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Solar Orbiter Trajectory Profile and Navigation Challenges

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ESA's Solar Orbiter mission, with NASA participation and scheduled to launch in October 2018 will, after a multi-year cruise phase, enter a series of elliptical orbits around the Sun with perihelion as close as 0.28 AU and increasing solar inclination that will reach over 32 degrees by the end of a 10-year mission. The probe will return unprecedented images of the solar polar regions and investigate the physical processes of the inner heliosphere. This paper describes in detail the trajectory profile that has been selected for implementation after a mission design process to maximize the science data return. Some aspects of the mission and spacecraft designs together with the near-Sun environment are unique to Solar Orbiter and present some challenges for the navigation.

Key Words: Solar Orbiter, ESA, Trajectory Design, Navigation

Nomenclature

η	:	Coefficient of absorption
μ	:	Coefficient of specular reflection
ν	:	Coefficient of diffuse reflection
R_S	:	Sun range
R_E	:	Earth range

1. Introduction

ESA's Solar Orbiter mission, implemented jointly with NASA and due to launch in 2018 will explore the processes that create and control the heliosphere. The heliosphere represents a uniquely accessible domain of space, where fundamental physical processes common to solar, astrophysical and laboratory plasmas can be studied under conditions impossible to reproduce on Earth and unfeasible to observe from astronomical distances. In particular, Solar Orbiter seeks to answer the following questions:[1-2)]

− What drives the solar wind and where does the coronal magnetic field originate from?

− How do solar transients drive heliospheric variability?

− How do solar eruptions produce energetic particle radiation that fills the heliosphere?

− How does the solar dynamo work and drive connections between the Sun and the heliosphere?

These questions represent fundamental challenges in solar and heliospheric physics. By addressing them, it is expected to achieve major breakthroughs in our understanding of how the inner solar system works and is driven by solar activity. To answer these questions, it is essential to make in-situ measurements of the solar wind plasma, fields, waves, and energetic particles close enough to the Sun so that they are still relatively pristine and have not had their properties modified by subsequent transport and propagation processes. This is one of the fundamental drivers for the Solar Orbiter mission, which will approach the Sun to as close as 0.28 AU (60 solar radii).

Relating the in-situ measurements back to their source regions on the Sun is one of the major scientific goals of the mission and requires simultaneous, high-resolution imaging and spectroscopic observations of the Sun in and out of the ecliptic plane. The resulting combination of in-situ and remote-sensing instruments on the same spacecraft (Table 1), together with its close perihelion distance and large solar inclinations up to about 32 deg, distinguishes Solar Orbiter from all previous and current missions, and promises to enable unprecedented science. Moreover the mission has unique science synergies with NASA's Solar Probe Plus,[3)] also scheduled to launch in 2018 and expected to go to the Sun for the first time within 10 solar radii remaining close to the ecliptic plane.

While in-situ instruments will be operating continuously throughout the mission, starting in the cruise phase, the use of the remote-sensing instruments will be restricted due to data return constraints to three 10-day "remote-sensing windows" (RSW) per orbit. Nominally, RSWs will be centred on perihelion and the extremes of northern and southern latitude of the orbit. Up to two RSWs can be contiguous, giving a single 20-day window of observations.

Table 1. Solar Orbiter instruments.

In-Situ	
MAG	Magnetometer
RPW	Radio and Plasma Waves Analyser
SWA	Solar Wind Plasma Analyser
Remote-Sensing	
EUI	Extreme Ultraviolet Imager
METIS	Coronagraph
PHI	Polarimetric and Helioseismic Imager
SoloHI	Solar Orbiter Heliospheric Imager
SPICE	Extreme UV Spectral Imager
STIX	Spectrometer/Telescope in X-rays

Figure 1 shows the reference design of the Solar Orbiter spacecraft together with the instrument payloads. A 2.5m x 3m heatshield will protect the spacecraft body (containing the platform units and the majority of the remote-sensing instruments) from the direct solar flux, thus providing a stable thermal environment and allowing the use of a conventional thermal control subsystem. Solar Orbiter is a 3-axis stabilized spacecraft that will mainly operate in Sun pointing attitude with the heatshield facing the Sun. In order to observe the solar disk through the heatshield, feed-through channels are installed in the shield with individual mechanical doors that can be opened or closed as required by the instruments. Two steerable solar arrays wings with total area of 14.7 m^2 provide the necessary power. The solar array temperature is managed through variation of the solar aspect angle via rotation around the longitudinal axis. The propulsion system includes a redundant set of 9 10-N thrusters to provide attitude control during safe mode, reaction wheel off-loading and delta-V manoeuvres for orbit control. Communications with the Earth will use X-band only. Nominal communications will use 2-axis articulated high gain antenna (HGA), while communications during contingencies will use the medium gain antenna (MGA) with 1-axis articulation.

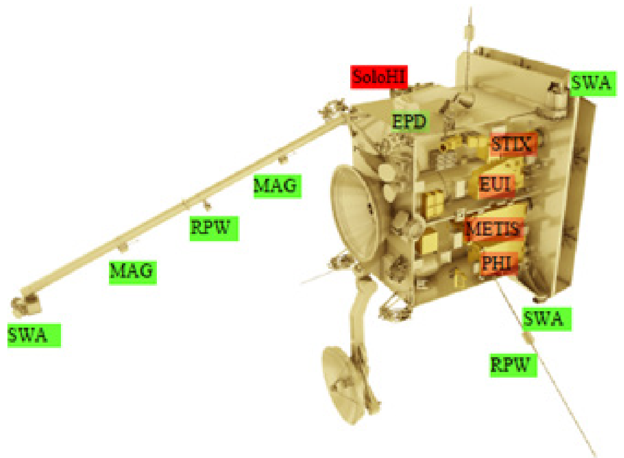

Fig. 1. Solar Orbiter Spacecraft. Remote-sensing instruments (red) are clustered within the body of the spacecraft behind the heatshield, and look through feed-through channels. In-situ instruments (green) are exposed to the space environment. Notes: SPICE instrument is mounted underneath the top panel, not visible in this view; solar arrays are not included.

