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HELIOSWARM: SWARM MISSION DESIGN IN HIGH ALTITUDE ORBIT FOR HELIOPHYSICS

Laura Plice,* Andres Dono Perez,† and Stephen West‡

Resolving the complex three-dimensional turbulent structures that characterize the solar wind requires contemporaneous spatially and temporally distributed measurements. HelioSwarm is a mission concept that will deploy multiple, co-orbiting satellites to use the solar wind as a natural laboratory for understanding the fundamental, universal process of plasma turbulence. The HelioSwarm transfer trajectory and science orbit use a lunar gravity assist to deliver the ESPA-class nodes attached to a large data transfer hub to a P/2 lunar resonant orbit. Once deployed in the science orbit, the free-flying, propulsive nodes use simple Cartesian relative motion patterns to establish baseline separations both along and across the solar wind flow direction.

INTRODUCTION

Plasmas are ubiquitous in the universe and are governed by highly dynamic, coupled physical processes operating on a vast range of scales from the microphysical (electron kinetics) to the galactic (magnetohydrodynamics, MHD). Spatially and temporally coordinated multi-point measurements are required for understanding these fundamental processes. The HelioSwarm mission, answering NASA's 2019 Heliophysics Medium Explorer (MIDEX) announcement of opportunity, will deploy multiple co-orbiting satellites to use the pristine solar wind as a natural laboratory for understanding the fundamental, universal process of plasma turbulence.

HelioSwarm's design requirements are similar to many missions conducting deep space studies: high altitudes for data collection with relatively brief returns to lower altitudes for expedient data downlink. Previous multi-satellite missions such as MMS[1] and Cluster II[2] have demonstrated the potential of multi-point, spatially and temporally resolved measurements. These missions form tetrahedra between four spacecraft to obtain measurements at a single scale at any one point in time. HelioSwarm represents a new capability for multi-satellite missions by flying a swarm of nine spacecraft (8 nodes, 1 hub) to make measurements at multiple scales simultaneously. The HelioSwarm mission concept uses a large hub for transport of the smallsat nodes to the mission orbit and data relay. Requirements for distributed observations address satellite-satellite "baseline" relationships: simultaneously sample baselines with components along and transverse to the solar wind flow direction. These components must span multiple spatial regimes: less than ion

* Systems Engineer V, Metis Technology Solutions, NASA Ames Research Center, Moffett Field, California 94035, U.S.A. MS 202/3
† Aerospace Engineer, Millennium Engineering and Integration Co., NASA Ames Research Center, Moffett Field, California 94035, U.S.A.
‡ Aerospace Engineer, Metis Technology Solutions, NASA Ames Research Center, California 94035, U.S.A.

kinetic scales (100 km), within the MHD regime (1200 km), and within the transition region between 100 and 1200 km. Requirements include 3D and polygonal configurations in three solar wind regions: the pristine solar wind, Earth's magnetosphere, and the magnetically connected region.

Engineering considerations impose further requirements on the mission design. All nodes Power and thermal limitations of the ESPA-class node spacecraft require limiting eclipses to no longer than 3 hours. Radiation concerns require avoidance of the Van Allen Belts in the science orbit. End of mission planning mandates that the science orbit permanently avoid crossing geostationary altitudes. Node relative motion design includes periodic approaches to the vicinity of the hub to increase hub-node crosslink data rate.

TRANSFER TRAJECTORY DESIGN

The HelioSwarm transfer trajectory consists of several phases that involve a Trans-Lunar Injection (TLI), phasing loops, and a lunar gravity assist that results in a perigee altitude raise to finally obtain a P/2 Earth-Moon resonant orbit. The resonant state is achieved by a Phasing Adjustment Maneuver (PAM) that occurs right after the lunar flyby at the next perigee. This maneuver sets the final science orbit lunar period to half the period of the Moon's orbit. This approach was previously introduced for the trajectory design of the Transiting Exoplanet Survey Satellite (TESS) mission[3,4], and it was also used for other designs such as the proposed mission Arcus, in that case to achieve a P/4 orbit.[5]

The trajectory design process starts with the selection of an appropriate launch date and time. The baseline launch site is set to NASA Kennedy Space Center and the requirements of the mission estipulate a launch date no later than February 2026. Several solutions were found for almost any launch date during the month of January, and the baseline Design Reference Mission (DRM) is set to the 4th of January of 2026. The selection of a given launch date and time is paramount for targeting the lunar gravity assist since the trajectory design involves the setting of the line of apsides, which needs to be in the Earth-Moon plane to ensure a lunar encounter. In addition to that, the magnitude given by the Trans-Lunar Injection (TLI) burn to gain enough C3 energy cannot be higher than -2.75±0.05 km^2/s^2, due to launch vehicle performance constraints. Hence, this parameter is targeted to avoid having a larger magnitude for the TLI burn. In addition, the coast time that the spacecraft is in Low Earth Orbit (LEO) before TLI, is also a design parameter that is adjusted to find the right orbit plane to target the lunar flyby to the desired outcome, which involves an inclination change and a perigee raise.

After TLI, the Hub spacecraft coasts for three eccentric orbits to time the lunar gravity assist, each of them have a high apogee altitude in the order of 25000-40000 km and a short perigee altitude in the order of 1000-3000 km. In order to time the lunar encounter and to achieve the necessary energy to set the apogee to the lunar distance, we budget a total of three Short Phasing Maneuvers (SPM), two nominal and one statistical. This technique, known as phasing loops, was successful in previous lunar gravity assist missions and has a strong heritage, tracking back to missions such as IBEX, LADEE, and TESS. Phasing loops have numerous benefits including wider launch window, more flexibility to launch vehicle performances, and higher tolerance to launch vehicle insertion dispersions.

In contrast to some other trajectory designs utilizing phasing loops in the past, this mission has the particularity to require high ecliptic inclinations during the science phase. The ecliptic inclination parameter is relative to the ecliptic plane, rather than the Earth equator, and it is a paramount parameter for this mission design since high ecliptic inclinations mean less likelihood of eclipses during the science phase. Reducing the number and the duration of the eclipses is fundamental for mission feasibility, since the nodes spacecraft can only tolerate up to 3 hours eclipse durations due to power and thermal limitations. High ecliptic inclinations set the line of apsides of the final orbit away from the ecliptic plane, meaning that the time spent in the Earth-Sun plane is minimized, and hence eclipses are less likely to occur and less frequent in general. In addition, the argument of periapsis with respect to the ecliptic plane is also critical since the nodes spacecraft need to minimize the time spent in that plane during apogee. The ecliptic argument of periapsis evolves in this type of orbit, therefore, a sustainable configuration that revolves around either 90 or 270 degrees is necessary. The initial ecliptic argument of periapsis, after the PAM, must be above certain threshold, which is usually equal or higher than 35 degrees, in order to minimize the eclipse duration during the first stage of the nominal mission. For the stable solutions we found, the ecliptic argument of periapsis tends to evolve towards 90 degrees, and then revolves around that in a periodic oscillation. More details can be found in the science orbit section. The same behavior applies to both the ecliptic and equatorial orbit inclinations, as well as the eccentricity, and as a consequence of the latter, to the altitudes of apogee and perigee. Therefore, it is important to target the right configuration of eccentricity and ecliptic argument of periapsis to ensure both stability and short eclipse durations. Due to DSN communications, we prioritize north hemisphere perigee passes, i.e., argument of periapsis close to 90 degrees, rather than 270 degrees that correspond to southern latitudes during the perigee passes. The reason is the existence of two DSN ground stations: Goldstone and Madrid, as compared to just one, in Canberra. The idea is that the perigee passes can be utilized to downlink data from the Hub spacecraft, since by definition, they offer shorter ranges that are beneficial for communications.

