International Symposium on Space Flight Dynamics

Mission Design for NASA's Inner Heliospheric Sentinels and ESA's Solar Orbiter Missions

John Downing, David Folta, Greg Marr (NASA GSFC)
Jose Rodriguez-Canabal (ESA/ESOC)
Rich Conde, Yanping Guo, Jeff Kelley, Karen Kirby (JHU APL)

Abstract

This paper will document the mission design and mission analysis performed for NASA's Inner Heliospheric Sentinels (IHS) and ESA's Solar Orbiter (SolO) missions, which were conceived to be launched on separate expendable launch vehicles. This paper will also document recent efforts to analyze the possibility of launching the Inner Heliospheric Sentinels and Solar Orbiter missions using a single expendable launch vehicle, nominally an Atlas V 551.

1.1 Baseline Sentinels Mission Design

The measurement requirements for the Inner Heliospheric Sentinels (IHS) call for multi-point in-situ observations of Solar Energetic Particles (SEPs) in the inner most Heliosphere. This objective can be achieved by four identical spinning spacecraft launched using a single launch vehicle into slightly different near ecliptic heliocentric orbits, approximately 0.25 by 0.74 AU, using Venus gravity assists. The first Venus flyby will occur approximately 3-6 months after launch, depending on the launch opportunity used. Two of the Sentinels spacecraft, denoted Sentinels-1 and Sentinels-2, will perform three Venus flybys; the first and third flybys will be separated by approximately 674 days (3 Venus orbits). The other two Sentinels spacecraft, denoted Sentinels-3 and Sentinels-4, will perform four Venus flybys; the first and fourth flybys will be separated by approximately 899 days (4 Venus orbits). Nominally, Venus flybys will be separated by integer multiples of the Venus orbital period. Post-flyby Sentinels heliocentric orbit periods prior to the final Venus flyby will be approximately 224.7 days (Venus orbital period) and 149.8 days. Perihelion of the final heliocentric orbit will be 0.25 AU for the Sentinels-1 and Sentinels-3 spacecraft. Perihelion will be slightly larger than 0.25 AU for the Sentinels-2 and Sentinels-4 spacecraft. The differences in the final heliocentric orbits will result in relative heliocentric motion of the four Sentinels spacecraft providing a number of scientifically desirable spacecraft configurations. The four IHS spacecraft will nominally be launched on an Atlas V 541 launch vehicle, but both Delta 4 and Atlas 5 launch vehicles have been considered.

1.2 Sentinels Launch Windows

Venus launch opportunities with a launch energy low enough to allow for sufficient payload mass occur every Earth-Venus synodic period (approximately every 584 days).

Data for launch opportunities in 2015, and 2017 follows in Table 1.2-1. C3 is twice the combined kinetic and potential energy per unit mass and is defined with respect to the departure body (Earth) below. Declination of launch asymptote (DLA) is the declination of post-launch hyperbolic excess velocity with respect to Earth. Hyperbolic excess velocity (VHP) is defined with respect to the arrival body (Venus) below. To achieve a heliocentric orbit with perihelion of 0.25 AU the VHP must be approximately 10 km/s or greater. The 2015 and 2017 launch trajectories below are Type 2 trajectories; Type 2 trajectories are more than half a heliocentric orbit on the Earth-Venus leg. The 2015 and 2017 launch opportunities which follow have relatively low ecliptic inclinations, VHPs of approximately 10.3 km/s or greater, and relatively low C3s (below the 30.0 km2/s2 target initially identified). These launch opportunities could be modified as necessary to meet constraints identified by more detailed future analysis. The current launch vehicle payload mass required for the 4 IHS spacecraft with margin is 3192 kg. Per KSC's ELV Performance Web Site a maximum Atlas V 541 payload mass of 3365 kg is possible with an Atlas V 541 with a C3 of 30.0 km2/s2 assuming a DLA between -28.5 and +28.5 degrees.

Depart Earth	Arrive Venus (Flyby 1)	C3 (km2/s2)	DLA (deg)	VHP Venus (km/s)	Heliocentric Ecliptic Inclination
8/26/2015	**2/14/2015**	**26.5**	**-23.0**	**11.22**	**0.0**
9/4/2015	**2/14/2015**	**23.6**	**-22.2**	**10.93**	**0.0**
9/15/2015	**2/14/2015**	**26.2**	**-20.6**	**10.91**	**0.0**
2/27/2017	**9/5/2017**	**27.7**	**20.4**	**10.32**	**0.5**
3/9/2017	**9/9/2017**	**24.4**	**18.1**	**10.28**	**0.9**
3/19/2017	**9/13/2017**	**22.7**	**16.0**	**10.32**	**1.3**

Table 1.2-1: Sentinels Launch Windows in 2015 and 2017

1.3 Sentinels Launch Scenarios

The four (4) IHS spacecraft will be launched using a single launch vehicle. The nominal release scenario starts at approximately L+2 hours and ends two hours later with spacecraft released every 40 minutes with the three adapter rings released in between. In this nominal scenario, the release of all of the spacecraft should occur within view of a DSN station. The spacecraft release (delta v) directions would nominally be 5-10 degrees apart to increase any possible close approach distances to an acceptable level (close to the separation at release). The release scenario above has been discussed with Kennedy

Space Center personnel and seems to be possible with an Atlas 5 or Delta 4 launch vehicle.