2. Mission Design and Trajectory Overview

A critical decision for the mission design is the choice of a ballistic trajectory to minimize the propellant on-board the spacecraft. Thus the trajectory does not contain sizeable deterministic deep space manoeuvres and propellant on-board the spacecraft will be dedicated to stochastic Trajectory Correction Manoeuvres (TCM) necessary to compensate navigation errors.

The science objectives call for an orbit close to the Sun and at large inclination with respect to the Sun equator to provide clear observations of the Sun's polar regions. Perceived critical technology capability at the inception of the project has limited the perihelion to be above 0.28AU (60 solar radii) thus allowing a maximum re-use of technologies developed for the ESA Bepi Colombo mission to Mercury. The high solar inclination goal is considered satisfied if more than 30 deg have been reached before the end of the 10-year mission lifetime.

In addition, the following requirements have a significant impact in the trajectory design:

- Solar superior conjunctions preventing communication with the spacecraft shall not occur in the proximity of gravity assist manoeuvres (GAMs).
- Duration of solar conjunction periods should be minimized in order to reduce the autonomy required to the spacecraft.
- Minimum altitude at closest approach at an Earth or Venus GAM shall be higher than 350 km in the nominal trajectory.

The required Sun close range and large solar inclination of the operational orbit can be reached by repeated Venus GAMs if the relative velocity of Solar Orbiter with Venus is around 18 km/s. A trajectory from Earth to Venus providing such relative velocities requires an Earth escape velocity above 10 km/s. This is beyond the capabilities of the launch vehicles regarded for Solar Orbiter. By using a sequence of Venus and Earth GAMs it is possible to leave Earth with an escape velocity compatible with the launcher performance and achieve the required Venus relative velocity. This strategy leads to a cruise phase of 2-3.5 years in which basically the Venus relative velocity is increased. This involves typically launching from Earth to Venus, coming back to the Earth with an increased infinite velocity and after 1 or 2 GAM at Earth going towards Venus again.[4-5]

Coming from a cruise phase close to the Ecliptic plane, the initial solar inclination of the operational orbit will not have the required high value. The inclination will be gradually increased by using a sequence of Venus GAMs, in which the period of the operational orbit is selected to be in resonance with the orbital period of Venus such that after a number of full orbit revolutions the spacecraft and Venus meet again in the same position to perform the next GAM. This "sequence of resonances" is carefully designed to maintain the perihelion close to the Sun as well.

The Solar Orbiter mission has been designed to be compatible with a set of trajectories with launch during the 2017-2018 timeframe that are expected to capture all the requirements in terms of the spacecraft and ground systems design.[4-5] In the meantime the baseline launch period has shifted to September-October 2018. A recent study has been carried out to improve the overall science data return for launch on this opportunity.[6-7] The most promising trajectory, internally known as Option E, provides several advantages like a shorter cruise with no excursion far from the Sun (no need for hibernation) and a much larger capability for data return. This trajectory has been endorsed by the Solar Orbiter Science Working Team in 2015 and it is currently regarded as the most likely for full implementation. The next sub-sections provide a detailed description of this trajectory.[8)]

2.1 Launch and launch window

Solar Orbiter will be launched in a NASA-provided Atlas V 411 from the Kennedy Space Centre. The launch period extends 29 days from 22nd September to 20th October 2018. The required escape velocity for each launch day varies from 5.2 to 5.5 km/s (max. required C3 30.25 km/s^2).

The launch period open is constrained by a geometry configuration in solar superior conjunction leading to a very long period (over 100 days) in which communications through the MGA would not be possible. As this exceeds the spacecraft autonomy capability to survive with no ground interaction, such trajectories are not acceptable. On the other end the launch period closes when the required hyperbolic excess velocity grows over the maximum value of 5.64 km/s compatible with the 1800-kg spacecraft.

2.2. Cruise phase

Finding appropriate cruise trajectories for Solar Orbiter implies an optimization process involving systematic search and branch/pruning of the combinations of sequences of Venus and Earth GAMs and the possible orbit types and resonances among them. This analysis is performed at ESOC with the in-house state-of-the-art SOURCE software.[9-10] The most promising cruise phases are typically based on EVEV, EVVEV and EVEEV profiles.

The Solar Orbiter trajectory regarded in this paper uses an EVVVEV profile. The 3 consecutive Venus GAMs involve 2 intermediate orbits in 1:1 resonance with Venus (n:m = n revolutions of the spacecraft, m Venus revolutions). The ecliptic projection of the cruise trajectory together with Earth and Venus orbits is given in Fig. 2. Also indicated in the figure are the "sweet" locations of the Venus orbit at which increasing the solar inclination via repeated Venus GAMs is more efficient (green lines). Notably Solar Orbiter arrives at the 4th Venus GAM (V4) almost half-way from the optimal to the worst location (red line).

