STEREO Trajectory and Maneuver Design

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Solar TErrestrial RElations Observatory (STEREO) is the third mission in NASA's Solar Terrestrial Probes Program. STEREO is the first mission to use phasing loops and multiple lunar flybys to alter the trajectories of more than one satellite. This paper describes the nine launch windows that were prepared for STEREO, the launch, operations during the critical first phasing orbit (including a deterministic ΔV maneuver near apogee that raised perigee to prevent catastrophic atmospheric reentry), and the lunar swingby targeting used to achieve accurate deployment into the planned heliocentric (solar) mission orbits after the two lunar swingbys. These swingbys were critical for ejecting the spacecraft from their highly elliptical Earth (phasing) orbits. In addition, the STEREO team had to make some interesting trajectory decisions to exploit opportunities to image a bright comet and an unusual lunar transit across the Sun.

INTRODUCTION

NASA's Solar TErrestrial RElations Observatory (STEREO) program provides coordinated observations of the Sun and the interplanetary medium by using a two-spacecraft formation in heliocentric orbit, as shown in Fig. 1. One spacecraft precedes the Earth in its orbit around the sun and is named Ahead; the other trails the Earth and is named Behind. The scientific goals and historic background of this interesting mission are described elsewhere.[1,2] The two spacecraft were launched with a single Delta II launch vehicle. Detailed launch windows were calculated for nine monthly opportunities, a record for a launch with a Delta II rocket. The launch window that was actually used, in October 2006, had the shortest parking orbit coast of any Delta II launch, making it necessary for the first time to deploy mobile assets to the Cape Verde Islands. The deployment was accomplished just in time for the first launch opportunity on 26 October. A flawless launch was performed on that first day at 0052:00.339 UTC, near the end of the 15-min daily window. During the first 10-day orbit, the STEREO

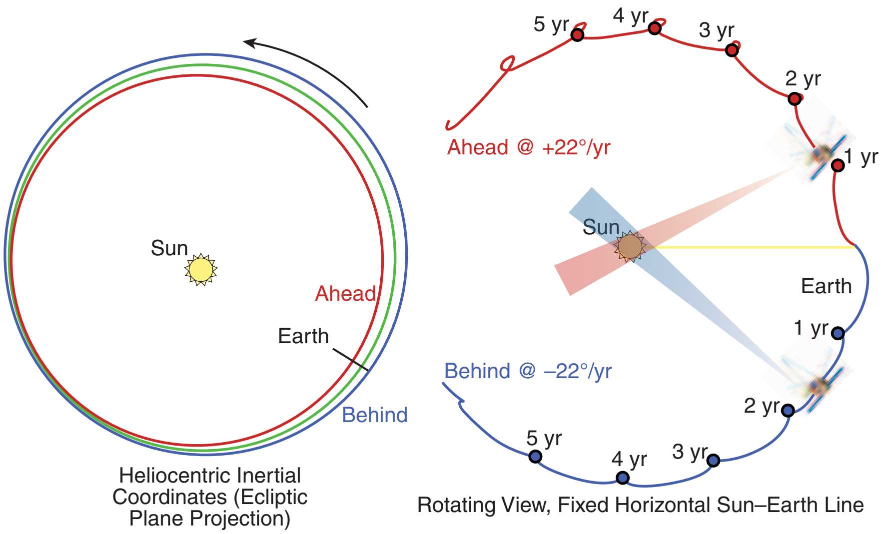

Figure 1. North-ecliptic-pole view of the STEREO spacecraft heliocentric orbits.

spacecraft was commanded to perform two 0.2-m/s engineering test maneuvers and an 11.7-m/s apogee maneuver to raise perigee, preventing atmospheric reentry at the first perigee after launch. The basic plan for the phasing orbit maneuvers is given below, including a description of guidelines and constraints. The second orbit also is described, especially the maneuvers in it that targeted the first lunar swingby (S1). In addition, the histories of three more maneuvers performed by the Behind spacecraft to target the second lunar swingby (S2) are discussed, along with other developments during this time, including Earth-based observations and images of the Moon and Comet McNaught taken by the spacecraft. A summary section describes the lunar transit imaged by Behind in February 2007 and discusses possible future options for the STEREO spacecraft.

LAUNCH VEHICLE AND SEQUENCE

The STEREO spacecraft mission used a Delta II 7925-10L launch vehicle. This vehicle consisted of a booster with a Rocketdyne RS-27A main engine augmented by nine Alliant Techsystems solid-propellant graphite-epoxy motors (GEMs) with extended nozzles on the air-lit GEMs, a second stage with an Aerojet AJ10-118K engine, and a Thiokol STAR 48B solid-motor third stage. A stretched 10-ft payload fairing enclosed the second stage, third stage, and payload during first-stage flight and the early portion of second-stage flight. The third stage used a 3712A payload attach fitting and a yo-yo despin system.

After coasting in a low-Earth parking orbit, the injection into the high-energy phasing orbit was accomplished by restarting the second-stage motor to initiate the transfer that is completed by firing the third-stage solid rocket motor. The spacecraft completed four revolutions in the phasing orbit before S1, giving time (for orbit determination as well as for designing and performing maneuvers) to accurately "phase" the orbit (change its period) to encounter the Moon at the right time and with the right geometry. The deployment sequence started shortly after the burnout of the Delta's third-stage solid rocket motor. The entire third-stage spacecraft stack was despun from an initial spin rate of near 60 revolutions per minute (rpm) to ~0.0 rpm by using a yo-yo device, which consisted of two weights on wires wound around the third stage that, when released, despun the stage. Table 1 details the timing of the ascent events.

After the TECO and despin, the two STEREO spacecraft were jettisoned from the third stage while stacked. The spacecraft then initiated a second separation that released the two stacked spacecraft from each other.

Table 1. Actual launch timeline.

Ascent Events for All Launch Dates	Time After Launch (s)	Injection Events for 26/27 Oct 2006 UTC	Time After Launch (s)
Liftoff	0.0	First Restart—Stage II	936.3
Main Engine Cutoff (MECO)	265.6	Second Cutoff—Stage II (SECO2)	1032.6
Stage I–II Separation	274.0	Fire Spin Rockets	1072.0
Stage II Ignition	279.5	Jettison Stage II	1075.1
Fairing Separation	283.5	Stage III Ignition	1113.4
First Cutoff—Stage II (SECO1)	609.9	Stage III Burnout (TECO)	1202.5
		Initiate Yo-Yo Despin	1500.1
		Jettison Stage III	1505.3

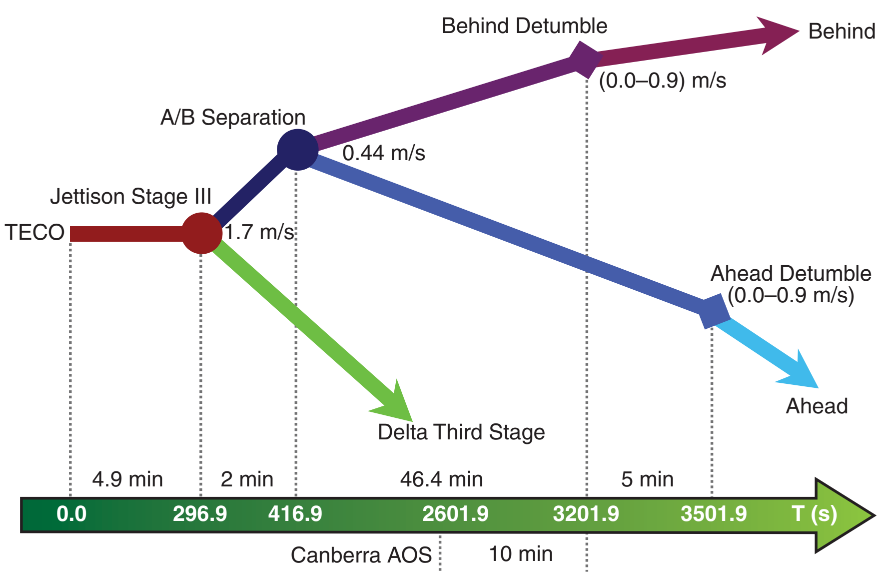

Figure 2. Nominal separation events, actual 26 October launch timeline. AOS, acquisition of signal; T, time.

The push-off forces for both the Delta–Ahead/Behind (A/B) and the A/B separation events were provided by springs. The spacecraft operated independently at all times and did not rely on any interspacecraft communications to coordinate their activities. After the A/B separation, timers started for turning on the traveling wave-tube amplifier (TWTA) and attitude system components. Also, a spacecraft timer was in place to inhibit a momentum dump for a prescribed period following an initiation of safe mode after A/B separation to reduce the collision probability. Two minutes after A/B separation, the spacecraft released their solar arrays and continued to drift apart.

Once the traveling wave-tube amplifiers were turned on and the spacecraft were in view of a ground station, the spacecraft activated their attitude control systems to dump any excess momentum and achieve a Sun-pointing attitude. The attitude is determined by using digital Sun sensors and three-axis rate information from an inertial measurement unit. Attitude control is provided by four reaction wheels and 12 thrusters. If momentum dumping is required, thrusters are used to provide the torque necessary to achieve the desired momentum state. The nominal separation events and their relative ΔV values are shown in Fig. 2.

LAUNCH WINDOW CONSTRUCTION

The STEREO launch windows were constructed by using the concepts of launch and coast times as clearly explained by Clarke[3] (according to Clarke, Krafft A. Ehricke was the first to propose the launch coast along the parking orbit). In our paper, coast time refers to the time between SECO1 and the first restart (see Table 1). The degrees of freedom provided by the launch and coast time variables coupled with the parameters of a launch from Cape Canaveral Air Force Station, Florida, facilitated the computation of trajectories that targeted the Moon. On the basis of the Moon's position, which had to be within a range of about 15° of ecliptic longitude difference from the direction to the Sun for the lunar swingbys to alter the trajectories in the desired way, software (using two-body problem dynamics) was set up to compute, classify, and organize launch opportunities in tables. These tables were used to quickly discard opportunities that violated ascent (including time to and duration of the first contact) and eclipse constraints. See Table 2 for the STEREO trajectory constraints. The data in the tables also were used as initial entries into higher-fidelity computations that were performed for each set of detailed test objectives (DTO) targets that were iterated with Boeing, the launch service provider, for the detailed launch computations.

Other constraints, such as the perihelion and aphelion distances in the heliocentric orbits, as well as the arrangement of mobile tracking assets required to observe the injection burns by the second and third stages, also had to be considered. The changing geometry of the Earth relative to both the Moon's orbit and the Earth–Sun line, as well as the fixed ground track of the parking orbit (since the launch azimuth was fixed at 93° for the Cape launch, resulting in a 28.5° inclination), resulted in a geometry similar to that of Fig. 3 for launches from late May to October. However, the geometry was reversed (i.e., rotate Fig. 3 180°, keeping the direction to the Sun fixed to the left) during the other months of the year; consequently, Behind had only one lunar swingby and Ahead had two of them. Note how the trajectories are very sensitive to the lunar swingby; in Fig. 3, a change of a few thousand kilometers in the swingby distance causes Ahead to escape directly into a heliocentric orbit, whereas Behind stays in a high-Earth orbit, falling back for S2 that ejects it into heliocentric orbit in the opposite direction. There are only ~2 days a month during which the Moon is in the right geometry relative to the Earth–Sun line for S1 to achieve the separate goals for both spacecraft.