The SPM maneuvers during the phasing loops phase occur at perigee and are always in the velocity direction. In order to account for launch dispersions and for maneuver execution errors, two other maneuvers are budgeted to correct them: an Orbit Correction Maneuver (OCM) near the second phasing loop apogee, and a final Trajectory Correction Maneuver (TCM) after the last perigee of the phasing loops and before the lunar flyby. The main orbit plane configuration is achieved at the time of launch and TLI, in order to set the necessary conditions for a desired lunar B-plane to arrive to the final inclined P/2 orbit. The lunar B-plane is therefore targeted with a B-theta that is typically higher than at least 25 degrees in magnitude, in order to change the final orbit ecliptic inclination of the P/2 orbit that initially, at the time of insertion after the PAM, ranges from 20-60 deg for the multiple solutions we found. The altitude of periselene at the flyby ranges between 5000-15000 km, in order to obtain enough energy to raise the perigee of the final orbit. The P/2 orbit perigees we found as a result, range between 60000-100000 km, depending on the launch date. Higher energies would mean that the PAM would be much larger and the transfer time much longer due to higher post-flyby apogees.

Table 1. Description of nominal transfer and science orbit. The parameters represent TLI, lunar gravity assist swingby, and the time of insertion into the science orbit.

<table>
<tr><th colspan="6">Transfer Trajectory</th></tr>
<tr><td>Parameter</td><td>B-theta at lunar periselene</td><td>Altitude of periselene</td><td>Vmag w.r.t. Moon</td><td colspan="2">C3 after TLI</td></tr>
<tr><td>Value</td><td>-31.09 deg</td><td>10489 km</td><td>1.22 km/s</td><td colspan="2">-2.75 km^2/s^2</td></tr>
<tr><th colspan="6">Science Orbit</th></tr>
<tr><td>Parameter</td><td>Perigee Altitude</td><td>Apogee Altitude</td><td>Eccentricity</td><td>Ecliptic Inclination</td><td>Equatorial Inclination</td></tr>
<tr><td>Value</td><td>78477 km</td><td>395555 km</td><td></td><td>31.03 deg</td><td>50.90 deg</td></tr>
</table>

For the nominal mission, these parameters appear in Table 1. The parameters correspond to the time of insertion, unless something else is specified in the table headings. Representative images of the nominal orbit and trajectory from and Earth Centered Inertial (ECI) point of view and from and Earth-Moon rotting frame are also represented in Figure 1. In both cases, the images show a view from the top-down perspective as well as sideways to appreciate the inclination with respect to the Earth-Moon plane and the ecliptic.

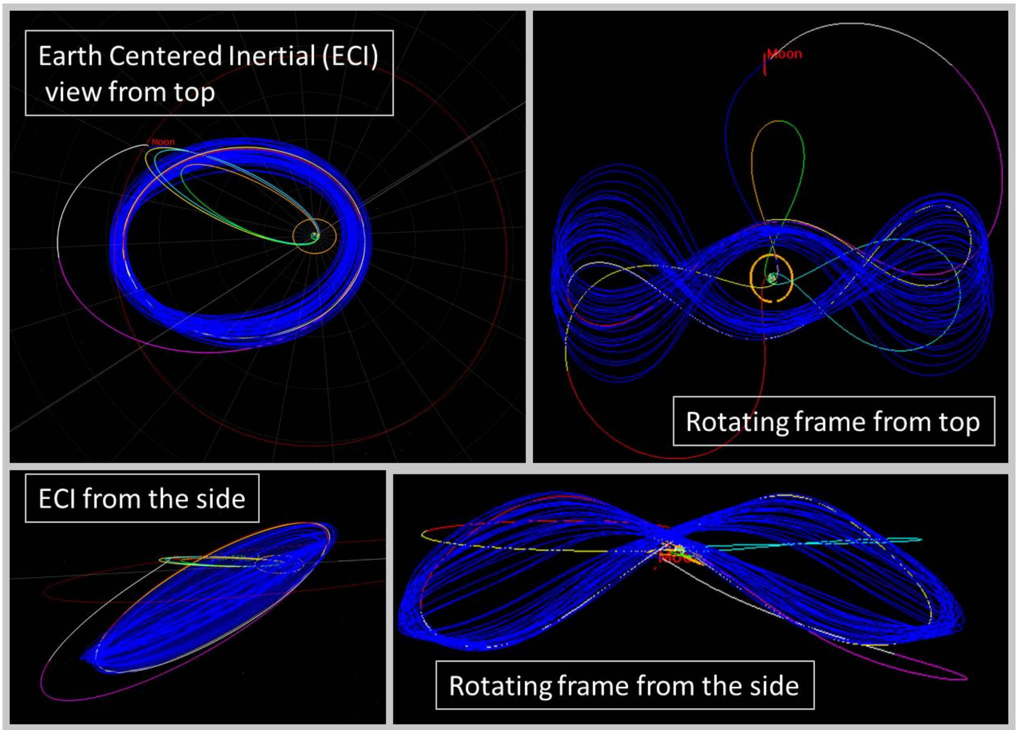

Figure 1. Images of the nominal trajectory and final science orbit (shown in blue). All the images show two years propagation from different perspectives. The upper figures show an Earth Centered Inertial frame and an Earth-Moon rotating frame viewed from the Z

Table 2 shows the nominal maneuver magnitudes to achieve this configuration in the nominal design and also the statistical assessment performed to obtain a delta-V budget to allocate enough margin. The allocation groups the phasing loops together due to their nature. Depending on the launch insertion, the first maneuvers and navigation performance, the subsequent phasing loops are recomputed, allowing for an overall maneuver budget assessment during this phase, since the nominal value of each SPM may vary but the whole allocation remain stable within a small tolerance. In addition, depending on the particular configuration of ephemeris, launch date, and TLI insertion from the launch vehicle, SPM1 may be smaller than SPM2, or vice versa. Hence they are complementary. For example, when SPM1 is small, then SPM2 may be high to achieve the desired flyby conditions, or the other way around. The high delta-V budget allocation with sufficient margin and contingency allows for design that is robust to changes in the launch date and to maneuver execution errors. Monte Carlo analyses assumed launch vehicle dispersion of up to 0.5

km^2/s^2 in magnitude and 0.1 deg in pointing offset. In addition, the budget accommodates maneuver execution errors up to 5% of the total magnitude and 5 deg pointing inaccuracies. Extra allocations cover launch date variation and launch window extension.

Table 2. ΔV budget, including statistical assessment. The Design Reference Mission (DRM) show a total magnitude of 101.26 m/s while the maximum statistical estimate based on Monte Carlo analysis, show a Maximum Expected Value (MEV) of 185.93 m/s. The assumptions appear in the right column. The total allocation leaves margin for potential ΔV growth as the design matures.

