*PROOF-OF-CONCEPT TRAJECTORY DESIGNS FOR A MULTI-SPACECRAFT, LOW-THRUST HELIOCENTRIC SOLAR WEATHER BUOY MISSION

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ABSTRACT

A new solar weather mission has been proposed, involving a dozen or more small spacecraft spaced at regular, constant intervals in a mutual heliocentric circular orbit between the orbits of Earth and Venus. These "solar weather buoys" (SWBs) would carry instrumentation to detect and measure the material in solar flares, solar energetic particle events, and coronal mass ejections as they flowed past the buoys, serving both as science probes and as a radiation early warning system for the Earth and interplanetary travelers to Mars.

The baseline concept involves placing a "mothercraft" carrying the SWBs into a staging orbit at the Sun-Earth L1 libration point. The mothercraft departs the L1 orbit at the proper time to execute a trailing-edge lunar flyby near New Moon, injecting it into a heliocentric orbit with its perihelion interior to Earth's orbit. An alternative approach would involve the use of a Double Lunar Swingby (DLS) orbit, rather than the L1 orbit, for staging prior to this flyby. After injection into heliocentric orbit, the mothercraft releases the SWBs — all equipped with low-thrust pulsed plasma thrusters (PPTs) — whereupon each SWB executes a multi-day low-thrust finite burn around perihelion, lowering aphelion such that each achieves an elliptical phasing orbit of different orbital period from its companions. The resulting differences in angular rates of motion cause the spacecraft to separate. While the lead SWB achieves the mission orbit following an insertion burn at its second perihelion passage, the remaining SWBs must complete several revolutions in their respective phasing orbits to establish them in the mission orbit with the desired longitudinal spacing. The complete configuration for a 14 SWB scenario using a single mothercraft is achieved in about 8 years, and the spacing remains stable for at least a further 6 years.

Flight operations can be simplified, and mission risk reduced, by employing two mothercraft instead of one. In this scenario, the second mothercraft stays in a libration-point or DLS staging orbit until the first mothercraft has achieved nearly 180º separation from the Earth. The timing of the second mothercraft's subsequent lunar flyby is planned such that this spacecraft will be located 180º from the first mothercraft upon completion of its heliocentric circularization maneuvers. Both groups of satellites then only have to spread out over 180º to obtain full 360º coverage around the Sun.

1 - INTRODUCTION

The fundamental concept for the Solar Weather Buoy (SWB) mission is to establish multiple spacecraft in an evenly-spaced formation in heliocentric ecliptic-plane orbits interior to the Earth's orbit (Fig. 1). Surrounding the Sun, the spacecraft would provide both solar wind science and advance warning of solar events — such as Coronal Mass Ejections (CMEs), which are clouds of ejected solar material denser and more energetic than the average solar wind — that might affect spacecraft in orbit near the Earth and elsewhere in the Solar System. Figure 1 shows a hypothetical formation of SWB spacecraft orbiting between the Earth and Venus in a circular ring around the Sun. For instance, the SWBs would be in a position to detect the CME shown in time to warn astronauts on the Earth-Mars transfer trajectory depicted.

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The goal of the SWB mission concept study was to explore methodologies for establishing a heliocentric constellation of multiple spacecraft interior to the Earth's orbit and with a fixed and equal longitudinal spacing of no more than 30° between them. (To attain a maximum angular spacing of 30°, at least 12 spacecraft would be required.) Further goals of the study were to seek solutions minimizing both individual SWB propulsion needs and the time to complete establishment of the constellation. It was assumed that the SWBs would be very small, low-mass spacecraft with a dry mass of no less than 30 kg. Fuel, if needed, would be additional. It was further assumed that the individual SWBs would be stored aboard a mothercraft and launched to a staging orbit via a single Delta II-class launch vehicle. The baseline staging orbit was assumed to be a Sun-Earth L1 libration-point orbit. From there, spacecraft could return Earthward on a transfer trajectory to a lunar flyby near the New Moon location. The flyby provides the escape energy needed to begin the heliocentric transfer to the mission orbit. This approach saves launch costs by requiring just one launch vehicle, and that single rocket has reduced launch energy requirements due to escape energy being provided by the Moon. While the L1 libration-point staging orbit was the baseline concept, other staging orbit possibilities exist and were briefly explored.

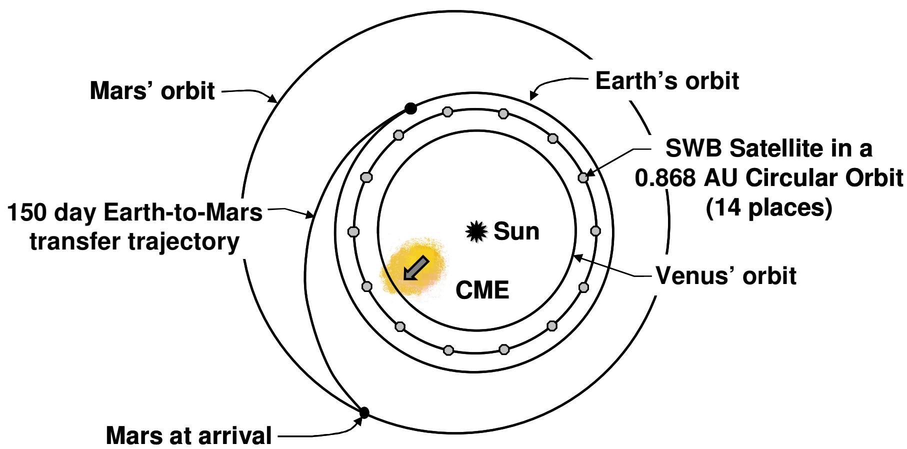

Figure 1. Solar Weather Buoy Mission Concept with Early Warning for Human Mission to Mars

Ultimately, the study proceeded through three primary stages. In the initial mission concept, each independent SWB probe would be equipped with only minimal propulsion capability. Each SWB would individually depart Earthward from a mothercraft at the L1 region at intervals of approximately 6 lunar synodic months, to enter 0.813×1 AU heliocentric orbits via lunar flyby. The supposition was that after 6 to 7 years, a completed constellation of spacecraft would revolve around the Sun with fixed-interval spacing between them. Analysis showed that such a scenario is not possible (as discussed below), since the elliptical orbits actually yield spacing that oscillates over time with large amplitudes. In the study's second stage, the mission concept evolved to having the SWB-laden mothercraft depart via lunar flyby, with each individual SWB carrying sufficient propulsive delta-V (ΔV) capability to insert itself into a constellation in a common circular heliocentric orbit. The mothercraft would release the SWBs during the transfer leading up to perihelion arrival. One SWB would begin circularization immediately, while the rest would establish individual phasing orbits to gain the required angular separation from its neighbors over several orbit revolutions prior to final circularization. A major science advantage of this scenario is that all the probes are placed in heliocentric orbits and returning science data early in the mission, rather than having some probes waiting in L1 orbit for their lunar flybys. In the third and last stage of the study, two mothercraft (each carrying half the fleet) with phased Earth departures were utilized to simplify mission operations and to eliminate the single point of failure that a single mothercraft represents. As in the single mothercraft scenario, each SWB would carry its own low-thrust propulsion system sufficient to circularize its own orbit. In the latter two stages of the study, a somewhat larger mission orbit of 0.868 AU was used to lower ΔV costs. The evolution of the mission concept is presented below, together with the results of scenarios from its latest form. Phasing orbit concepts and heliocentric injection via the lunar flyby technique are also discussed.