Every month, on the basis of the lunar motion, the desired escape trajectories, and the onboard propellant budget, 14–16 days were selected for launch. In Fig. 3, the apogees of the four phasing orbits are labeled A1 through A4. By varying the launch energy of the initial phasing orbit, its period can range from just under 8 to more than 13 days, but a small maneuver near the second perigee (after the first two phasing orbits are completed) results in a period of 11 or 12 days for the last two phasing orbits. In this way, the total time in the phasing orbits can vary by 16 days, allowing a launch window this long, although the S1 time must remain fixed, within a day. The monthly launch window geometry could have the perigees on the night side or on the day side. The selection of the perigees on the night or day side was based on the trajectory and ground coverage constraints (the time to move ground coverage assets from one location to another was also considered). Once the selection had been made between night or day side perigees, daily windows were computed. Both descending and ascending solutions were available and were again selected subject to the trajectory and ground-coverage constraints. In total, there were 112–128 trajectories to consider every month (14–16 days × 2 night or day side perigees × 2 descending or ascending × 2 spacecraft). Again, software was set up to compute, classify, and organize these opportunities in tables. A typical plot showing the coast time behavior for orbits in which Ahead gets to its mission orbit first (lead > lag parking orbit coast time) is shown in Fig. 4. Recall from Table 2 that, for power and operational reasons, the maximum coast time allowed is 55 min, with a maximum eclipse time of 40 min, and with the first acquisition of signal taking place within 100 min of launch. Figure 5 displays the expected contact times and their durations as a function of coast time.

Within a given daily window, the rocket ascent trajectory remains the same, and only the time of launch varies. For a given launch window, there is one time (sometimes two,

Table 2. STEREO trajectory constraints.

Profile	Parameter	Minimum	Maximum
Ascent	Coast time	300 s[a]	55 min
	Eclipse time	N/A	40 min
	First station contact	N/A	100 min
Maneuvers	$+x$ body to Sun	N/A	45°
	$+/-z$ body to Earth	N/A	90°
	ΔV budget	N/A	182 m/s
Phasing + Flybys	Perigee altitudes	500 km	N/A
	Periselene altitudes	200 km	N/A
Heliocentric	Sun distance: Ahead	0.909 AU	1.022 AU
	Sun distance: Behind	0.983 AU	1.089 AU
	Earth distance: Ahead	Height: 500 km	0.750 AU
	Earth distance: Behind	Height: 500 km	0.881 AU

AU, astronomical unit; N/A, not applicable.
[a]Absolute minimum is 200 s.

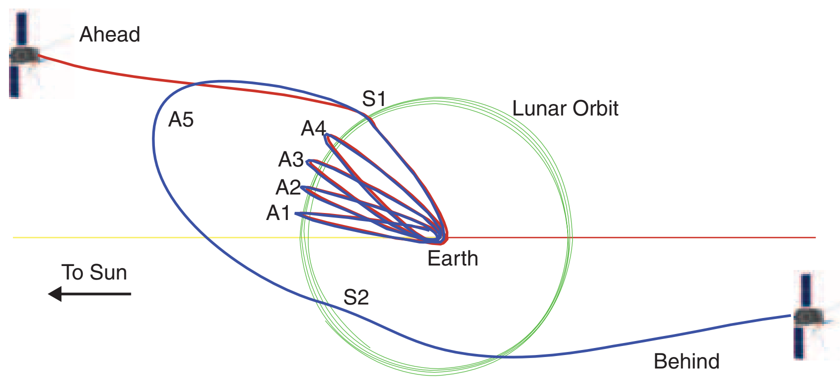

Figure 3. Phasing orbit and departure trajectory (fixed Earth–Sun line, ecliptic projection).

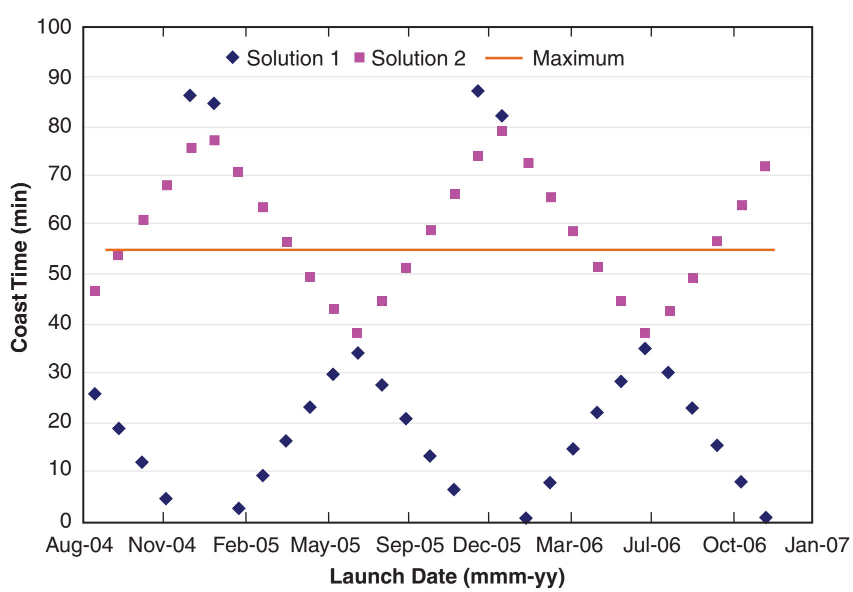

Figure 4. Parking orbit coast time (lead > lag).

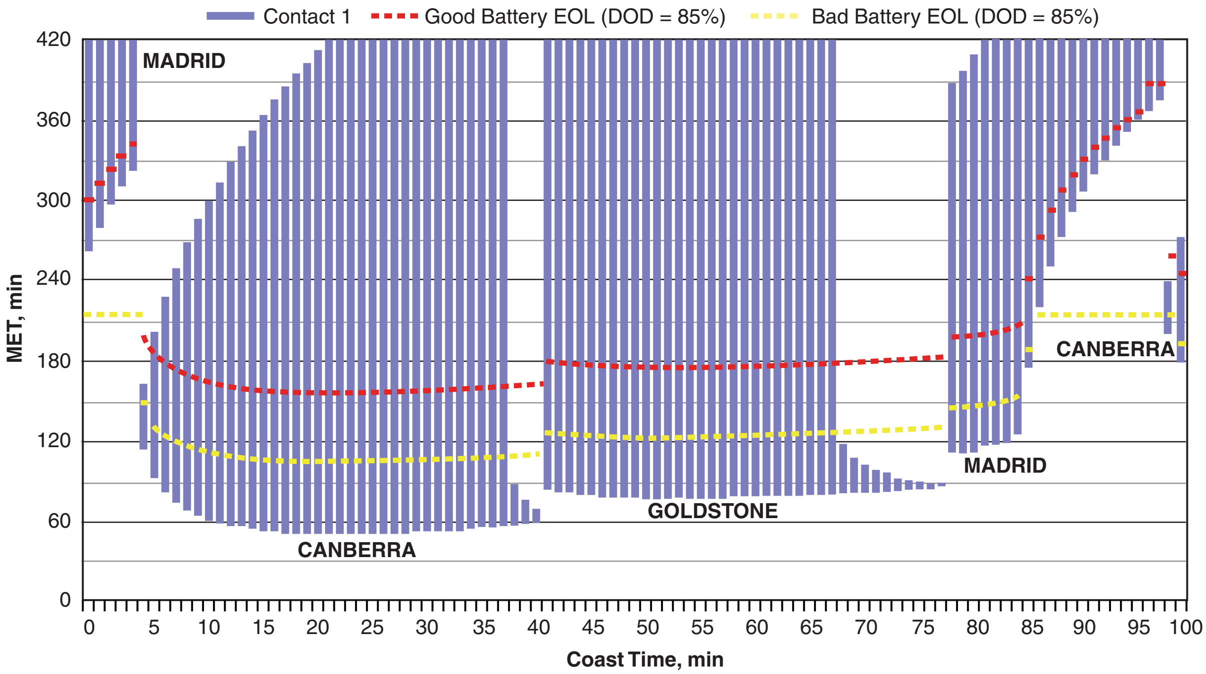

Figure 5. First station contact times as a function of coast time, EOL, end of life; DOD, depth of discharge; MET, maximum eclipse time.

one for short-coast and one for long-coast, or ascending/descending node, solutions) when the target point near the Moon at S1 is in the parking orbit plane, which is fixed to the rotating Earth. The phasing loop maneuvers are then used to absorb the offset in the first lunar flyby created by the different launch time. An allocation of 20 m/s per second is allocated for 15-min daily window maintenance. (Longer or shorter windows might be prescribed, depending on the particular window.) Monte Carlo runs[4] were used to allocate propellant for launch error corrections and trajectory correction maneuvers (TCMs). Figure 6 illustrates a typical window for one of the spacecraft with the prescribed allocations and the deterministic ΔV (with finite burn penalties).

It also is possible to have a second daily window by shifting the location of S1. An example of the dual daily geometry is shown in Fig. 7 for the first day of a 14-day window and in Fig. 8 for the last day. Note that the first two phasing orbits of Fig. 7 are larger (with periods of almost 14 days) than the last two phasing orbits, whereas the opposite is true (first phasing orbits have a period of about 8 days) in Fig. 8, at the end of the launch window. The small maneuver at the second perigee (after completing the first two orbits), in opposite directions in the two figures, causes this behavior, resulting in virtually the same S1 across the whole 14-day launch window.

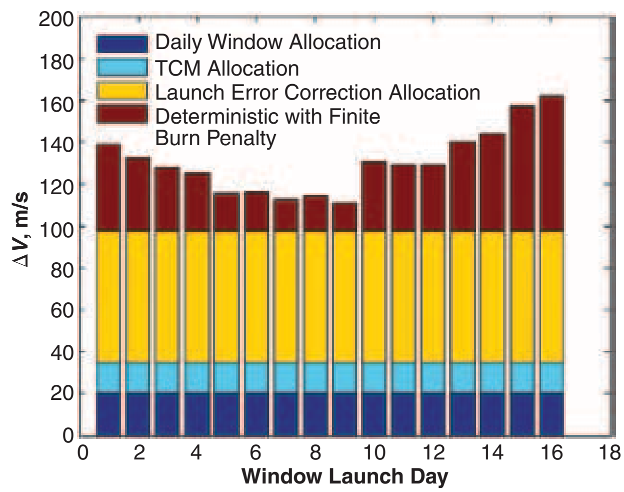

Figure 6. ΔV use for each launch day of a typical 16-day launch window (with maximum allocations from Monte Carlo runs).

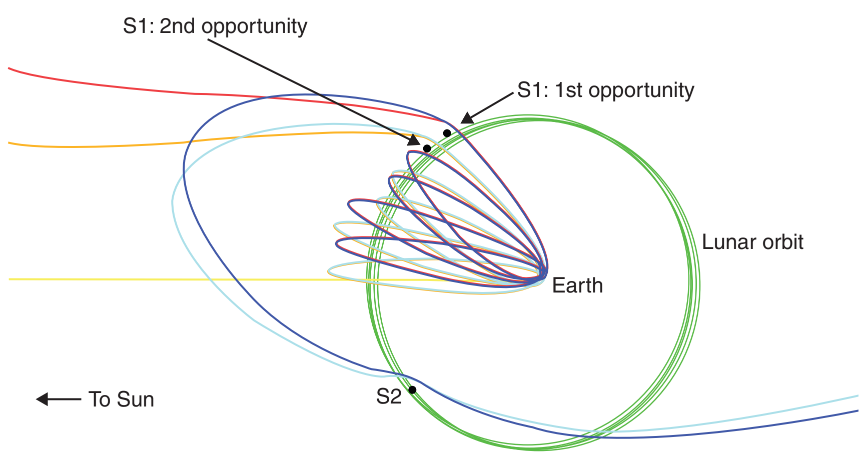

Figure 7. Typical dual daily trajectories, day 1 of 14.