ΔV Event	DRM Case (m/s)	Statistical Current Estimate (m/s)	ΔV Allocation (m/s)	Notes
Apogee engineering burn	2	1	5	Calibration burn
Sum of Phasing Loop Maneuvers	30.63 SPM1=30.20 SPM2=0.43	30.63 to 79.7 Phasing Loop Total Dv	80	Dispersions in each propulsive event corrected in subsequent maneuvers Launch vehicle dispersion, energy: up to 0.5 km^2/s^2 Launch vehicle dispersion, plane offset: up to 0.1 deg. Maneuvers: up to 5% magnitude error, 5 deg pointing error, TCM correction at SPM3 +24hr
Variation for different launch dates	-	-8.4 to +9.6	10	Nominal cases through 1 lunar cycle, compared to baseline
Launch window extension	-	0	10	Analysis cases resilient to launch times within 15 minutes of nominal
Total for Phasing Loops	32.63	25.23 to 92.3	105	
Sum of Transfer Orbit Burns (PAM, OCM)	68.63	-25 to +25	155	Estimation based on Monte Carlo analysis
TOTAL	101.26	68.86 to 185.93	260	

SCIENCE ORBIT DESIGN

The selection of a P/2 lunar resonant orbit as the final candidate for science operations fulfills the altitude requirements, the design heritage, the ΔV savings, and the stability configuration. This orbit, while still being Earth centered, can provide high apogee altitude to comply with mission requirements, and still have relatively low perigee passes that are important for ground control communications, due to a moderately high eccentricity. The ΔV savings come from the targeted lunar gravity assist that allows for a free perigee raise and an inclination change, that are followed by a final Period Adjust Maneuver (PAM) to establish permanent lunar resonance. The nominal design apogee altitude ranges from 60.47-66.9 Re and is suitable for the science needs, while the perigee altitude ranges from 7.75-14.1 Re, which implies a sufficient buffer to avoid GEO and the high radiation areas of the outer Van Allen belts.

Our P/2 science orbit is a similar but modified version of the solution used on the TESS mission. HelioSwarm's cubesat-based nodes requirement of 3 hr maximum eclipse duration implies an intended small design change with respect to the heritage solution to achieve short eclipse durations. The node smallsats get released in the final P/2 science orbit and they must stay there for almost two years (considering both Commissioning and Science Phases), surviving with their own capabilities. Therefore, minimizing eclipse duration becomes paramount and that has implications in the characteristics of the final orbit. The HelioSwarm science orbit has the line of apsides off the ecliptic plane in order to avoid slow velocity at the plane crossing.

In this type of particular P/2 orbit, the argument of periapsis becomes fundamental too since the nature of the lunar resonance implies significant drift over time of orbital parameters. The precession is quite significant and values around 0 or 180 degrees imply long duration eclipses. To avoid that, we followed a similar approach than in the TESS mission by attempting orbits that revolve around ecliptic argument of periapsis of either 90 or 270 degrees. Those solutions tend to be more stable according to the Kozai theory[6] and, moreover, they help to avoid long eclipse durations. Some of our solutions do not particularly arrive to 90 degrees at the moment of insertion but the trend occurs in that direction, avoiding eclipses in later phases of the mission. Therefore it is important to make sure the signature of the ecliptic argument of periapsis versus the eccentricity follows that trend.

Figure 2 shows three different plots corresponding to the nominal mission, the signature of the eccentricity-argument of periapsis evolution after 25 years propagation, and lastly also after 100 years propagation. During the nominal mission, the eccentricity remains quite stable, averaging about 0.75 while the argument of periapsis with respect to the ecliptic plane starts slightly above 30 degrees to evolve naturally towards the polar region of 90 degrees. This is beneficial for the swarm mission since higher values of this orbital element means lower likelihood of eclipse encountering as well as shorter durations when they occur. For the nominal mission, the highest eclipse during the 2 years nominal mission is just 1.87 hours, meeting the 3 hour requirement with sufficient margin. Other solutions we found for different launch dates follow the same pattern and they all comply with the same eclipse requirement.

The eccentricity plays also an important role with respect to orbit stability, its variation implies changes in the altitude of apogee and perigee which is important to the mission in terms of science operations and collision avoidance. The orbit needs to remain clear of the GEO altitude in order to ensure safety and it needs to have sufficient orbit altitude during apogee to enable science measurements. Figure 3 shows a similar figure with the trajectory phase, with a description of the maneuvers previously explained in the trajectory section. This image illustrates the science orbit acquisition. Figure 3 also shows in the middle, the nominal science orbit evolution in terms of altitude of apogee and perigee that remains stable for this period of time. In the grand scheme, after 100 years of propagation, the periodic behavior as a consequence of the eccentricity variation is easier to distinguish, with peaks and depths in both of the altitudes. The orbit remains stable for the long term propagation, cleared of GEO altitudes, implying a perpetual orbit stability due to this periodic motion.

The other main parameter which defines the particularity of these families of P/2 resonant orbit solutions is the high ecliptic inclination, which is above 25 degrees at the time of insertion for most of our results. This inclination, linked with the previously explained argument of periapsis with respect to the ecliptic, has an influence in the reduction of eclipse durations. As introduced in the trajectory section, the ecliptic inclination implies a line of apsides that is set away from the ecliptic, minimizing the time the spacecraft spends in that plane. According to the Kozai theory, the inclination also follows a pattern that is shown for the nominal case in Figure 4 for both the science phase duration and for a 100 years propagation where the trends and periodicity can also be found.

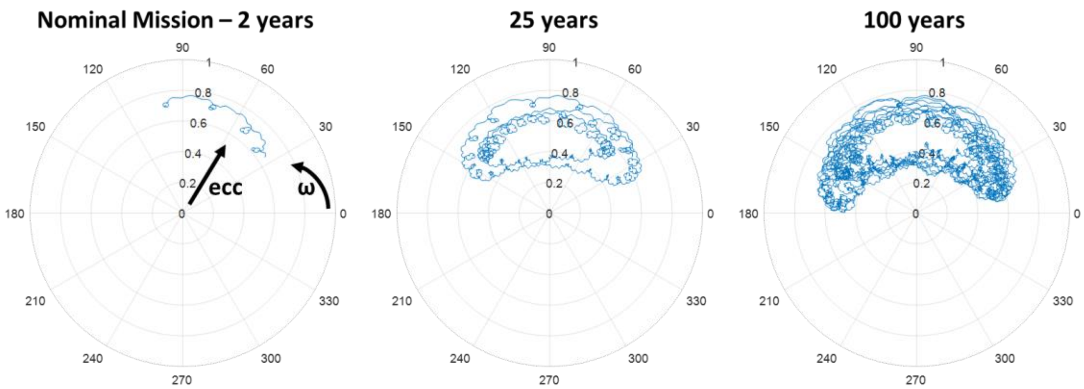

Figure 2. Eccentricity (r axis) versus Ecliptic argument of periapsis (angle axis) for the nominal mission, and after 25 and 100 years of propagation respectively. The signature of these two orbital elements that revolves around 90 degrees and stays within 0.7-0.8 is fundamental to achieve short eclipse durations and orbit stability during the science phase. Later, after more than 25 years propagation, the pattern evolves to a higher extent, but the stability remains locked as seen in Figure 5, although more eclipse may be found due to argument of perigees closer to 0 and 180 degrees.