2 - TRAJECTORY SCENARIOS EMPLOYING A SINGLE MOTHERCRAFT

The SWB mission scenarios presented here began with the assumption that the Solar Buoy mothercraft, carrying 12 or more spacecraft, would initially be placed in orbit at the Sun-Earth L1 libration point. Following an L1-to-Moon transfer, each spacecraft would then be sent into heliocentric orbit by a lunar gravity assist flyby. The Sun-Earth-Moon geometry at the time of each flyby would control the orbital parameters achieved at the ensuing perihelion. (Incidentally, the 0.813 AU perihelion is close to the limit that can be achieved with the lunar flyby departure technique.)

The trajectories were calculated with the GSFC Mission Design and Analysis Tool (Swingby) program.[1] The force model included the gravitational effects of the Earth (4×4 JGM-2 geopotential field for the L1 orbit and flyby phases, then point mass for the heliocentric orbit phase), Sun, Moon, Venus, Mars, and Jupiter, as well as solar radiation pressure. Initially, the dry mass for each SWB was assumed to be 30 kg.

2.1 Scenario #1: Individual lunar flyby for each spacecraft, with minimal propulsion systems

In the initial trajectory concept, the Solar Buoy spacecraft would exploit the natural periodicity of the L1 libration-point orbit by releasing one spacecraft each revolution from L1 orbit to perform a lunar flyby. This would create an approximate lag time of 6 months between successive spacecraft entries into the heliocentric orbit. Figure 2 shows a sample trajectory from the L1 orbit to a trailing-edge lunar flyby near New Moon, yielding escape into heliocentric orbit with a perihelion of less than 1 AU. Examining the least expensive option first, it was initially assumed that each spacecraft would carry only the minimal propulsion required to target the lunar flyby, but not enough fuel to execute circularization burns in the solar orbit. Instead, the spacecraft would be left in their individual elliptical orbits around the Sun.

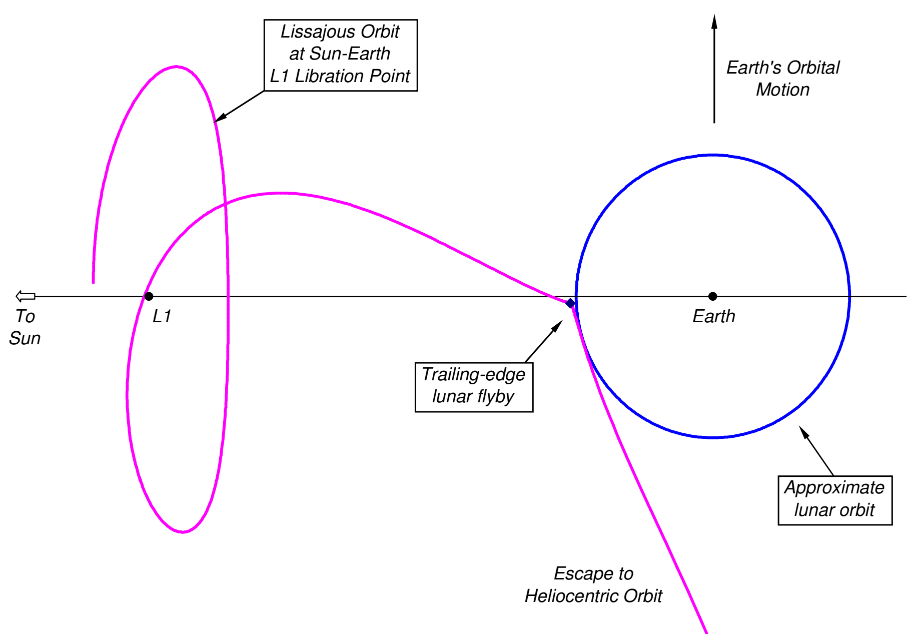

Figure 2. Representative Earth Departure Scenario (Ecliptic Plane Projection in L1-centered Rotating Libration Point Coordinates)

To minimize the fuel required for targeting the lunar flyby for the first SWB, the initial L1 Lissajous orbit was first constructed by propagating backward from the desired flyby location. While this minimized the ΔV for the first spacecraft of the formation, the evolution rate of the Lissajous orbit was such that subsequent spacecraft departing from L1 incurred a wide range of ΔV costs, from 7 to 57 m/s, to target the lunar flyby. In addition, the differences in Sun-Earth-Moon geometry at the flybys for these cases created significant variability in orbital parameters of the resulting heliocentric orbits. Perihelion distances for these cases ranged from about 120.0×10^6 to 130.5×10^6 km (0.802 to 0.872 AU).

A hypothesis was formulated that starting from a more slowly-evolving Lissajous orbit or a halo orbit at L1 would allow tighter control over the flyby geometry and resulting heliocentric orbits. Specifically, cases were computed with initial states corresponding to the Wind and SOHO spacecraft, which are both in orbit at the Sun-Earth L1 point.[2, 3] Wind is currently in a slowly-evolving Lissajous orbit, while SOHO is in a quasi-periodic halo orbit. The use of these orbits as the initial state did produce more uniform flyby and heliocentric orbit geometry across the formation. Note that the current Wind and SOHO orbit states were used for hypothetical analysis purposes only, as the actual launch date for SWB would be several years in the future. For the trajectories starting from Wind's orbit, ΔV costs for targeting the lunar flybys were 30 to 45 m/s, and perihelion distances ranged from 126.8×10^6 to 128.2×10^6 km (0.848 to 0.857 AU). Starting from SOHO's orbit, lunar flyby targeting required ΔV from 0.1 to 35 m/s, with resulting perihelion distances of 124.3×10^6 to 128.0×10^6 km (0.831 to 0.856 AU).

Simple Keplerian calculations based on a circular Earth orbit, and ignoring the 5.2° ecliptic inclination of the Moon's orbit, indicated that this strategy should produce an approximate spacing of 30° between spacecraft in the heliocentric formation. However, propagation of the spacecraft using a realistic force model showed that the angular spacing in the elliptical orbits for two Solar Buoys inserted 6 months apart displays a sinusoidal variation with amplitude of about 25°. The amplitude of variation for spacecraft inserted a year apart is only about 5°, but the average spacing between them would be about 40°, rather than the desired 30°.