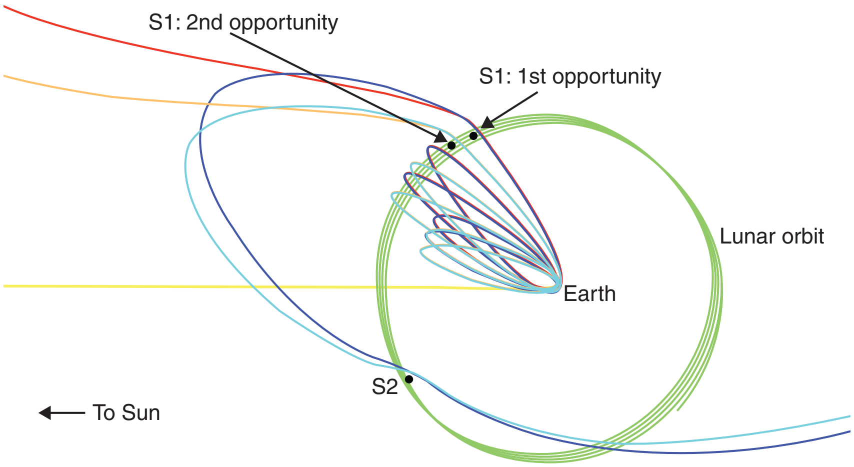

Figure 8. Typical dual daily trajectories, day 14 of 14.

For most of the DTO computations, dual daily opportunities were employed. Each daily opportunity used a different ascent trajectory. Figure 9 shows a schematic with this information. To provide the best chance to launch at the first opportunity even if something went wrong (e.g., weather, range intrusions, etc.), the time between the daily opportunities needed to be as wide as possible. The first window was short so that the battery would not be discharged very much. Boeing and NASA Kennedy Space Center (KSC) recommend a 2-min first window rather than a 1-min window to address any collision avoidance issues that usually clear in 1 min. During collision avoidance analysis, the trajectories were screened against a catalog of satellites and debris by the Joint Space Operations Center because some launch times would not be used if close approaches were present. Then, for operational reasons (including fuel loading), Boeing specified that 66 min was the maximum time available between opportunities. Moreover, the 66 min between windows would allow the spacecraft battery to be reconditioned for the longer second window. With a longer first window, the battery would be discharged more and could not be reconditioned to prime for the second window. Table 3 displays the DTO inputs provided by the mission design team to the Boeing launch team.

The actual window in which STEREO launched had only one opportunity because the coast time was too short to support two opportunities. Instrument delays, a strike by the International Association of Machinists and Aerospace Workers, and a NASA-mandated detailed inspection of the second stage [after second-stage flaws were discovered during a close inspection of the second stage for the THEMIS (Time History of Events and Macroscale Interactions during Substorms) mission

Table 3. DTO windows provided to Boeing.

DTO No.	2006 Launch Windows	No. of Daily Opportunities	No. of Days	Perigee Side	Spacecraft with Two Flybys	Reason Window Was Not Used (Except 8)
1	11–24 Apr	2	14	Day	Ahead	Strike, instrument delay
2	26 May–8 Jun	1[a]	14	Night	Behind	Strike, instrument delay
3	23 Jun–7 Jul	2	15	Night	Behind	Strike, instrument delay
4	22 Jul–6 Aug	2	16	Night	Behind	Spin balance, etc.
5	20 Aug–4 Sept	2	16	Night	Behind	Second-stage inspection
6	18 Sept–4 Oct	2	16[b]	Night	Behind	Second-stage inspection
7	7–20 Oct	2	14	Day	Ahead	Second-stage inspection
8	**19 Oct–2 Nov**	**1[c]**	**16**	**Night**	**Behind**	**Launch!**
9	4–17 Nov	2	14	Day	Ahead	After launch

[a]Dual daily not possible because of lunar orbit plane inclination and ascent coverage.
[b]For 18 Sept, the second opportunity could not be calculated (no viable trajectories were found).
[c]Only one opportunity for optimized ground coverage.

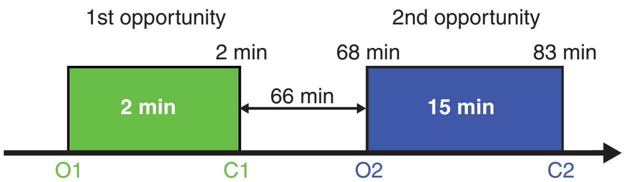

Figure 9. Dual daily launch window time schematic. O1/2, open 1/2; C1/2, close 1/2.

launch vehicle] were among the reasons the earlier launch opportunities could not be used.

During each launch window, the launch energy C_3 (twice the orbital specific energy from two-body problem dynamics) started near −1.65 km^2/s^2 and then decreased gradually to about −2.55 km^2/s^2 at the end 14–16 days later. The total deterministic ΔV would start near 20 m/s, gradually drop to ~10 m/s near the middle of the window, and then increase to ~40 m/s at the end. For operational reasons and to decrease the very detailed DTO calculations, launch blocks were created, in which the same launch trajectory (including coast time and second/third-stage injection to the desired C_3 after the coast) was fixed within the block, each block lasted 2–4 days, and only the launch time varied on the different days within the block. More information about the launch blocks, graphs showing details of the computed launch windows, the DTO iteration process with Boeing, and more information about the various launch delays have been published by Guzmán et al.[5]

October–November Launch Window: A Very Short, Fixed Coast

At the end of July 2006, the APL mission design and Boeing teams had performed preliminary computations for a window starting in October. Options in mid-October were causing problems because the descending solution (short coast) had long (120-min) eclipses in the phasing loops, and the ascending (long coast) solutions had first contact at Canberra about 79 min after launch. At this point, Dunham, the leader of the mission design team, suggested the computation of a night-side perigee launch window that would have very short coasts with first contact at Canberra and only one opportunity per day. Our colleagues at Boeing were concerned with the short coast times (on the order of 512.7 s in the mission design software), and thus a sample trajectory was provided to Boeing to verify that the short coast times were feasible. The Boeing team had no problems targeting the trajectory but had to shorten the coast time a bit further to ~475 s to have enough velocity reserves to meet the probability of command shutdown reserve requirements. All of the Boeing subsystems had to be reviewed because no previous mission had attempted such a short coast time. The Boeing team estimated the absolute minimum coast time between SECO1 and first restart to be 200 s.

Meanwhile, the short coast time of this window imposed additional challenges on the APL operations and on the NASA (Flight Dynamics Facility) navigation team. Specifically, for the Deep Space Network (DSN) stations, the first contact was at Canberra ~68 min after launch. However, the pass there was short, only about 3 h and 20 min, followed by a 100-min gap with no coverage until the second contact, at Madrid, 6.2 h after launch; the Madrid pass was rather normal, lasting 7.1–7.2 h, and coverage after that point was normal as the spacecraft rose higher. The first Canberra pass was too short to perform all of the planned first-pass operations or to determine the orbit well. Priority was given to just ensure spacecraft survival through the 100-min gap to the Madrid pass, which could then be used to complete the normal first-pass activities. Also, the operations team advised the navigation team that as much tracking as possible should be obtained during the Canberra passes. However, for navigation, the first pass would contain only Doppler data because the attenuators needed on the antennas preclude ranging.[6]

In addition, the NASA launch services team (at KSC) was concerned with the ground-coverage assets. The Big Crow (an instrumented tracking airplane) would not be available for the second week of the window. Thus, only assets that would be fixed during the window could be used for the entire window. To further simplify the logistics of ground coverage, the first opportunity on each day was given up, and the coast time was fixed at 364.1 s for the entire window. The official letters with the (DTO no. 8) target inputs were provided to Boeing on 6 and 13 September. See Table 4 for the mission design inputs; the maneuver designated as A3+ takes place shortly after the third apogee at 200° of true anomaly, which was found to be the location that minimized the ΔV.

After careful consideration of all of the ascent maneuvers and constraints, the Boeing team in conjunction with the NASA launch services team in KSC agreed that it would be feasible to use the October–November window. Tracking assets would be placed only in São Tomé and Cape Verde. Eventually, all of the requirements were met but not without some additional excitement. To get the tracking equipment to Cape Verde, the equipment was flown first in a commercial flight from Johannesburg, South Africa (where it was used for the previous window), to Dakar, Senegal. Then, to transport the equipment from Dakar to Cape Verde, a produce flight had to be used. It also took some time to obtain permission from the government of Cape Verde to import the equipment. In fact, permission was granted the day the flight had to leave from Johannesburg. Fortunately, during the 3 days that the crew had to set up the equipment at Cape Verde, no problems requiring new parts were found; otherwise, the launch would have been delayed. Figure 10 illustrates the ground track plot for the 26–27 October DTO trajectory.

Table 4. October–November window Behind basic information (data at window start).

Date	Block No.	Launch, UTC	C_3 (km^2/s^2)	Days to P1	P1 h (km)	A1 ΔV	P2 ΔV	A3+ ΔV	Total ΔV (m/s)
18 Oct	1	2346:37	−1.6504	13.2	3917	0.0	−20.5	0.0	20.5
19 Oct	1	2350:38	−1.6504	13.2	2193	0.0	−17.2	0.0	17.2
20 Oct	1	2353:03	−1.6504	13.2	1257	0.0	−14.9	0.0	14.9
21 Oct	1	2355:25	−1.6504	13.3	757	0.0	−13.3	0.0	13.3
22 Oct	2	2349:12	−1.7278	12.7	485	0.0	−8.7	0.0	8.7
23 Oct	2	2351:52	−1.7278	12.7	362	1.7	−8.0	0.0	9.7
25 Oct	3	0042:28	−1.8320	11.2	−195	9.2	2.3	0.0	11.5
26 Oct	3	0037:52	−1.8320	11.6	−188	9.1	3.3	0.0	12.4
27 Oct	3	0026:11	−1.8320	11.2	−12	6.8	2.7	0.0	9.5
28 Oct	4	0031:42	−2.0313	10.0	−84	8.4	11.6	0.0	20.0
29 Oct	4	0026:26	−2.0313	10.0	73	6.1	12.4	0.0	18.5
30 Oct	4	0016:50	−2.0313	10.0	−48[a]	8.0	11.4	0.0	19.4
31 Oct	5	0019:01	−2.1901	8.9	307	2.9	19.5	0.0	22.4
1 Nov	5	0010:15	−2.1901	8.9	−30[a]	8.3	18.5	0.0	26.8
2 Nov	6	0012:20	−2.3762	7.9	319[a]	3.0	27.9	0.0	30.9
3 Nov	6	0003:55	−2.3762	7.9	−43[a]	9.2	27.0	0.0	36.1

The launch times were rounded to the nearest minute, as described in Ref. 5. Prime days of blocks are shown in boldface; there is no launch on 24 October UTC.

[a]For theses dates, h is for P2, not P1, whose h is greater.

LAUNCH DAY

On the day of launch, weather balloons were released from the launch site starting 5 h before launch. Wind speed and azimuth, temperature, and pressure data were transmitted to Boeing in California and inserted in the 5-degree-of-freedom computer simulations run by the Trajectory Analysis, Controls, Guidance, and Structures groups to confirm that the vehicle could safely launch and obtain the desired orbit. On 26 October at 0038 UTC, the launch window opened. Initially, the countdown was holding at T − 4 min because the Range Safety Officer at the Cape indicated that the launch was "no go." The wind data were creating concerns that a launch explosion would make toxic gases drift over populated areas. Then, at 0043 UTC, the range announced that it was ready to go and that the launch sequence could resume at 0048 UTC. The countdown resumed, and the launch took place on 26 October at 0052:00.339 UTC (14 min and 0.339 s into the 15-min window).

The first stage was augmented with nine strap-on solid rockets, six ground-lit and three air-lit. Table 1 lists the times of the major launch events, and Fig. 10 shows both the ground track up to spacecraft separation and the coverage by mobile and fixed ground stations. The times given in the legend of Fig. 10 are based on DTO calculations in September and are superseded by the best estimate trajectory times in Table 1. The rocket configuration is shown in fig. 2 of Ref. 5. Telemetry showed that the ascent and injection into the parking orbit was nominal; the actual times were within a second of the best estimate trajectory times until the second stage was jettisoned (2 s early) and the last two events of Table 1, which were almost 5 s late. The third stage was spin-stabilized at about 55 rpm. Past performance of the third stage showed good reliability so that we could expect errors in its performance of only a small fraction of a percent.

At 0110 UTC, the Delta second stage was jettisoned from the third stage. At 0219 UTC, the second stage burned off its remaining propellant and then entered an orbit with a perigee height of 194 km, an apogee height of 3154 km, and an inclination of 24.7°. Atmospheric drag slowly decreased both the perigee and apogee heights to the point where the second stage finally broke apart and burned up in the Earth's atmosphere on 1 November 2007.

