GENESIS TRAJECTORY AND MANEUVER DESIGN STRATEGIES DURING EARLY FLIGHT

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Abstract

As the fifth Discovery mission, the Genesis spacecraft was launched on August 8, 2001 with a science objective to collect solar wind samples for a period of approximately two and a half years while in orbit in the vicinity of the Sun-Earth L_1 libration point. These samples will eventually be delivered back to the Earth for analysis, posing a formidable challenge in terms of both mission design and navigation. This paper discusses trajectory and maneuver design strategies employed during the early phases of flight to accommodate spacecraft and instrument design constraints, while achieving the science objectives of the mission. Topics to be discussed include: mission overview, spacecraft design and constraints, maneuver analyses and trajectory re-optimization studies, and operational flight experience to date.

INTRODUCTION

Mission Overview

Genesis is the fifth mission selected by NASA under its low cost Discovery program. The primary goal of the Genesis mission is to collect solar wind particles over a 2.5-year period and return them safely back to Earth. As such, Genesis will be the first mission to return extra-terrestrial samples since the Apollo missions nearly 30 years ago. After years of development, Genesis was successfully launched from Kennedy Space Center on August 8, 2001. The trajectory of the spacecraft takes it out to the vicinity of the Sun-Earth L_1 libration point (along the line between the Sun and the Earth). After insertion into a large amplitude Lissajous (or halo orbit) about the L_1 point, the collection arrays were deployed to begin sampling the solar wind. Genesis will collect samples for five revolutions, or approximately 2.5 years, about the libration point. After completing its time near L_1, Genesis follows a free return trajectory home via a looping return about the Sun-Earth L_2 point to set up a daylight entry and mid-air capture over the western United States in September 2004. After a safe return to Earth, the samples will be curated at Johnson Space Center and made available for study to scientists throughout the world. This type of sample return has never been attempted before and presents a challenge to both mission design and navigation that will be discussed in this paper.

The Genesis mission is lead by Principal Investigator Dr. Donald Burnett of the California Institute of Technology. The Genesis team consists of members from the Jet Propulsion Laboratory, Lockheed Martin Astronautics, Los Alamos National Laboratory, and the Johnson Space Center. Project management resides at JPL along with mission planning, navigation, and sequencing. The spacecraft and operations teams are headed by Lockheed Martin, and the science team primarily resides at Los Alamos. The handling and curation of the returned samples will be lead by the Johnson Space Center.

Science Overview

The science objective for the Genesis mission is to precisely determine the elemental and isotopic composition of the solar wind. The solar wind is thought to be compositionally identical to the Sun's photosphere. Furthermore, it is believed that the photosphere is representative of the solar nebula from which the solar system was formed. By studying the solar wind then, scientists can, in essence, study the very material that formed the sun and all the planets, moons, asteroids, and comets.

The solar wind is not constant in speed or composition. It is affected by activity on the surface of the Sun, such as coronal mass ejections. The Genesis spacecraft has electron and ion monitors to help classify the nature of the solar wind at any given instant. Based up this classification, the spacecraft deploys various collector arrays that have been specifically designed for that particular solar wind regime. Genesis also has an electrostatic concentrator instrument that is tuned to optimize collection of O, N, and C ions, while rejecting most of the H ions from the solar wind. A portion of the samples collected will be made available to scientists immediately, while the rest is held in reserve for future, perhaps as yet undeveloped, analyses.

Because the science objective is to collect pristine solar wind samples, it is critical that the collection be performed away from any interaction with the Earth's electromagnetic environment. This requirement drove the selection of a Lissajous or halo orbit about the Sun-Earth L_1 point. Trajectories in this region remain in front of the bowshock interaction of the solar wind with the Earth's magnetic field and suffer no eclipsing issues.