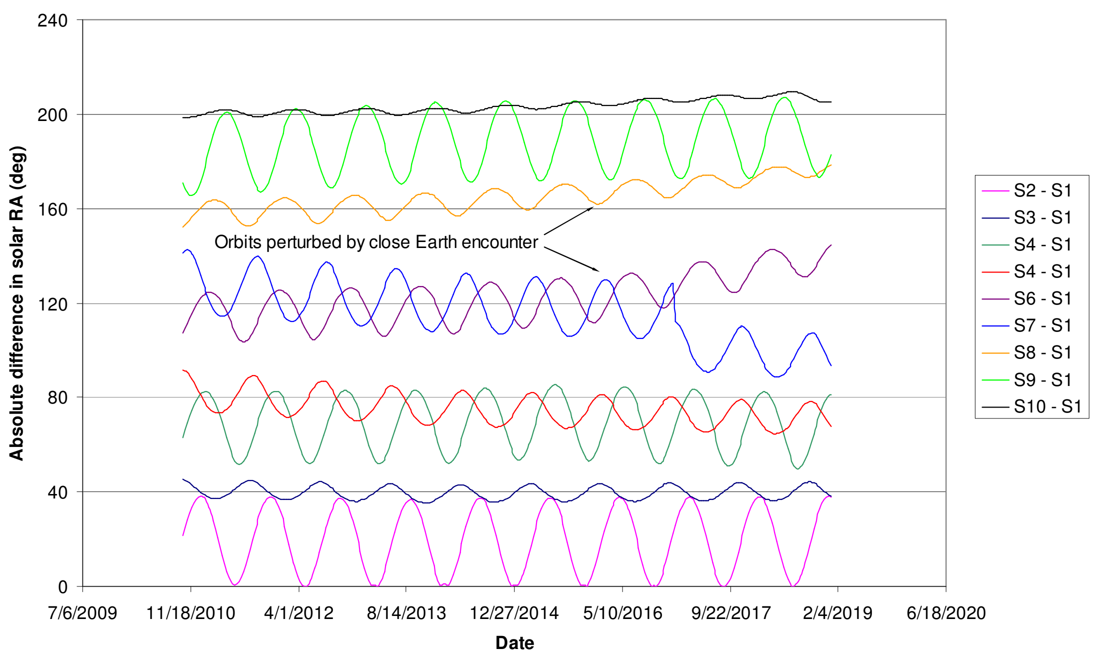

Figure 3. Separation Between Solar Buoys with Elliptical Orbits

Figure 3 shows the angular spacing between the first SWB and 9 subsequent probes, given as the difference in heliocentric right ascension. These trajectories began with a SOHO-type initial L1 halo orbit, with one spacecraft released from the L1 orbit toward the Moon every revolution. Each spacecraft passed approximately 42 km above the lunar surface during its flyby. No ΔV was applied in the heliocentric orbits, leaving each SWB in an elliptical orbit. As seen in the figure, spacecraft inserted into the heliocentric orbit 6 months apart show much greater variation in spacing than those inserted a year apart. There is also a correlation in ΔV costs for the sets of trajectories inserted at similar seasons of the year. The odd-numbered spacecraft had lunar flyby targeting ΔV of 32.6 to 34.5 m/s, while the even-numbered spacecraft required ΔV of 0.1 to 11.0 m/s. Spacing between the even-numbered spacecraft only is shown in Figure 4.

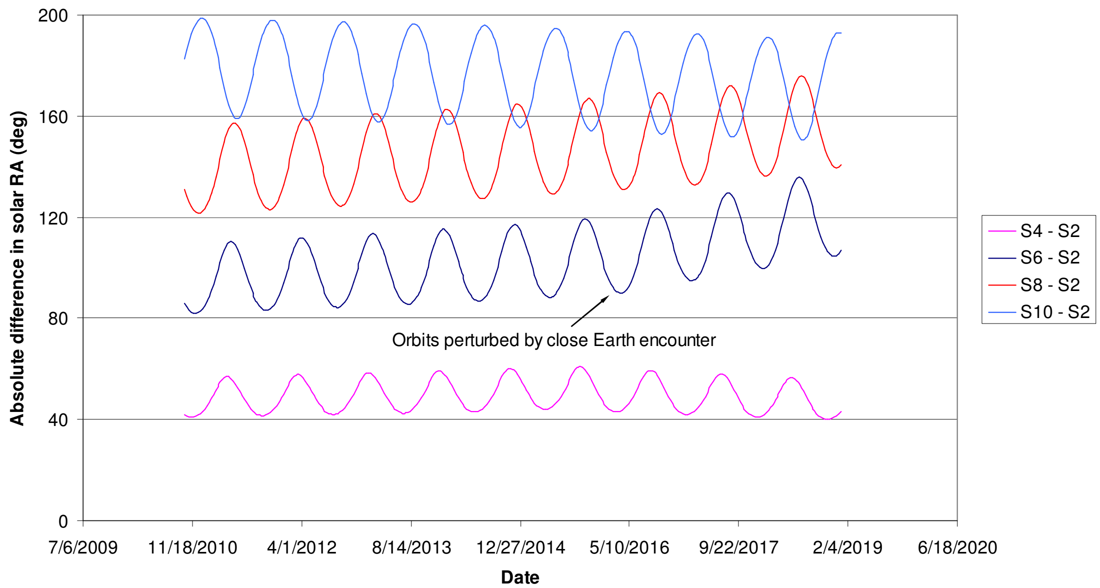

Figure 4. Separation Between Even-Numbered Solar Buoys with Elliptical Orbits

Another problem with the elliptical heliocentric orbits is that they are subject to strong perturbations from the Earth, as seen in Figure 3 around the year 2016. At this time, some of the Solar Buoy probes would pass within a million kilometers of the Earth in this hypothetical scenario, causing significant disturbance to the formation during the following years. This result displays one important reason to equip each spacecraft with sufficient propulsive ΔV capacity to reduce its aphelion distance below the Earth's sphere of influence.

2.2 Scenario #2: Single lunar flyby for the mothercraft, with heliocentric phasing orbits

For the second major scenario considered, the mothercraft was initially placed into a planar L1 Lissajous orbit, with virtually no motion out of the ecliptic. This orbit was selected to ensure reasonable ΔV costs for targeting lunar flybys from the L1 orbit, but no further effort was expended to optimize each scenario. To truly minimize the ΔV for a specific trajectory SWB case, an out-of-ecliptic component may be required in the Lissajous orbit. While still carrying a full load of SWB probes, the mothercraft would perform a lunar flyby to enter heliocentric orbit. During the 165-day transfer trajectory from the Moon to perihelion, all probes would be released from the mothercraft and prepared for their ultimate entry into formation in a common, circular orbit around the Sun. This formation would be achieved by initially placing each probe (except the first, which would be circularized upon arrival at perihelion) into a heliocentric phasing orbit of slightly different period and allowing the differences in mean motion to create the desired angular separation between spacecraft over several revolutions. Ultimately, though, the SWBs need to be in "circular" orbits all of nearly the same radius magnitude as possible. Deviations from this ideal scenario would result in constant offsets in mean motion between them, causing their separations to grow linearly with time, rather than remain fixed as desired.

For this scenario, it was assumed that each spacecraft would be equipped with low-thrust plasma engines (I_{sp} = 1150 sec) to enable entry into its own heliocentric orbit. Trajectories were computed specifically for a formation consisting of 14 SWBs, with nominal spacing of approximately 25.7°. The desired heliocentric orbit had a target radius of 0.868 AU, which would have the advantage of ΔV costs lower than the 0.813 AU case yet be far enough away from the Earth's orbit to avoid unwanted perturbations to the formation.

The period of the 0.868 AU orbit is 295.44 days. To establish the formation, one spacecraft (designated S1) would enter the desired circular orbit at the first perihelion passage. It was initially assumed that the engine on each spacecraft would be able to operate at a maximum power of 100 W, providing a thrust of 0.00123 N. With this level of thrust, the maneuvers required for orbit circularization would be too long, as the total theoretical burn duration needed to circularize the heliocentric orbit would be greater than the orbit period. Because of this, further analysis was performed with the assumption of plasma engines operating at 400 W, for a thrust of 0.00492 N. The additional hardware, such as larger solar arrays, required for the 400 W thrusters increased the dry mass estimate from 30 kg to 53.4 kg. Fuel mass was revised to be 7 kg.

By splitting the total ΔV into multiple burns, the burn duration was reduced to a maximum of 90 days per orbit for the 400 W thruster cases, which still may prove too long to be operationally feasible. Since the circularization maneuvers using the low-thrust engines were so long, significant finite burn effects were encountered. For example, the perihelion maneuvers have the undesirable effect of decreasing perihelion distance. To counteract this, the lunar flyby was designed so as to bias the first perihelion passage to a higher value than desired for the final orbit. The biased perihelion target of 131.0×10^6 km was selected based on the change in perihelion calculated during the circularization burns. Starting from this initial perihelion, the first spacecraft achieved the desired circular orbit at 0.868 AU upon completion of its maneuvers. The total ΔV required for circularization was divided into two burns, which were executed at the first and second perihelion passages. Note that this strategy eliminates the need for a perihelion-raising maneuver by the mothercraft. However, due to the different phasing orbit periods for the individual SWB probes, some of them still require perihelion-adjustment maneuvers to establish their phasing orbits. Future stages of the mission design process should include an examination of the optimal strategy for initial perihelion biasing, to minimize the total required ΔV across the constellation.

