecraft separation from the launcher and ends at the start of scientific operations (some science may be performed during the cruise phase).

2. The nominal mission phase, during which the main scientific mission is performed.

3. The extended mission phase, when further gravity assist manoeuvres will allow higher-inclination observations.
During the cruise phase (0 to 1.86 years) there are SEP-thrust phases, with durations ranging from 6 to 105 days, and Venus swing-bys driving the semi-major axis changes and the inclination increase. During the nominal mission (1.86 to 4.74 years, duration: 2.88 years), the orbit is typically of order 150 days, taking the spacecraft from about 0.2 AU to 0.9 AU. There are two Venus swing-bys in this 7-orbit phase taking the orbital inclination to over 30°. During the extended mission (4.74 to 7.02 years, duration: 2.28 years), there are two further Venus swing-bys over six orbits which provide a continued increase in the orbital inclination. The important orbital parameters (perihelion distance and heliographic latitude) are shown in Figures 28.2 and 28.3. The orbit assures a close distance to the Sun relatively early during the mission, while a high, scientifically satisfactory, heliographic latitude is achieved towards the final phase of the mission. The celestial coordinates of the Sun, Venus and Earth yield a launch window of three weeks in every nineteen months. Figure 28.4 shows the overall configuration of the Solar Orbiter spacecraft during scientific observations. The spacecraft body has a cross-section of 1.6 m ($Y$) × 1.2 m ($Z$) and its length is 3.0 m ($X$). The $+X$ side, which faces the Sun (±30°), is covered by a thermal shield shadowing the spacecraft body. In order to minimize the energy input from the Sun, the cross-section of the spacecraft in the $Y$, $Z$ plane has been minimized. The spacecraft is of modular design, with a Service Module (SVM) and a Payload Module (PLM). The ±$Y$ sides of the PLM accommodate the cruise solar arrays (two wings, three panels per wing) and the top-shield radiators. The optical instruments are right beneath...
the top shield and are pointed in the +X direction. They are isostatically attached to the central cylinder and may be cooled by radiators to cold space on the ±Z sides. The instruments’ electronic boxes are mainly mounted on the bottom panel of the PLM. The SVM accommodates the orbit solar panels and the solar electric propulsion equipment radiators on the ±Y panels. The high gain antenna (HGA) and the thrusters are attached to the +Z panel. In order to cope with different attitudes and distances from the Sun, both solar arrays incorporate a one-degree-of-freedom (1-DOF) driving mechanism. The cruise solar array can be jettisoned. The HGA mast (2 m long) is mounted on a 2-DOF mechanism and ensures coverage throughout the mission. The spacecraft is 3-axis stabilized and always Sun pointed (X-axis), except during SEP firing. The Sun-pointing face of the Solar
Orbiter is as small as is possible in order to minimize the thermal input from the Sun. The remaining spacecraft surfaces will carry radiators. The extreme environments encountered by the spacecraft throughout the whole mission drive the thermal design of the Solar Orbiter. At one extreme, the spacecraft is orbiting the Sun at distances as close as 0.21 AU. At the other extreme, the spacecraft is as far as 1.21 AU from the Sun. Another important factor is the electrical propulsion that intermittently generates a considerable amount of heat inside the spacecraft. Therefore the thermal design has to accommodate a wide range of heat load levels and locations. The basic design concept consists of shading as much as possible every part of the spacecraft from the Sun at the closest approach to the Sun and during the ion-thruster firings in the cruise phase. To this aim, a sunshield has been included at the +X side of the spacecraft. It extends on the ±Z sides to make sure that the spacecraft is shadowed even when the +X axis is offset by as much as 10° from the Sun. (The ±Z offset of 10° occurs during the SPT firing at the closest approach to the Sun (0.33 AU) during the cruise phase.)

28.4 The Strawman Payload

The payload must be state-of-the-art, withstand the considerable thermal load at 45 solar radii, comply with the general requirements of a low-mass, compact and integrated
Table 28.1: In-situ heliospheric instrumentation.