Spacecraft Overview

The Genesis spacecraft is depicted in Figure 1. The spacecraft consists of basically two parts. The first part is the bus that comprises the solar panels, propulsion system, attitude control system, avionics system, communications systems, as well as the electron and ion monitors. The spacecraft is spin-stabilized with a nominal spin rate of about 2.6 RPM. Due to the need to minimize sample contamination on the top of the spacecraft, all of the thrusters are on the aft side (as shown in the right portion of Figure 1). This configuration produces unbalanced thrusting that must be accounted for in terms of both attitude control and propulsive maneuver design. The implications to trajectory and maneuver design are discussed in detail later in the paper.

The second part of the spacecraft is the Sample Return Capsule or SRC. The SRC is the portion of the spacecraft that is designed to re-enter the atmosphere and separates from the spacecraft bus prior to re-entry. The SRC consists of a hinged capsule (as shown in Figure 2) that contains the sample collection canister, including the collection arrays and the electro-static concentrator. The SRC backshell is covered with an ablative material to protect the capsule during re-entry, while the foreshell contains a parafoil to help slow down the capsule during the atmospheric portion of its descent. After the SRC has reached its terminal descent velocity, it will be captured in mid-air by a helicopter and then taken safely to the processing facility.

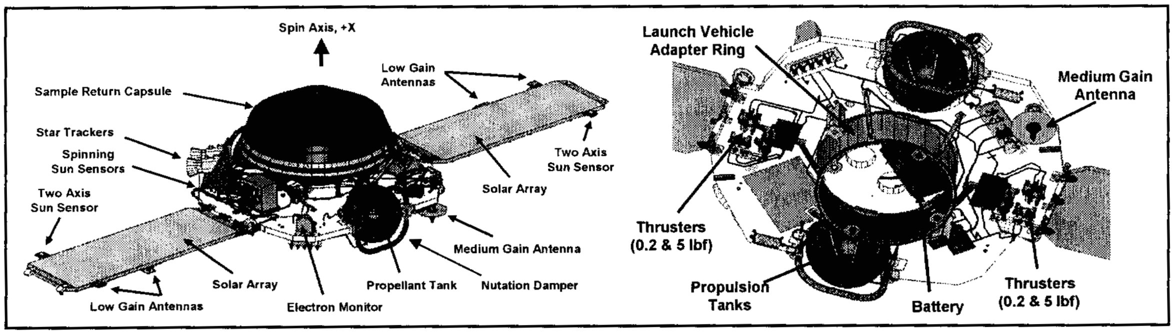


Figure 1. Fore and Aft Views of the Genesis Spacecraft

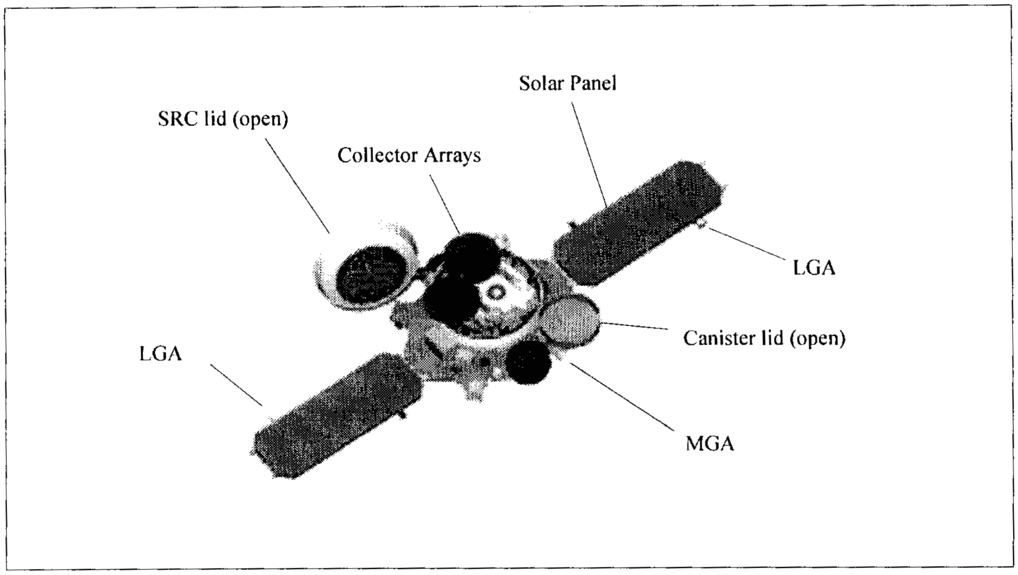


Figure 2. Genesis Spacecraft in Science Collection Configuration

Trajectory Overview

The Genesis trajectory was designed using Dynamical Systems Theory coupled with a two-level differential corrections process. This combination of techniques allowed all of the various top-level mission constraints to be satisfied, such as: no interference from the Earth's electro-magnetic field, at least 23 months of science collection, and daylight entry over Utah for mid-air capture. These top-level constraints have driven the design and re-design of the trajectory. Earlier papers provide details on procedures employed to determine suitable trajectory solutions (see references).