The remaining 13 SWB probes also performed maneuvers at the first and second perihelion passages, but rather than achieving a circular orbit right away, they were put into phasing orbits that would produce nominal spacing of about 25.7° between adjacent spacecraft after several orbit revolutions. Upon reaching the desired angular spacing, another pair of maneuvers would be performed to circularize the orbits. The target phasing orbit period for each spacecraft is shown in Table 1. The table also gives the impulsive ΔV required to establish each spacecraft's phasing orbit. Note that the total ΔV for each SWB is 1.109 km/s, and that the period for SWB #14 is equal to that of the heliocentric transfer trajectory.

Table 1. Ideal Phasing Orbit and ΔV Parameters Based on Keplerian Approximation and Impulsive Burn Calculations

Spacecraft	Ideal Phasing Orbit Period (days)	Ideal ΔV for Phasing Orbit (km/s)	Ideal ΔV for Circular Orbit (km/s)	Ideal Angular Spacing from S1 after 8 phasing orbits (deg)
S1	295.4 (circular orbit)	0	1.109	0
S2	298.1	1.015	0.094	25.7
S3	300.7	0.922	0.187	51.4
S4	303.3	0.831	0.278	77.1
S5	306.0	0.742	0.367	102.9
S6	308.6	0.654	0.455	128.6
S7	311.3	0.567	0.542	154.3
S8	313.9	0.482	0.627	180.0
S9	316.5	0.399	0.711	205.7
S10	319.2	0.316	0.793	231.4
S11	321.8	0.235	0.874	257.1
S12	324.5	0.156	0.954	282.9
S13	327.1	0.077	1.032	308.6
S14	329.7	0	1.109	334.3

2.2.1 Detailed Development of the Phasing Orbits and Mission Orbit Circularization

The general idea is to use elliptical phasing orbits — each of unique semimajor axis — to build up a desired angular separation between each of the SWBs. This process exploits the differences in mean motion between the phasing orbits for each of the SWBs as well as that of the circular mission orbit. A limiting consideration is that the phasing orbits ideally should not be larger than the transfer trajectory itself, in order to avoid undue ΔV costs. The effect of this limitation is that multiple revolutions in the phasing orbits will be required to gradually accrue the needed angular spacing.

To start the process, the details of the heliocentric transfer trajectory and the intended "circular" mission orbit can be estimated with sufficient accuracy via the usual Keplerian formulas. (Ultimately, the mission orbit can be made only near-circular, as perfectly circular orbits cannot be achieved in nature.) Next, the details of the individual phasing orbits for each SWB need to be estimated. We begin by noting that the desired mission orbit spacing between each spacecraft is simply $360°/n$ where n is the number of solar probes to be used. In the case developed here, n = 14, so the desired spacing was 25.7°. Furthermore, to find the spacing, s, between any SWB in the series and S1, it is simply $s = (S\# - 1)\,(360/n)$. For example, the spacing between S3 and S1 is just (3 − 1) 25.7° = 51.4°.

Given that the difference in mean motion between the phasing orbit and the intended mission orbit will cause the angular separation between two spacecraft that were initially co-orbiting to increase linearly over time, the question is how many revolutions in a phasing orbit are necessary. The answer to that question helps determine the size and period of the individual phasing orbit for each SWB. One factor constraining the answer is that it is undesirable to have a phasing orbit larger than the transfer orbit from Earth because of the added ΔV costs of the aphelion raising that would be required. Thus, the last SWB of a given group — the one with the most angular separation to generate — should not have a phasing orbit period larger than the transfer orbit period, nor an aphelion radius larger than 1 AU. Given that the phasing orbit perihelion radius is necessarily fixed (0.868 AU for the cases studied here), the phasing orbit aphelion radius must be between 0.868 AU and 1 AU. Viewed another way, its eccentricity, e, must observe $0 < e \leq 0.0706638$. Since the phasing orbit has a period larger than that of the circular mission orbit, the first time the spacecraft returns to perihelion, it will be some angular distance behind the swifter-moving spacecraft already established in the mission orbit. The angular separation will increase by the same amount with each succeeding phasing revolution. Upon reaching the revolution where sufficient separation has accumulated, the spacecraft may finally proceed with its circularization maneuvers. In this way, each SWB falls into its allotted mission orbit slot — with the planned angular separation — to the west of those preceding it.

To determine a reasonable number of phasing revolutions, we first determine the number of days, ΔT, required for a spacecraft in the circular mission orbit to traverse an arc equal to the desired angular spacing, s, between the SWB in question and S1 (which is circularized upon arrival). That is, $\Delta T = s/m_{mo}$, where m_{mo} is the mission orbit mean motion (for the 0.868 AU case, m_{mo} is 1.219°/day). To test an initial guess at a number of phasing revolutions, r, we calculate a "gain" of angular arc per revolution by simply computing s/r. The number of days of orbital period in excess of the circular orbit period that the phasing orbit requires to achieve that gain is simply $\Delta T/r$. Finally, the period for a phasing orbit, $P_{phasing}$, with r revolutions would be $P_{phasing} = P_{mo} + \Delta T/r$, where P_{mo} is the period of the mission orbit. For the 0.868 AU case, P_{mo} = 295.4 days, and the period of the transfer orbit, P_{trans}, is 329.732 days. Thus, if $P_{phasing}$ does not exceed 329.732 days, the phasing orbit qualifies as acceptable. For example, the spacing between S1 and S14 is s = 334.286° (see Table 1). If r = 8, then $P_{phasing}$ = 329.726 days, which just qualifies as an acceptable phasing orbit by a slim margin.

Once the period of the desired phasing orbit is known, other phasing orbit properties such as semimajor axis, radius of aphelion, and velocity at perihelion, may be calculated via the usual Keplerian formulas. It is then similarly straightforward to calculate the ΔVs necessary to insert into the phasing orbit from the transfer trajectory, and to insert into the final circular mission orbit from the phasing orbit.

Once the analytical estimates have been developed, the next step is to develop the numerically integrated and targeted SWB phasing orbits and mission orbits with the Swingby mission design program. The first SWB of a given group begins its circularization upon arrival at perihelion. Initially, the retro circularization burn is targeted impulsively; then it is re-modeled as a low-thrust, long-duration finite burn with the spacecraft modeled as rotating so that the thrust vector follows the velocity vector. Given that the ΔV

required is large and the thrust level very low, it is practical to circularize using two burns; approximately half the required ΔV is performed at first perihelion, and the remainder is imparted at the second perihelion passage. These multi-day burns are centered on perihelion and targeted on the desired mission circular orbit period using Swingby's differential corrector targeter. As discussed in the previous section, the lunar flybys are designed to bias the arrival perihelion high so that upon completion of the first SWB probe's two circularization burns, the result is a circular orbit of the desired radius.

The remaining SWBs of the group must be placed in their unique elliptical phasing orbits to produce the required angular separation, as described above. In general, this entails simply lowering aphelion from the transfer orbit level to the needed radius. From an energy perspective, the process is merely that of circularization in stages, beginning at perihelion arrival. To initiate phasing, an impulsive retro ΔV is targeted on the desired period of the phasing orbit. Given that the finite burns can be very long, the phasing ΔV is performed in two parts at successive perihelion passages, just like the circularization burns described above. The revolution between the two phasing burns counts as the first phasing revolution; after propagating through the remaining needed phasing revolutions, two circularization burns are computed, similar to those of the first SWB of the group. In general, it was found that burns at the aphelion between both the phasing burn pairs and the circularization pairs were needed to control the perihelion radius during the process. Again, later and more refined stages of mission design should include efforts to optimize the initial perihelion radius across the entire constellation, and not just the mothercraft. These sequences for each SWB were refined iteratively to yield mission orbits with the desired radius, period, and very small eccentricity.