At 0117 UTC, the Delta third stage completed the injection of the STEREO stack into its highly elliptical orbit and the two separated from each other by the

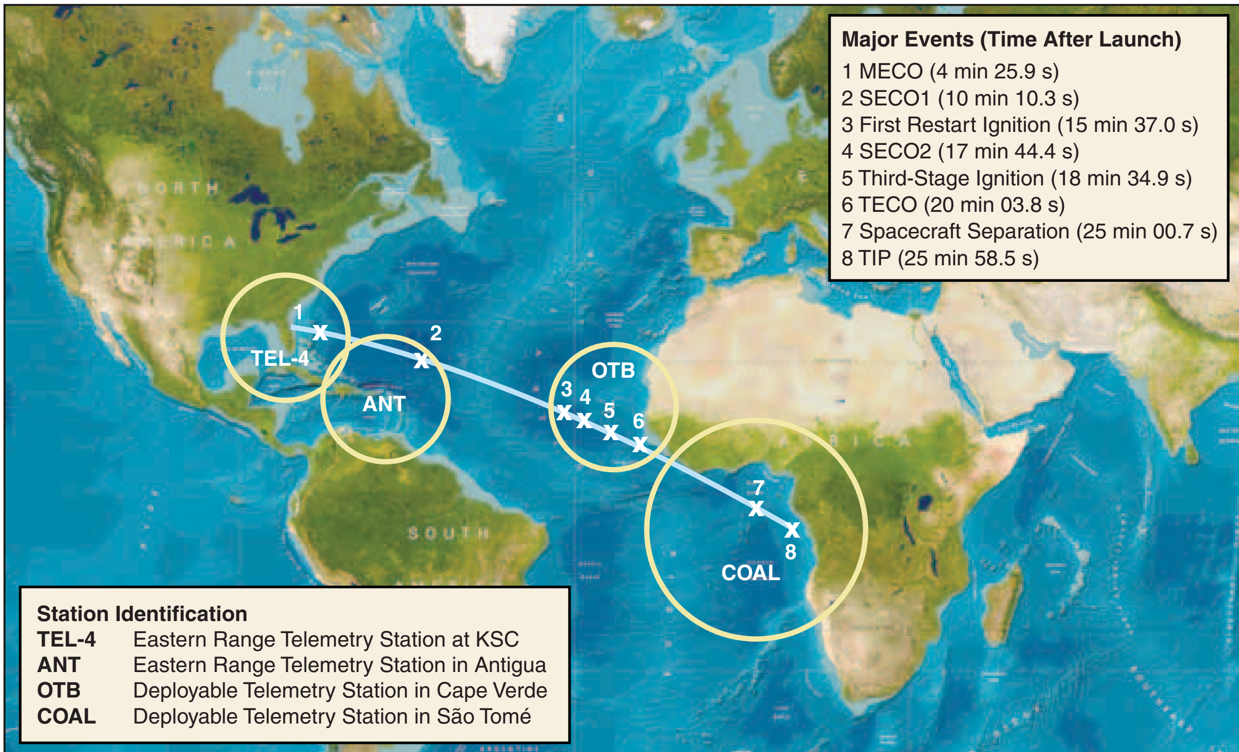

Figure 10. Ground track for the 26–27 October 2006 DTO trajectory. ANT, Antigua; COAL, São Tomé; OTB, Cape Verde; TEL-4, radar site near Cape Canaveral; TIP, target interface point.

separation springs. Two minutes later, more separation springs separated the STEREO A and STEREO B spacecraft from each other, and at 0120 UTC, all three objects emerged from the Earth's shadow into sunlight. At 0121:39 UTC, over half an hour before the spacecraft rose above the horizon at the DSN tracking station at Canberra, Australia, a retired professional astronomer, Greg Roberts, near Cape Town, South Africa, started taking images of the spacecraft. As far as we know, he was the first person to observe the spacecraft after their injection. Roberts provided several astrometric measurements of the two spacecraft, which had not yet separated enough to be resolved. Figure 11 shows one of Roberts' images, with STEREO shown as a dot in the center and the stars trailed. The two bright stars are μ and η Geminorum. There were some clouds (irregular diffuse light patches) and a nearby (out of focus) aerial telephone cable.

By using the trajectory determined from the early DSN tracking data,[6] the launch injection itself was found to be extremely accurate, with an estimated spacecraft-stack geocentric inertial velocity of only 0.393 m/s less than planned, only about 0.1 σ. The geocentric inertial velocity error is related to the C_3 error of Table 5 by the orbital energy (*vis viva*) equation.

After TECO and despin, the two STEREO spacecraft were jettisoned from the third stage while the spacecraft were still stacked. Again, the nominal separation events and their relative ΔV values are shown in Fig. 2. Following the separation of the two spacecraft from each other, timers on the spacecraft waited long enough to be sure that they had emerged in sunlight. Then thrusters were fired to automatically detumble and stabilize the spacecraft, orient them to point toward the Sun, and then deploy the solar panels to provide a continuous source of power. These early operations are described in Ref. 7.

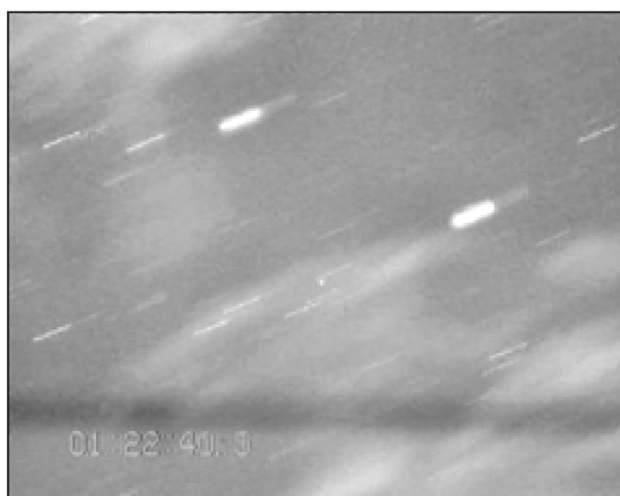

Figure 11. Image of STEREO several minutes after injection (at 0122:40 UTC), by Greg Roberts.

Table 5. STEREO orbit injection errors.

Orbit Parameter	Actual	Targeted	Error	3-σ Accuracy
C_3 (km^2/s^2)	−1.843	−1.835	−0.008	−0.72
Inclination (°)	28.44	28.46	−0.02	−0.18
Argument of perigee (°)	156.76	156.78	−0.02	−0.48
True anomaly (°)	38.43	38.29	+0.14	+0.49
RAAN (°)	230.55	230.53	+0.02	+0.15

RAAN, right ascension of the ascending node.

PLAN FOR THE PHASING ORBIT MANEUVERS

Each STEREO spacecraft has twelve 4.5-N thrusters for performing ΔV and attitude-adjustment (usually momentum-dump) maneuvers. The thrusters are arranged in three groups (A, B, and C) of four each, which are placed near the corners of the spacecraft box.[8]

Key events in the phasing orbits and locations for actual and planned maneuvers are shown in Fig. 12. As described previously, two small engineering burns were performed in the first orbit to assess the performance of the B and C thrusters. Some information about the A thrusters was obtained during the detumble maneuvers shortly after injection. The first engineering burn (E1) was deemed most important because the B thrusters it tested would be used for the critical perigee raise maneuver near A1. The maneuvers near A2 and P2 targeted S1 on 15 December 2006. As shown below, they were so accurate for Ahead that no further ΔV maneuvers were needed for that spacecraft. A small TCM was needed by Behind near A4 to improve the S1 targeting enough so that its S2 would produce a heliocentric orbit drift rate within the design constrains. Shortly after the A4 maneuver, the STEREO science team decided to change the aim point at S2 to achieve a transit of the Moon across the Sun as seen from the receding spacecraft on 25 February 2007. Two TCMs, one 6 days after S1 and one near A5, were needed to accomplish that objective. Additional DSN tracking for STEREO was scheduled for possible TCMs near the P1, P3, and P4 perigees, and near A3, but both the launch injection and previous ΔVs were accurate enough that no TCMs were needed at those locations. Maneuver design constraints are described below.

The +*x* axis had to be within 45° of the direction to the Sun, to ensure that the spacecraft had enough power from the solar panels. This means that if the ΔV was within 45° of the solar direction, the A thrusters were used, whereas if it was within 45° of the anti-Sun direction, the C thrusters were used. Most of the time, neither of these conditions were met, in which case, the B thrusters were used, and the spacecraft was rolled around the *z* axis to minimize the +*x* direction to the Sun to meet the constraint.

Telemetry was needed during the maneuver to monitor its real-time progress, to make sure that the systems were operating properly, to allow an emergency abort command if needed, and to measure the change in the Doppler

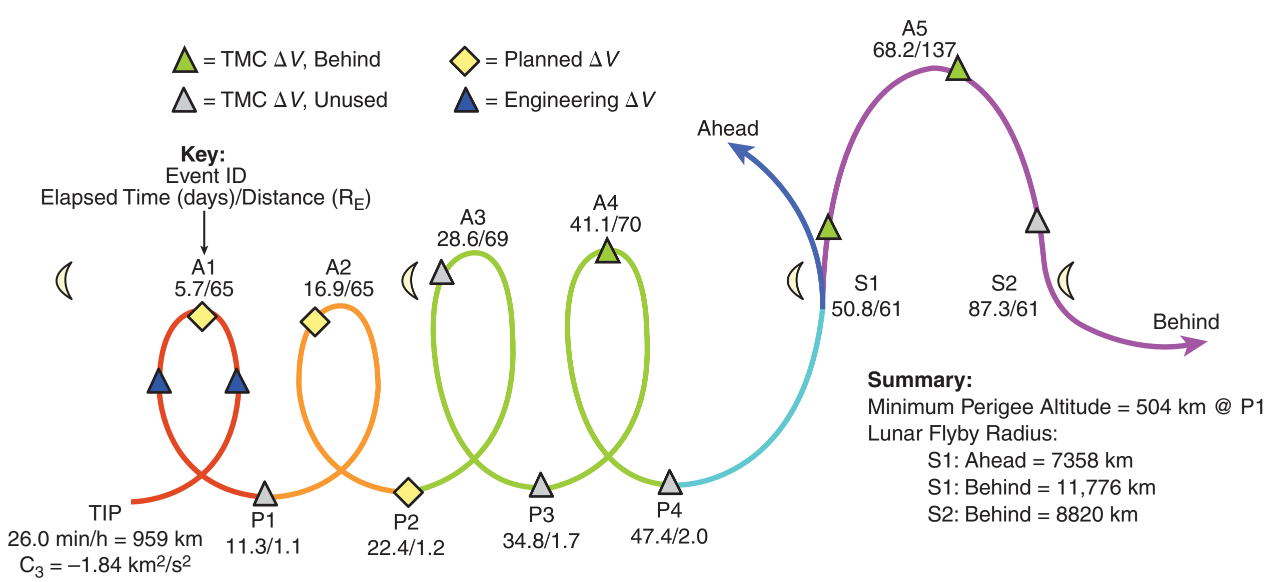

Figure 12. Schematic of STEREO's phasing orbits showing key events. A*x*, apogee *x*; P*x*, perigee *x*; R_E, Earth radii.

shift of the radio signal to see how well it matched the predicted shift. Therefore, the spacecraft had to be within one of the DSN visibility periods (the stations are near Madrid, Spain; Goldstone, California; and Canberra, Australia). This timing was important for perigee maneuvers when there was usually no DSN visibility within about an hour of perigee. During the phasing orbits, the distance to the Earth was relatively small, so that the hemispherical low-gain antennas (LGAs) on the $+z$ and $-z$ sides of the spacecraft could be used. The $-z$ LGA was preferred because it was used during nonmaneuver periods and there was a desire not to reconfigure to the $+z$ LGA unless it was necessary. We normally designed, when possible, to keep the $+z$-to-Earth angle >110° to further improve the link margin.