<table>
<thead>
<tr>
<th>Name</th>
<th>Acronym</th>
<th>Measurement</th>
<th>Specifications</th>
<th>Mass / kg</th>
<th>Size / (cm×cm×cm)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Solar Wind Plasma Analyser</td>
<td>SWA</td>
<td>Thermal ions and electrons</td>
<td>0 to 30 keV/Q; 0 to 10 keV</td>
<td>6</td>
<td>20 × 20 × 20</td>
</tr>
<tr>
<td>Radio and Plasma Wave Analyser</td>
<td>RPW</td>
<td>AC electric and magnetic fields</td>
<td>μV/m to V/m; 0.1 nT to μT</td>
<td>10</td>
<td>15 × 20 × 30</td>
</tr>
<tr>
<td>Radio Sounding</td>
<td>CRS</td>
<td>Wind density and velocity</td>
<td>X-band; Ka-band</td>
<td>1</td>
<td>10 × 10 × 10</td>
</tr>
<tr>
<td>Magnetometer</td>
<td>MAG</td>
<td>DC magnetic field</td>
<td>to 500 Hz</td>
<td>1</td>
<td>10 × 10 × 10</td>
</tr>
<tr>
<td>Energetic Particle Detector</td>
<td>EPD</td>
<td>Solar and cosmic-ray particles</td>
<td>Ions and Electrons 0.01 to 10 MeV</td>
<td>4</td>
<td>10 × 20 × 20</td>
</tr>
<tr>
<td>Dust Detector</td>
<td>DUD</td>
<td>Interplanetary dust particles</td>
<td>Mass (g): 10⁻¹⁶ to 10⁻⁶</td>
<td>1</td>
<td>10 × 10 × 10</td>
</tr>
<tr>
<td>Neutral Particle Detector</td>
<td>NPD</td>
<td>Neutral hydrogen and atoms</td>
<td>0.6 to 100 keV</td>
<td>1</td>
<td>10 × 10 × 20</td>
</tr>
<tr>
<td>Neutron detector</td>
<td>NED</td>
<td>Solar neutrons</td>
<td>e &gt; 1 MeV</td>
<td>2</td>
<td>10 × 10 × 10</td>
</tr>
</tbody>
</table>

design, make use of on-board data compression/storage and require a modest data transmission rate. The selected payload, which meets the solar and heliospheric science objectives of the mission, encompasses two instrument packages: in-situ heliospheric instruments and remote-sensing instruments. The design of these instrument packages builds upon heritage of the previous Helios, Yohkoh, SOHO, Ulysses, WIND, ACE and TRACE missions.

28.4.1 In-situ Heliospheric Instruments

The in-situ instruments on Solar Orbiter constitute a package of sensors suited to a comprehensive study of the solar-wind plasma, its constituents, motions, and energetic processes. Additional instrumentation is included for the detection of neutral atoms, neutrons, circumsolar and interplanetary dust, solar radio emission and local magnetic fields. A list of the in-situ instruments with their main specifications is given in Table 28.1.

28.4.2 Remote-sensing Solar Instruments

An integrated ensemble of high-resolution remote sensing instruments is proposed for the Solar Orbiter: an EUV full-Sun and high-resolution imager, a high-resolution EUV spectrometer covering selected emission lines from the chromosphere to the corona, a high-resolution visible-light telescope and magnetograph, an EUV and visible-light coronagraph and a radiometer. Table 28.2 lists the instruments of the remote-sensing package and their main specifications.

All the disk-observing instruments will resolve small-scale dynamical processes in the solar atmosphere. The combination of instruments proposed provides a complete set of
28.5. Implications and Constraints for the Payload

Technical challenges to the instruments of the Solar Orbiter payload are mainly imposed by the mission design. We see four major causes for potential technical problems:

1. The varying orbital distance to the Sun results in a thermally-variable and thus difficult environment.
2. The close approach to the Sun causes severe technical problems because of the increased radiation flux.
3. Limited communication with the spacecraft requires autonomous operation and storage of large data volumes.
4. The mass resources are limited, yet high-resolution instrumentation requires additional thermal hardware (to cope with the thermal environment mentioned under point 1).

Resolution of these problems must be achieved while aiming at stability of the spacecraft and, as a result, stability of calibration.