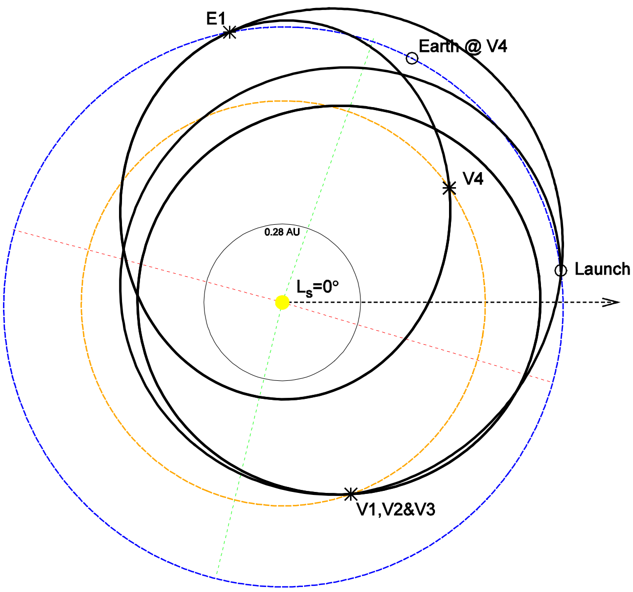

Fig. 2. Ecliptic projection of cruise phase.

The Earth GAM in January 2021 sets the perihelion below 0.35 AU allowing the start of the remote sensing observations. This event is actually defining the end of the cruise phase 2.25 years after launch and the start of the remote sensing observations.

In the current trajectory a 1.12 AU maximum Sun range is reached during the Venus to Earth arc. Being below 1.2 AU Solar Orbiter will not require a hibernation phase as it was planned for several of the previously regarded trajectories involving maximum Sun ranges up to 1.48 AU.[4-5]

2.3. Science phase

The spacecraft reaches V4 in an outbound arc after perihelion with an arrival hyperbolic velocity of 17.46 km/s. During the science phase Solar Orbiter will implement a sequence of resonances with Venus. This sequence of resonances is carefully designed to provide close perihelions, maximizing the resolution of the remote sensing observations, and a gradual increase of the solar inclination, to reach clear views of the Sun's poles at the end of the mission. An additional important design consideration is the maximization of the science data return.[6]

The sequence of resonances 4:3 3:2 3:2 5:3 3:2 has been found the most promising. The science mission is split in a 4-year nominal mission phase (NMP) from the Earth GAM until V6 and an extended mission phase (EMP) from V6 for 4.3 years until the mission end. The overall trajectory duration is therefore 10.5 years.

Solar latitude and distance profiles in Fig. 3 and Fig. 4 show the mission's approach for getting close to the Sun while gradually increasing the inclination. During the cruise phase Solar Orbiter remains close to the ecliptic (solar latitude within ±5 degrees) and with perihelion above 0.5 AU. The solar inclination increase strategy starts with V4, and continues with each Venus GAM until the mission end. At each step the spacecraft increases the solar inclination and stays at the new solar inclination for a few orbits.

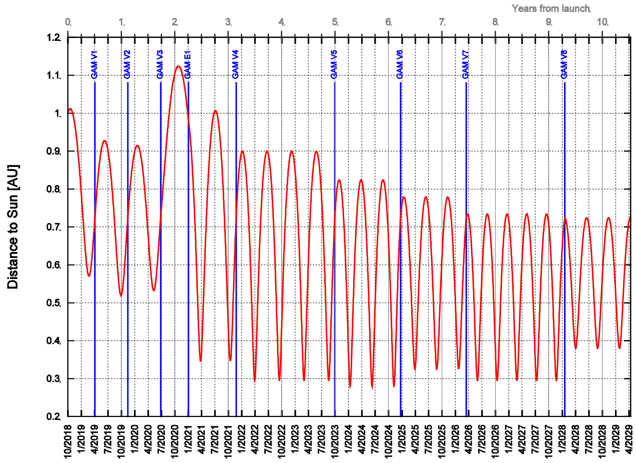

Fig. 3. Evolution of Sun range.

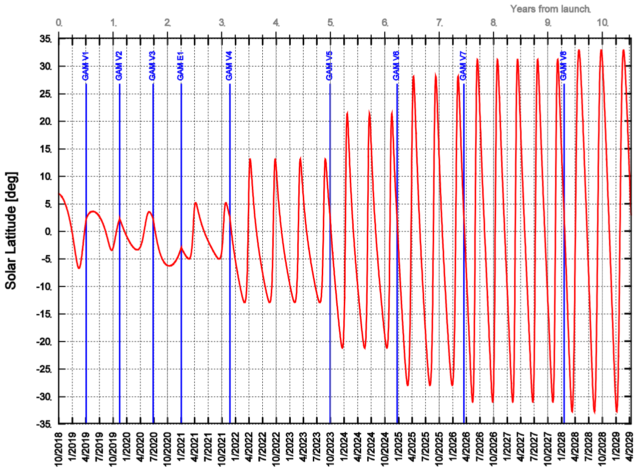

Fig. 4. Evolution of solar latitude.

Figure 5 provides an orbit representation that illustrates the increase of the solar inclination of the science orbits during NMP and EMP. The X-axis shows the modulus of the projection of the spacecraft position vector on the Sun Equator plane; while the Y-axis shows the projection of the spacecraft position on the Sun's poles direction. In this plot the Sun is located at the origin of coordinates, grey circles show the Sun range and orange lines show the solar latitude. From the start of the science phase at the Earth GAM, the plot shows the evolution of the orbit after each Venus GAM. Solar Orbiter will describe the orbits in the plot in clockwise direction. The first orbit (from Earth GAM to V4) is the closest to the Sun equator. Venus GAMs occurring at the common intersection of all orbits increase the solar inclination and at the same time increase the latitude at perihelion, which is located in the Sun Northern hemisphere.

The figure shows as well the location of all the 10-day remote sensing windows (green lines). As can be seen, the maximum latitude RSW tends to be closer to the Sun than the minimum latitude RSW, except for the last of the science orbits.