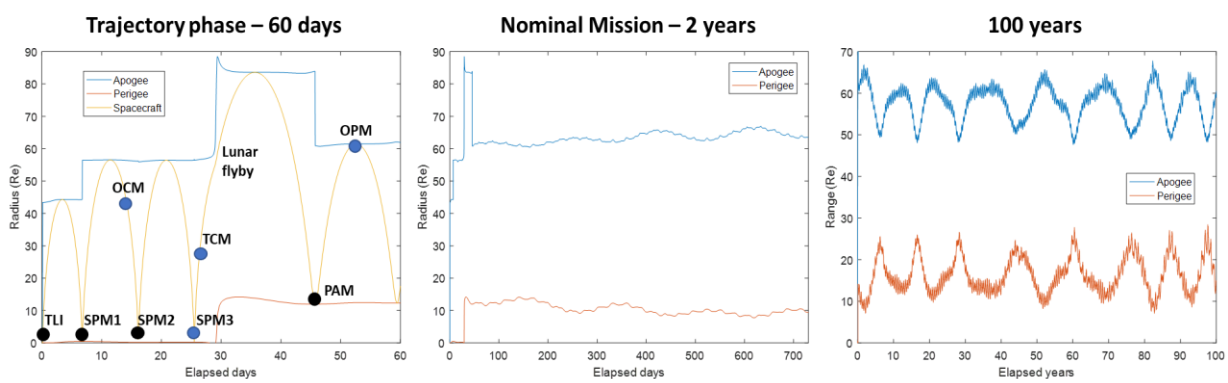

Figure 3: Radius of perigee and apogee during the trajectory phase, nominal mission, and for a total propagation of 100 years. The trajectory phase plot shows the list of maneuver and their placement. The black colored dots show the nominal maneuvers, while the blue colored dots are the statistical maneuvers. The nominal mission show sufficient clearance with respect the GEO belt altitudes. Longer propagation for 100 years, show a stable configuration, where no GEO crossing occurs.

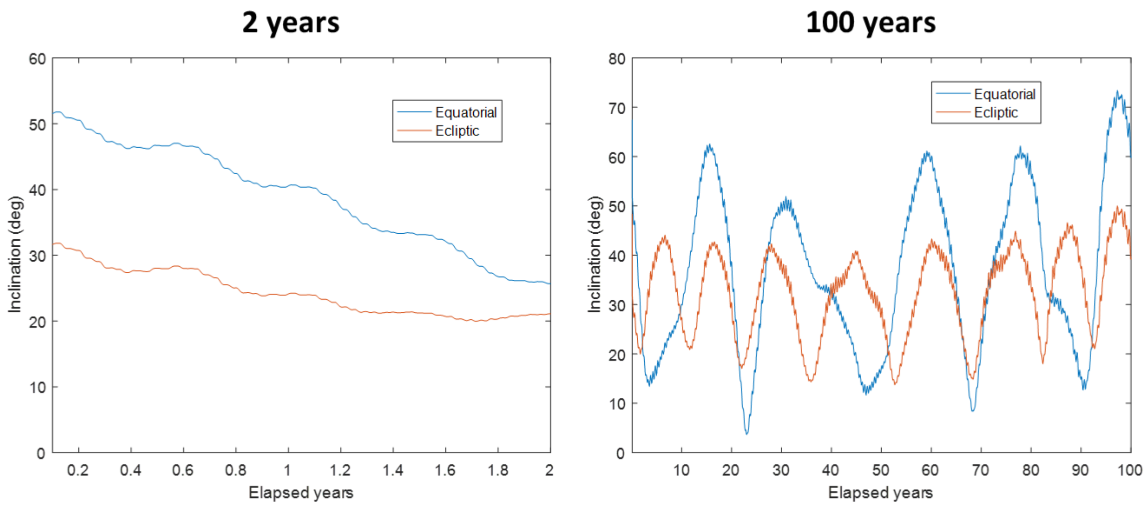

Figure 4: Evolution of the ecliptic and equatorial inclination during the nominal mission, and during a total propagation of 100 years, which shows the periodic evolution that they follow.

Science Orbit Application to Mission Requirements

Orbit and swarm designs coordinate the annual movement of the inertial orbit around the Sun.

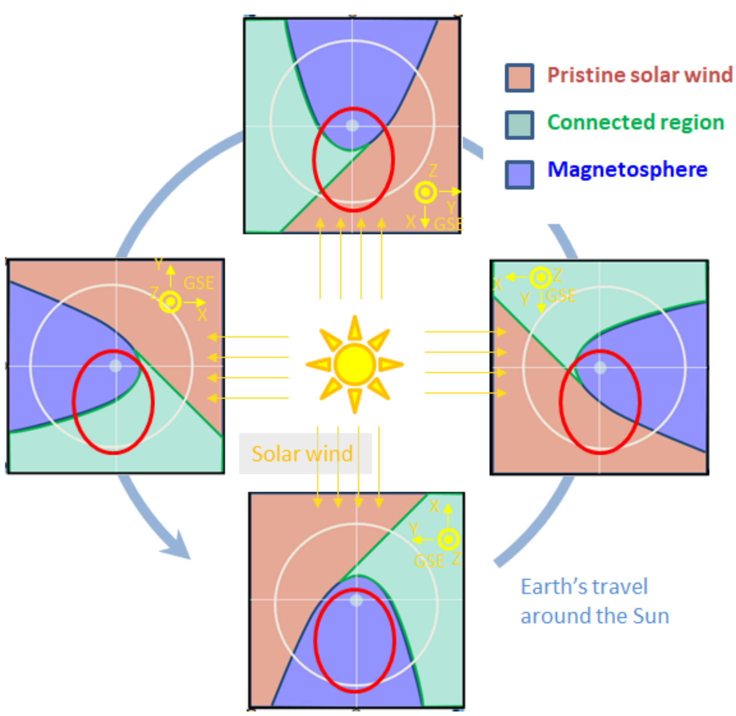

Figure 5. The inertially fixed HelioSwarm science orbit progresses through all regions of solar wind/ magnetosphere interaction.

As the Earth travels around the Sun and the swarm co-orbits with the hub around the Earth, the orientation of the swarm to the Sun vector covers large and small dimensions oriented along and across the solar wind direction, Figure 5.

Fulfillment of science requirements relies on moderate orbit eccentricity (in the reference case averaging approximately 0.75 during the mission's 18 month science phase). Science data collection occurs primarily at high altitudes where relative motion has lowest velocity. Figure 5 shows the apogee of the inertial orbit progressing through the solar wind regions. Data collection hours accrue in each region successively.

Relative motion in eccentric three-body orbit is less familiar than in LEO or GEO, however predictable patterns occur with simple application of fundamental dynamics. Figure 6 provides an informative look at analysis cases of relative motion in the HelioSwarm P/2 orbit for an exemplar, fixed +1 m/s conormal, impulsive delta-v. For enhancing 3D coverage, HelioSwarm leverages the offset available in the conormal dimension by applying the insertion impulse at true anomaly values of 120 or 240 degrees, and avoids large separations around perigee, such as would result from conormal dv application near apogee.

