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THE SOLAR ORBITER

A. Lyngvi
Science Payload & Advanced Concepts Office, European Space Agency, ESTEC, The Netherlands
aleksander.lyngvi@esa.int

N. Rando[1], R. Marsden[3] A. Jeanes[1], A. Owens, L. Gerlach[2], G. Janin[4] and A. Peacock[1].
[1]Science Payload & Advanced Concepts Office, European Space Agency, ESTEC, The Netherlands
[2]Solar Generators Section, European Space Agency, ESTEC, The Netherlands
[3]Research and Scientific Support Department, European Space Agency, ESTEC, The Netherlands
[4]Mission Analysis Office, European Space Agency, ESOC, Darmstadt, Germany

ABSTRACT

The Solar Orbiter mission is part of ESA's Cosmic Vision science program. In the last year this mission has been studied in an assessment phase aiming at demonstrating technical feasibility and defining the mission at systems level. The Solar Orbiter will explore the innermost regions of the heliosphere from high heliopsheric latitudes. It will reach a distance of 0.22 AU from the Sun and obtain an inclination of 35 degrees with respect to the Sun's equator. In these areas a series of in-situ and remote sensing measurements will be performed, providing first time in-situ measurements of regions so close to the Sun and at the same time provide 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. Two mission profiles have been studied during the assessment; one utilizing solar electric propulsion and one using chemical propulsion. Both these scenarios will be discussed in this paper and the respective spacecraft design and the current Strawman payload will be presented. The strong link to BepiColombo through reuse of components will be outlined and specific technology development needs for the Solar Orbiter will be described.

INTRODUCTION

In 2000 the Solar Orbiter mission was submitted to ESA and selected by ESA's Science Programme Committee (SPC) 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 provided that final approval is given by the SPC.

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. It incorporates both a near-Sun and a high-latitude phase. The near-Sun phase of the mission enables the Orbiter spacecraft to approach the Sun as close as 48 solar radii (~0.22 AU) during part of its orbit, thereby permitting observations from a quasi-helio-synchronous vantage point (so-called co-rotation.). At these distances, the angular speed of a spacecraft near its perihelion approximately matches the rotation rate of the Sun, enabling instruments to track a given point on the Sun surface for several days. During the out-of-ecliptic phase of the mission (extended mission), the Orbiter will reach modest solar latitudes (up to 35º in the extended phase), making possible detailed studies of the Sun's polar caps by the remote-sensing instruments.

SCIENCE GOALS

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 under conditions impossible to reproduce on Earth or to study from astronomical distances. The results from missions such as Helios, Ulysses, Yohkoh, SOHO, TRACE and RHESSI have advanced significantly our understanding of the solar corona, the associated solar wind and the three-dimensional heliosphere. Further progress is to be expected with the launch of STEREO, Solar-B, and the first of NASA's Living With a Star (LWS) missions, the Solar Dynamics Observatory (SDO). Each of these missions has a specific focus, being part of an overall strategy of coordinated solar and heliospheric research. An important element of this strategy, however, has yet to be implemented. 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 major breakthroughs in solar and heliospheric physics. The Solar Orbiter will, through a novel orbital design and an advanced suite of scientific instruments, provide the required observations. The unique mission profile of Solar Orbiter will, for the first time, make it possible to:

- Explore the uncharted innermost regions of our solar system;
- Study the Sun from close-up;
- Fly by the Sun tuned to its rotation, examine solar surface and space above from a co-rotating vantage point;
- Provide images & spectral observations of the Sun polar regions from out of the ecliptic

Within the framework of the global strategy outlined above, the top-level scientific goals [3] of the Solar Orbiter mission are to:

- Determine the properties, dynamics and interactions of plasma, fields and particles in the near-Sun heliosphere;
- Investigate the links between the solar surface, corona and inner heliosphere;
- Explore, at all latitudes, the energetics, dynamics and fine-scale structure of the Sun's magnetized atmosphere;
- Probe the solar dynamo by observing the Sun's high-latitude field, flows and seismic waves.