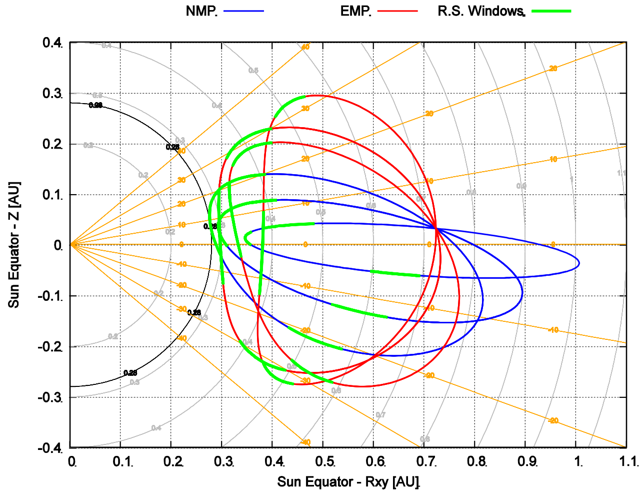

Fig. 5. Projection of science orbits in Sun's Equator and Pole.

2.4. GAMs timeline

Table 2 shows the timeline of all Venus and Earth GAMs implemented during the mission, along with several geometric magnitudes (Vinf = infinite velocity, Hmin = minimum fly-by altitude, ESV = Earth-Sun-Venus angle) and durations of eclipse and occultation. Since Solar Orbiter trajectory is ballistic, all GAMs are unpowered. Thus the result of a GAM is a deflection of the relative velocity vector, while its modulus is preserved.

The minimum altitudes during the GAMs are the result of the trajectory optimization process. The first 3 Venus GAMs will be performed at relatively high altitude and on the sunlight side, thus with no eclipse. The altitude of the GAMs decreases progressively. This will allow gaining knowledge of the spacecraft performance during the GAMs, before trying to accomplish any of the low-altitude GAMs required during the science phase.

The Earth GAM and all the Venus GAMs after V4 will be performed close to the minimum allowed 350-km nominal altitude. This allows maximizing the solar inclination at end of the mission.

From V4 onwards the Solar Orbiter will enter an eclipse of up to 19 minutes located in the vicinity of the closest approach to Venus. Occultation of the Earth behind Venus will occur for V7 and V8. Communications will be interrupted by as much as 11 minutes. Solar superior conjunctions close to a Venus GAM are avoided by trajectory design. The minimum SES angle at a GAM is 16.4 degrees at V7.

Table 2. Timeline and geometry of Venus and Earth GAMs.

GAM	Date	Vinf (km/s)	Hmin (km)	Eclipse (min)	Occult. (min)	Re (AU)	SSE (deg)	SES (deg)	ESV (deg)
V1	2019-04-04	9.82	15070	0	0	1.300	50.3	34.0	95.7
V2	2019-11-14	9.83	11000	0	0	1.511	33.6	24.0	-122.3
V3	2020-06-26	9.83	5831	0	0	0.362	135.3	30.2	14.5
E1	2021-01-01	9.69	387	0	0	-	-	-	-
V4	2021-11-23	17.46	363	18.7	0	0.478	108.5	44.1	-27.4
V5	2023-09-28	17.46	350	14.1	0	0.508	107.5	43.6	29.0
V6	2024-12-21	17.46	440	10.7	0	0.829	78.3	46.1	-55.5
V7	2026-03-15	17.46	561	10.6	10.0	1.621	22.9	16.4	-140.6
V8	2028-01-18	17.46	715	8.5	10.8	1.157	57.7	38.5	-83.7

2.5 Communications and Data Return

Daily communication with Solar Orbiter is essential to download the huge amounts of science data produced by the instruments. Communication links between ground and spacecraft are required to guarantee the mission operations, providing the spacecraft telemetry and allowing sending commands to be executed on-board, and in addition providing radiometric measurements to perform orbit determination for navigation. Solar Orbiter communications will be supported with the deep space antennas of ESA's ESTRACK network located at Malargüe in Argentina, Cebreros in Spain and New Norcia in Australia. The operation plan foresees the use of Malargüe as baseline ground station with Cebreros acting as backup.

Being a mission towards the inner solar system, Solar Orbiter will be affected by frequent solar conjunction periods, which can be identified in the Sun-S/C-Earth (SSE) and Sun-Earth-S/C (SES) angles profile over the mission shown in Fig. 6.

When both SES and SSE angles are small and the Sun is between Earth and the spacecraft, the Sun's plasma and other electromagnetic noises produce disruption of the communication between Earth-based antennas and the spacecraft. Solar Orbiter communication is assumed unavailable for both HGA and MGA when the SES angle is below 3 degrees. Solar Orbiter will encounter 8 periods of superior solar conjunction that will result in outage or degradation of the X-band communications with the spacecraft. The longest solar conjunction will start on 7th April 2027 and have a duration of 25 days.

Furthermore, in contingency communications using the MGA close to a superior solar conjunction the link can be severely degraded or interrupted by the geometric blocking of the MGA boresight by the heatshield. Communication via the MGA is considered unfeasible if SES is less than 5 deg or

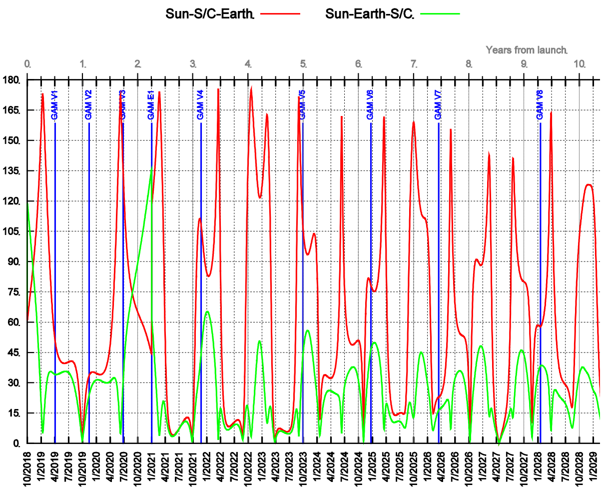

Fig. 6. Sun-S/C-Earth and Sun-Earth-S/C angles (in deg).