The Genesis trajectory for the July/August 2001 launch opportunity is shown in Figure 3. This view of the trajectory is in a frame that rotates with the Earth about the Sun. Thus, the Sun is always to the left along the –X axis.

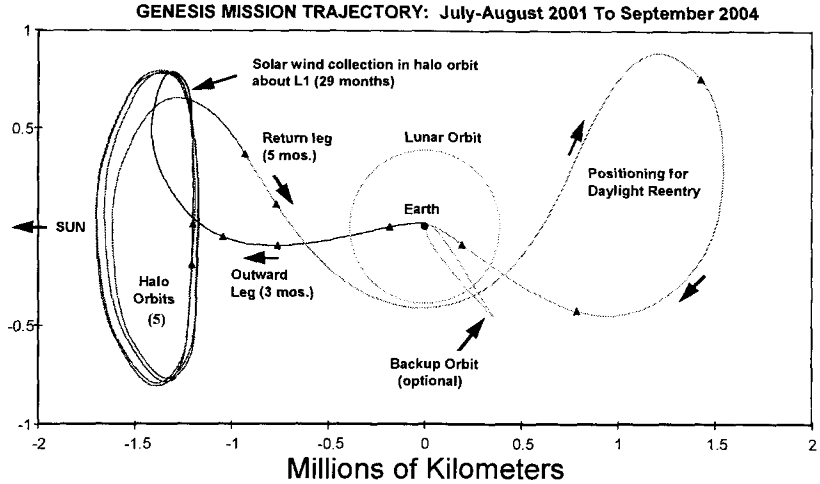


Figure 3. Genesis Trajectory in Sun-Earth Rotating Frame

The trajectory launches in the sunward direction towards the Sun-Earth L_1 point and injects onto the Lissajous or halo orbit near the first downward crossing of the Sun-Earth line. This maneuver is called the Lissajous Injection Maneuver or LOI. Genesis successfully performed its LOI maneuver on November 16, 2001 and is currently about half way through its planned stay near L_1. After completing five revolutions in the Lissajous orbit, the natural dynamics of the motion cause the trajectory to depart L_1, pass close to the lunar orbit, and make a loop around L_2. (Note that the Moon actually only plays a small role in this return trajectory; the closest approach is over 300,000 km away from the Moon. The primary dynamics at work is the Sun-Earth-Spacecraft three body problem.) The L_2 loop positions the spacecraft for a daylight entry over Utah from the northwest. It is of note that the Earth's terminator roughly coincides with the X=0 line in the plot. Trajectories returning from L_2 cross the terminator into daylight as they re-enter, while trajectories returning from L_1 cross into darkness. This fact necessitated the return from L_2 to allow the helicopter pilots to be able to adequately see the SRC for retrieval.

EARLY TRAJECTORY CORRECTIONS

Design

The launch period for Genesis opened on July 30, 2001 and extended through August 14. In all cases, each launch date assumed a direct assent trajectory leading to the same LOI point on November 16, 2001. The direct assent, as well as collision avoidance requirements, limited the length of each launch opportunity to no more than 2 minutes on each launch date. For convenience, a common trajectory could be assumed for 2-3 day periods, known as launch blocks, centered on July 30, August 2, August 5, August 8, August 11 and August 14, respectively. Injection into the transfer orbit was achieved with a Delta II 7326 launch vehicle with a Star 37 third stage.