THE MODEL PAYLOAD

The actual scientific payload for the Solar Orbiter mission will be selected on a competitive basis, following an Announcement of Opportunity that will be open to the international scientific community. The model payload described in this paper is used as to progress with the mission definition before selection of actual instruments and comprises units (in-situ and remote-sensing measurements) defined on the basis of input received from the scientific community. As to maintain compatibility with the boundary conditions of a medium size mission, a resource effective payload is required (max total allocated mass of 180 kg, including maturity margins).

A summary of the Solar Orbiter reference payload [4] is provided in Table 1. In order to optimize the use of resources, several of the smaller in-situ sensors have been grouped into so-called *suites*. In this way, four categories are identified: a) In-Situ sensor units (sharing common DPU); b) In-Situ suite common elements (providing the common resources required by each suite); c) '*1 arcsec, 1m class'* Remote Sensing instruments (representing the maximum allowed envelope for the biggest units; d) Payload Support Elements (e.g. boom, rotating platform, etc.). The table refers to the *core payload complement*, reflecting the science prioritization given in [3]. All figures reported in the table include design maturity margins (depending on heritage).

Instrument	**Science Objective**	**Mass [kg]**	**Power [W]**
a) In-Situ instruments			
Solar Wind Plasma Analyzer	Investigation of kinetic properties and composition (mass and charge states) of solar wind plasma	14.0	13
Radio and Plasma Wave Analyzer	Investigation of radio and plasma waves including coronal and interplanetary emissions	9.6	6.4
Magnetometer	Investigation of the solar wind magnetic field	1.5	1.5
Energetic Particle Detector	Investigation of the origin, acceleration and propagation of solar energetic particles	5.7	8.5
Dust Particle Detector	Investigation of the flux, mass and major elemental composition of near-Sun dust	1.8	6
Neutron Gamma ray Detector	Investigation of the characteristics of low energy solar neutrons, and solar flare processes	4.2	4
b) IS suites items	Items include DPU, DC/DC, enclosures	11.3	8
c) Remote-Sensing instruments			
Visible Imager & Magnetograph	Investigation of the magnetic and velocity fields in the photosphere	30	35
EUV Spectrometer	Investigation of properties of the solar atmosphere	18.0	25
EUV Imager	Investigation of the solar atmosphere using high resolution imaging in the EUV	20.4	25
VIS Coronagraph	Investigation of coronal structures using polarized brightness measurements in the VIS	18.0	25
Spectrometer Telescope Imaging X-ray	Investigation of energetic electrons near the Sun, and solar x-ray emission	4.4	4
d) Payload Support Elements	Scanning platform, boom, doors/windows and specific P/L thermal HW	27.6	20
TOTAL		**167.2**	**175.9**

Table 1 The Solar Orbiter reference payload and the mass and power budgets. In the spacecraft design an allocation of 180 kg and 180 W has been used for the spacecraft definition

MISSION ALTERNATIVES

Obtaining an orbit with high solar 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 (GAM). The Solar Orbiter will use Venus GAMs to obtain the high inclinations reaching 35 degrees with respect to the Sun's equator at the end of the mission.

To reach the final inclination of 35 degrees the Solar Orbiter will be in a resonant orbit with Venus making the possibility to perform frequent GAMs for raising the inclination. A short time between Venus encounters would allow the orbit to raise its inclination at a high pace. However a 2:1 resonance would mean that the perihelion would be less than 0.2 AU and this implies a too challenging thermal design. It was therefore decided that the science orbit should be a 3:2 resonant orbit with the perihelion at about 0.22 AU.

In achieving this science orbit several alternatives were investigated during the assessment phase. A common constraint for all investigated alternatives was that that they would need to be compatible with a launch on a Soyuz-Fregat 2-1B launch vehicle from Centre Spatial Guyanais (CSG) in 2013. The alternatives included the use of Solar Electric Propulsion (SEP), chemical propulsion or hybrid solutions. As a hybrid solution utilizing both chemical and electric propulsion would add both complexity and cost it was decided to discard this option. The chemical option and the SEP option were therefore studied in detail.

Chemical Propulsion Scenario

Using chemical propulsion the minimum cruise time to reach the science orbit would be about 3.4 years. To reach the required science orbit the spacecraft would do Venus-Earth-Earth-Venus gravity assist manoeuvres before insertion. This orbit is shown in Figure 1.