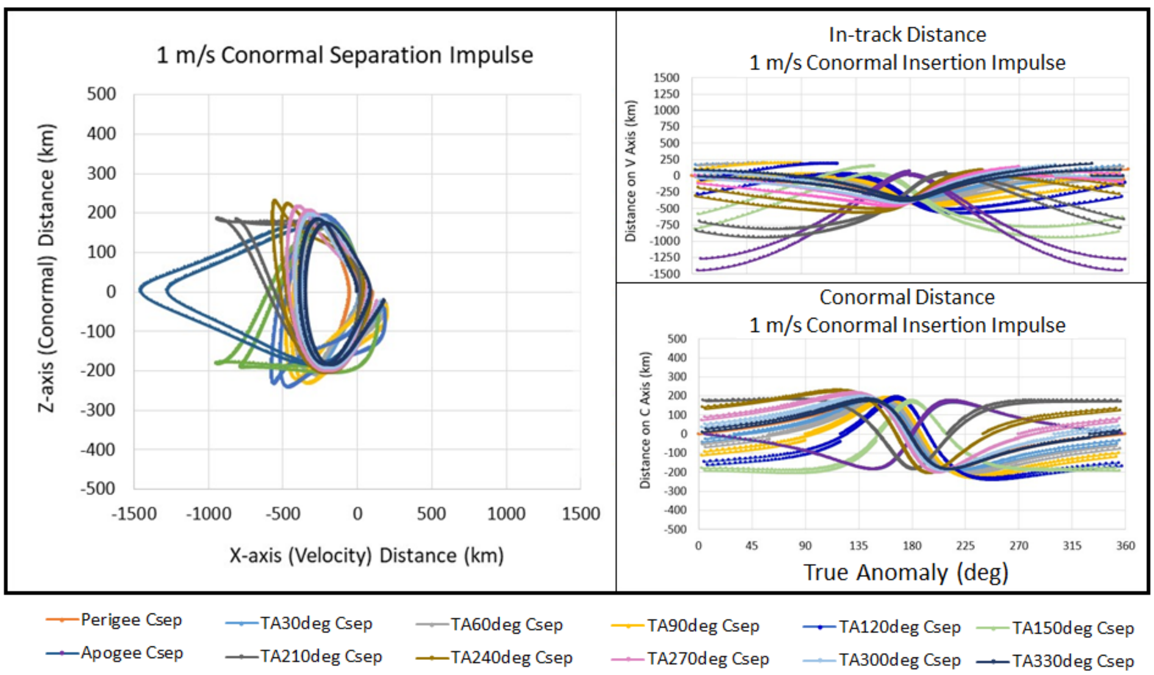

Figure 6. Studies of basic patterns of relative motion in three body, eccentric orbit. Data illustrate two consecutive orbits for each maneuver case.

Once selected, the relative motion of each node in reference to the hub remains consistent over time with the use of very small trim maneuvers, usually once per orbit. Figure 7 shows the 18 month HelioSwarm science phase, highlighting repeating configurations with true anomaly on the scale of the swarm's motion in orbit and the relative motion between swarm members..

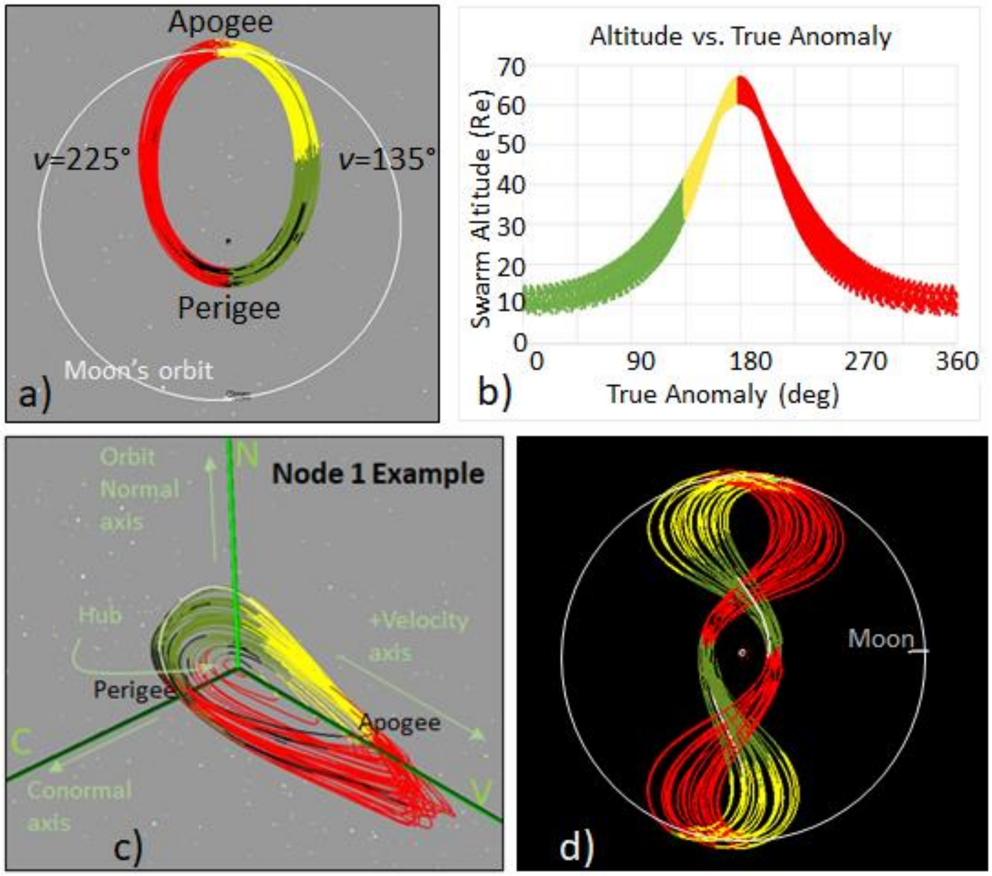

Figure 7. HelioSwarm leverages repeating configurations preceding apogee and surrounding apogee. Part a) shows the orbit in the inertial reference frame, b) provides the correlation of true anomaly and altitude, c) illustrates an example of repeating motion relative to the hub spacecraft at the origin of the VNC axes, and d) shows the P/2 lunar resonant science orbit in the rotating reference frame. Color codes reflect ranges of true anomaly, while black traces represent low thrust maneuver durations.

The desired spatial relationships for science measurements use the Geocentric Solar Ecliptic (GSE) rotating reference frame, where the x-axis is the Earth-Sun vector and the z-axis is ecliptic normal. The HelioSwarm reference mission transforms requirements to VNC coordinates. Figure 8 demonstrates an example relationship between VNC and GSE coordinates, at a mement when the in-track axis aligns with the Sun direction at apogee. The coupled nature of motion in the orbit plane means that GSE-X and GSE-Y distributions will result from the combination of Velocity and Conormal separations as the swarm travels around the orbit (green and blue data lines in the top two frames of Figure 7). GSE-Z or ecliptic normal excursions are simply the orbit normal motion due to relative inclination, multiplied by the cosine of the ecliptic inclination (orange data line in the bottom frame in Figure 8).

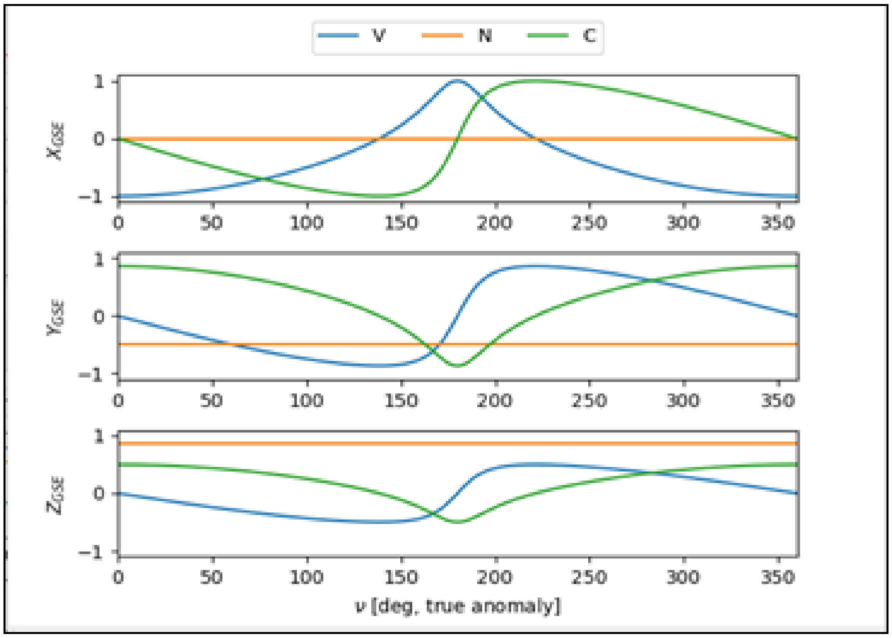

Figure 8. Example transformation of VNC coordinates to GSE, where the ecliptic inclination of the orbit is 30 deg and the line of apsides is perpendicular to the Sun direction.