Up to five trajectory correction maneuver (TCM) opportunities were scheduled between launch and LOI to correct transfer orbit injection errors and set up the proper conditions for performing the LOI maneuver. These maneuvers were designed in accordance with an early TCM strategy, summarized in Table 1, which attempted to account for limitations in spacecraft capabilities and to minimize operational complexity in the critical period immediately following launch.

Maneuver	Primary Location	Backup Location(s)	Conditions/Explanations
All Transfer	See below	See below	Maneuver strategy expanded to accommodate ACS delta-design and ground s/w development. All transfer maneuvers targeted to LOI location near L1 (minor targeting variations possible).
TCM-0 (Contingency)	L+24h	Same as TCM-1 primary/ backup(s)	Emergency TCM used in place of TCM-1 only if maximum maneuver size allowed by spacecraft power and thermal constraints exceeded at L+48h (> 110 m/s anti-sunward or > 130 m/s sunward design same as TCM-1; highly unlikely contingency.
TCM-1	L+48h	L+3d, L+4d, L+5d, L+6d... (see below)	Performed only if required maneuver at L+48h > 5 m/s but <110 m/s if anti-sunward or <130 m/s if sunward (option to not perform if as high as 30 m/s, based on operational assessment of cost of TCM-3 without earlier maneuvers); fixed aimpoints in plane containing Sun and injection (TIP) attitude near Ecliptic Plane. Two-maneuver optimization with TCM-3 assumed. Maneuver on spinning sun sensors (SSS) only, assuming injection attitude near Earth-based velocity vector (Boeing's PMA and DTO data provide the actual injection attitude); must observe ACS-defined keepout zones per flight rule 0001-A-ACS. Use one of two fixed aimpoints in either sunward or anti-sunward direction; only three inertial aimpoints sufficient to cover nominal blocks. Daily backup opportunities (if delayed before execution); minimum 48 hour delay if execution aborted to reestablish OD and verify s/c health and safety; switch to TCM-2 on L+7d if TCM-1 was required but was unable to be executed.
TCM-2 (Backup Only)	N/A	L+7d and L+18d	Highly unlikely maneuver (e.g., if TCM-1 required but not executed by L+7d or extremely large injection error could not be corrected sufficiently by TCM-0/1). In the very unlikely event that TCM-2 is needed, consider the following: • Two-maneuver optimization with TCM-3. • Employ best possible design (i.e., greatest flexibility in burn direction that can be afforded) based on SSS-only attitude control with appropriate observation of KOZs. • Backup opportunity at L+18d avoid periods of low target adjustment capability. • Only need to use OD-supported determination of initial attitude when clock angle knowledge has been lost or severely degraded by previous attitude maneuvers.
TCM-3	L+35d	L+49d	Performed only if > 1.25 m/s to avoid use of smaller thruster off-sun maneuver modes (relief to maneuver decomposition software development schedule pre-launch. Spin Track must be operational by this point (otherwise, drastic mission re-design anticipated). Prime location at L+35d and backup opportunity at L+49d avoid OD quiet period; if backup used, may require replanning of small forces calibration and science start activities.
TCM-4 (Contingency)	L+65d	L+72d	Contingency maneuver (no optimization), highly unlikely given transfer phase minimum maneuver size constraint. Retain opportunity as backup to TCM-3. Prime location at L+65d to afford better target adjustment capability. Backup opportunity at L+72d; prior to start of science (opening of canister, deployment of arrays, etc.) to avoid possible additional cycle on mechanisms.