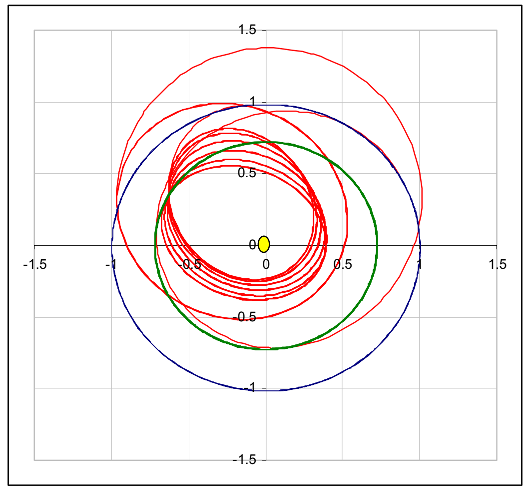

Figure 1 Ecliptic view of the Solar Orbiter trajectory for the 2013 chemcial scenario. Blue line is Earth Orbit, green line is venus orbit. The trajectory of the Solar Orbiter is shown in red.

The Delta-V for obtaining this orbit will vary greatly depending on the launch window, as is shown in Table 2. The current baseline launch date is in 2013 with a back up in 2015. For these two cases, the 2013 is clearly more challenging in terms of Delta-V and hence this launch window is used as the sizing case.

The configuration for a chemical scenario is shown in Figure 2. The Delta-V required to

Launch window	Propulsion system	Transfer duration	DSM Delta-V[1] (m/s)	Escape velocity (km/s)	Launch mass (kg)
Oct-2013	SEP	1.8	3996	3.010	1515
Nov-2013	CP	3.4	277	3.522	1356
May-2015	SEP	1.8	3534	2.985	1522
May-2015	CP	3.4	77	3.557	1345

Table 2 Potential Solar Orbiter launch dates and typical mission paramters. [1]Deep Space Manoeuvres (DSM) do not include any margins or the Delta-V required for GAM correction.

achieve this orbit is 562 m/s. This includes 15 m/s allocated for correction and preparation for each GAM.

As some of the instruments are doing sensitive remote sensing in the UV there are very strict cleanliness requirements for the spacecraft and the payload. Hence, also contamination originating from the thrusters would need to be taken into consideration. A bipropellant system is less clean than a monopropellant system. Furthermore, a bipropellant system would add both complexity and cost compared to a simple monopropellant system. Hence, monopropellant hydrazine was selected as the propulsion system for the chemical scenario. The selection of hydrazine allows the use of the same system for AOCS as for the navigation manoeuvres with slightly larger thrusters needed for the main Deep Space Manoeuvres (DSM).

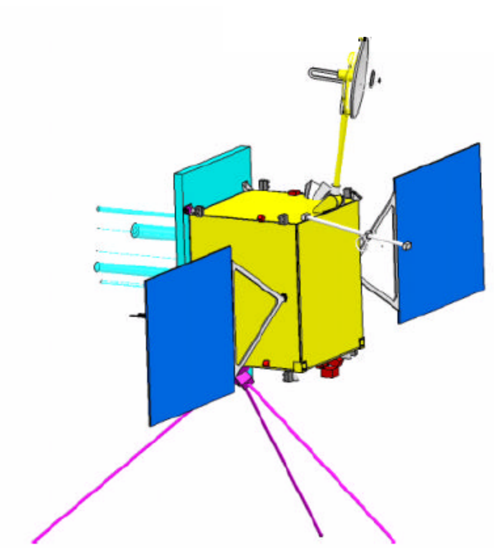

Figure 2 A typical configuration for a chemical scenario.

Solar Electric Propulsion Scenario

The use of SEP for Solar Orbiter leads to a shortened cruise time. The science orbit could then be achieved after only 1.8 years. The trajectory for the SEP option is a Venus-Earth-Venus gravity assist sequence, shown in Figure 1. This is one less GAM compared to the chemical option. The Delta-V required for the SEP is about 4700 m/s (including all margins and Delta-V for correction after GAM), which implies about 147 kg of propellant. The different launch options for the SEP scenario are shown in Table 2.