SWARM DESIGN

Multi-satellite swarms provide spatially and temporally coordinated measurements and the function of the swarm design process is to create configurations advantageous to pertinent science questions while maintaining stable swarm configurations. For HelioSwarm's studies of plasma turbulence, simultaneous multi-scale measurements are desirable, where a single scale requires a polygon with number of members, n, ≥ 4.

For streamlined mission planning and operations, HelioSwarm uses simple Velocity-Normal-Conormal (VNC) Cartesian relative motion patterns for the swarm members to establish separations parallel and perpendicular to the Sun vector. After arrival in the science orbit attached to the hub, the 8 node spacecraft deploy from the hub by lightband release and insert into assigned relative patterns with small maneuvers. HelioSwarm leverages the orbit eccentricity and applies combinations of normal and conormal insertion vectors to create recurring patterns that expand at apogee for distributed science observations, and condense near perigee for efficient data crosslink to the hub for downlink to the ground.

Figure 9 shows the swarm motion in the reference mission design. The hub is at the origin of the VNC axes while the nodes use trim maneuvers, approximately one per orbit, to establish repeated relative patterns. Four of the nodes travel in regions establishing orthogonal pairwise separations, while the inner four nodes utilize the coupled nature of relative motion in the orbit plane to achieve three dimensional distribution sufficient for polyhedral configurations.

Pairwise baseline components use the orthogonal geocentric solar ecliptic (GSE) In addition to the 3D "baseline" requirement, HelioSwarm must form two geometrically sound polyhedra of different scales at the same time.

The relative motion design for HelioSwarm focuses on n=4 tetrahedra. There are 136 unique combinations of 4 satellites using HelioSwarm's 9 members. Lower level requirements limit planarity (P), or "flatness," and elongation (E), or "pointiness," so that the usable combinations tend to have visual resemblance to regular tetrahedra with all sides of equal length. For valid HelioSwarm data points, simultaneous acceptable tetrahedra must differ in scale (S), (i.e. size), by at least a factor of 3:1. The science and flight dynamics teams collaborated closely to implement measures of observatory quality into the mission design toolset. This enabled requirements to be directly levied on the polyhedron quality factor [$(E^2+P^2)^{1/2} < 0.6$], size ratio [>3:1], and total accrued hours of "polyhedral" observations. Evaluations of polyhedral geometry parameters follow the methods outlined in Paschmann and Daly, 1998[7].

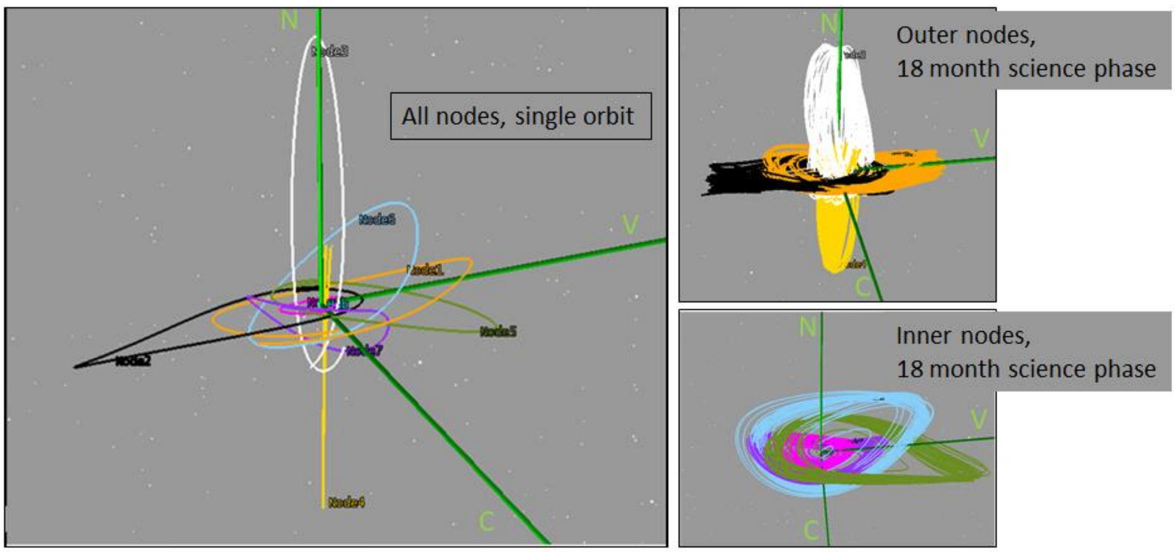

Figure 9. Swarm member motion in VNC axes.

There are 36 unique pairwise combinations of 9 satellites. Projections of satellite-satellite relative positions onto the GSE axes create short, medium, and long components parallel and perpendicular to the incoming solar wind. Figure 10 shows the 36 unique pairings with their GSE axis projections sorted into distance bins delineated by 100 km and 1200 km, with a 3 day trace of the swarm's position in orbit, and examples of pairwise relative position vectors.

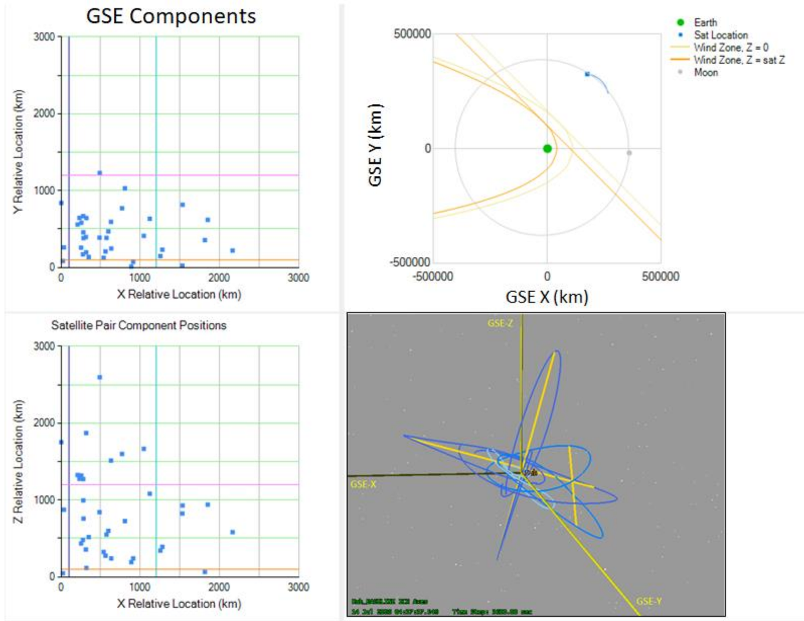

Figure 10. Example configuration with swarm members occupying pairwise relationships projecting onto GSE axes in dimensions under 100 km, between 100 and 1200 km, and larger than 1200 km. (Only 3 example pairs illustrated out of 36)

Tetrahedral and polyhedral, n > 4, combinations conforming to low planarity and elongation constraints form primarily during the portion of the orbit between 135 and 180 degrees of true anomaly. Figure 11 illustrates an example configuration where nodes occupy positions that establish simultaneous tetrahedra with a scale ratio just over 4:1.