Table 1. Summary of Early TCM Strategy

During the first few days after launch, the attitude control subsystem (ACS) was effectively limited only to sun sensors to support spacecraft attitude determination. Star trackers, required to support the spin track mode with full three-axis attitude determination, could not be relied upon to support the ACS until a calibration was performed a few days after launch. For a majority of interplanetary missions, this would not be a major issue, since the first TCM is typically performed weeks after launch, allowing plenty of time for checkout and calibration of various spacecraft systems. However, as shown in Figure 4, the assumed injection covariance was large enough for Genesis to necessitate a potentially large post-launch TCM. The data shown in the figure are statistical delta-v magnitudes, including mean and 95% probable maximum delta-v, based on monte-carlo simulation results. These magnitudes grow rapidly as time past injection increases. Such potentially exorbitant delta-v costs precluded delaying the first TCM more than a few days after launch.

On the other hand, the TCM could not be planned until tracking data from two, or preferably three, ground stations had been collected and processed to determine the actual spacecraft orbit resulting from launch injection errors. Also, it was considered prudent to allow sufficient time to deal with a variety of potential operational difficulties that could readily arise during this period. Consequently, the first TCM (designated TCM-1) was scheduled to be performed nominally at 48 hours after the post-injection target interface point (TIP), which occurred about 40 minutes after launch. As a contingency, if launch errors were extremely large, owing to about a 1% probability of early shutdown of the second stage of the launch vehicle, then obtaining highly accurate orbit determination information was less critical than performing some sort of trajectory correction as quickly as possible. In this case, the first TCM would be executed at 24 hours past TIP. This contingency TCM (designated TCM-0) would replace TCM-1 in such an event.