SEP is the baseline propulsion for the BepiColombo mission and the Solar Orbiter would use the technology developed in the frame of this programme, thus reducing the development cost of the Solar Orbiter.

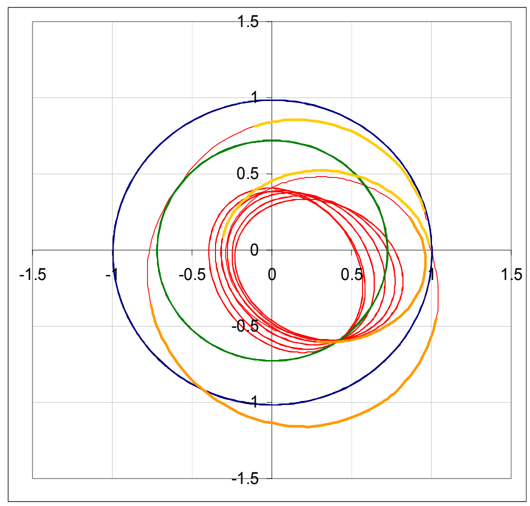

Figure 3 Ecliptic view of the Solar Orbiter trajectory for the 2013 SEP scenario. Blue line is Earth Orbit, green line is venus orbit. The trajectory of the Solar Orbiter is shown in red where the thrust arcs are shown in bold orange line.

One of the drawbacks of the SEP is that it would require very large amounts of power to obtain the required thrust and the necessary specific impulse. The thrust levels required for Solar Orbiter would be about 160 mN and the Isp would be about 4600 s, which implies a total power demand for the SEPM in excess of 7 kW. This is compatible with the current developments of the BepiColombo and AlphaBus thrusters.

On BepiColombo a separate Solar Electric Propulsion Module (SEPM) is being used. It was decided that this would also be beneficial for the Solar Orbiter. Using the thrusters from BepiColombo the power demand would result in a solar array requirement of about 30 m^2. When in science orbit only limited propulsive manoeuvres are necessary, mainly to correct for disturbance during GAM and to off-load any reaction wheels. There is therefore no need to keep the SEPM during science. As the use of the SEPM in science orbit would also need to have a spacecraft compatible with the Sun flux at 0.22 AU it was decided to jettison the SEPM before the second GAM.

As the SEPM would be jettisoned after the second Venus GAM the SEP would only need to sustain temperatures down to 0.3 AU. This would significantly ease the thermal control of the SEPM and allow for similar design to BepiColombo, which would also need to be compatible with a similar Sun distance.

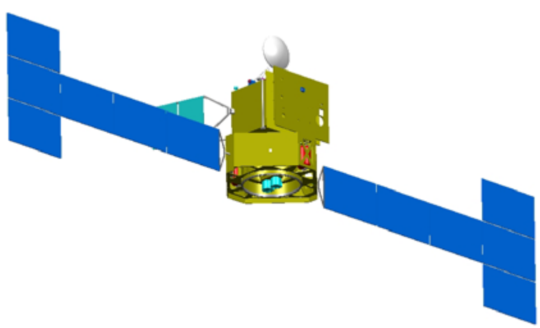

Figure 4 Possible spacecraft configuration using SEP. The figure shows the spacecraft with the SEPM still attached

The Orbiter Design

After jettisoning the SEPM, the SEP orbiter will be very similar to the spacecraft used in the chemical propulsion scenario. The main difference will be the solar array size, the overall structural size and the propulsion module sizing. The remaining elements will be very similar.

The main challenge for the orbiter is to sustain the thermal environment. The Sun flux experienced at 0.22 AU will be about 28000 W/m^2. This is almost twice the Sun flux that the BepiColombo mission will experience and the orbiter is largely driven by this thermal environment. In addition to the high Sun flux the plasma density caused by the solar wind will be higher resulting in possible higher degradation of the components used.