Spacecraft release and separation analysis was performed for the release scenario above. Four spacecraft with masses of 750 kg each were released from a stacked configuration. An Atlas 5 second stage mass of 2200 kg was assumed for this analysis. It was assumed the spring release mechanism nominally provided a delta v of 1.0 m/s to a single spacecraft in the direction of ecliptic normal (relative to the pre-release state). It was assumed the spring applied an equal total impulse in opposite directions during release. If the spacecraft release delta v error could be reduced to less than approximately 5-6% for the scenario above, there should be no post-release close approaches of the spacecraft even in a case where the spacecraft were all released in the same direction. Larger release delta v errors can result in post-release close approaches if the spacecraft release delta v's are applied in the same direction. However, if the spacecraft release (delta v) directions are offset by 5-10 degrees, post release close approach distances can be increased significantly to a level not much smaller than the release distance, and this is the release scenario that would nominally be used.

The first accurate orbit determination solution would nominally be obtained between approximately L+12 and L+24 hours. The first trajectory correction maneuver (TCM1) on the Earth-Venus leg would nominally occur between L+10 and L+13 days for the four Sentinels spacecraft. The TCM1 maneuvers will be performed to insure the spacecraft arrive at Venus at different epochs. Small changes to the TCM1 maneuvers should allow separations of the Venus flyby epochs of 15-30 minutes. This will insure spacecraft separation at flyby 1 while insuring the separation is not larger than the approximately 100,000 km desired for (science) checkout during the Earth to Venus transfer. For two Sentinels spacecraft (September 4, 2015 launch case) with Venus flyby 1 periapsis epochs separated by approximately 30 minutes, the spacecraft were separated by approximately 20,000 km at Venus flyby 1 periapsis and approximately 50,000 km maximum during the Earth to Venus transfer.

1.4 Sentinels Baseline Orbit Trajectories

Figure 1.4-1 is a heliocentric view of a Sentinels-1 trajectory (9/4/2015 launch). There are 3 Venus flybys. The trajectory is green from launch through the first Venus flyby, purple after the first Venus flyby, cyan (light blue) after the second Venus flyby, and gold after the third Venus flyby. The Earth and Venus orbits are blue and red respectively.

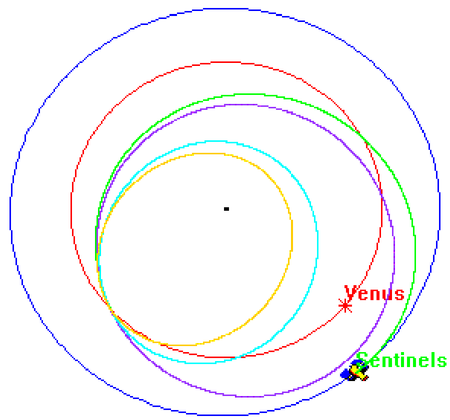

Figure 1.4-1: Heliocentric View of Sentinels-1 Trajectory for September 4, 2015 Launch
(green: launch to Venus flyby1, purple: Venus flyby 1 to flyby 2, cyan: Venus flyby 2 to flyby 3, gold: post-Venus flyby 3, blue: Earth, red: Venus)

An ~ 0.25 x 0.74 AU heliocentric orbit can be achieved with a C3 of ~ 20-30 km2/s2 and VHP of more than approximately 10 km/s using 3 Venus flybys. Nominally, the heliocentric ecliptic inclination of the spacecraft will be the minimum possible, the absolute value of the heliocentric ecliptic declination of Venus at the epoch of the flybys.

Details concerning the orbits of the 4 Sentinels spacecraft for the 9/4/2015 launch opportunity follow. With the exception of the portion of the Sentinels-3 and Sentinels-4 trajectories between flybys 2 and 3, the heliocentric inclination of the orbits is zero degrees.

- Sentinels-1
  - 0.25 x 0.74 AU final orbit, 127.15 day period, 0 degree ecliptic inclination
  - Three Venus flybys (2/14/2016, 9/26/2016 (flyby 1+224.7 days), 12/20/2017 (flyby 2+449.4 days)
  - Minimum flyby altitude 1079 km
- Sentinels-2
  - 0.28 x 0.76 AU, final orbit 136.93 day period (127.15*7.0/6.5) , 0 degree ecliptic inclination
  - Three Venus flybys (2/14/2016, 9/26/2016, 12/20/2017)
  - Drifts app. 180 degrees from Sentinels-1 in 2.5 years (drift rate not constant over orbit because of orbit eccentricity)
  - Minimum flyby altitude 1079 km
- Sentinels-3
  - 0.26 x 0.74 AU final orbit, 129.23 day period, 0 degree ecliptic inclination
  - Four Venus flybys (2/14/2016, 9/26/2016, 5/9/2017 (flyby 2+224.7 days), 8/1/2018 (flyby 3+449.4 days)
  - Ecliptic inclination of 2 degrees between flybys 2 and 3

- Will drift toward Sentinels-1 over a 5 year period after Sentinels-1 final flyby (drift rate not constant over orbit because of orbit eccentricity)
- Minimum flyby altitude 902 km

- Sentinels-4
    - 0.25 x 0.74 AU final orbit, 127.15 day period, 0 degree ecliptic inclination
    - Four Venus flybys (2/14/2016, 9/26/2016, 5/9/2017, 8/1/2018)
    - Ecliptic inclination of 2 degrees between flybys 2 and 3
    - Minimum flyby altitude 902 km
    - App. 44.0 degrees ahead of Sentinels-1 at flyby 4 (heliocentric)

Selected data for the 2015, and 2017 launch trajectories follows in Table 1.4-1. The Sentinels-1 and Sentinels-4 perihelion radius is 0.25 AU for all cases after the final flyby.