SSE is less than 3 deg. This geometry is particularly relevant for the recovery after safe mode as the MGA is then the primary means to access the spacecraft. The two longest of these periods will occur in 2021 April-June and 2027 March-May with a duration of 45 and 42 days, respectively. In 2023 another 2 of these periods extending barely a month will take place.

Science data downlink is critical for the success of the Solar Orbiter mission. Downlink data rate scales approximately with $1/R_E^2$. With the current communication assumptions, using GMSK modulation to improve the performance when close to Earth,[7] Solar Orbiter supports a data rate of 202 kbps at 1 AU. Closer to Earth the data rate grows up until a maximum of about 1 Mbps at 0.45 AU. Obviously, improving the data return capability involves communicating closer to Earth and this can be favoured by selecting geometries of the

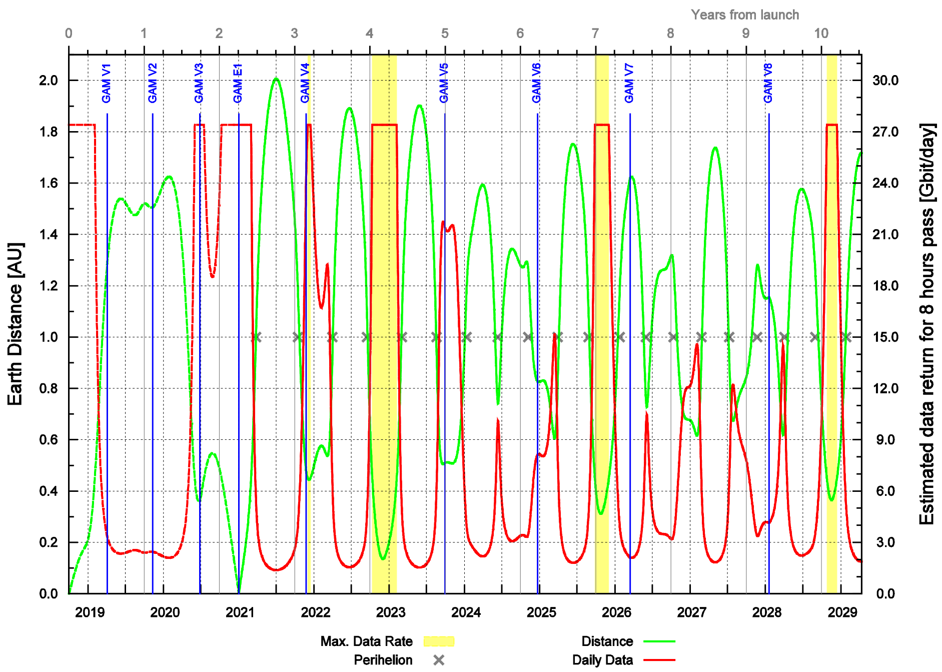

Fig. 7. Solar Orbiter Earth range and daily data return.

science orbits in which the aphelion is closest to Earth and avoiding geometries in which the closest point in the orbit is the perihelion.

Figure 7 shows the Earth distance and the potential daily data return profiles over the mission, assuming regular 8-hour visibility passes per day (solar conjunction periods are not considered). When the Earth distance is below 0.45 AU the transmitter saturates leading to the maximum possible daily data return. 4 of such periods (during the science phase, not around the Earth GAM) are indicated with a yellow shadow. 3 of such periods have long durations well over 2 months and will provide a huge downlink capability.

Figure 8 shows the projection of each of the science orbits in the Sun-Earth rotating frame. V3 at an Earth-Sun-Venus (ESV) angle of -27 deg (negative sign indicates that Venus lags behind Earth) is close to a downlink best case for a 4:3 orbit: an aphelion at 0.45 AU from Earth comes immediately after the GAM and the second next aphelion is shifted to an inferior conjunction geometry getting as close as 0.15 AU from Earth. The maximum data rate can be sustained for about 4 months (this is seen also in Fig. 7).

The first of the 3:2 resonances starts with V5 at 29 deg ESV angle providing again good downlink conditions during the immediately next aphelion. The orbit then drifts to a geometry not favourable for data downlink with minimum Earth distance at perihelion. The second 3:2 orbit after V6 is well suited again for downlink thanks to the Earth-close aphelion that provides a 2-month period with maximum data rate. Of the last two resonances, the 5:3 after V7 is less favourable for downlink and the last 3:2 resonance after V8 also provides an aphelion close to Earth and a period of about 45 days with the maximum data rate.

4. Navigation Challenges

This section describes some challenging navigation aspects of the Solar Orbiter mission. Some are related to the environment encountered by the spacecraft in the nominal trajectory with frequent passages close to the Sun, while others are related to the platform of the Solar Orbiter spacecraft. Non-gravitational forces will need to be properly modelled in order to ensure that Solar Orbiter can be navigated with the planned TCM strategy and within the available propellant budget.

4.1. Spacecraft background

The Solar Orbiter spacecraft consists of the heatshield, 2 steerable solar panels, the spacecraft bus protected behind the heatshield, a 2-degrees-of-freedom articulable high-gain antenna and some appendages and booms holding instrument payloads. The body coordinate frame (axes X, Y, Z as illustrated in Fig. 9) is defined by the following:

X normal to the heatshield surface,

Y in the longitudinal direction of the solar arrays, and

Z to complete a triorthogonal set.