2.2.2 Finite Maneuver Modeling Results

There are many finite burn strategies that could achieve the desired angular separation between SWBs, as mentioned above. Note that no effort was made to minimize the ΔV required for these scenarios, other than the obvious practice of centering the burns on perihelion or aphelion. S2 through S4 performed most of their aphelion-reducing ΔV at the first two perihelion passages. As a result, the perihelion distances were reduced substantially prior to performing their phasing orbits. However, because the phasing orbit periods did not exactly match those predicted by the impulsive ΔV calculations, the approximate desired spacing between these satellites was achieved after only 5 additional revolutions in the phasing orbit. At this point, an additional pair of perihelion burns was executed to circularize the orbits. A maneuver was also inserted at the aphelion between these final perihelion burns, to ensure that the final orbit dimensions matched those of S1.

S5 through S14 performed progressively smaller maneuvers at their first two perihelion passages, with the result that the phasing orbit periods were too far from ideal to achieve the desired orbital spacing. Because of this, a maneuver was added at the first aphelion to adjust the phasing orbit perihelion distance. Following the second perihelion passage, each spacecraft completed 7 phasing orbit revolutions before beginning its final sequence of burns for orbit circularization.

The ΔV and approximate burn duration computed for each spacecraft are given in Table 2. Note that this table only gives the ΔV required in the heliocentric orbit, and does not include the additional fuel required for targeting the entire fleet, on the mother ship, through the lunar flyby. The total ΔV for all but the first probe exceeded the impulsive estimate of 1.109 km/s. The largest ΔV of 1.2519 km/s was required for S6, which required 98.4 m/s for perihelion adjustment burns. This additional ΔV might be reduced somewhat through optimization exercises, but the realities imposed by the low-thrust propulsion system would nevertheless require additional fuel for all spacecraft beyond that predicted by the impulsive approximation. The longest burn was 89.2 days, required for the first perihelion maneuver of S1. At this time it is not clear whether such burns are operationally feasible for the proposed PPTs.

The angular separation between probes as a function of time is illustrated graphically in Figure 5, beginning 30 days after the lunar flyby. The different phasing orbit strategy utilized for S2 through S4 as compared to the remaining probes is evident in the figure; recall that for these three probes, 8 phasing revolutions were not needed, and they achieve their circular orbits two revolutions sooner than the rest of the fleet. While the angular separations achieved with the 400 W thruster scenarios do not exactly match the ideal spacing of 25.7° between adjacent spacecraft, the values are within 5° of the nominal values. Once again, no effort was expended to force the spacing to match the ideal targets, although this could be achieved with further effort.

Table 2. Single Mothercraft Scenario: Maneuvers Required for 0.868 AU Circular Orbits Assuming 30 kg SWBs, 8 Phasing Orbit Revolutions, and 400 W Thrusters

Maneuver	S1	S2	S3	S4	S5	S6	S7
ΔV (km/s)							
Phasing ΔV #1		0.5075	0.4612	0.4156	0.3709	0.3269	0.2837
Initial Aphelion ΔV					0.0714	0.0751	0.0768
Phasing ΔV #2		0.5079	0.4612	0.4156	0.4058	0.3685	0.3302
Circ. ΔV #1	0.5600	0.0530	0.0935	0.1390	0.1835	0.2275	0.2710
Final Aphelion ΔV		0.0204	0.0411	0.0534	0.0164	0.0233	0.0095
Circ. ΔV #2	0.5374	0.0449	0.1085	0.1660	0.1819	0.2306	0.2635
Total ΔV	1.0974	1.1337	1.1655	1.1896	1.2299	1.2519	1.2347
Total Fuel Used (kg)	6.3388	6.3234	6.3208	6.3297	6.4784	6.4465	6.4300
Burn duration (days)							
Phasing ΔV #1		78.5	69.9	61.9	54.6	47.6	41.0
Initial Aphelion ΔV					9.8	10.3	10.6
Phasing ΔV #2		74.2	66.5	59.3	57.7	52.1	46.5
Circ. ΔV #1	89.2	6.8	12.1	18.3	24.2	30.3	36.6
Final Aphelion ΔV		2.6	5.3	6.9	2.1	3.0	1.2
Circ. ΔV #2	79.0	5.7	13.9	21.5	23.5	30.0	34.6
Total burn duration	168.2	167.8	167.7	167.9	171.9	173.3	170.5

Maneuver	S8	S9	S10	S11	S12	S13	S14
ΔV (km/s)							
Phasing ΔV #1	0.2411	0.1993	0.1582	0.1177	0.0779	0.0387	0.0002
Initial Aphelion ΔV	0.0724	0.0657	0.0739	0.0742	0.0742	0.0741	0.0740
Phasing ΔV #2	0.2915	0.2523	0.2131	0.1738	0.1347	0.0959	0.0573
Circ. ΔV #1	0.3135	0.3555	0.3965	0.4370	0.4770	0.5160	0.5545
Final Aphelion ΔV	0.0116	0.0231	0.0133	0.0284	0.0328	0.0470	0.0655
Circ. ΔV #2	0.3037	0.3490	0.3755	0.4100	0.4402	0.4676	0.4942
Total ΔV	1.2338	1.2449	1.2305	1.2411	1.2368	1.2393	1.2457
Total Fuel Used (kg)	6.4249	6.5041	6.4706	6.5868	6.6521	6.7714	6.9400
Burn duration (days)							
Phasing ΔV #1	34.6	28.5	22.5	16.7	11.1	5.5	0.027
Initial Aphelion ΔV	10.0	9.2	10.3	10.4	10.5	10.5	10.5
Phasing ΔV #2	41.0	35.4	29.8	24.3	18.9	13.5	8.1
Circ. ΔV #1	43.0	49.6	56.3	63.4	70.8	78.6	87.1
Final Aphelion ΔV	1.5	3.0	1.7	3.7	4.3	6.2	8.7
Circ. ΔV #2	40.3	46.9	50.9	56.1	60.9	65.4	69.8
Total burn duration	170.4	172.6	171.5	174.6	176.5	179.7	184.2

Further analysis was completed to estimate the difference in maneuver profiles between 400 W and 800 W thrusters. A trajectory was modeled using an 800 W thruster (assumed thrust of 0.00984 N and I_{sp} of 1150 sec, dry mass of 58.7 kg, initial fuel mass 7.7 kg) for the first SWB only, following the targeting methodology described above. The total ΔV required for circularization was divided between two burns, performed at the first and second perihelion passages after the lunar flyby. The ΔV and burn duration for each maneuver were roughly half those of the 400 W case. Unfortunately, increasing the power requirements drives up dry mass requirements beyond the allowed envelope for this mission concept.