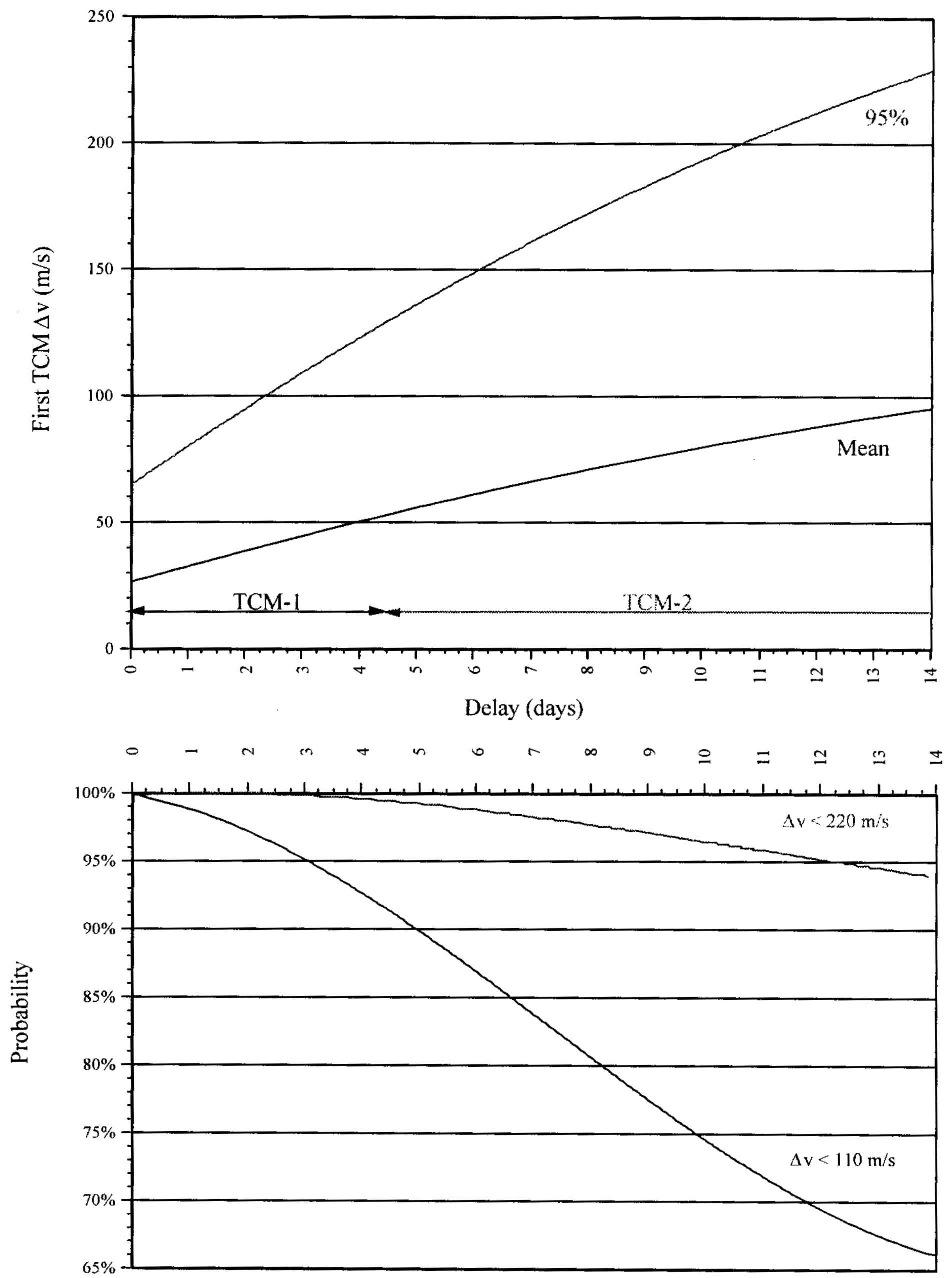


Figure 4. Potential ΔV Costs for First TCM

Due to the limited ACS capability, the implementation of TCM-0 or 1 had the potential to be quite complex with considerable ground-spacecraft interaction. Fortunately, this first TCM was needed mainly to correct the launch energy or C_3, so the required maneuver orientation would be ideally in the direction of, or opposite to, the spacecraft velocity. Also, because Genesis was injected from the launch vehicle in a spinning state, an injection attitude in the direction of the velocity prior to injection could be assumed. So if burn directions were selected to lie in a plane which included the injection attitude and the Sun, the spacecraft could be oriented in a pre-planned, fixed pointing direction merely by dead-reckoning with sun sensors. Two burn directions particular to each launch block were chosen, one sunward to compensate for an injection underburn, the other antisunward in the event of an injection overburn.

The sun sensors, primarily the spinning sun sensors (SSS) operating at post-injection spin rates of 9.5 RPM and higher, impose constraints on spacecraft attitude relative to sunward and anti-sunward directions. These constraints, known as keep-out zones (KOZ) are designed to prevent a potentially mission-fatal situation where the ACS cannot determine spin rate due to false Sun crossing indications in the presence of nutation and allowing for the possibility of single thruster failures. Consequently, the fixed aimpoints selected for various launch blocks were always on the edge or beyond the KOZ's, albeit as close to the ideal velocity or anti-velocity direction as allowed operationally. For operational simplicity, the number of inertial orientations for the entire launch period could be reduced to just three directions. These were specified in separate Maneuver Profile Files (MPF's), one sunward and two anti-sunward (the second providing an update for the first starting with the August 8 launch block). Each MPF was processed into a specific maneuver sequence before launch. After launch, the Navigation Team needed only to determine which of three cases (sunward or antisunward TCM-1 or sunward TCM-0) would apply and provide a simple burn magnitude update in accordance with the estimated injection C_3 error. An illustration of the guideline used in this process for the August 8 launch block is indicated in Figure 5.

Following the aforementioned launch energy correction, a TCM at 35 days after launch (designated TCM-3) was scheduled to correct pointing errors arising from injection and earlier TCM's. Further TCM's were included for contingency purposes only, at 7 days after launch (TCM-2) and 65 days after launch (TCM-4). These TCM's would only be needed in the event of severe spacecraft anomalies associated with abort or delay of TCM-0/1 and/or TCM-3.

Execution

Because of bad weather on July 30 and hardware concerns, the actual launch did not occur until August 8. In reality, the injection provided by the Delta Star 37 third stage was so accurate that only a small anti-sunward burn of about 5 m/s was needed to correct a slight injection overburn. This was achieved with an overall magnitude of about 8 m/s, including turns and spin changes, with an error of about +4% (overburn).