Thermal Design

The thermal control of the orbiter is based on using a Sun shield to obtain a benign environment for the spacecraft bus [5]. The Sun shield would potentially reach very high temperatures as the front layer would be directly exposed to the Sun. A further complication is the instruments which in many cases would need a direct view to the Sun hence requiring openings in the shield. The shield temperature will be dictated by the front layer absorption and emissivity. In order to minimize the heat into the spacecraft the shield would need to combine the use of radiative coupling to space and limit of the conductive coupling to the spacecraft by the use of MLI or other materials. As the Sun shield is such a mission critical component two alternatives of the shield design has been kept. One of these alternatives would use a white or grey front layer, while the other would use a black front cover. This would allow reducing the development risk of the Sun shield.

Power System

The orbiter will be Sun pointing throughout the mission and eclipses would only last for very short periods during GAM or during Launch and Early Operations (LEOP). A battery would therefore only need limited capacity and it might be possible to use an unregulated 28 V bus for power regulation.

The main challenge for the power system is the solar arrays as these would need to operate at 0.22 AU. For the SEP option there would be an additional need for very large solar arrays during the cruise phase.

To use typical aluminium substrate for the solar arrays would not be possible due to the very high temperature that the solar array would operate at during perihelion passes. This is also an issue for the BepiColombo mission, which has started a solar array development. This solar array is likely to based on a carbon-carbon substrate with triple junction GaAs solar cells. Even with this substrate the maximum temperature would be about 230 degrees. In order to achieve this temperature the Solar Orbiter would tilt its solar arrays. However, to simplify issues like edge effects, internal reflection in the solar array and uncertainties in the degradation of the cells, the solar arrays would be limited to not tilt more than 70 degrees. This would lead to the use of a mixture of solar arrays and optical solar reflectors (OSR). And a typical cell to OSR ratio would be about 40 -50 %.

Communication

The communication subsystem of the Solar Orbiter is based on the use of X-band and Ka-band in downlink and X-band to uplink. The ground station is assumed to be New Norcia, which would need to be upgraded to Ka-band receive capability.

In order to fully meet the science goals, the Solar Orbiter would require a data rate of about 200 kbps at 1 AU distance from the Earth. The communication subsystem is largely based on the current BepiColombo design, where components such as the transponder and the amplifiers will be re-used. This would give an RF power of about 30 W and an antenna of about 1 m in diameter.

The HGA would also be based on the BepiColombo design. For BepiColombo the minimum distance to the Sun would be about 0.3 AU. However, for Solar Orbiter using this down to the lowest perihelion of 0.22 AU would be of benefit as it would ensure contact with the spacecraft over most of its mission, allow for downlink in the closest science windows and minimize mass memory needs. The baseline is therefore to perform the required developments for allowing the HGA to be operated at 0.22 AU.

Data Handling System

The data handling system is largely based on the ongoing developments in the BepiColombo programme. The assumption is to use a Highly Integrated Control and Data System (HICDS), which is an ongoing data system development based on a LEON2-FT processor. The interface to the instruments would be using SpaceWire where it is assumed that the instruments would use Remote Terminal Controllers (RTC). The RTC is a remote terminal unit (RTU) that is implemented using a SpaceWire ASIC. Each RTC has its own processor and many features adding considerable processing power to such units, thus they would be capable of covering all required functions. The embedded LEON processor could be used as an instrument controller or even for data compression.

The mass memory sizing is driven by the 10 day science windows where both the data rate would be at its peak. As the qualification of the HGA antenna for use at 0.22 AU is unsure, the mass memory is sized for the scenario where the antenna could only be used at distances down to 0.3 AU. The mass memory required for the mission is at least 24 Gbytes.

AOCS

The Solar Orbiter will be a three-axis stabilized spacecraft with the pointing requirements as shown in Table 3. The pointing is driven by the Relative Pointing Error (RPE) requirement that drives the components selection of the Attitude and Orbit determination and Control System (AOCS).

Some of the instruments would require even better short term stability, however this is ensured through instrument specific internal stabilization systems rather than putting excessive requirements on the spacecraft.

Pointing Parameter	Line of Sight	Around Line of Sight
APE	< 2 arcmin	< 20 arcmin
PDE	< 1 arcmin / 10 days	< 10 arcmin / 10 days
RPE	< 1 arcsec / 10 sec	< 2 arcsec / 10 sec (TBC)
AMA	No requirement	< 3 arcmin in 10 sec

Table 3 Pointing requirements for the Solar Orbiter. APE is Absolute Pointing Error, PDE is Pointing Drift Error, RPE is Relative Pointing Error and AMA is Absolute Measuring Accuracy.