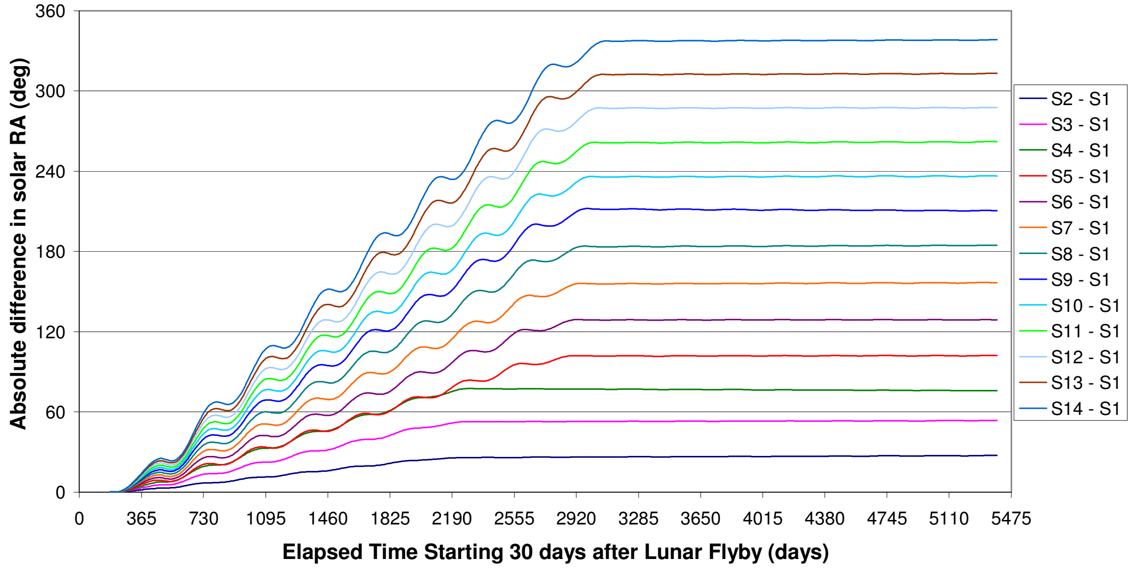

Figure 5. Separation Between Solar Buoy Probes for the Single-Mothercraft Scenario

Figure 6 depicts a sample transfer (this example belongs to S6) to a circular mission orbit of 0.868 AU via a multi-revolution phasing orbit modified by six separate multi-day low-thrust finite burns (see Table 2). The time from Earth departure to final circularization of the mission orbit is 8.2 years.

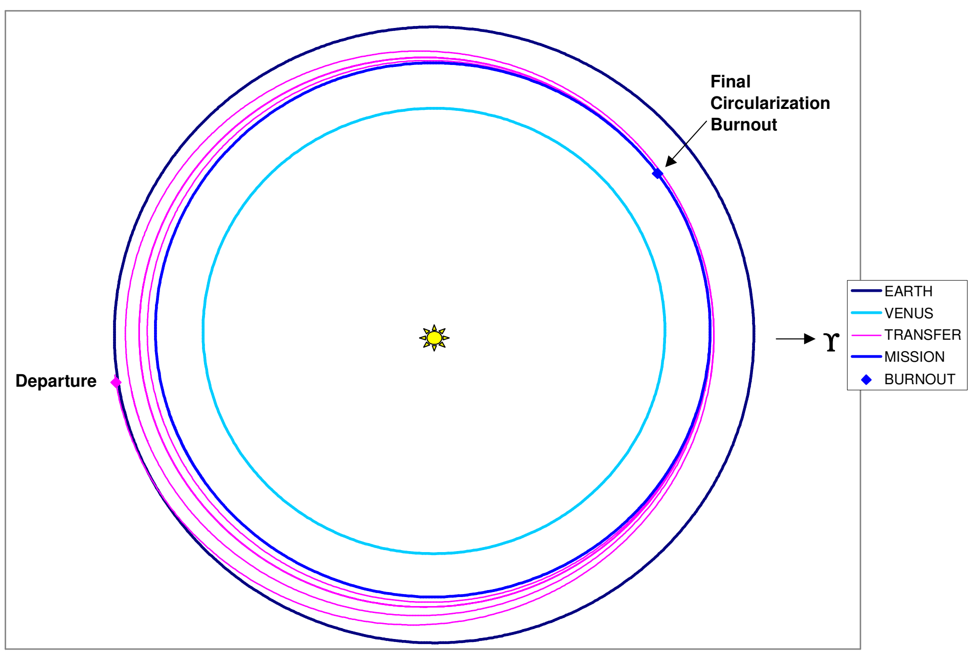

Figure 6. Representative Heliocentric Transfer to a Multi-revolution Phasing Orbit and Mission Orbit Circularization (Ecliptic Plane Projection in Heliocentric Inertial Coordinates)

2.3 Other Scenarios Considered

Additional means of establishing the Solar Buoy formation were also considered, incorporating different pre-flyby trajectories for spacing the satellites' entry into heliocentric orbit. For example, a Distant Prograde Orbit (DPO) around the Earth-Moon system with period of approximately 6 months could be used as the initial orbit instead of the L1 libration point orbit.[4] Similar to the method employed with the libration-point orbit cases, one spacecraft could theoretically depart the DPO for a lunar flyby each revolution, resulting in heliocentric orbit entries as frequent as approximately 6 months apart, if desired. As with the L1 orbit scenarios, however, trajectories departing from the DPO into solar orbit are also subject to effects due to the eccentricity of the Earth's orbit and the inclination of the Moon's orbit, resulting in variations in heliocentric orbital parameters. Thus the DPO was not found to have significant advantages over an L1 libration-point orbit, optimized for lunar geometry at the planned flyby time. This is especially true for the single mothercraft scenario described in Section 2.2.

A Double Lunar Swingby (DLS) orbit could also be utilized for the staging orbit prior to the lunar flyby, rather than a libration-point orbit.[5] This actually would provide greater flexibility in the timing of the lunar flyby than the L1 orbit departure scenarios, a fact that was exploited in the mission design using two mothercraft, discussed below.

3 – TRAJECTORY SCENARIOS EMPLOYING TWO MOTHERCRAFT

In this scenario, the second mothercraft stays in a libration-point or DLS staging orbit until the first mothercraft has achieved nearly 180º separation from the Earth. The timing of the second mothercraft's subsequent lunar flyby is planned such that this spacecraft will be located 180º from the first mothercraft upon completion of its heliocentric circularization maneuvers. Both groups of satellites then only have to spread out over 180º to obtain full 360º coverage around the sun. Advantages of this scenario include operational simplification, by not needing to execute maneuvers for all 14 SWBs at the same initial perihelion, and avoidance of the risk incurred by carrying all probes on a single mothercraft vehicle.

Because the first mothercraft would enter the desired heliocentric mission orbit through maneuvers at its first and second perihelion passages, the time lag between lunar flybys of the two mothercraft can be roughly calculated based on the difference in mean motion of the Earth's orbit and the mission orbit. With a heliocentric orbit of 0.868 AU, the lag time required between completion of circularization burns of the first and second mothercraft is approximately 2.1 years. The correct phasing for the second mothercraft is accomplished through a combination of additional revolutions in the initial L1 Lissajous orbit, followed by a DLS cycle. Use of the DLS orbit provided greater flexibility in the timing of the final lunar flyby of the second mothercraft, allowing it to arrive at perihelion close to 180° out of phase with the first mothercraft.

Figure 7 shows the transfer trajectory from the L1 orbit, through the DLS cycle, and into heliocentric orbit for the second mothercraft. The DLS cycle begins with a leading edge lunar flyby that establishes a phasing orbit of 18.9-day period, with dimensions of 27,875 km × 571,100 km. After one full revolution in this phasing orbit, a trailing edge flyby sends the spacecraft into a DLS outer loop of approximately 1-month duration. Details of the DLS orbit concept may be found in Reference 5. The outer loop is terminated by a trailing edge lunar flyby that sends the mothercraft into heliocentric orbit.

Prior to arrival at its first perihelion, the second mothercraft (designated S8) releases its contingent of SWB probes and performs the first of its two circularization burns. The sequence of events that follows is identical to that of the first mothercraft and its probes, which are already well along the way toward establishing the desired constellation. The individual probes S9 through S14 execute a similar sequence of maneuvers and phasing orbits to those of analogous probes S2 through S7 of the first mothercraft group.