The maneuver had to be in sunlight (there were eclipses at the first three perigees) and ≥30 min from the start or end of an eclipse so that the spacecraft had time to return to its nominal ($+x$) Sun-pointing attitude before the maneuver or before the eclipse began.

There had to be an opportunity for a backup maneuver to achieve the same goals as the primary maneuver for a similar cost a day or more after the primary maneuver. In the case of maneuvers near perigee, the primary maneuver was performed far enough before perigee that there was DSN coverage and no eclipse, with the backup opportunity at the first chance after perigee. Details of the backup maneuver were worked out in advance, and commands to execute it were uploaded and stored on the spacecraft in a disabled state. If for any reason the primary burn failed to execute, a simple command could have been sent to enable the backup maneuver to execute.

In addition, there were orbit design constraints that might have limited the maneuver, to avoid a low perigee, etc., as described previously in Table 2. The maneuvers were designed to meet a specific goal, for example, the heliocentric drift rate from the Earth of +22.0°/year for Ahead and −22.0°/year for Behind, as described in more detail below.

FIRST ORBIT MANEUVERS

The launch vehicle placed both STEREO observatories, and the third stage, into a highly elliptical phasing orbit with a period of approximately 11.3 days and an apogee altitude beyond lunar orbit. During the first orbit, engineering maneuvers of 0.2 m/s were performed to test the B set of thrusters for both spacecraft in preparation for the critical first maneuvers at apogee because those maneuvers would also use the B thrusters. Figure 13 shows the geometry for E1 for Ahead, performed at 1330 UTC on 28 October, 2.5 days after launch. The circumstances for Behind's E1 burn, performed 3 h later, were nearly identical.

Both spacecraft needed apogee (A1) maneuvers of 11.7 m/s to raise their periapsis altitudes to 500 km; without them, lunisolar perturbations would have caused them to reenter and burn up in the atmosphere over the Atlantic Ocean, which is what happened to the Delta third stage. Using the nominal injection conditions with the actual launch time, the propagated trajectory for the third stage showed that it would reenter the atmosphere as a brilliant meteor about 200 km north of Puerto Rico, and some of the mission design team members considered traveling there to video-record this interesting event. But using the planned separation from the spacecraft, and the actual trajectories for them based on the navigation team's processing of the DSN tracking data[6] showed that the reentry would occur about 1000 km east of the Virgin Islands on 6 November 2006 at 0048 UTC. Because the reentry point was over the open ocean with no islands close enough to view the reentry, the reentry was not observed, as far as we know. The geometry for

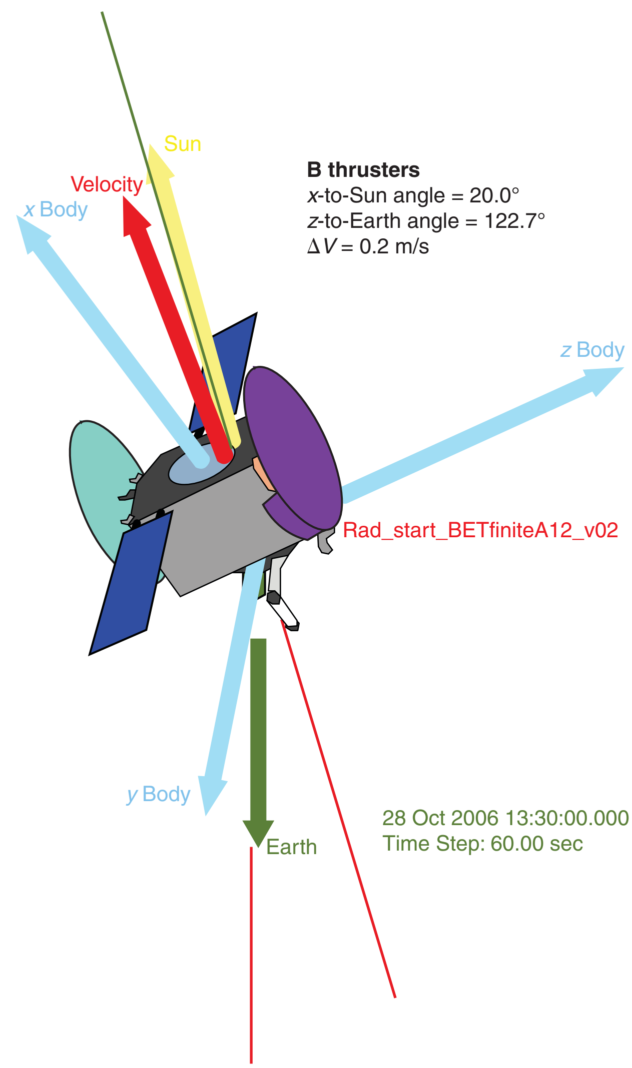

Figure 13. E1 ΔV geometry for Ahead, 28 October 2006.

Ahead's A1 burn, performed at 1800 UTC on 30 October, is shown in Fig. 14. Behind's A1 ΔV, performed 3 h after Ahead's, was only 0.1 m/s greater than Ahead's, with an orientation within 0.2° of that for Ahead. The performance of the maneuvers was very good, with magnitude errors <0.1% and pointing errors just under 1°.[8] As a result, the first perigee height was raised by almost 700 km, from about 200 km below the Earth's surface to 504 km for both spacecraft, just 4 km over the target.

On 2 November, exactly 3 days after the A1 maneuvers, the last set of thrusters (set C) was tested with a second engineering burn, called E2, that, like E1, had a ΔV of 0.2 m/s.[5] Its geometry for Ahead is shown in Fig. 15. Behind's E2 burn, again performed 3 h after Ahead's, had a spacecraft +*x*-to-Sun angle of 23.2° and a +*z*-to-Earth angle of 110.4°. For all of these first-orbit maneuvers, the spacecraft +*x*-to-Sun angle was well under the 45° maximum limit, and the +*z*-to-Earth angle was ≥110°, allowing communication with DSN stations on the Earth using the –*z* LGA without any obstruction by spacecraft structures. Because the –*z* LGA was also used for normal operations before and after the maneuvers, there was no need to reconfigure either spacecraft's telemetry system to switch to another antenna.

SECOND ORBIT, TARGETING THE S1 LUNAR SWINGBY

Because the launch occurred near the end of the daily launch window, large apogee maneuvers were required to adjust the out-of-plane component of the lunar B-plane for both spacecraft. The B-plane is used to define the target conditions for the lunar swingby; it is a plane passing through the Moon perpendicular to the incoming velocity direction.[9] In the launch window studies, the apogee maneuvers were always performed after the third apogee and were called A3+ maneuvers.[5] But with the large maneuvers that were now required near apogee, the mission design team felt that it might be better to perform these maneuvers an orbit earlier to allow A3+ to be used to fine-tune the targeting and to clean up any errors of an earlier attempt. Calculations soon showed that there was no penalty or significant difference in placing the maneuver just after A2 rather than after A3. The team also considered splitting the burn between the two apogees to perform two possibly easier-to-manage smaller burns, but the others on the STEREO project said this

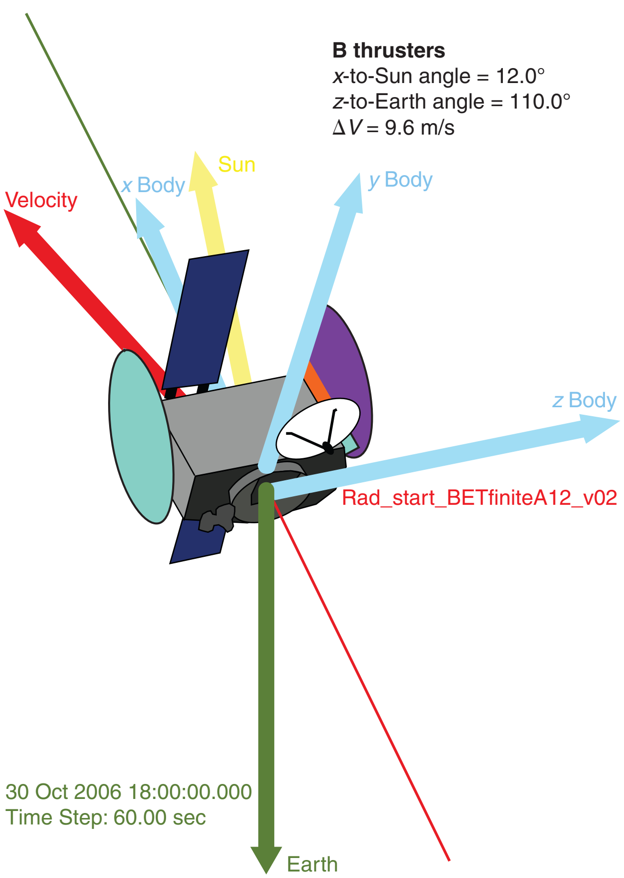

Figure 14. A1 Δ*V* geometry for Ahead, 30 October 2006.

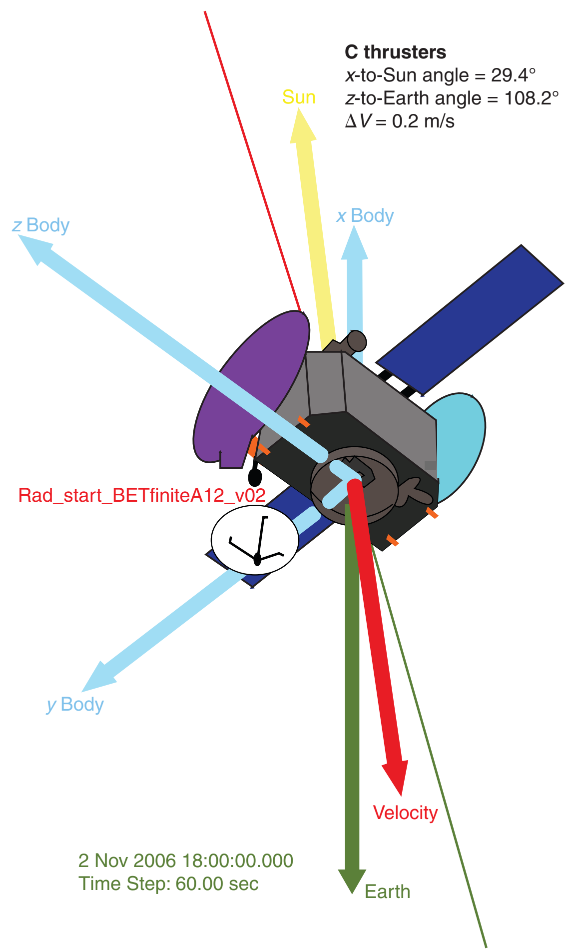

Figure 15. E2 Δ*V* geometry for Ahead, 2 November 2006.

was not necessary and would complicate things; the desire was to perform the least number of maneuvers to target the spacecraft to their desired heliocentric orbits. The mission design team recommended, and the STEREO Project concurred, that these maneuvers be performed shortly after A2 rather than A3. The original DSN tracking request did not include as much coverage near A2 as desired for a maneuver, and a possible backup, but with some minimal swapping of DSN time with other projects, suitable coverage was arranged. The hope was that if the A2+ maneuvers, as they were called at that point, were performed anywhere near as accurately as the A1 maneuvers, there would be no need for maneuvers near A3, and this result was realized.[6] The A2+ maneuvers were scheduled for 14 November with a backup opportunity the next day. The geometry of Ahead's A2+ maneuver is shown in Fig. 16. The geometry for Behind's A2+ maneuver was similar, with a +*x*-to-Sun angle of 35.2° and a +*z*-to-Earth angle of 111.5° but with a ΔV of only 28.4 m/s.