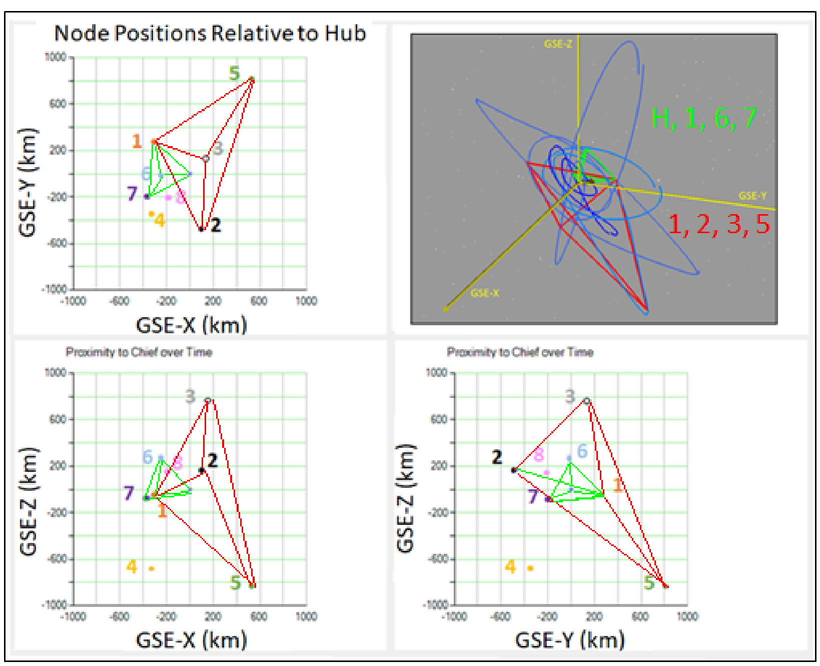

Figure 11. Example swarm configuration with members forming multiple, simultaneous tetrahedra. (Only two relevant tetrahedra shown out of 136 subsets of four satellites)

For the requirement for the nodes to have close approaches to the hub for data crosslink in each orbit, the swarm design applies the same principle as the lunar resonant orbit: larger distances at apogee for data collection and shorter distances in the perigee portion of the orbit for higher data rates. Figure 12 shows an example perigee configuration on the left, with crosslink ranges reduced for higher data rates and illustrates each node's time spent within 500 km of the hub data relay on the right.

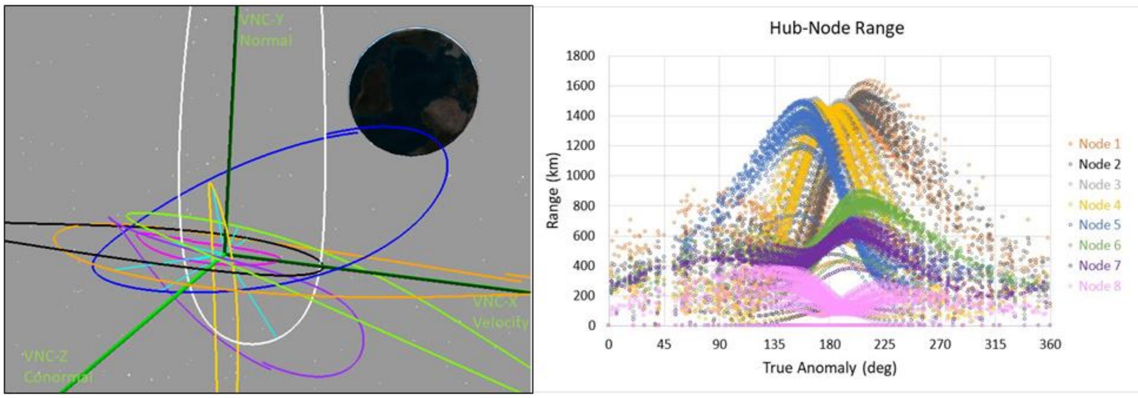

Figure 12. Swarm members approach close to the hub comm relay during the perigee portion of the orbit. Node positions relative to the hub appear in cyan.

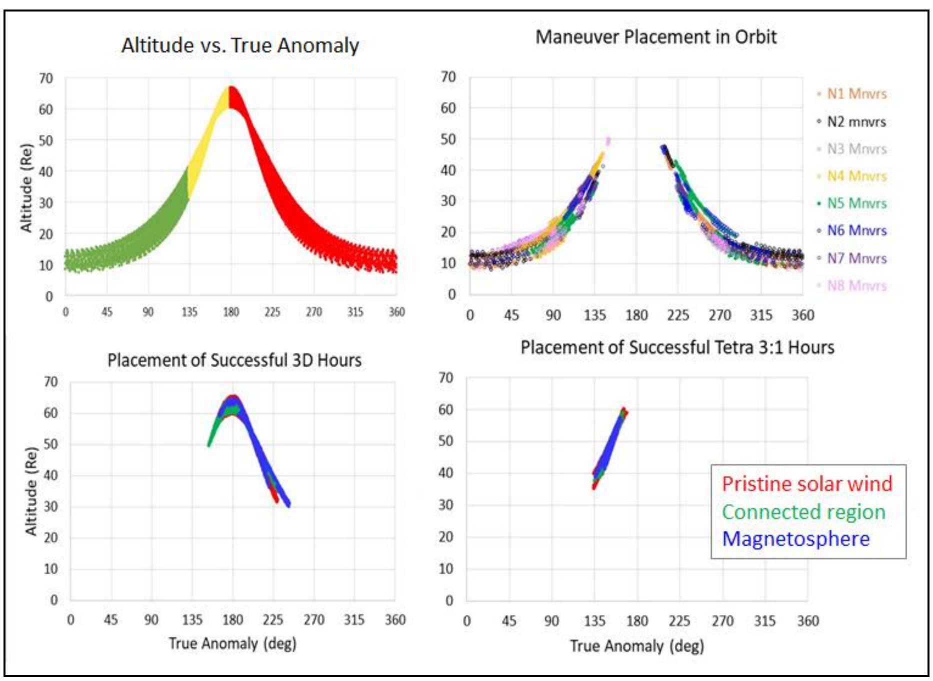

Figure 13. Repeated orbit events displayed as they occur in true anomaly and altitude.

The HelioSwarm relative motion design provides repeating passes through configurations for pairwise 3D measurements, simultaneous tetrahedral combinations, and for facilitating data crosslink. Figure 13 demonstrates the recurring mission activities in the science orbit. The Upper left highlights active portions of the orbit; upper right shows maneuvers occurring almost entirely in the perigee portion, reducing outage time from participation in science data collection; the lower right shows the pre-apogee portion of the orbit where the successful polyhedra form; and the lower left graph shows the true anomaly region around and just after apogee where the 3D measurments align along and across the solar wind.

Operational Factors in Swarm Design

Relative position knowledge derives from a space-based ranging between hub and nodes.[8] Files containing hub to node time of flight measurements downlink to the ground for conventional processing. Position uncertainty in predicted orbits for all spacecraft is well under 10 km and routine trim maneuvers control minimum separations to just under 100 km. Collisions are not a concern while the spacecraft are functioning nominally and executing routine trim maneuvers. In contingency situations, gravitational perturbations and differential solar radiation pressure in the absence of routine trim maneuvers cause spacecraft to drift away from the swarm within a few orbits, allowing enough time for recovery by ground operations and establishing a low risk outcome if no recovery occurs.