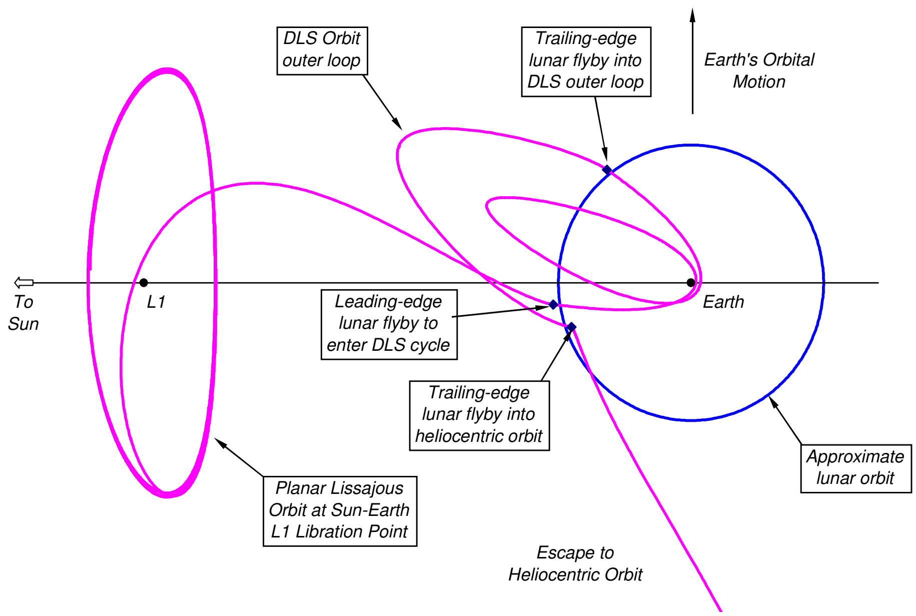

Figure 7. Earth Departure Trajectory via Lunar Flyby for the Second Mothercraft (Ecliptic Plane Projection in L1-centered Rotating Libration Point Coordinates)

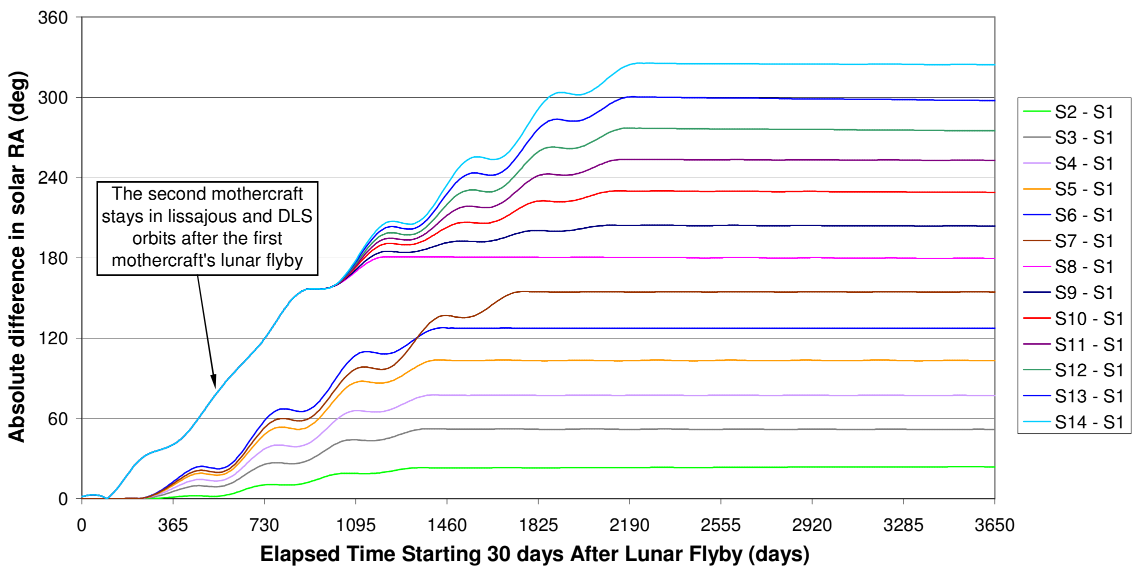

Figure 8. Separation Between Solar Buoy Probes for the Two-Mothercraft Scenario

To reduce the amount of time required to establish the constellation, the two-mothercraft scenario presented here was calculated with only 3 phasing orbit revolutions rather than 8, as in the single mothercraft case described above. Figure 8 shows the spacing between satellites of the SWB constellation, given as the angular distance of each spacecraft from S1. The figure begins 30 days after completion of the first mothercraft's lunar flyby. Probes S1 through S7 begin separating at their first perihelion passage, where they perform their first circularization or phasing maneuvers. The second contingent (S8 through S14) stays in Lissajous and DLS orbits for approximately 2 years before performing its lunar flyby into heliocentric orbit. The remaining lag time between mission orbit entry of S1 and S8 is incorporated into the orbit between the circularization maneuvers of S8. The heliocentric right ascension of S8 at its second perihelion is 40.5°; at the same time, S1 has a heliocentric right ascension of 221.2°, for approximately 180° spacing between these two mothercraft.

Table 3. Two-Mothercraft Scenario: Maneuvers Required for 0.868 AU Circular Orbits Assuming 60 kg SWBs, 3 Phasing Orbit Revolutions, and 400 W Thrusters

Maneuver	S1	S2	S3	S4	S5	S6	S7
ΔV (km/s)							
Phasing ΔV #1		0.4929	0.3334	0.2263	0.1193	0.0260	0.0674
Initial Aphelion ΔV		0.0297	0.0518	0.0629	0.0645	0.0664	0.0665
Phasing ΔV #2		0.3880	0.3343	0.2254	0.1195	0.0848	0.0675
Circ. ΔV #1	0.5433	0.1221	0.2383	0.3552	0.4423	0.5280	0.4866
Final Aphelion ΔV	0.0120	0.0040	0.0033	0.0159	0.0291	0.0628	0.0632
Circ. ΔV #2	0.5528	0.1221	0.2406	0.3395	0.4437	0.4775	0.4992
Total ΔV	1.1081	1.1588	1.2017	1.2252	1.2184	1.2455	1.2504
Total Fuel Used (kg)	6.3774	6.4538	6.4587	6.5814	6.7126	6.9793	7.0946
Burn duration (days)							
Phasing ΔV #1		78.2	50.0	33.3	17.4	3.8	9.8
Initial Aphelion ΔV		4.1	7.3	9.0	9.3	9.6	9.6
Phasing ΔV #2		55.9	48.4	32.3	17.2	12.1	9.7
Circ. ΔV #1	85.6	16.3	32.8	51.0	66.7	83.0	75.6
Final Aphelion ΔV	1.6	0.5	0.4	2.2	4.0	8.5	8.8
Circ. ΔV #2	82.0	16.1	32.4	46.9	63.6	68.3	74.7
Total burn duration	169.2	171.2	171.4	174.6	178.1	185.2	188.2

Maneuver	S8	S9	S10	S11	S12	S13	S14
ΔV (km/s)							
Phasing ΔV #1		0.4750	0.3500	0.2600	0.1650	0.0600	0.0300
Initial Aphelion ΔV		0.0977	0.1133	0.1222	0.1300	0.1280	0.1275
Phasing ΔV #2		0.4505	0.3773	0.2671	0.1594	0.0715	0.0273
Circ. ΔV #1	0.5600	0.1150	0.2290	0.3350	0.4380	0.5400	0.5880
Final Aphelion ΔV	0.0430	0.0119	0.0006	0.0090	0.0242	0.0530	0.1079
Circ. ΔV #2	0.5691	0.1276	0.2242	0.3227	0.4131	0.4816	0.5855
Total ΔV	1.1721	1.2777	1.2944	1.3160	1.3297	1.3341	1.4662
Total Fuel Used (kg)	6.7525	6.8584	6.7646	6.8428	7.0242	7.1873	8.4827
Burn duration (days)							
Phasing ΔV #1		72.2	51.2	37.4	23.5	8.5	4.260
Initial Aphelion ΔV		13.2	15.6	16.9	18.2	0.01	18.0
Phasing ΔV #2		64.0	53.2	37.1	22.1	10.0	3.8
Circ. ΔV #1	88.8	14.8	30.3	46.0	62.9	82.7	93.5
Final Aphelion ΔV	5.8	1.5	0.07	1.2	3.2	7.0	14.2
Circ. ΔV #2	84.6	16.2	29.2	43.0	56.5	70.0	91.3
Total burn duration	179.1	182.0	179.5	181.6	186.4	178.2	225.1