In the nominal reference attitude, the +X axis is maintained Sun pointed, while the –Y axis is contained in the orbit plane and is oriented towards the spacecraft velocity vector at perihelion and aphelion.

Each solar array drive mechanism provides a one-axis steering capability, called the cant angle. In the canonical position (θ=0 deg) the normal to the solar array cells face is aligned with the +X axis. A positive cant angle implies a rotation of the solar panel around the –Y axis (see Fig. 9).

Only fixed values of the cant angles are allowed which ensure that yoke reflection does not damage the external payloads. In addition, power and solar cell temperature limitations constrain the range of operational cant angles as a function of the Sun range. For the lowest Sun range of 0.28

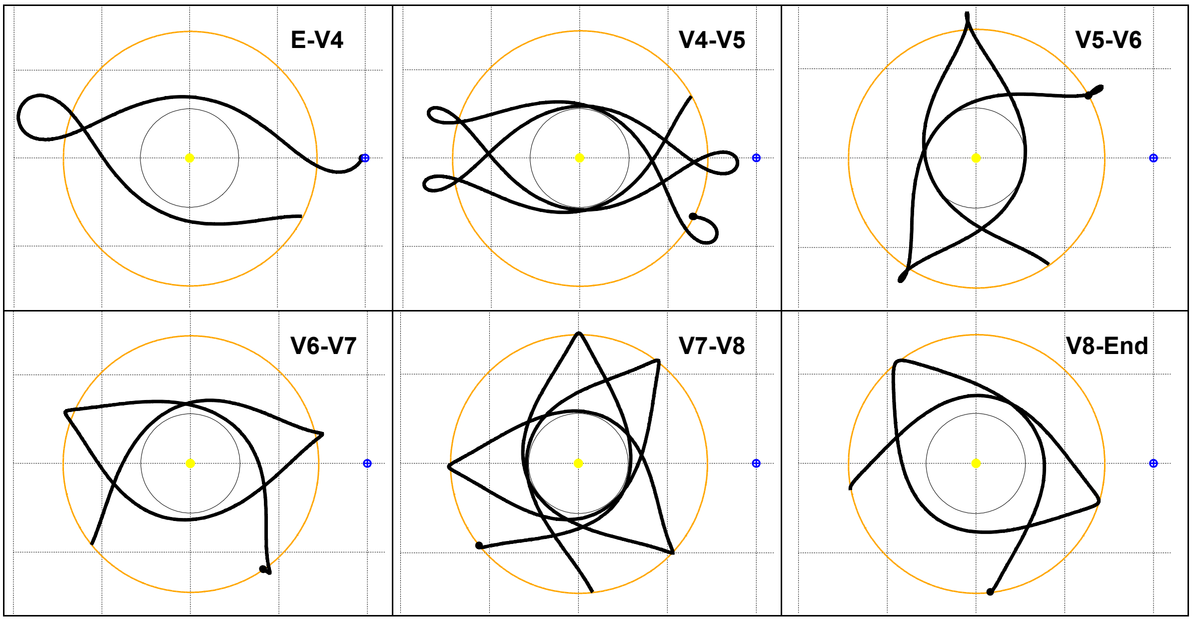

Fig. 8. Science orbits projection on the Sun-Earth rotating frame.

AU a cant angle of 86 deg is required, thus the solar panels are almost parallel to the incident solar radiation. These constraints are reflected in a solar array steering profile defined by industry and that will be followed in the real operations.

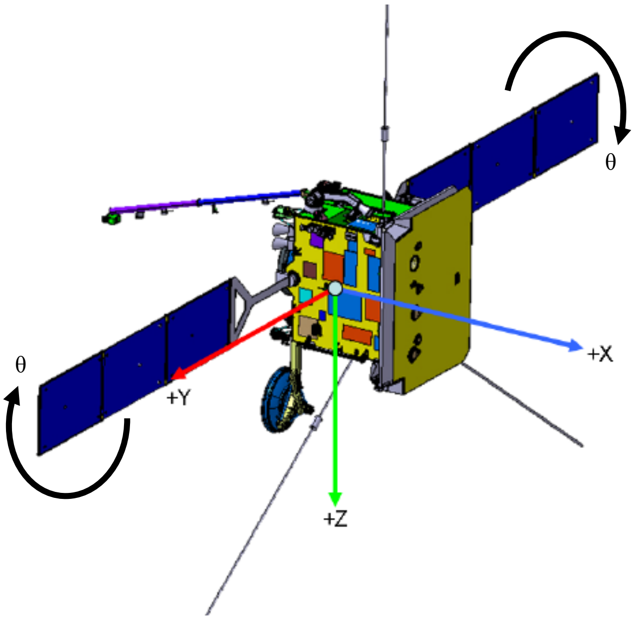

Fig. 9. Solar Orbiter body frame and solar arrays rotation directions

4.2 Solar radiation pressure force modelling

Solar radiation pressure (SRP) is the effect of solar photons striking on the spacecraft and imparting a force on it. The force results from the combination of photons that are reflected, either with specular or diffuse reflection, and photons that are absorbed, in either case transmitting energy to the spacecraft. Photons can also heat up the spacecraft changing its thermal radiative characteristics and an additional force is originated by the energy that is re-radiated via thermal emission. This latter is often assumed equivalent to a force produced by diffuse reflection.

An appropriate modelling of the SRP force is fundamental for the correct navigation of Solar Orbiter as it will become one of the largest non-gravitational accelerations when reaching the closest Sun range of 0.28 AU. In addition, the steering of the solar panels will introduce a non-radial SRP force component that is normally absent in most of other interplanetary spacecraft.