Less than 3 days after the A2+ maneuver, the P2 maneuver had to be performed. The A2+ and P2 maneuvers were out-of-plane and in-plane (timing) maneuvers, respectively, that were needed to target the spacecraft to the proper points in the B-plane near the Moon at the first lunar swingby, S1, on 15 December. The B-plane is shown in Fig. 17, with drift rate contours plotted. The B-plane components shown in the figure, *B***T* and *B***R*, are in the ecliptic plane and normal to it, respectively. The contours were generated by a large series of trajectory calculations, propagating the trajectory beyond the swingby for 1 year and evaluating the drift rate, if the trajectory is heliocentric (that is, if the distance from the Earth is >2 million km, comfortably beyond the L1 and L2 libration point distances of the Sun–Earth system). If the trajectory is not "heliocentric" by this definition, it is still bound to the Earth (or has impacted the Moon or the Earth) and is shown as green. The yellow-to-red areas show where positive drift rates are achieved; the 22°/year contour is the one desired for Ahead, and the red nominal aim point for it is on this curve. On 9–10 November 2006, shortly after the designs for the A2+ and P2 maneuvers were finalized,

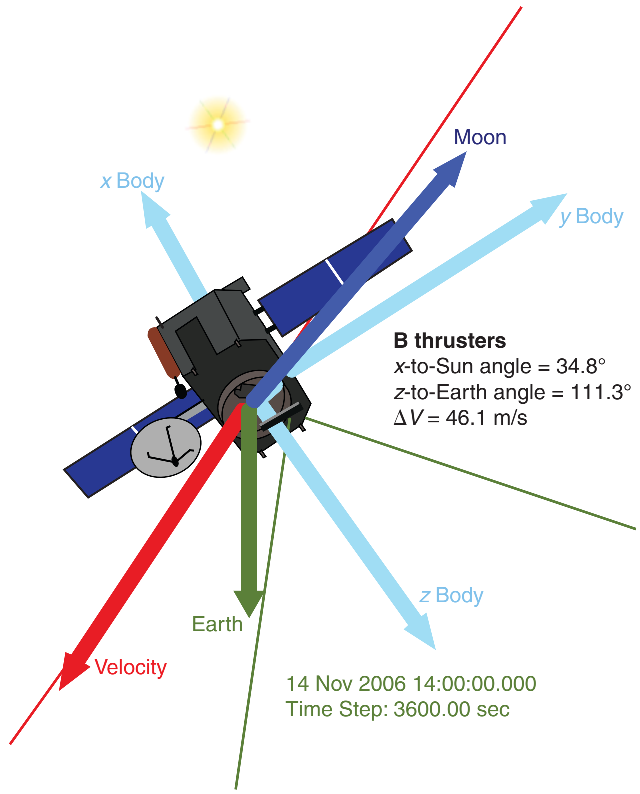

Figure 16. A2+ ΔV geometry for Ahead, 14 November 2006.

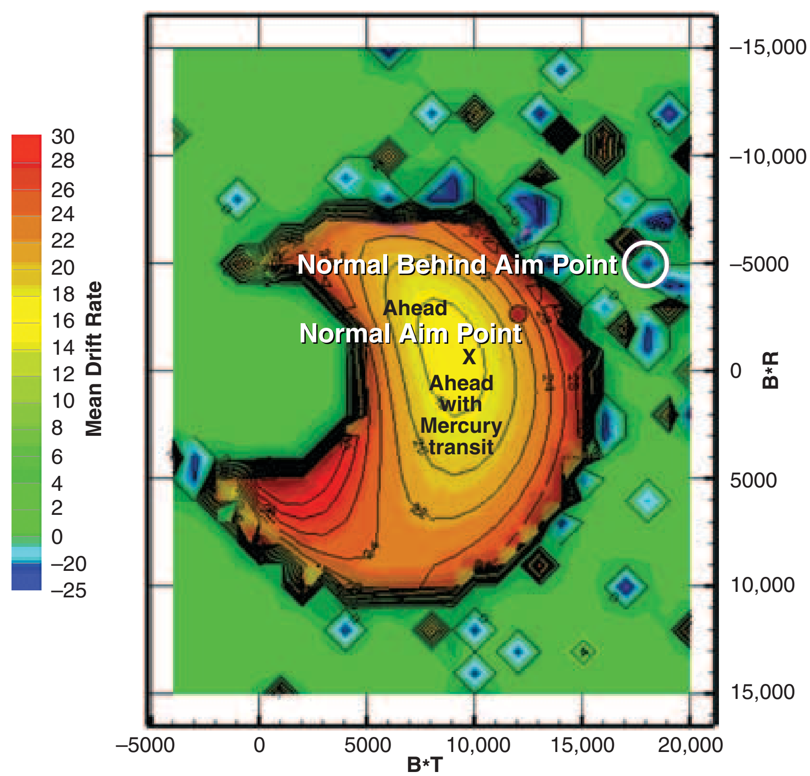

Figure 17. B-plane for the 15 December 2006 S1 lunar swingby. Positions in the B-plane are approximate. With the current trajectory, the actual minimum drift rate in the yellow area is near the 21.78°/year rate for the Mercury transit trajectory.

a transit of Mercury was seen from the Earth, so some STEREO scientists asked whether one of the STEREO spacecraft might be able to observe a Mercury transit. The authors found that by changing the B-plane target shown in Fig. 17, a Mercury transit could be arranged for Ahead. However, an additional maneuver would be needed, and the STEREO Project concluded that Ahead observations of a Mercury transit would have no scientific value, so the nominal trajectory without the transit was maintained.

There are also several small blue "islands" on the B-plane plot. These are places where, 1 or more months after the S1 swingby, another close swingby of the Moon occurred, causing the spacecraft to escape with a negative drift rate. This is precisely what is desired for Behind, and its nominal aim point is indeed on one of these islands.

The P2 maneuver could not be performed at perigee because the trajectory was in the Earth's shadow there, and there was also no DSN contact at the time. Consequently, the maneuver was scheduled so that it would finish at least 30 min before entering the Earth's shadow. For Ahead, the maneuver started 47 min before perigee, less than the 60 min used for determining the penalty factor of 2 used for the prelaunch ΔV budget calculations.[5] For Behind, the maneuver began 78 min before perigee, incurring a larger gravitational penalty factor, so the ΔV was nearly twice as large as for Ahead. However, with the small size of the P2 maneuvers by virtue of the 26 October launch being near the middle of the launch window, this posed no problem; the small (2-m/s) difference was more than compensated for by the much larger (by 18 m/s) A2+ maneuver for Ahead. The geometry for the P2 maneuver is shown in Fig. 18. For Behind, the geometry was similar, with a +x-to-Sun angle of 24.0°, a +z-to-Earth angle of 123.2°, and a ΔV of 5.0 m/s.

Using a 50-cm telescope, John Broughton, an amateur astronomer, obtained the charge-coupled device (CCD) image of Ahead, showing as an 8th-magnitude streak, shown in Fig. 19.

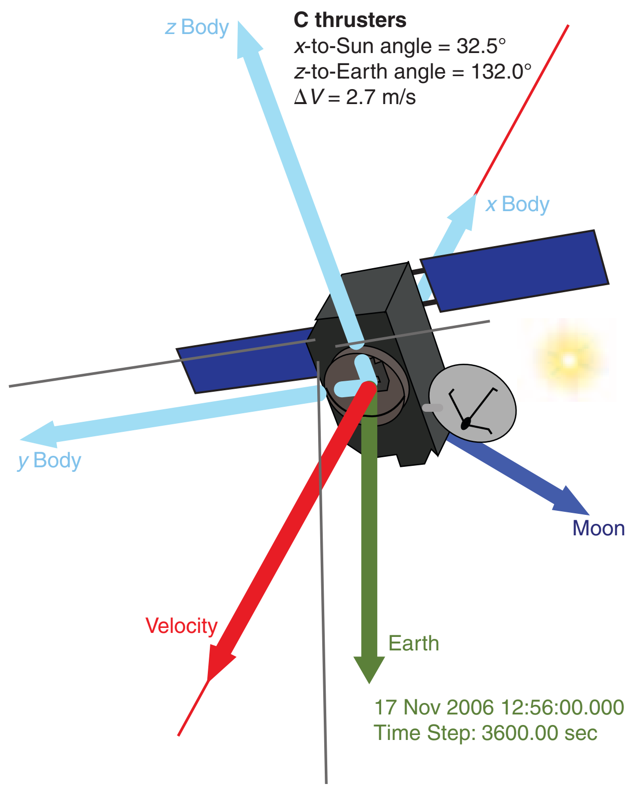

Figure 18. P2 ΔV geometry for Ahead, 17 November 2006.

Figure 19. Ahead image near P2 by John Broughton, Reedy Creek, Queensland, Australia.

THE LAST PHASING ORBITS, TO BEHIND'S LUNAR TRANSIT

The orbit determinations[6] after P2 showed that the actual trajectories were close to the planned one, proving that the A2+ and P2 maneuvers had been performed accurately. For Ahead, the S1 swingby was targeted accurately so that the drift rate would be +21.58°/year, well within the 2° tolerance from the planned +22.0°/year. No more trajectory ΔV maneuvers were needed, and none were executed by Ahead after the P2 maneuver. Behind also had an S1 swingby that was accurate enough to already cause it to have an S2 swingby, but the resulting drift rate was a few degrees from the −22.0°/year target; another small maneuver would be needed. The first thought was to perform it near P3, but the trajectory was too sensitive there; the guidance and control team couldn't guarantee that the small maneuver there could be executed accurately enough to achieve the goal. Also, a maneuver on 29 November would require Operations personnel to work over the Thanksgiving holiday. It was found that a maneuver near A4 in early December could easily correct Behind's drift rate; 6 December was selected because STEREO already had good DSN tracking coverage scheduled then.

Early on, an unfortunate aspect of Behind's trajectory was that it violated the solar distance constraints of Table 2. Between the S1 and S2 lunar swingbys, with the Earth near perihelion, the minimum distance was violated much of the time, but this was just an artifact of the phasing orbits, and there was no violation of the minimum distance in the heliocentric orbit after S2. The solar distance constraint is scientific, based on the sizing of the occulting disk for the Sun–Earth Connection Coronal and Heliospheric Investigation (SECCHI) instrument suite's coronagraph for covering the desired part of the Sun; there were no thermal or communications problems out to the A5 apogee. Because science operations were not guaranteed until after the phasing orbits were completed, the minimum distance "violation" posed no problem. However, the aphelion of Behind's heliocentric orbit would cause the spacecraft to exceed the maximum solar distance of 1.089 AU for ~1 month each year. Again, this distant aphelion posed no thermal or communications problems, and the science team said they could live with it. However, an official waiver was needed because of this constraint violation, which the Project preferred to avoid.

The mission design team decided to search for trajectories that might have lower aphelia. They constructed a drift rate B-plane plot for S2 similar to that of Fig. 17; the result is shown in Fig. 20. Besides the nominal trajectory, nine trajectories were selected along the horseshoe-shaped −22.0°/year for further study, numbered and color-coded in Fig. 21. Trajectory 3 turned out to be on the outer side of the "horseshoe," but a point that was closer to the Moon yet still on the inside of the horseshoe was desired. A 10th trajectory, called 2a, was calculated to meet the goal. The resulting trajectories are shown in two views in Figs. 21 and 22. In addition to the −22.0°/year horseshoe that was considered, there were also trajectories with the right drift rate along the outer dark edges of the negative drift rate region, but they were in a sensitive area and difficult to calculate; they all dwelled near the L1 and L2 libration points, properties that were not wanted for operational and scientific reasons.