28.5.1 What Can Be Learned from SOHO? Recommendations for Solar Orbiter

The stability of the calibration of SOHO’s optical and UV instruments is a result of the cleanliness program imposed on instruments and spacecraft. Among others, the material selection and design features were the main contributors to success: see the “Summary of Cleanliness Discussion” in this volume [Schühle et al., 2002]. Although in most cases it is

### Table 28.2: Remote-sensing instrumentation.

<table>
<thead>
<tr>
<th>Name</th>
<th>Acronym</th>
<th>Measurement</th>
<th>Specifications</th>
<th>Mass / kg</th>
<th>Size / (cm×cm×cm)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Visible-light Imager and Magnetograph</td>
<td>VIM</td>
<td>High-res. disk imaging and polarimetry</td>
<td>Fe 630 line</td>
<td>26</td>
<td>30 × 40 × 120</td>
</tr>
<tr>
<td>EUV Imager and Spectrometer</td>
<td>EUS</td>
<td>Imaging and diagnostics of TR and corona</td>
<td>EUV emission lines</td>
<td>22</td>
<td>30 × 15 × 140</td>
</tr>
<tr>
<td>EUV Imager</td>
<td>EXI</td>
<td>Corona imaging</td>
<td>He and Fe Ion lines</td>
<td>36</td>
<td>40 × 40 × 250</td>
</tr>
<tr>
<td>Ultraviolet and Visible Light Coronagraph</td>
<td>UVC</td>
<td>Imaging and diagnostics of the corona</td>
<td>Coated mirror coronagraph</td>
<td>17</td>
<td>20 × 20 × 50</td>
</tr>
<tr>
<td>Radiometer</td>
<td>RAD</td>
<td>Solar constant</td>
<td>Visible light</td>
<td>4</td>
<td>11 × 11 × 22</td>
</tr>
</tbody>
</table>

multi-wavelength measurements required to understand the Sun’s magnetized atmosphere from below the photosphere up into the extended corona.
not possible to remove all actuators from the optically critical volumes, the materials used for these actuators can be critically screened and space-conditioned and separated from the components of the control electronics. SEM’s degradation, though small, underlines the importance of the correct location of instrument apertures with respect to other spacecraft hardware. Also within individual instrument enclosures, proper hardware placement is of importance. For instance, where a rotating shutter is used close to a cooled CCD with provision for periodic warm up, provision must be made to ensure that the shutter sector used for this latter phase differs from that normally used during observations.

It is surprising that the design of the Solar Orbiter spacecraft as described in the reference did not build more on the SOHO experience. A good feature of the latter was the PLM box structure supporting the instruments on five sides, with a sunshield over the upper end protecting the top-mounted instruments and the PLM structure from excessive solar heating. The space between the upper end of the structure and that shield was closed by thermal blankets and vented at a controlled location away from any instrument aperture. All instrument thermal blankets were designed to eject outgassed products to the $-X$ direction (or outboard where that was impossible).

Solar Orbiter presently has a cylindrical structure with the optical instruments mounted on its surface under the shield, but apparently with their front faces coincident with that of the shield. The non-optical instruments are mounted on the forward face of the cylinder. It seems that any outgassing from these latter instruments, the structure and the hot shield may have an easy path to the optical instrument apertures. We would suggest that the sunshield be located, say, 30 cm ahead of the optical instrument aperture plane and the top end of the structure, which should be closed at $+X$ and positively vented near $-X$. The space between each optical instrument and the shield should have a structure to deflect outgassing from the central instruments to the sides of the optical apertures; this structure need not be more substantial than a membrane (kapton, if thermal considerations are acceptable). If each optical instrument has an external door, then the actuator should be mounted outboard of the aperture. The instrument could carry the outgassing shield mentioned in this paragraph mounted on the two inboard sides. Outgassing from the $-X$ side of the sunshield will depend on material selection, which needs attention. However, taking account of the long time between launch and first observations should do much to clean up the shield.

The mission profile shows periods when the internal instruments’ heaters could be powered to “bake-out” optics before observations, there being ample power available when the SEP is not operating. Figure 28.5 shows the possible temperature evolution for the shield, based on tables 7-1 and 17-1 and figure 11-3 of the reference. Notice that real-time data telemetry and commanding are not possible during periods when the Sun-spacecraft distance is less than 0.5 AU. Ideally, in this bake-out phase, venting of the products to space should be enhanced. The entrance aperture is not suitable for this as its area is small and, in molecular flow path terms, it will be far from the baked detector. A large, baffled vent offering a flow rate of $1000 \, \text{l s}^{-1}$, or even a venting door close to the baked optics, should be considered. Some of the remote sensing instruments have large primary mirrors in full Sun. This means they will face heat input of $952 \, \text{W m}^{-2}$ at aphelion (1.2 AU) to $34 \, \text{kW m}^{-2}$ at perihelion (0.2 AU) and will be radiatively cooled for the hot case. The large temperature excursion in flight is going to be a severe test of the mountings and interfaces between the mirrors and the metal supports and perhaps of the thermal stability of the mirrors themselves. Compensation for the thermal excursions cannot be accomplished by
28.5. Implications and Constraints for the Payload

Figure 28.5: Possible temperature evolution for the shield. Also shown are periods (example 600 to 620 days) when internal instruments’ heaters could be powered to “bake out” optics before observations, there being ample power available when the SEP is not operating.