The presence of significant secular drift for inactive spacecraft presents a design option for future investigation. All the node spacecraft are identical but the hub has substantially different ballistic coefficient, causing much of the node maneuvering to be for the effect of overcoming differential forces, rather than strictly for maintaining the swarm configuration. A "virtual hub" with the same ballistic coefficient as the nodes could serve as a useful reference point to establish relative motion for all swarm members, hub and nodes alike.

Attitude design for the HelioSwarm spacecraft is very streamlined as all spacecraft use sun-pointing attitude throughout the science phase with the only exceptions of maneuvers, momentum desaturation, and hub downlink to the ground. Crosslink communications use omnidirectional antennas and do not impose attitude requirements.

MANEUVER DESIGN

The process for establishing the desired relative motion patterns begins with a small (< 1 m/s) mechanical impulse imparted by the spring-driven separation system. All Nodes separate from the hub on vectors constrained to the NC plane. Minimizing any in-track component of the initial separation velocity, mitigates large separations due to secular drift. The node spacecraft will drift in close proximity to the hub for approximately one orbital period (13.7 days) while commissioning and initial relative orbit determination activities proceed. After commissioning, each node spacecraft performs a series of one to three insertion maneuvers to increase its separation from the Hub and begin the desired relative motion. For operational considerations, these maneuvers occur infrequently, typically with a single maneuver in each two-week orbit. After completing the insertion maneuvers, each Node continues to perform, on average, one maneuver per orbit to maintain its separation and relative motion.

These orbit trim maneuvers use targeting to achieve three goals:

1) Form geometrically satisfactory tetrahedra between true anomaly 135-170°
2) Achieve the requisite maximum separation from the Hub, in the correct VNC quadrant, between true anomaly 155-230°

3) Return to the vicinity of the Hub at perigee while remaining outside the 10 km keep out zone around the Hub

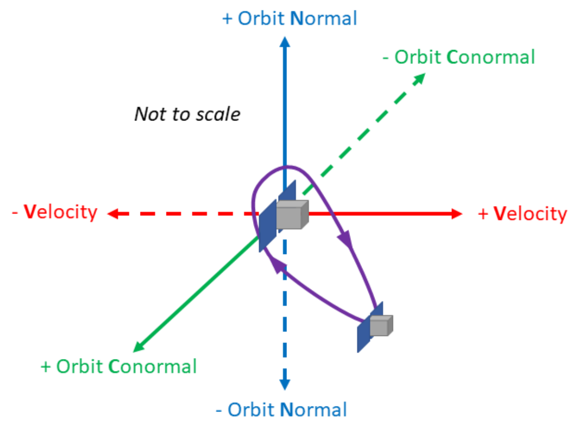

Figure 14. Cartoon of Node orbit relative to Hub shown in the Hub's VNC (Velocity, Nor-mal, Conormal) frame.

Each Node's orbit is designed to maximize occupy of a quadrant of space in the Hub's VNC frame (Figure 14). By providing symmetrical coverage of the V and N axes, the swarm meets the 3D baseline requirement. Asymmetrical distribution of nodes between the N and C axes also sets up the tetrahedral geometry for science measurements. Table 3 summarizes the configuration of each Node in the swarm as well as the typical placement of their ΔV events. Nominally, separation, insertion, and orbit trimming all occur around the same true anomaly though mission planning may relocate select events due to operational constraints.

Table 3: Summary of Node relative motion design goals and placement of ΔV events

Node	Desired Motion	Target Max. Range from Hub	Maneuver Placement
1	In plane, +V/C	1600 km	TA90
2	In plane, -V/C	1600 km	TA90
3	Out of plane, +N	1500 km	TA90
4	Out of plane, -N	1500 km	TA90
5	Asymmetric, +V/-N/+C	1500 km	TA240
6	Asymmetric, +V/+N/-C	900 km	TA120
7	Asymmetric, +V/-N/-C	700 km	TA120
8	Hub Orbiter	400 km	TA60

Node maneuvers are designed for a thrust of 0.76 mN at 4000s I_{sp}, well below the Node spacecraft's maximum thrust of 1.6 mN. For the design reference mission, calculations assume spacecraft mass of 85 kg (above the Node spacecraft's maximum expected mass to provide performance margin). Each maneuver utilizes a fixed maneuver attitude in the spacecraft's VNC frame. Figure 15 shows the accumulation of thruster operation time and propellant consumption over the duration of the mission. Table 4 summarizes the ΔV budget for the Nodes.

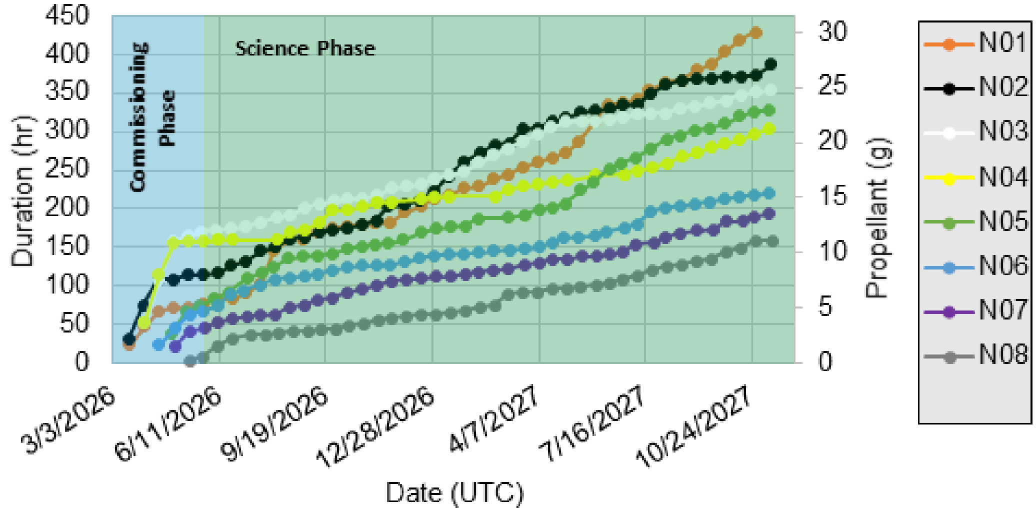

Figure 15: Cumulative thruster operation time and propellant consumption over mission duration

Table 4: ΔV budget for HelioSwarm Nodes

Node	DRM ΔV (m/s)			Allocation (m/s)	
	SIM	OTM	Total	Total	Margin
1	2.2	25.9	28.1	50.0	43.8%
2	3.5	19.7	23.2	50.0	53.5%
3	5.1	13.6	18.7	50.0	62.6%
4	5.0	10.3	15.3	50.0	69.4%
5	2.2	18.2	20.4	50.0	59.3%
6	0.8	13.8	14.6	50.0	70.8%
7	1.4	10.6	12.0	50.0	76.1%
8	0.1	10.9	11.0	50.0	78.0%

CONCLUSION

The HelioSwarm mission design delivers a swarm of ESPA-class small satellites to a high-altitude Earth orbit with prolonged access to the pristine solar wind. The transfer trajectory and science orbit leverage a P/2 lunar resonant orbit with the line of apsides rotated out of the ecliptic plane to mitigate lengthy eclipses. Once established in the science orbit, the ESPA-class node spacecraft use electric propulsion to establish relative motion patterns that provide distributed coverage of the along and across Sun directions. HelioSwarm's unique mission design enables novel, multi-point measurements of the pristine solar wind that will reveal the fundamental physics of turbulent plasmas in our solar system and beyond.

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