Launch Date	**Minimum Flyby Altitude (km)**	**Sentinels-1,4 Post-Final Flyby Heliocentric Period (days)**	**Sentinels-1,4 Heliocentric Angular Separation at Sentinels-4 Final Flyby (deg)**
8/26/2015	**810**	**128.02**	**49.4**
9/4/2015	**902**	**127.15**	**44.0**
3/9/2017	**554**	**123.59**	**28.8**
3/19/2017	**737**	**123.61**	**29.1**

Table 1.4-1: Selected data for 2015 and 2017 launch trajectories

With a Delta 4 or Atlas 5 launch vehicle (where Delta 4 represents the worst case errors), the first trajectory correction maneuver (TCM1, Earth-Venus leg) delta v is estimated at 30 m/s; this maneuver will nominally be performed at L+10,11,12,13 days for Sentinels-1,2,3,4. The TCM1 analysis was performed using the larger of the Delta 4 and Atlas 5 errors provided (Delta 4). The final pre-Venus flyby trajectory correction maneuvers will nominally be performed at flyby-15,14,13,12 days for Sentinels-1,2,3,4. The first post-Venus flyby trajectory correction maneuvers will nominally be performed at flyby+12,13,14,15 days for Sentinels-1,2,3,4. Orbit maneuvers required to achieve different Venus flyby geometries from the similar inbound trajectories should nominally be small if the maneuvers are done well in advance of the Venus flyby. The pre-Venus flyby trajectory correction maneuver delta v allocation is estimated at 10 m/s per flyby based on previous mission analysis. The Sentinels-3 and Sentinels-4 spacecraft will reach heliocentric ecliptic inclinations approximately 2 degrees greater than nominal between flybys 2 and 3. The spacecraft nominally have a 100 m/s delta v capability per spacecraft, sufficient to meet requirements above. The propulsion system will be capable of applying an orbit maneuver delta v in any direction. The Sentinels delta v requirements will be studied in more detail in the future.

The number of Venus flybys might be reduced by one for each Sentinels spacecraft by targeting an approximately 168.5 day post-flyby heliocentric orbit period. The time required to achieve the final orbits and the final orbits would be unchanged. There are potential problems with this approach. This approach would reduce the minimum Venus flyby altitude to ~ 350 km for the 2015 launch window. It would not be possible to achieve a 0.25 AU perihelion for the identified 2017 launch window using this approach.

For the trajectories discussed so far the Venus flybys are separated by integer multiples of the Venus orbital period; therefore, there is not significant separation of the perihelion Right Ascensions of the Sentinels spacecraft. A significant separation of the perihelion Right Ascensions of different Sentinels spacecraft could be desirable from a science perspective. This perihelion separation can be achieved if Venus flybys are separated by non-integer multiples of the Venus orbital period. Such a scenario would be expected to also result in a different heliocentric inclination, which could be desirable for launch cases with high heliocentric ecliptic inclinations (for example, the 2012 Type 2 launch opportunities). This analysis was done for the September 4, 2015 launch case. Data for that orbit follows:

- 0.25 x 0.74 AU final orbit (post-Venus flyby 3), 127.52 day period, 3.21 degree ecliptic inclination
- Three Venus flybys (2/14/2016, 12/4/2016, 2/26/2018 (flyby 2+449.4 days)
- Minimum flyby altitude 4929 km

The heliocentric ecliptic right ascension of perihelion for this spacecraft is app. 86 degrees greater than the Sentinels-1 spacecraft with a final flyby epoch of December 20, 2017. See Figure 1.4-2 is for a heliocentric view of this trajectory (9/4/2015 launch).

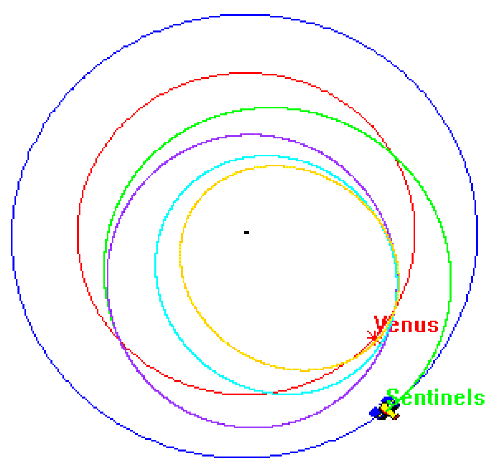

Figure 1.4-2: Heliocentric View of a Possible Sentinels Trajectory for September 4, 2015 Launch
(green: launch to Venus flyby1, purple: Venus flyby 1 to flyby 2, cyan: Venus flyby 2 to flyby 3, gold: post-Venus flyby 3, blue: Earth, red: Venus)

In Figure 1.4-2 notice the separation of post-Venus flyby 3 perihelion compared with Figure 1.4-1. Similar results would be expected for the other Type 2 trajectories identified. For larger VHPs, the angular separation of perihelion would be expected to be slightly larger; and for smaller VHPs, the angular separation of perihelion would be expected to be slightly smaller.

The angle between the heliocentric orbit plane and the spacecraft to Earth line is a parameter of interest for spacecraft antenna design. This parameter was analyzed for the 2/18/2014, 8/26/2015, 9/4/2015, 3/9/2017, and 3/19/2017 launch trajectory cases. For the spacecraft with the largest heliocentric ecliptic inclinations (2/8/2014, 3/9/2017, and 3/19/2017 launch cases), that angle was approximately 5-9 degrees in the days after launch and decreased to less than 1 degree at the first Venus encounter. Between Venus flybys 2 and 3 of Sentinels-3 and Sentinels-4 (the period of higher ecliptic inclination) that angle was approximately 6.4 degrees maximum. With a heliocentric ecliptic inclination of 1.3 degrees and with maximum heliocentric ecliptic declination near aphelion, the maximum value of that angle after the final flyby would be approximately 5.4 degrees.