Heater power. Concepts for variable heat transfer from telescopes to radiators are needed, for example regulated heat pipes. Heat-rejection filters (different for EUV than for visible), which reduce heat input to instruments but are stable to radiation, could also limit the ultraviolet bandpass inside the instrument, resulting in reduced contamination sensitivity. In the same way, selective mirror coatings, reflecting only the wanted part of the solar spectrum while transmitting most of the unused spectrum into a heat sink, would separate the heat and ultraviolet radiation from mirrors further down in the optical path and from the focal planes. Offpointing in cruise phases can be as much as 30° at 1 AU and 10° at the closest SEP operation with some risk of asymmetric heating of the instrument bodies. Such excursions are similar to those experienced by SOHO during the loss-of-attitude, which resulted in redistribution of contaminants inside instruments. In the case of SUMER this resulted in a loss of responsivity of 31% (on the average), which otherwise would have not experienced any degradation. Thus a clean internal design is not sufficient under these circumstances. A large venting area without UV illumination, preferably with instrument heating, can reduce this degradation effect.

One solution to reduce the large difference in heat input between open and closed aperture doors is to use SUMER-type doors which transmit visible light (ideally longer than 300 nm) at all times and so remove the transient thermal effects from the exposures. Yet the offpointing problem remains, as does the variation of input between aphelion and perihelion in most operational orbits, assuming that the primary mirror is cooled by a radiator that can dump the entire heat load irrespective of the orbit position. The volumes of the instruments are large and naturally fall into two or more separated compartments, the largest containing the primary and secondary mirrors and being difficult to keep clean while the other(s) can be more easily isolated from the spacecraft, with purging possible on the ground. Large instruments could fold their optical paths and thus shorten and stiffen
their optical benches. This would reduce the thermal-control problems and, as a bonus, assist in minimizing the mass. For EUV instruments, however, there is a high price to pay in throughput for each reflection, which might not be acceptable for any resolving instrument measuring radiances.

Dry-gas purging is difficult to guarantee on a continuous basis at spacecraft level, but we would recommend that provision be made for this. Purging should be foreseen as a weekly operation for, say, one hour and on exit from thermal vacuum tests or other tests resulting in cold detectors. This applies to most optical instruments, and we expect that more frequent or even continuous purging may be necessary for some detector types.

The instruments of the Solar Orbiter should be expected to have a high duty cycle in order to catch small scale dynamic processes, but one of the most serious bottlenecks of the mission will indeed be the limited telemetry. On the other hand, one cannot afford to undersample or even miss interesting events. This concern can be resolved by having the capacity to autonomously trigger on pre-determined categories of phenomena. This implies taking a number of observations that will not be down-linked. This solution also demands advanced processing on board. Assuming that stored data will normally be read out on a first-in, first-out basis, it might prove useful to apply on-board intelligence to tag some data for earliest possible read-out.

The stability of the instrument responsivities, and particularly of the focal plane flat-fields, must be monitored carefully if it is to be maintained. For instruments imaging the full solar disk, the diameter will indeed vary throughout the orbit, resulting in a variable exposure and thus in a less deterministic evolution of the exposure and potential degradation of the detector. The latter effects should therefore be minimized by new technological developments for focal-plane arrays (see “New Detector Concepts” in this volume [Hochedez et al., 2002]). Simultaneously, the instruments should explore means for recalibration (calibration lamps, assessment of the mirror changes).

The Solar Orbiter mission profile makes the operational conditions much more variable, making stability of calibration a very important issue. The stability that was possible on a mission like SOHO may not be achievable. However, we are optimistic that in spite of the severe thermal problems to be faced, but given the success of SOHO, Solar Orbiter will not present undue cleanliness problems.

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Bibliography