The upper trajectories of Fig. 21, looping over the L2 libration point, correspond to the more distant parts of the −22.0°/year contour; they were not wanted because the spacecraft would be on the "wrong" side of the spacecraft–Sun line for the first few months, hindering communications and scientific operations at a time when scientists were eager to prove the STEREO system's 3-D solar imaging capabilities. So the closer, inner trajectories were examined in more detail. Trajectories 2a and 8 did not cross above L2 and had lower aphelia distances than the nominal trajectory; in fact, they were less than the 1.089 AU constraint. These trajectories were tested for eclipses, and a long one by the Moon was found for 2a; it occurred on 25 February

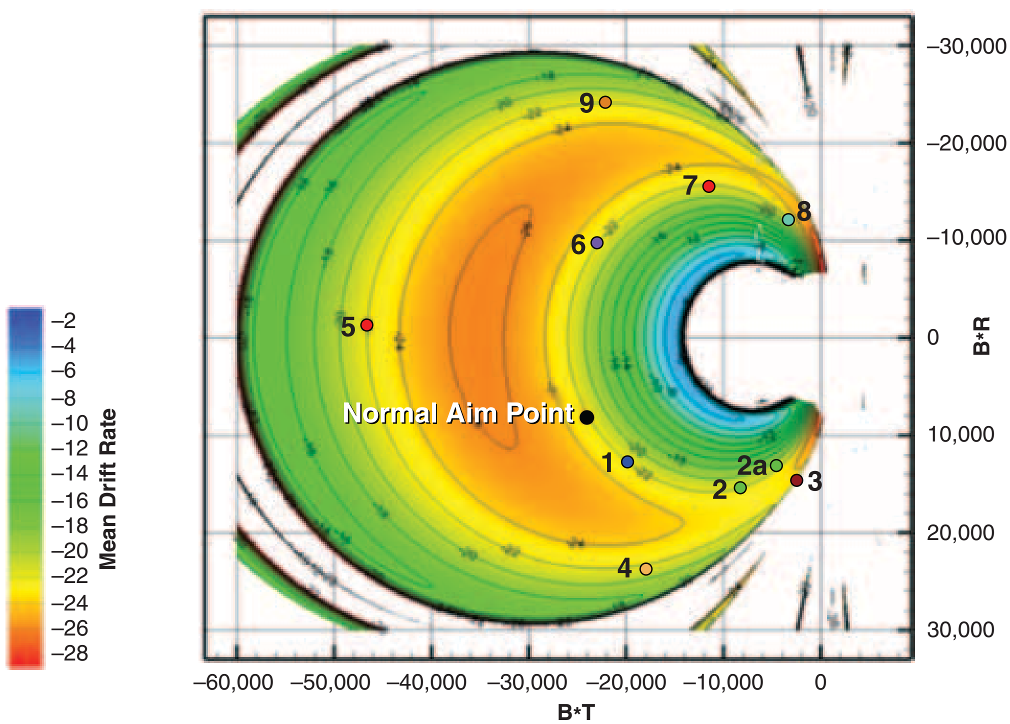

Figure 20. B-plane for the 21 January 2007 S2 lunar swingby for Behind.

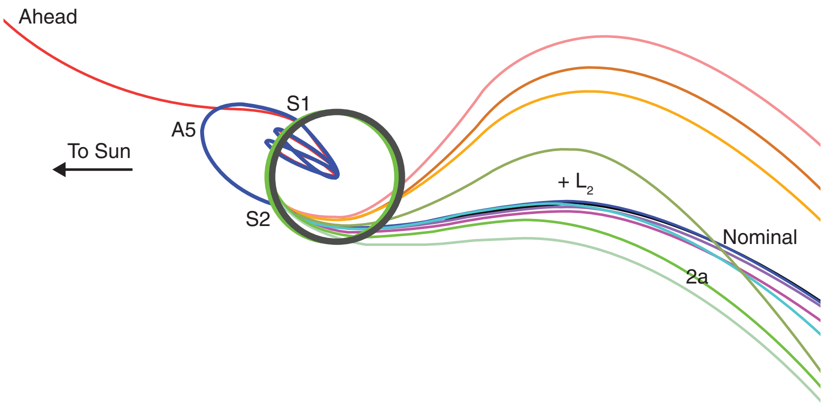

Figure 21. Rotating ecliptic-plane view of 11 trajectories with a −22°/year drift rate.

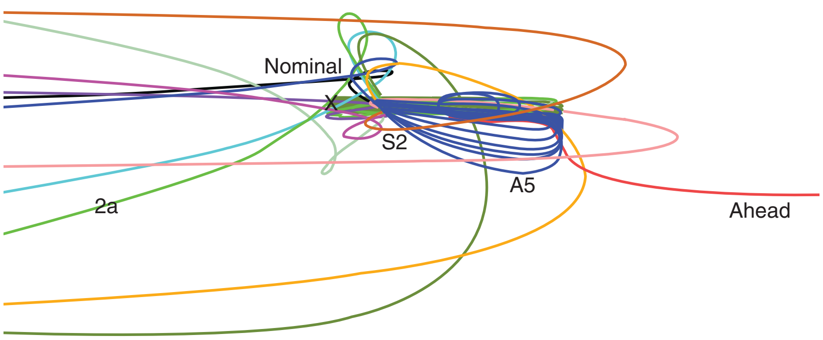

Figure 22. Rotating view toward the Sun of 11 trajectories with a −22°/year drift rate. The lunar transit occurs at the "X" on the 2a trajectory.

2007 and is shown with an "X" in Fig. 22. At first, this eclipse was thought to be a show stopper until it was realized that <5% of the Sun's disk would be eclipsed, a change in the thermal environment well within spacecraft tolerances. Rather than an eclipse, it was more like a big transit. When informed of the possibility, some of the SECCHI scientists became excited; the transit could provide an opportunity to make good measurements of stray light to help calibrate other SECCHI images. At first, some scientists were worried that the higher inclination of the 2a trajectory to the ecliptic would cause some problems in correlating Behind's images with those by Ahead, although this would be a temporary problem that would go away as soon as the spacecraft drifted farther from the Earth. The operations team did not want to change the A4 maneuver that was already planned, so the maneuver was executed to target the nominal trajectory. Shortly after that, however, the STEREO scientists reached a consensus that the 2a trajectory with the transit was preferred. Work began on design of an "S1+" maneuver 6 days after S1, on 21 December, to change the S2 B-plane point to achieve the 2a trajectory.

In the meantime, the STEREO spacecraft were headed toward their last (P4) perigee and the Moon. Near P4, the spacecraft would be higher than the previous perigees, and because there was no eclipse, they would be visible from a wide area, including western North America and most of the Pacific Ocean. However, because of the distance, they would be relatively faint. About a week before perigee, the mission design team remembered the "sunglint" maneuvers performed by the Near Earth Asteroid Rendezvous (NEAR) spacecraft as it approached the Earth in January 1998. The spacecraft's momentum wheels were used to change the spacecraft's orientation by small amounts, causing its 10 m^2 of solar panels to sweep a beam of sunlight across the United States in a pattern that included most large cities.[5] Because the STEREO spacecraft have solar panels that are similar in size to NEAR's and because the P4 perigee was lower than it was when NEAR performed its sunglint maneuvers, the team estimated that STEREO sunglints could appear as bright as first- or second-magnitude stars. The NEAR software was modified to calculate a series of sunglint pointings for both STEREO spacecraft that could have beamed sunlight to 16 metropolitan areas, starting at Honolulu and ending at Laurel, Maryland (where the STEREO operations center is located) during the half hour after the P4 perigee. The guidance and control team verified that these pointings could be performed by the spacecraft, and Operations personnel concluded that the spacecraft could safely do them. Early on 8 December, the mission design team sent the quaternion files for the sunglint maneuvers to the operations team, which began generating the spacecraft commands needed to implement the sequences. A couple of hours later, the Project decided not to perform the maneuvers, and the operations team work on them was stopped. Still, the spacecraft did fly over North America, but they were quite faint. The only known optical observation of STEREO near P4 was made, again of Ahead, by Bill Keel of the Astronomy Department of the University of Alabama by using remotely the SARA (Southeastern Association for Research in Astronomy) 0.9-m telescope on Kitt Peak in Arizona.

In Ahead's normal Sun-pointing attitude, momentum started to build toward unacceptable limits; the Guidance and Control team designed a momentum dump to take care of this problem. It was successfully executed on 13 December, 1 day after P4. Shortly afterward, because no more Ahead ΔV maneuvers were envisioned, the heliospheric imager's door was opened, and the spacecraft obtained an interesting series of images of the Moon just after the S1 swingby (one example is shown in Fig. 23). The dark side of the nearly new (from Earth's perspective) Moon is seen illuminated by earthshine; the CCD sensor was overwhelmed by the sunlit part of the Moon, and only a small of it visible at the top of the figure.

After S1, on 21 December, Behind performed the maneuver needed to change the S2 B-plane point to the 2a trajectory. As can be seen in Fig. 24, the size of the maneuver and its geometry were very similar to the A1 maneuver. Although the S1+ maneuver was executed very accurately in both magnitude and direction, the post-S1 orbit solutions had some instability; by 4 days after the maneuver, it was determined that the new trajectory would just barely miss the lunar transit. Another maneuver of 0.8 m/s, performed on 8 January 2007, a few days after the A5 apogee, successfully targeted the lunar transit. However, the success of the targeting was not discovered for a few days, during which time the STEREO scientists and Project anxiously awaited the opening of Behind's instrument doors. Finally, on 11 January, the mission design team concluded that the trajectory was good for the lunar transit and abandoned plans for any more maneuvers. Just in time, Behind's instrument doors were opened, and immediately an impressive series of images of Comet McNaught, the brightest comet to appear in 30 years, were taken. One of the images is shown in Fig. 25. The comet's overexposed head, saturating the vertical lines of the CCD, is on the right. Venus is at the bottom on the left with a vertical saturation line that is not nearly as strong as McNaught's. Many background stars and the comet's impressive tail, showing much detailed structure, completes the interesting view.

As noted above, Behind's last maneuvers enabled Behind to image a lunar transit of the Sun. Some images of that transit, taken just over a month after the last swingby, are shown in Fig. 26.

PHASING ORBIT TABLES, MANEUVER RECONSTRUCTIONS

Information about key points in the phasing orbits of the STEREO spacecraft is given in Table 6. Details of

Figure 23. The Moon imaged with the heliospheric imager by Ahead just after S1 on 15 December 2006.

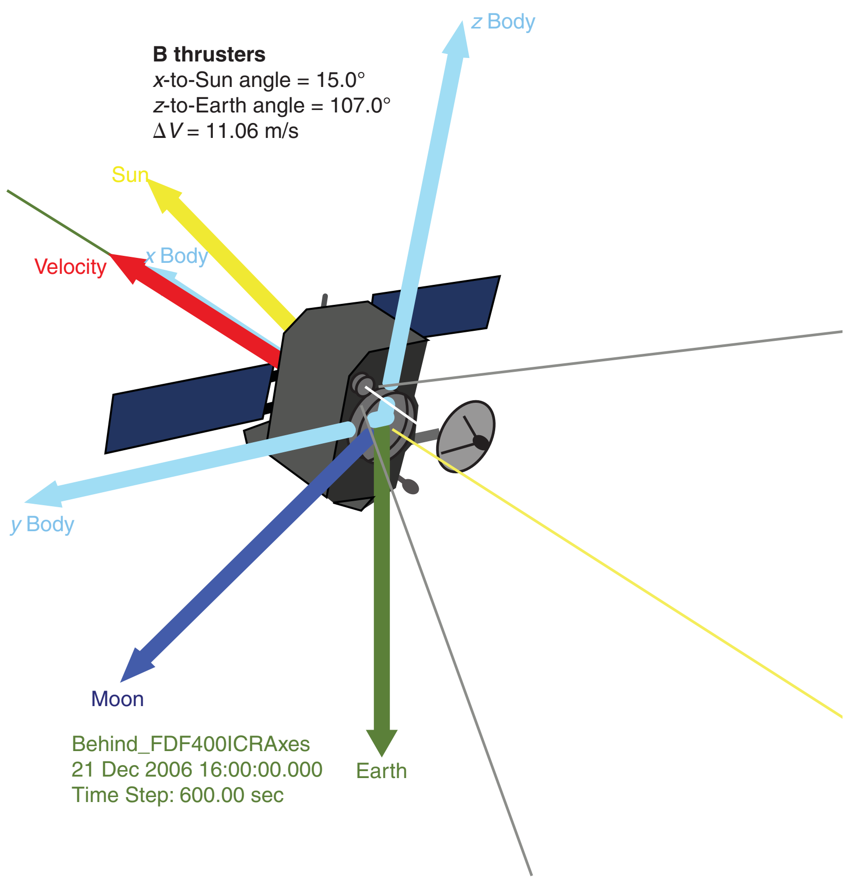

Figure 24. S1+ ΔV geometry for Behind, 21 December 2006.

the eclipses that occurred during the first three perigees are given in Table 7.