1.5 Sentinels Navigation

Nominally, DSN 2-way range and Doppler data will be used for spacecraft navigation. DSN Delta-DOR data may be used prior to Venus flybys. There will nominally be no need for optical navigation. There are no science-related orbit determination requirements beyond the orbit determination necessary to meet the mission requirements. From a navigation perspective it is desirable to minimize thruster perturbations.

Post launch (per spacecraft) there will nominally be near continuous tracking from launch to launch+2 weeks. There will nominally be 5 DSN passes per week from launch+2 to launch+4 weeks. There will nominally be 3 DSN passes per week from launch+4 to launch+6 weeks.

For Venus flybys (per spacecraft), there will nominally be 3 DSN passes per week from flyby-5 weeks to flyby-1 week. There will nominally be 1 DSN pass per week from flyby-1 to flyby+1 week.

During the nominal mission phase, there will nominally be two 8 hour DSN tracking passes every 3 weeks per spacecraft. A Solar conjunction of 50-60 days should not be a problem assuming accurate orbit determination beforehand. In addition, the duration of the effects of the Solar conjunction would be expected to be longer for transmission of high rate science data than for tracking data.

Given the demands of deep space navigation and the fact that there are 4 Sentinels spacecraft, DSN loading is an issue that will need to be studied, especially after launch and prior to Venus flybys.

2.1 Baseline Solar Orbiter Mission Design

The Solar Orbiter mission design is based on a transfer phase and an inclination-raising phase (core of the science operations), which will provide unprecedented views of the Solar poles. The overall mission design aims to reduce the perihelion distance to approximately 0.22 AU (permitting the Orbiter to move in near-synchronism with the solar surface for periods of a few days), and to gradually increase the orbital inclination to more than 30 degrees with respect to the solar equator through repeated Venus flybys. The nominal Solar Orbiter trajectory utilizes a Venus flyby followed by two Earth flybys and a series of Venus flybys to achieve the mission objectives.

To satisfy the scientific goals of Solar Orbiter (SolO) it is required to perform near-Sun interplanetary measurements as close as 48 – 50 solar radii (about 0.22 AU, permitting the Orbiter to move in near-synchronism with the solar surface for periods of a few days), and, over extended periods, observations of the polar regions from heliographic latitudes as high as 32º - 35º. Also, the far side of the Sun not visible from the Earth shall be observed.

The mission is based on a single 3-axis stabilized spacecraft launched by a Soyuz launcher. The SolO mission design is based on a transfer phase and an inclination-raising phase. In both phases Gravity Assist Maneuvers (GAM) with Venus and Earth are used. The nominal SolO transfer trajectory utilizes a Venus flyby followed by two non-resonant Earth flybys and a Venus flyby at more than 18 km/s relative velocity. This high relative velocity is needed to increase the orbital inclination during the operational phase. At the end of the transfer phase, the perihelion distance is between 48 and 50 solar radii (approximately 0.22 AU), and the period about 150 days. The duration of the transfer phase is about 3.5 years. At the end of that phase, the orbital inclination with respect to the solar equator is a few degrees, but, during the inclination raising phase, a series of resonant 3:2 Venus flybys, gradually increase the orbital inclination to more than 30 º.

Deep Space Maneuvers (DSM) with Chemical Propulsion may be required depending on the day of launch.

The first Venus flyby will occur approximately 6 months after launch. After 10.6 months it follows an Earth GAM, and 22.3 months later a non-resonant Earth GAM. 3.5 years after launch, a Venus GAM injects the spacecraft into an orbit of 0.23 x 0.88 AU and low inclination relative to the solar equator. After the second Venus GAM, a sequence of high energy resonant GAM with Venus will gradually increase the inclination of the orbit. About 7 to 9 years after launch the orbit inclination relative to the solar equator will reach a value greater than 32 º.

Launch is foreseen from Kourou using a Soyuz/ST version 2-1b with a Fregat upper stage.

2.2 Solar Orbiter Launch Windows

The nominal launch of SolO is 2015, and a back up possibility occurs one Earth-Venus synodic period (approximately every 584 days) in 2017.

Data for launch in 2015 follows in Table 2.2-1. C3 is the square of the post-launch hyperbolic excess velocity (VHP) with respect to Earth. The VHP with respect to the Earth has been fixed to 3.59 km/s. Declination of launch asymptote (DLA) is the Declination of post-launch hyperbolic excess velocity with respect to Earth. The DLA is allowed to vary when optimizing the transfer phase. To achieve a Heliocentric orbit with perihelion of 0.22 AU, and to increase the inclination, the VHP at second Venus flyby must be approximately 18 km/s or greater. The total DSM for launch in the period May 7 to 24, 2015 is less than 90 m/s.

The current launch vehicle payload mass required for the SolO spacecraft with margin is 1250 kg. This payload can be achieved by the Soyuz-Fregat launcher system for DLA between -20º and -10 º.