Table 3 gives the ΔV, burn durations, and fuel required for the finite burns of the scenario utilizing two mothercraft. The ΔV and fuel required for S8 through S14 were higher than the corresponding probes for the case using a single mothercraft. One reason for this is that the arrival conditions of S8 at perihelion are more constrained than those of S1, because of S8's requirement to achieve 180° spacing with respect to S1. This also imposes constraints on the phasing orbits of S9 through S14, which necessarily begin their phasing orbit maneuvers at the same perihelion as S8. In addition, because the two mothercraft performed their lunar flybys from different starting orbits and at different times, the transfer trajectory for S8 differs from that of S1. Note also that the phasing orbits for S14 in this representative case actually have aphelion slightly outside the Earth's orbit, as required to achieve the desired angular spacing in 3 phasing orbit revolutions. This is admittedly non-optimal in terms of ΔV, but as the goal of this exercise was simply a proof-of-concept and not an optimal trajectory design, no effort was expended to reduce the ΔV for S14. However, the results for S7 illustrate a possible method for accomplishing this. Rather than using 3 phasing orbits, which would have required the same increase in aphelion distance as S14, the trajectory for S7 used 4 revolutions in a smaller phasing orbit, reducing the total ΔV. Additional ΔV reductions could potentially be achieved by altering the initial perihelion bias, although the final design is constrained by the angular spacing requirements between S1 and S8. More refined stages of the mission design would include efforts to optimize these sorts of parameters.

For the single mothercraft scenario with 8 phasing orbits, it would take 8.7 years after the lunar flyby to achieve the desired SWB formation. By using 3 phasing orbits instead of 8, the time required to achieve the desired angular separation between probes may be substantially reduced. If the ΔV budget was not a constraint and this could be accomplished with a single mothercraft, the constellation would be in place only 4.2 years after the lunar flyby. However, if all 14 SWBs were carried on a single mothercraft, several of them would require phasing orbit aphelion distances either exterior to the Earth's orbit or interior to the mission orbit in order to move into place over only 3 phasing orbits. This would prohibitively increase the ΔV costs. By using two mothercraft instead of one, the use of 3 phasing orbits becomes feasible. The group of probes carried on the first mothercraft would achieve their final orbits 4.2 years after the flyby. Because of the 2.1-year lag time required for phasing the flyby of the second mothercraft, the entire constellation would be in place 6.3 years after the first mothercraft's lunar flyby. Even with the additional waiting time between lunar flybys, this scenario still would establish the entire 14-probe formation 2.4 years earlier than the case with 8 phasing orbits. (Note that using the alternative targeting strategy for S7 and S14, as described in the preceding paragraph, would increase the time required for establishing the constellation due to the extra phasing orbit revolution.)

Lastly, as this study progressed, the conception of the propulsion system and size of the spacecraft evolved along with the trajectory concepts. This culminated in a conception assuming each SWB configured with four 100 W PPTs — each producing thrust of 0.00124 N — yielding a total thrust of 0.00496 N for orbit maneuvers. Given the 400 W power requirement that four PPTs would entail, the baseline SWB dry mass estimate increased to 53.1 kg. The fuel mass baseline is a nominal 7 kg; hence, the total SWB mass would be approximately 60 kg each. For the two mothercraft scenario, each mothercraft would have a mass of approximately 60 kg also. Factoring in an assumed contingency or mass growth factor of 30%, the aggregate launch mass could be as much as 1259 kg. Boosting a payload of this mass to L1 is well within the performance capability of a Delta-II class launch vehicle. Given that margin exists, there is room for at least some of the SWBs to carry somewhat more than 7 kg of fuel if necessary.

4 - CONCLUSIONS

Two primary scenarios were examined for the Solar Weather Buoy mission concept, which would place 12 or more small spacecraft into an evenly-spaced formation in a circular heliocentric orbit by way of lunar gravity assist flybys. For the first scenario, it was assumed that the individual Solar Buoy spacecraft would not carry sufficient propulsion capacity for circularizing the heliocentric orbit, but would simply remain in the elliptical transfer trajectories following their respective lunar flybys. The variations in Sun-Earth-Moon geometry between these trajectories, the significant oscillatory spacing behavior due to the ellipticities of the individual mission orbits, and the potential for the probes to experience significant Earth perturbations once in heliocentric orbit all illustrated the need for substantial propulsive ΔV capability on each spacecraft to meet the proposed requirements of the SWB mission. However, it is important to note that for a variation of this mission concept that did not require the constant spacecraft spacing, the elliptical mission orbit solution might well suffice.

Other scenarios, involving a formation with all spacecraft in the same circular heliocentric orbit, examined the use of low-thrust plasma engines operating at 400 W or 800 W power levels. Assuming a 400 W capacity, the finite burns required for circularization of a heliocentric orbit of 0.868 AU were up to 89.2

days in duration, which may not be operationally feasible with the proposed engines. With an 800 W power level, the maximum burn duration would be reduced to 44.1 days. In any case, very long burn times are needed by all spacecraft to establish circularity in a scenario involving low-thrust engines with the specified performance characteristics. Nevertheless, the study did show that placing all spacecraft in the same circular orbit of given radius, with a given longitudinal spacing between them, will yield a stable formation that maintains constant spacing over an extended period of time.

The desired constellation could be accomplished through the use of a single mothercraft carrying 13 additional probes or two mothercraft, each carrying 7 probes. Advantages of using two mothercraft — with the second leaving Earth after sufficient delay for it to arrive at perihelion 180° from the first — include risk reduction and operational simplicity. The study also showed that, given the constraints on the allowable size and period of the phasing orbits, a scenario involving two mothercraft can be designed to establish the entire constellation significantly faster than a single mothercraft. Given the more favorable results and apparent advantages of this scenario, it must be considered the best of the approaches examined.

The study discussed here was largely limited to establishing the feasibility of such a mission, which was demonstrated. The 0.868 AU mission orbit that was studied is not necessarily ideal; it was just a working case. Directions for follow-on studies could include examination of orbits besides the 0.868 AU orbit, and the trade-offs between the time to full establishment of the constellation, deterministic ΔV requirements, and finite burn characteristics such as especially burn duration. There are relationships between the size of the intended mission orbit and the ΔV costs to achieve it, and between the size of the orbit and the time needed to establish the full constellation. The ΔV costs increase with smaller mission orbits, as Hohmann transfer calculations readily show. However, the time needed to establish the full constellation decreases with decreasing orbit size, because the desired separation can be achieved faster. The smallest mission orbit achievable using a lunar flyby departure technique is for practical purposes approximately 0.81 AU. There also exists a practical upper bound on mission orbit size, due to the need to avoid perturbations from the Earth's sphere of influence. Lastly, while angular spacing requirements and ΔV limitations dictate a certain minimum number of phasing orbit revolutions for the final SWB of a given scenario, not all SWBs of a given group need to execute this same number of revolutions. Within certain limits, earlier members of the group could be established sooner — with fewer revolutions — simply by opting for a larger phasing orbit. Hence, trades between various mission orbit sizes within a practical range, the number of phasing revolutions for each SWB, and the impacts on spacecraft propulsion of given specifications, could be explored.

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