For Solar Orbiter ESOC's Mission Analysis has developed a refined SRP force model that considers the three main components exposed to the Sun radiation: heatshield, solar panels and HGA, as flat plate elements. In a first approximation the HGA is assumed in a fixed configuration completely normal to the incident Sun radiation. The SRP model adds up the contribution of every element and does not consider shadowing effects.

Coefficients of absorption η, specular and diffuse reflection, μ and ν, respectively, have been provided by industry for each of the spacecraft components. Assuming zero transmissivity the relationship $\eta + \mu + \nu = 1$ holds. For heatshield and HGA the coefficients are expected to vary with the spacecraft aging, thus values at begin and end of life are provided. For the solar array in addition to the variation with the spacecraft aging, the coefficient of absorption of the solar cells (populating more than 50% of each solar panel) changes also significantly with the solar incidence angle (θ). The absorption coefficient remains above 0.9 for incidence angles below 50 deg, then drops continuously until a value of 0.36 at the maximum allowed incidence of 86 deg. The decay of the absorption implies an increase of both reflection coefficients.

The force contribution from a flat plate element is computed with the expression:[11-12)]

$$\bar{f} = \frac{CA}{R_S^2}\left[(2\mu - 1)\cos\alpha\,\hat{u}_r - (2\nu + 4\mu\cos\alpha)\cos\alpha\,\hat{u}_n\right] \quad (1)$$

Where C is the average solar flux at 1 AU, A is the reference area of the flat element, $\hat{u}_r$ is the unit vector in the direction from spacecraft to the Sun, $\hat{u}_n$ is the unit vector normal to the plate and α is the angle between these two vectors.

Taking into account the solar array steering profile as function of the Sun range, the variation of optical properties of the solar cells with the solar array cant angle and the reference attitude with –Y in the orbit plane and aligned to the velocity vector at perihelion and aphelion, the SRP force can be calculated as shown in Fig. 10. The steps seen in the curves are caused by the transition between the fixed steering angles. The steering away from the Sun reduces the contribution of solar panels to the force radial component, while at the same time introduces a non-radial force opposite to the angular momentum vector in the reference attitude. The magnitude of the SRP force radial component reaches up to 3.3E-10 km/s^2 at the minimum allowed 0.28 AU Sun range and goes down to 8.6E-11 km/s^2 at 1 AU. The magnitude of the non-radial SRP force can reach up to 4.2E-11 km/s^2 at 0.28 AU.

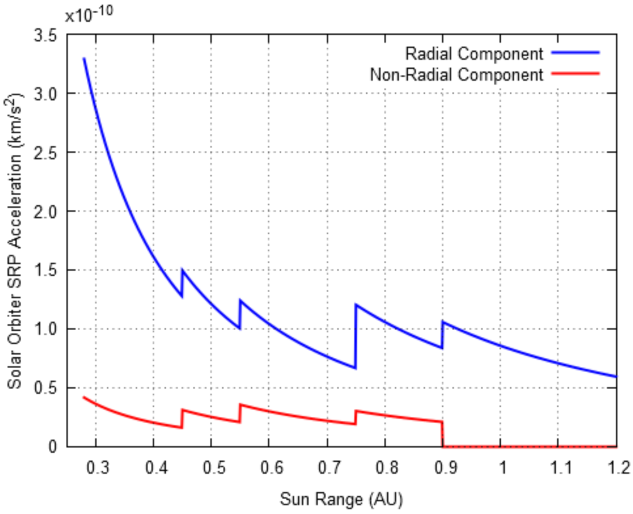

Fig. 10. Nominal SRP force acting on the Solar Orbiter spacecraft as function of the Sun range.

In the actual operations perturbations in the SRP force modelling will come from the following factors:

- Uncertainties in the optical properties of the different elements and its variation during the mission lifetime. Also it will be challenging to estimate the amount of thermal emission, which for simplicity has been omitted in the computation of this reference SRP force.
- Differences between the a-priori assumed solar array steering profile and the one actually applied in the real operations.
- Difference in the spacecraft attitude from the a-priori reference attitude. In particular the spacecraft will be obliged to abandon the reference attitude if the HGA to Earth line is blocked by any of the spacecraft appendages and external surfaces. To maintain communications the spacecraft will roll around the Sun line (HGA communication roll). By mission requirement the roll angle must remain below 5 deg more than 85% of the overall mission lifetime, but in exceptional temporary transits a maximum roll angle of about 80 deg will be needed. Therefore, the non-radial component of the SRP force from the solar panels, which is nominally in the out-of-plane direction during reference attitude periods, will rotate during communications rolls and exert a perturbation force also in the direction along the velocity.
- The SRP force on the HGA will actually depend on the articulation angle that is required to maintain the communications link.
- Science requests to off-point the spacecraft so that the remote sensing instruments can track specific features on the Sun up to the edge of the solar disk. This requests will be result of the short-term science planning and therefore it is not possible to include them in the longer term trajectory computation cycle.

Therefore the estimation of the SRP force in the real operations will be a rather complex task. The trajectory computations for the spacecraft guidance will have to rely on a-priori profiles for solar array steering and communication rolls. Deviations from the profiles in the actual operations and the unknown off-pointings for science introduce perturbing accelerations that will contribute to the navigation errors that need to be corrected when targeting the next GAM.

An analysis at mission analysis level has been performed to understand the impact of ignoring HGA communication rolls in the trajectory computation during the navigation process. As a worst case (unrealistic) scenario we consider the long phase in the 4:3 resonance (1.8 years) with the spacecraft maintaining a non-reference attitude. Two cases have been regarded:

1. Spacecraft attitude 180 deg from reference attitude
2. Spacecraft attitude 90 deg from reference attitude

In both cases the radial component of the SRP perturbation force is the same as the one with the reference attitude, so that it produces no error contribution. In case 1) the SRP perturbation non-radial force is still normal to the orbital plane, but in the opposite direction. With respect to the reference trajectory this produces a cross-track dispersion when arriving to the next Venus GAM. All the accumulated effect is easily corrected at the TCM at GAM-14 days by a delta-V in the order of 1 m/s.