Depart Earth	C3 (km2/s2)	DLA (deg)	Venus Flyby 1	VHP Venus Flyby 1 (km/s)	Venus Flyby 2	VHP Venus Flyby 2 (km/s)	Heliocentric Ecliptic Inclination
5/07/2015	**12.9**	**-12.2**	**11/17/2015**	**5.4**	**10/10/2018**	**18.4**	**33.4**
5/17/2015	**12.9**	**-17.5**	**11/24/2015**	**5.6**	**10/10/2018**	**18.5**	**33.6**
5/24/2015	**12.9**	**-20.0**	**11/27/2015**	**5.8**	**10/10/2018**	**18.6**	**33.8**

Table 2.2-1: Solar Orbiter Launch Windows in 2015

2.3 Solar Orbiter Baseline Orbit Trajectories

Figure 2.3-1 presents the ecliptic projection of the SolO trajectory for a launch on 5/22/2015. The trajectory extends up to March 2026 and includes 7 Venus flybys. After the 2nd Venus flyby, the heliocentric orbit is 0.23 x 0.88 AU with an inclination respect the solar equator of 10º. The orbit progressively changes to 0.36 x 0.75 with 34º at the end of mission life.

Details concerning the orbit of the SolO spacecraft for the 5/22/2015 launch opportunity follow:

- Launch 5/22/2015
- Venus flyby on 11/27/2015 with VHP = 5.4 km/s, altitude 8300 km

- Earth flyby 1 & 2 on 10/9/2016 and 8/8/2018, respectively, with VHP of 9.3 km/s. Altitude of 4860 km, 500 km, respectively.
- End of transfer, Venus flyby on 10/10/2018.
- Operational phase and inclination raising in orbit with a period of 150 days, and Venus flyby every 450 days (resonance 3:2)

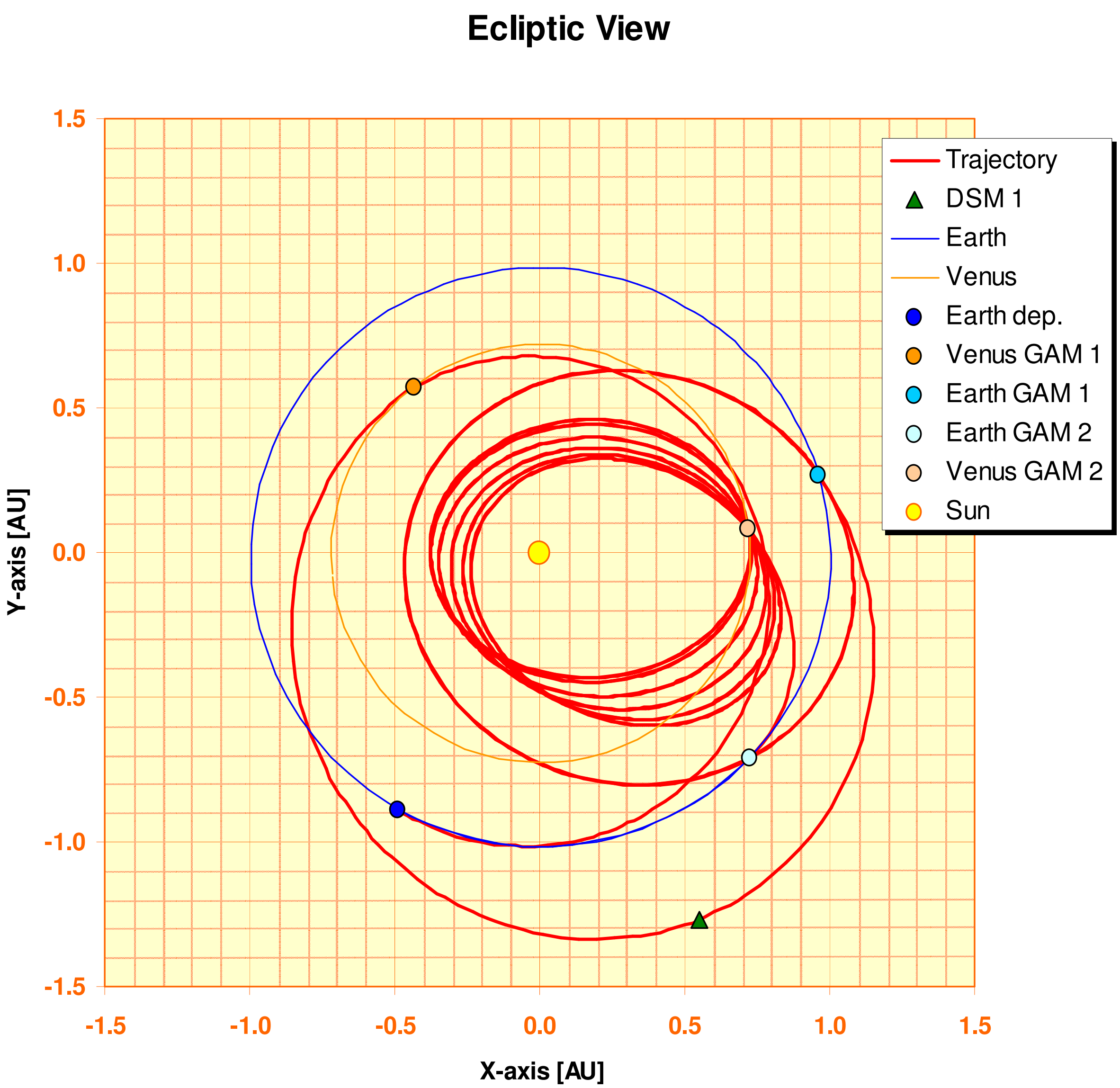

Figure 2.3-1: Ecliptic View of SolO Trajectory for May 22, 2015 Launch

With a Soyuz+Fregat launch vehicle, the first trajectory correction maneuver (TCM1, Earth-Venus leg) delta v is estimated at 17 m/s; this maneuver will nominally be performed around 2 weeks after launch. The trajectory correction maneuver allocation is estimated at 135 m/s based on previous mission analysis. DSM and navigation maneuvers are covered within the spacecraft maneuvering capability of 250 m/s, sufficient to meet requirements above.