Details of the ΔV maneuvers are given in Tables 8 and 9. They give quantities described above, as well as the numbers of the pre- and post-ΔV trajectories, provided by the Flight Dynamics Facility (FDF)[6] of the Goddard Space Flight Center, that were used[10] for the maneuver reconstructions given in Table 10.

The "actual" ΔV magnitudes, J2000 Earth equatorial right ascensions, and declinations (in the third, fourth, and fifth columns, respectively, of Table 10) are calculated from a 6-degree-of-freedom adjustment of the initial position and the ΔV, by using its observed duration, to match the post-maneuver state vector. The errors in magnitude (in meters per second and in percent) and pointing are formed by subtracting the actual values from the planned values given in the maneuver plan files, which were used by the guidance and control and the operations teams.

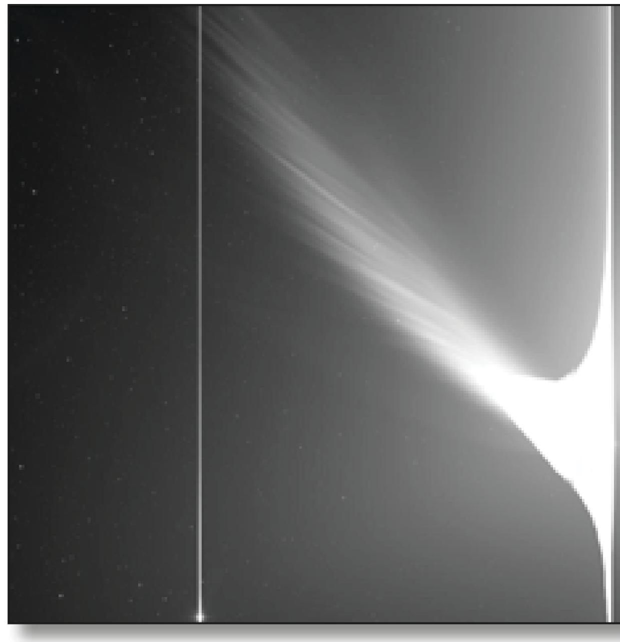

Figure 25. Comet McNaught imaged by Behind, January 2007.

STEREO'S FUTURE

As detailed above, all of the STEREO maneuvers were performed extremely accurately. This performance combined with STEREO's accurate launch have left the spacecraft with a generous supply of fuel, with an ~60-m/s ΔV capacity that could be used for future spacecraft operations. Some scientists would like to stop the STEREO spacecraft's drift near the L4 and/or L5 libration points of the Sun–Earth system, but to do that directly would require 600 m/s, about 10 times the remaining capacity. The additional maneuvers for Behind ended up targeting its drift rate to –22.000°/year, exactly the desired value. Last December, the mission design team suggested modifying the maneuvers to achieve a –22.5°/year drift rate, which would add up to 360° and an Earth return in 2023, 16 years after launch. But with the current drift rates, Ahead's closest approach to Earth in 2023 will be 8.2 million km on 20 August, whereas Behind's will be 10.0 million km on 14 July. Using approximately half of the remaining propellant

Figure 26. Lunar transit as observed by SECCHI on Behind, 25 February 2007.

Table 6. Phasing orbit key events.

		Ahead		Behind	
Event	Date	UTC	*h* (km)	UTC	*h* (km)
A1	31 Oct 2006	1639	411,554	1557	410,146
P1	06 Nov 2006	0909	504	0744	504
A2	11 Nov 2006	2346	408,870	2145	407,578
P2	17 Nov 2006	1343	1,898	1118	1,533
A3	23 Nov 2006	1624	432,551	1452	434,610
P3	29 Nov 2006	1958	4,425	1921	4,244
A4	06 Dec 2006	0208	435,833	0229	437,965
P4	12 Dec 2006	0831	6,668	0955	6,666
S1[a]	15 Dec 2006	2128	7,358	2103	11,776
A5	02 Jan 2007			0602	867,843
S2[a]	21 Jan 2007			0904	8,820

[a]Lunar swingby values in the *h* column are actually radii from the Moon's center.

Table 7. Phasing orbit eclipses.

Event, S/C	Date	Start				End			
		UTC	*h* (km)	λ (°)	ϕ (°)	UTC	*h* (km)	λ (°)	ϕ (°)
P1, STA	31 Oct 2006	0859	1,665	+166	+27	0920	2,208	−108	−15
P1, STB		0735	1,662	−173	+27	0756	2,210	−87	−15
P2, STA	17 Nov 2006	1336	2,506	+103	+28	1357	3,506	+165	−17
P2, STB		1111	2,180	+137	+26	1132	3,274	−155	−16
P3, STA	29 Nov 2006	1956	4,448	+21	+14	2015	5,944	+57	−9
P3, STB		1919	4,270	+30	+14	1939	6,015	+71	−8
P4, STA[a]	12 Dec 2006	0831	6,668	−175	+11				
P4, STB[a]		0955	6,666	+164	+11				

S/C, spacecraft; STA, STEREO A (Ahead); STB, STEREO B (Behind); λ, longitude positive east of Greenwich; ϕ, geodetic latitude, positive to the north.

[a]There was no eclipse at P4, so the information for perigee is given in the Start columns.

could change the current drift rates to 22.5°/year. A possible trajectory with that drift rate is shown in Fig. 27, including an S3 lunar swingby that could put Behind in an L2 halo orbit. With Earth and/or lunar swingbys in 2023, a significant change in the STEREO drift rate would be possible, if scientists of that time might want to do that rather than continue the mission with their current, interesting trajectories.

CONCLUSIONS

STEREO was the first mission to use lunar swingbys to place two spacecraft launched on one rocket into very different (and in this case, oppositely directed) heliocentric orbits. The STEREO mission design and Boeing launch teams computed nine detailed (14- to 16-day) windows. This paper described the methodology employed to compute the launch windows that were prepared. At the end, a window with a very short coast between the Delta II second-stage burns was employed. Furthermore, to limit the number of tracking assets needed during the launch ascent, the coast time was fixed for the entire window, and only one daily opportunity (15 min) was attempted. This strategy worked, and after almost a year of trajectory computations, STEREO was successfully launched on 26 October at 0052:00.339 UTC. The STEREO Earth orbit activities were conducted without any serious anomalies or incidents and the prelaunch plan for the Earth orbit activities was followed almost exactly as it was written. The experience, although intense, was a joy for the whole STEREO team. As detailed above, all the maneuvers were performed extremely accurately. This fact combined with an accurate launch left each STEREO spacecraft with a generous supply of fuel (at the end of the mission design activities), with ~60-m/s Δ*V* capacity remaining.

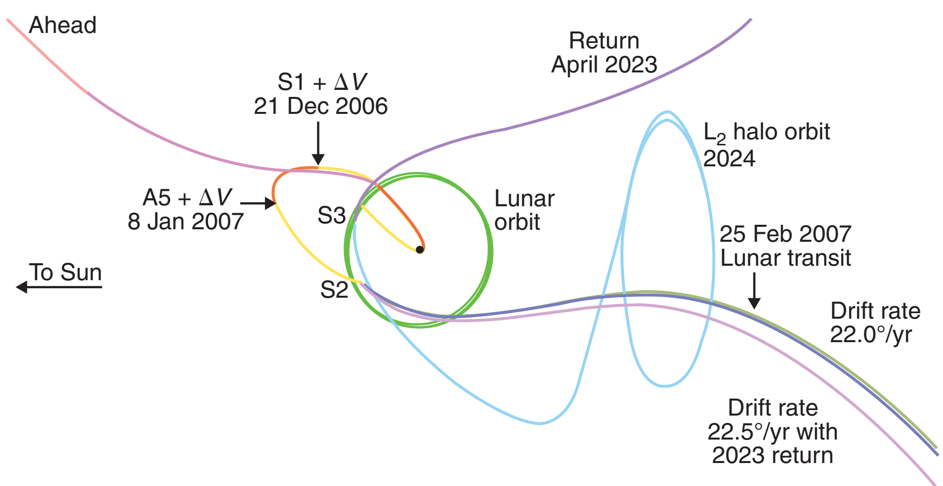

Figure 27. Behind's possible return trajectory in 2023.

Table 8. Parameters of ΔV maneuvers for Ahead.

ΔV Name	Date, UTC	Start, UTC	ΔV Magnitude (m/s)	Thruster Group	x-to-Sun Angle (°)	z-to-Earth Angle (°)	LGA	Pre-ΔV FDF Orbit	Post-ΔV FDF Orbit
E1	28 Oct 2006	1330	0.196	B	20.0	119.3	$-z$	STA476	STA553
A1	30 Oct 2006	1800	11.709	B	15.3	110.0	$-z$	STA553	STA582
E2	2 Nov 2006	1800	0.202	C	33.4	118.4	$-z$	STA582	STA002
A2+	14 Nov 2006	1400	46.054	B	34.8	111.3	$-z$	STA096	STA142
P2	17 Nov 2006	1256	2.735	C	32.5	132.0	$-z$	STA142	STA218

Table 9. Parameters of ΔV maneuvers for Behind.

ΔV Name	Date, UTC	Start, UTC	ΔV Magnitude (m/s)	Thruster Group	x-to-Sun Angle (°)	z-to-Earth Angle (°)	LGA	Pre-ΔV FDF Orbit	Post-ΔV FDF Orbit
E1	28 Oct 2006	1630	0.199	B	20.0	119.3	$-z$	STB474	STB553
A1	30 Oct 2006	2100	11.851	B	15.5	110.0	$-z$	STB553	STB586
E2	2 Nov 2006	2100	0.206	C	23.2	110.4	$-z$	STB586	STB609
A2+	14 Nov 2006	1600	28.422	B	35.2	111.5	$-z$	STB102	STB136
P2	17 Nov 2006	1000	4.950	C	24.0	123.2	$-z$	STB136	STB225
A4	6 Dec 2006	2000	0.205	B	0.4	61.3	$+z$	STB262	STB313
S1+	21 Dec 2006	1600	11.071	B	15.0	107.0	$-z$	STB427	STB488
A5+	8 Jan 2007	1900	0.791	B	27.7	107.6	$-z$	STB544	STB599

Table 10. ΔV maneuver reconstructions.

Spacecraft	Maneuver	ΔV Mag. (m/s)	ΔV RA (°)	ΔV Declination (°)	Error in ΔV Mag. (mm/s)	% Error in ΔV Mag.	Pointing Error (°)
Ahead	E1	0.196	−36.67	9.23	−3.85	−1.924	2.56
	A1	11.709	−46.35	29.04	−8.04	−0.069	0.97
	E2	0.201	8.04	2.09	0.90	0.448	1.69
	A2+	46.054	85.27	69.30	−3.66	−0.008	0.36
	P2	2.735	63.24	−11.77	2.69	0.099	0.13
Behind	E1	0.199	−36.56	6.84	−1.04	−0.520	3.17
	A1	11.650	−45.93	28.98	−9.49	−0.081	0.89
	E2	0.202	14.25	14.39	2.37	1.184	0.68
	A2+	28.422	84.93	68.65	0.99	0.003	0.17
	P2	4.950	56.20	−4.78	7.61	0.154	0.09
	A4	0.205	153.87	−25.04	4.16	2.068	2.15
	S1+	11.071	61.77	−47.05	5.78	0.052	0.15
	A5+	0.791	179.41	27.17	4.22	0.537	0.64

Mag., magnitude; RA, right ascension.

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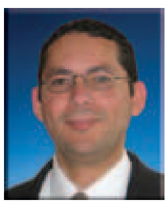
José J. Guzmán


Peter J. Sharer