To meet the RPE requirement a high accuracy gyro is required and the spacecraft micro vibration would need to be analysed. Preliminary analyses show that the spacecraft is able to meet the current requirements without the need for specific disturbance measures.

The pointing requirements cannot be met during reaction wheel off-loading, or any movement off appendages, such as solar array or HGA. These operations would therefore need to be minimized during the science windows.

The thermal design of the Solar Orbiter, based on a Sun shield, implies that the spacecraft must remain Sun pointing at all times to limit the Sun flux on the lateral panels. This calls for a very robust Failure, Detection, Isolation and Recovery (FDIR) mode that would quickly put the spacecraft in a Sun pointing safe hold mode in case of failure.

TEST APPROACH

The Solar Orbiter design is driven by the Solar Flux at 0.22 AU which also drives the test approach required for the mission. Currently there is no facility with the necessary specifications available for a system level thermal test in a relevant sun environment. To perform such a test would require a facility with a sun flux of 28 000 W/m^2 with a beam width of the order of 3 m in diameter.

The current assumption for the Solar Orbiter is therefore to perform dedicated testing on materials in order to properly test their characteristics and then to do the thermal testing of the Solar Orbiter on element level rather than on the system. For the sun shield testing could be done on representative models of the critical areas such as the instrument apertures. Using this approach would greatly reduce the required size of a test facility.

TECHNOLOGY DEVELOPMENT

A key aspect of the assessment study has been the re-use of functional elements from other ESA missions. The commonality with BepiColombo has been promoted throughout all activities. The study has clearly confirmed that it is possible to re-use a large number of units, thus allowing a significant saving in development cost.

The re-use of components has led to a significant decrease in required technology developments. Furthermore, the remaining

Activity	Remarks
Orbiter Solar Array	Customization of BepiColombo design to survive the 22 SC flux
High Temperature HGA	Delta development from BepiColombo to extend operations below 0.3 AU from the Sun
SAS-AAD glasses	Protective glasses to be applied to existing SAS and AAD
High Temperature MLI	Delta development activities on BepiColombo HTMLI to match environment
Heat Shield Material Testing	Qualification of heat shield material in representative environment
Dedicated test facility	'1 m^3 class' test facility to create representative test environment

Table 4 Required spacecaft related technology developments for Solar Orbiter

technology developments that are still required can in many cases be largely based on ongoing activities that would lead to a shorter development time and a lower development risk.

The number of developments required is strictly related to the thermal environment at the closest distance to the Sun. These developments are shown in Table 4.

PROGRAMMATIC ISSUES

The present working assumptions is the release of the instruments AO in the second quarter of 2006 and a launch date in late 2013. In this case the following milestones have been identified:

- Instrument AO phase (1 year, 2nd quarter 2006 – 2nd quarter 2007)
- Definition phase (~1.5 year, 3rd quarter 2007 – 4th quarter 2008)
- Implementation phase (~5 year, 2nd quarter 2009 – 4th quarter 2013, including 6 month contingency)

A detailed cost assessment has been performed in order to estimate the Cost at Completion of the Solar Orbiter mission, based on the documentation provided by both the industrial teams. A cost-risk analysis has also been applied. The assessment shows a lower development risk and lower cost for the chemical profile.

CONCLUSION

The Solar Orbiter assessment phase has been completed and two technical feasible solutions have been identified; a Solar Electric Propulsion (SEP) option and a chemical option. The SEP option would consist of an orbiter and a Solar Electric Propulsion Module, while the chemical option would only consist of an orbiter. In both options the orbiter will be very similar utilizing very similar subsystems.

The Solar Orbiter assessment study has shown that a strong link to the BepiColombo mission will be ensured through re-use of many of the elements. This has also allowed a significant reduction of required technology development activities.

Under the present assumption of a launch in 2013, the Announcement of Opportunity for the payload will go out in the second quarter of 2006 with the start of the definition phase in the third quarter of 2007.

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