During the period leading up to the scheduled TCM-3 on September 12, there was an indication of excessive temperature of the batteries needed for SRC recovery at the end the mission. This was most probably due to contamination of adjacent surfaces internal to the SRC itself, which had the potential to greatly increase the temperature of these batteries over the course of the mission and compromise the recovery of the SRC. This anomaly necessitated that scheduled activities be delayed or postponed to allow for remedial actions to be carried out. Fortunately, it was determined that the direction of the LOI maneuver could be improved, from the standpoint of Earth visibility, by canceling TCM-3 and other transfer TCM's altogether. This led to some re-optimization of the post-LOI trajectory as well. These considerations are explained in more detail in the following section on LOI re-design.

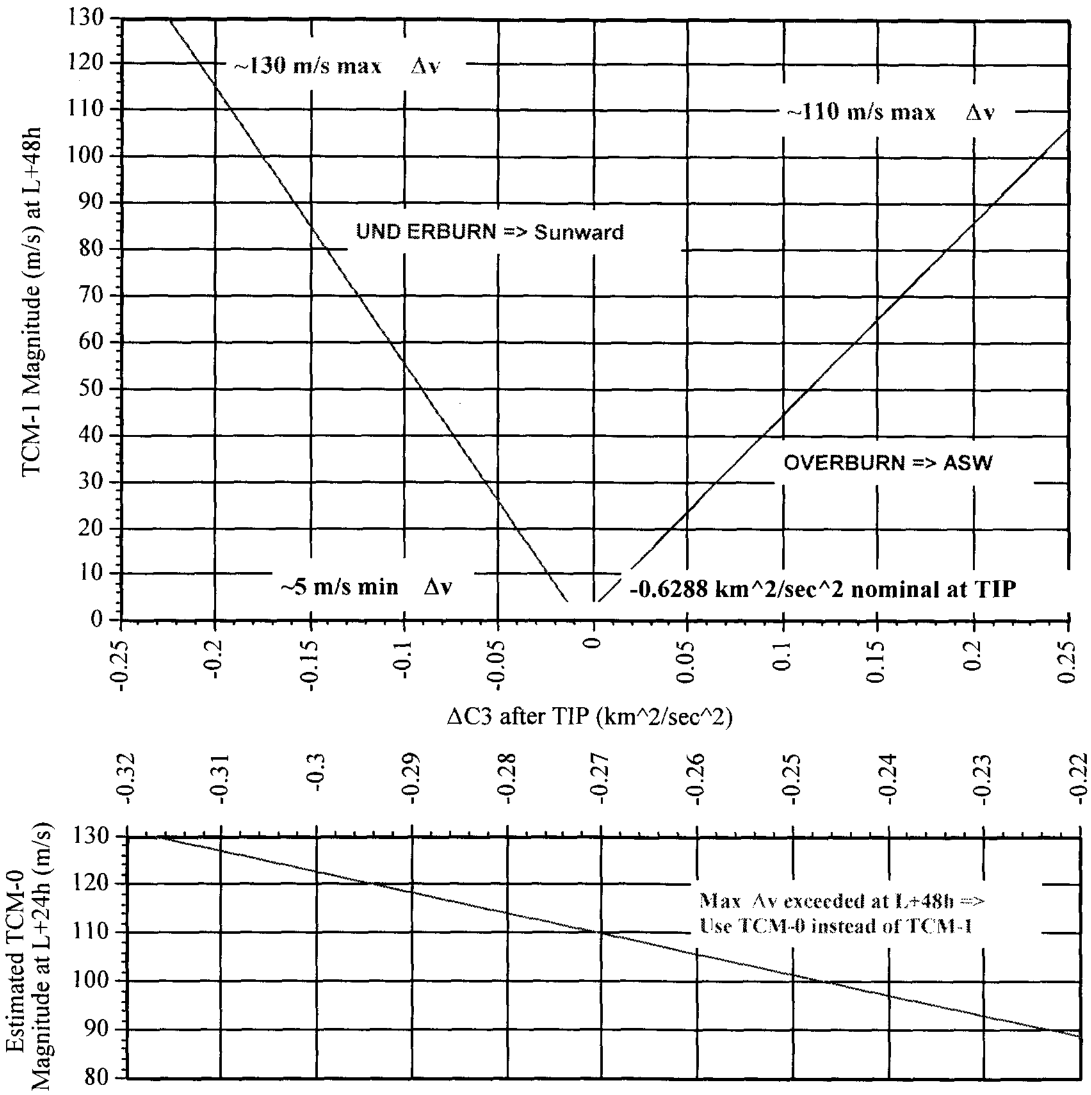


Figure 5. Determination of TCM-0/1 Delta-V Magnitude based on C_3 from Post-Injection Orbit Determination

LOI MANEUVER

Re-Design

After reconstructing TCM-1, as discussed above, the TCM-3 maneuver size and direction relative to the Sun were estimated based on a monte-carlo simulation with 5000 samples, as shown in Figure 6. The data shown assume execution of a delayed TCM-3 on September 18 or about 41 days after launch. However, the contamination issue discussed earlier complicated potential execution of TCM-3. To deal with this problem, it was decided that the SRC should be partially closed to avoid further exposure to the Sun that might result in further annealing of contaminant particles to SRC