In case 2, however, the real SRP non-radial force has a component along the velocity vector that introduces a significant error in phasing when approaching the next Venus GAM. Correcting the total accumulated effect at the TCM at GAM-14 days requires unacceptable delta-V well above 10 m/s. However, the phasing error is corrected more efficiently close to the last aphelion before the GAM. By doing so the required delta-V comes down to about 2-3 m/s.

In the real operations HGA communication rolls are expected to change sign within one orbit as the spacecraft crosses the ecliptic. Therefore periods with the SRP non-radial force producing a component along the velocity will be compensated by periods producing a component against the velocity. It is therefore expected that this will cancel out a large part of the accumulated effect, so that even if the HGA communication roll profile is ignored in the trajectory computation, the correction delta-V required will be much less than for the worst case mentioned above.

4.3 Other sources of dynamic noise

Frequent wheel off-loadings (WOLs) will be required for the momentum desaturation especially in the phases when the spacecraft orbits closer to the Sun. Each WOL produces a residual delta-V, which based on ESOC operational experience is currently assumed up to 1 mm/s.[8)]

The spacecraft has the capability to perform the WOLs autonomously, which will add more uncertainty to the prediction of the spacecraft state. In addition in order to limit as much as possible the disturbances during the RSWs, the spacecraft is required during these periods to perform the WOLs with a frequency not exceeding one WOL every 3 days. This limit in the frequency can lead to an increase of the residual delta-V of the WOLs, especially for the RSW at perihelion where the perturbing SRP torque compensated by the reaction wheels is largest.

An exhaustive survey of additional disturbing forces is given by Ref. 12 for the case of Solar Probe Plus. These include among others plasma drag, Lorentz force, aberration of incoming light, that become more relevant for spacecraft orbiting very close to the Sun. Noticing the Solar Orbiter minimum distance of 60 solar radii instead of 10 solar radii for Solar Probe Plus, these disturbances are preliminarily estimated to be very small compared with the effects of SRP force and the WOL residual delta-V. Other disturbing forces like outgassing and the atmospheric drag during Earth and Venus flyby will affect the spacecraft for short periods of time and their impact in the navigation is expected to be small.

4.4 Constraints to the TCM timeline

The TCM timeline for Solar Orbiter needs to consider the requirement to maintain the reference attitude as much as possible during the entire mission. Especially the RSW shall be periods fully devoted to the science observations, therefore free of correction manoeuvres. In addition for thermal reasons the spacecraft will not be able to operate the thrusters when the Sun range is below about 0.5 AU, and due to ESOC operational rules a TCM shall not be executed during a solar superior conjunction. Therefore, when considering all these constraints the location of the TCMs might not be exactly at those times that are optimal for the navigation in terms of delta-V consumption.

A TCM operational strategy has been carefully designed with the aim to minimize the number of required TCMs and concentrate them around the GAMs. This ensures TCM-free RSWs. In principle a GAM will be navigated using 4 TCMs prior to the closest approach and at GAM-30, -14, -7 and -3 days followed by an additional clean-up TCM 7 days after the GAM. The pre-GAM TCMs are used to finely target the GAM B-plane conditions, while the clean-up TCM is meant to correct the navigation error incurred during the GAM.

In case of an outbound Venus GAM with the spacecraft arriving from perihelion passage, the hypothetical TCM slot at GAM-30 days falls close to perihelion, where it is not allowed to execute a manoeuvre. This issue is solved by advancing this TCM slot to the previous aphelion. Indeed to optimize propellant expenditure, this TCM will be implemented N days after the aphelion, with N around 25-35 days typically providing the best performance.[8)]

4.5 Manoeuvre implementation

As a consequence of the need to maintain the reference attitude of the spacecraft with the heatshield oriented towards the Sun, the TCMs for Solar Orbiter are divided into two categories:

- Type 1 TCM: far from the Sun, above 0.95 AU, the spacecraft can abandon temporarily the nominal Sun pointing attitude, slew to the most efficient burn attitude aligning the +X axis with the desired velocity vector and firing the thrusters on the –X panel at high efficiency.
- Type 2 TCM: below 0.95 AU the attitude is constrained to remain Sun pointing with the heatshield towards the Sun. To achieve pure force authority in any direction in inertial space the thruster configuration has been designed to provide a semi-circle of delta-V authority in the spacecraft XY plane (through combination of +/-X and -Y direction thrust vectors). This together with the freedom to roll around the Sun-line provides access to delta-V in any direction.

For Type 2 burns the propellant efficiency of the manoeuvre depends strongly on the solar aspect angle of the desired delta-V direction. Directions along the +/-X axes are more efficient than those with a significant component along the –Y axis. While this basically affects the propellant consumption, another effect more relevant for the navigation will be the dependency of the manoeuvre execution errors with the solar aspect angle. Type 1 burns and Type 2 burns along the +/-X axes are implemented more accurately and will lead to smaller execution errors than Type 2 burns along the –Y axis.

7. Conclusion

Solar Orbiter, an ESA-NASA joint mission, will be a next step in the exploration of the Sun, both from close-up distances and by clear observation of its poles, and will answer fundamental questions on how the heliosphere works. The trajectory profile of Solar Orbiter has been described in detail showing how the maximization of science return drives the trajectory design. Several aspects affecting the navigation of the Solar Orbiter spacecraft have been addressed, in particular the importance of accurately modelling the SRP force and how the trajectory correction manoeuvres will be affected by mission, operational and platform constraints.

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