3.1 Launching Sentinels and Solar Orbiter on the Same Launch Vehicle, Mission Design

Preliminary analysis has been performed to assess the feasibility of launching 3 IHS spacecraft and Solar Orbiter on a single expendable launch vehicle, nominally an Atlas V 551, with launch occurring between 2015 and 2018. Since this abstract was submitted it was determined that the IHS and Solar Orbiter spacecraft would be launched on separate launch vehicles. This paper documents the analysis results obtained at the time that decision was made.

This analysis focused on trajectories which meet the IHS requirements at the first Venus flyby with no deterministic IHS delta v requirement and which meet the SolO requirements using additional Earth and Venus and possibly Mercury flybys. The Venus flyby 1 VHPs of the nominal Solar Orbiter trajectories were too low (less than 6 km/s) to meet the IHS requirements at that epoch.

The minimum Solar Orbiter requirements for a joint IHS/Solar Orbiter launch were defined by the ESA science team. The Solar Orbiter spacecraft would need to achieve a minimum perihelion radius of between 0.225 and 0.25 AU and a maximum Heliographic latitude of 25.0 degrees or more. Per reference 3 the nominal Solar Orbiter trajectory for a 2015 launch would achieve a minimum perihelion radius of 0.225 AU and a maximum Heliographic latitude of 34.2 degrees. In addition, the NASA and ESA teams determined that a maximum C3 of 26.5 km2/s2 would allow Solar Orbiter and the 3 IHS spacecraft to be launched on the same Atlas V 551 (with margin) assuming a launch DLA between -28.5 and +28.5 degrees. For this study the Sentinels Science Team requested a determination be made of whether or not a final IHS Heliographic inclination of 1.0 degrees or less would be possible for the launch opportunities identified and whether or not a Solar Orbiter Heliographic inclination of 1.0 degrees or less would be possible for the portion of the Solar Orbiter trajectory with minimum perihelion radius.

A number of candidate trajectories were determined. This effort initially focused on Solar Orbiter trajectories requiring no deterministic delta v. However, Solar Orbiter has the capability to apply a deterministic delta v, and these Solar Orbiter trajectories could have been analyzed at a future date. Two trajectories were deemed most promising by the IHS and Solar Orbiter science teams and were analyzed most extensively, and those results are presented in this paper. In addition to the Venus flybys, those two cases analyzed using two Earth flybys (Section 3.2) and one Earth flyby with and without a Mercury flyby (Section 3.3).

3.2 Mission Design Using Two Earth Flybys

Trajectories meeting the requirements for launching the IHS and Solar Orbiter spacecraft on the same launch vehicle were determined for launches in 2015 and 2017; the Solar Orbiter spacecraft used a Venus flyby followed by two Earth flybys and a series of Venus flybys. No deterministic delta v's were required for any of these Solar Orbiter trajectories. Starting with Venus flyby 3 the spacecraft performs a series of Venus flybys every 2 Venus orbits (every 3 spacecraft orbits, 449.4 days). Minimum Solar Orbiter perihelion distance occurred after Venus flyby 3 (2 perihelion passes). A minimum flyby altitude of 500 km for (Solar Orbiter) Venus flybys was assumed. No issues were noted with the IHS trajectories. Most of the remainder of Section 3 will focus on the Solar Orbiter trajectories.

For the 2015 launch 2 Earth flyby Solar Orbiter case a 20 day launch window with a maximum C3 of app. 21.6 km2/s2 would be possible. Heliographic inclinations of less than 1.0 degrees will be possible for IHS and for SolO (during minimum perihelion phase). An increase in launch C3 can decrease SolO minimum perihelion radius and slightly increase SolO maximum Heliographic inclination; the analysis initially focused on these cases. Table 3.2-1 and Figure 3.2-1 provide data for the 2015 launch opportunity.

Launch (MM/DD/YYYY)	**08/20/2015**	**08/30/2015**	**08/30/2015**	**09/09/2015**
C3 (km^2/s^2)	**25.16**	**24.13**	**21.54**	**24.39**
DLA (deg)	**-24.66**	**-22.85**	**-23.64**	**-21.61**
Venus flyby 1 (V1)	**02/10/2016**	**02/14/2016**	**02/06/2016**	**02/10/2016**
V1 VHP (km/s)	**10.92**	**11.00**	**10.40**	**10.66**
Post-V1 aphelion (AU)	**1.51**	**1.52**	**1.50**	**1.51**
Earth flyby 1 (E1)	**12/24/2016**	**01/01/2017**	**12/23/2016**	**12/28/2016**
E2	**10/13/2018**	**10/20/2018**	**10/12/2018**	**10/17/2018**
V2	**05/08/2019**	**05/13/2019**	**05/07/2019**	**05/11/2019**
V2 periapsis alt (km)	**720**	**1030**	**860**	**1000**
V3	**02/12/2020**	**02/17/2020**	**02/12/2020**	**02/15/2020**
V3 VHP	**17.71**	**17.88**	**17.45**	**17.64**
Post-V3 perihelion x aphelion (AU)	**0.238 x 0.87**	**0.236 x 0.87**	**0.241 x 0.86**	**0.239 x 0.87**
Post-V3 Heliographic Inclination	**0.16 deg**	**0.35 deg**	**0.21 deg**	**0.14 deg**
IHS Minimum Heliographic Inclination	**0.40 deg**	**0.04 deg (0.25 x 0.74 AU)**	**0.74 deg (0.25 x 0.73 AU)**	**0.32 deg**
SOLO Maximum Heliographic Inclination	**34.0 deg**	**34.3 deg**	**33.5 deg**	**33.9 deg**

Table 3.2-1: 2015 Launch Opportunities with Two Earth Flybys

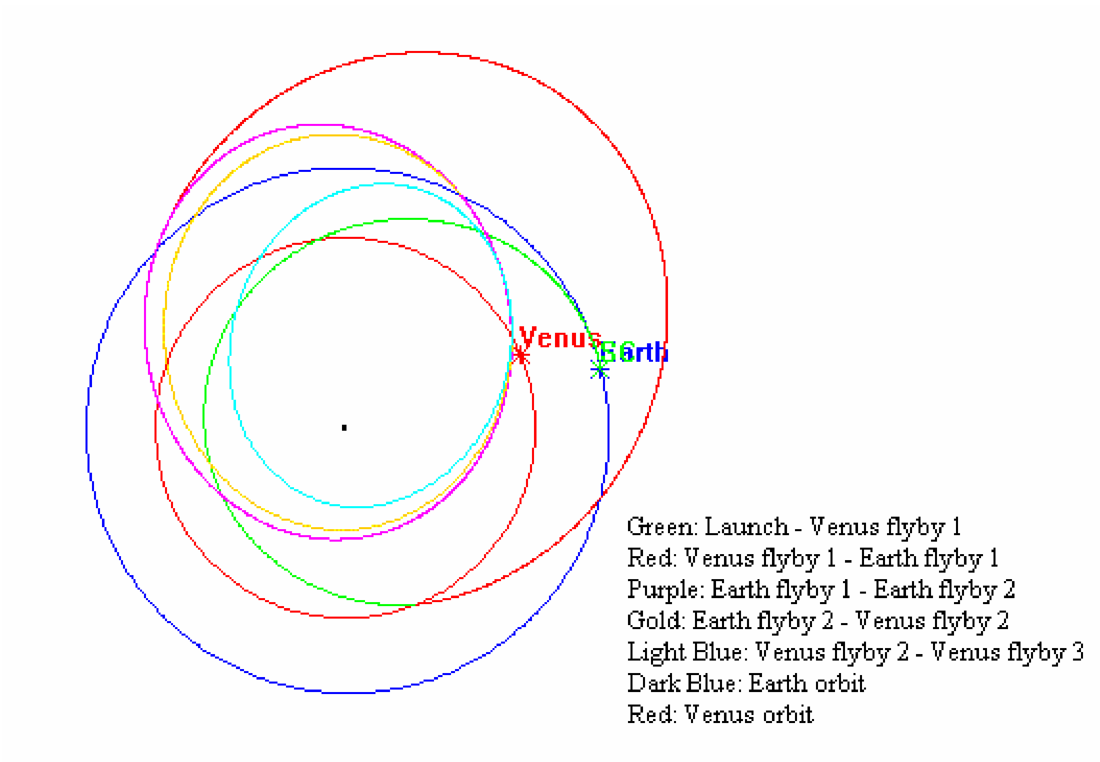

Figure 3.2-1: 2 Two Earth Flyby Trajectory (20150830 Launch, 20160214 Venus flyby 1, Heliocentric View through Venus flyby 3)

For the 2017 launch 2 Earth Flyby Solar Orbiter case a 20 day launch window with a maximum C3 of app. 23.0 km2/s2 should be possible. However, Heliographic inclinations of less than 1.0 degrees will be possible for IHS and for SolO will not be possible for this launch opportunity. Preliminary analysis indicated a one Earth flyby trajectory similar to the trajectory described in Section 3.3 with the Mercury flyby eliminated could also meet requirements.

3.3 Mission Design Using Earth and Mercury Flybys

Trajectories meeting the requirements for launching the IHS and Solar Orbiter spacecraft on the same launch vehicle were determined for launches in 2015; the Solar Orbiter spacecraft used a Venus flyby followed by an Earth flyby, a Mercury flyby, and a series of Venus flybys. No deterministic delta v's were required for any of these Solar Orbiter trajectories. For the 2015 launch Earth and Mercury flyby Solar Orbiter case a 20 day launch window with a maximum C3 of app. 21.9 km2/s2 would be possible. A minimum perihelion radius of 0.237 AU, and a maximum heliographic inclination of 35.5 degrees would be possible. Figure 3.3-1 provides data for the 2015 launch opportunity. This option achieves a higher final Venus VHP and therefore a higher maximum Heliographic latitude than the two Earth flyby approach. This option also achieves the first perihelion of less than 0.30 AU (a parameter of interest to the Sentinels Science Team) 2.83 years after launch, compared with 4.08 years for the two Earth flyby approach. With this option Heliographic inclinations of less than 1.0 degrees will not be possible for IHS and for SolO. However, the use of a Mercury flyby would have required additional analysis of thermal, geometry, and navigation issues associated with the Mercury flyby because the original SolO mission design did not include a Mercury flyby.

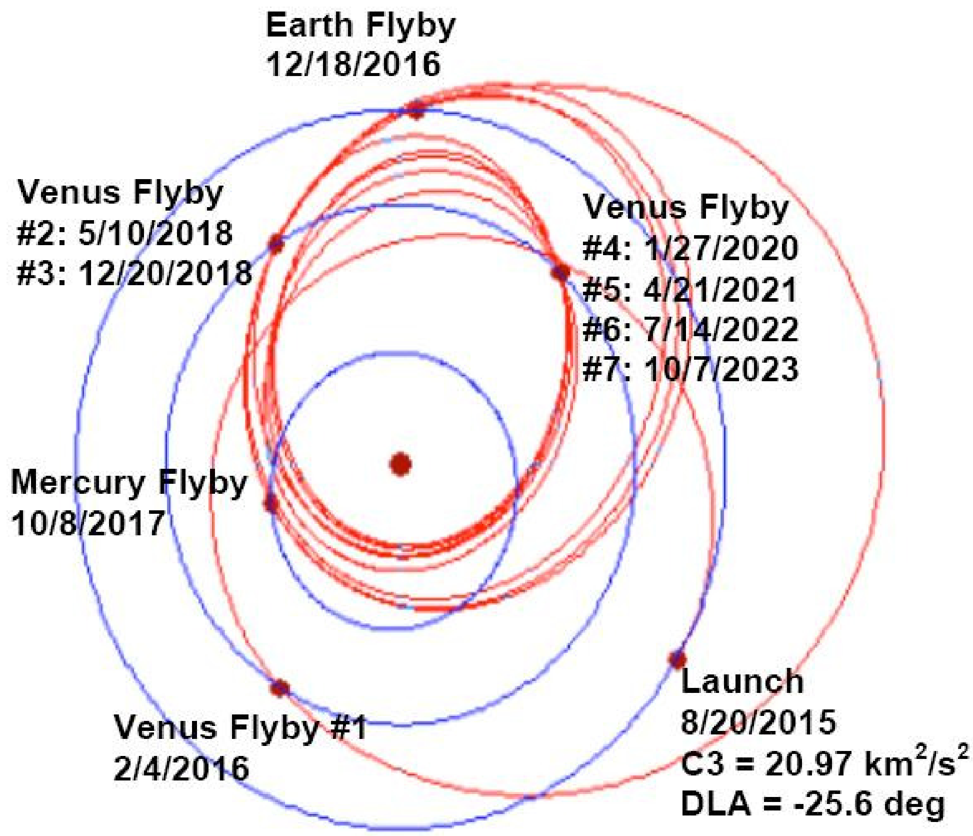

Figure 3.3-1: 2 Earth and Mercury Flyby Trajectory

3.3 Launching Sentinels and Solar Orbiter on the Same Launch Vehicle, Mass, Volume, Clampband

Several payload and launch vehicle configurations were examined to meet the minimum mission parameters and payload configurations. The Atlas V (551) launch vehicle was the most viable U.S. Launch Vehicle able to meet the required launch mass and C3 needed to achieve proper mission orbit parameters. A Delta IV Heavy ELV can meet the orbit parameters but at a significant additional monetary cost.

In order to meet launch vehicle mass and structural load requirements with adequate margins, one of the original four Sentinels would be replaced by Solar Orbiter with a maximum of 1250 kg (wet mass). Each Sentinels spacecraft has been uniquely designed for structural loading depending on its placement in the stack. This approach optimizes mass but requires a specific stacking configuration (see Table 1). Therefore spacecraft are not interchangeable in terms of physical placement for launch although they are identical in functionality. The estimated combined mass of the IHS-Solar Orbiter configuration is 3867 kg (with margin). The maximum C3 possible while accommodating the prescribed 1250 kg Solar Orbiter spacecraft is 26.5 $km^2/sec^{2.}$ The Solar Orbiter was placed on top of Sentinel spacecraft stack with an assumed Center of Mass (CM) in the geometric center of the spacecraft. The mechanical structural analyses were static calculations. Mass margins of 30% were included in all mass estimates and a structural Factor of Safety of 1.25 was applied.

The baseline Saab 66-inch/1666 (mm)VS clampband for the baseline Sentinels mission cannot bear the moment loads with the heavier Solar Orbiter at the top of the stack. Saab has indicated that they are developing an improved version of this clamp system that can provide adequate margins as calculated in the chart below. Volumetrically, all four spacecraft fit well within the Atlas 5m payload medium faring per Appendix D of the Altas Mission Planner's Guide. Using the Sentinels–SolO stack configuration with a CG of 3.9m, the moment loads exceed the existing launch vehicle clamp interface even with minimum margins/factors of safety. However, with appropriate design factors and clamp band load margins, the stack remains just under the CG envelope for the new prototype Saab 1666S clamp band with 2% design margin remaining. This case is inclusive of 30% mass margins, an additional 1.25 launch vehicle load design margins and an additional 1.25 design margin on clampband line load capability

References

1. Solar Sentinels STDT, "Solar Sentinels: Report of the Science and Technology Definition Team", NASA/TM-2006-214137, 2006

2. Solar Orbiter Final Executive Report, ESA, December 15, 2005

3. Conde, R., Szabo, A., Maldonado, H., Marr, G., et al, "Optimization of Inner Heliospheric Sentinels Spacecraft Conceptual Design", Solar Orbiter Workshop, Athens, Greece from October 15 through October 20, 2006

4. Labunsky, Alexei V., Multiple Gravity Assist Interplanetary Trajectories, 1998

5. Kaplan, Marshall H., *Modern Spacecraft Dynamics and Control*, 1976

6. Wertz, James R., *Spacecraft Attitude Determination and Control*, 1984