interior surfaces. This partial closure would also permit contaminants to be baked off and vented away using all available spacecraft heaters for a short enough period to minimize risk to the SRC batteries. Differences in mass properties among SRC backshell configurations produced different maneuver decompositions required to achieve the total desired delta-v in the presence of unbalanced thrusters. With the SRC backshell open, this decomposition would yield a single turn-burn-turn sequence with the burn direction outside the KOZ's prescribed for the sun sensors. However, now that the SRC backshell would be closed, the directions indicated in Figure 6 would produce a violation of the antisunward KOZ if performed as a single burn. To avoid such a constraint violation would require execution of TCM-3 as a double or "dogleg" maneuver in two parts on consecutive days. Such an implementation would entail considerable operational complexity and could interfere with efforts to resolve the contamination problem.

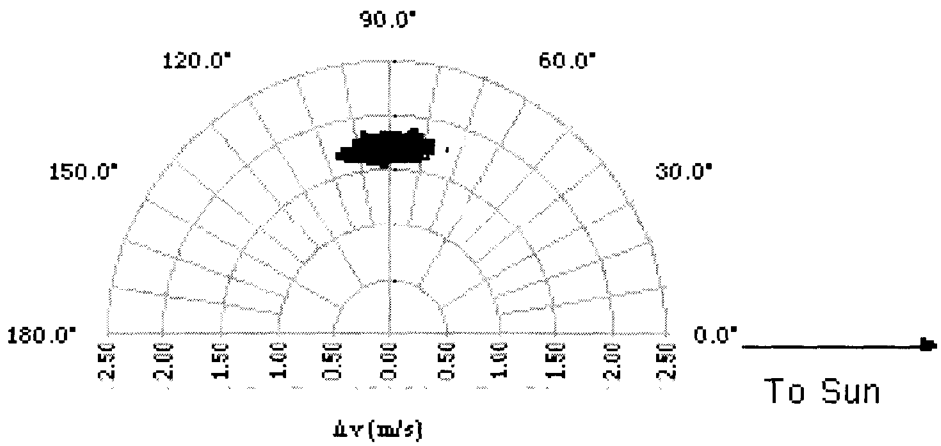


Figure 6. TCM-3 Magnitude and Direction per TCM-1 Reconstruction

Happily, an alternative arose which provided additional operational benefits for LOI execution. From pre-launch analysis, the LOI had a large prospect of being executed at ~90 deg from the Earth-Sun line. In this attitude, the burn would be completely out of view of the Earth since the orientation of the spacecraft would be such that both LGA coverage patterns would be blocked effectively by the solar panels. However, if TCM-3 were canceled, the required LOI maneuver would be affected as shown in Figure 7. Although the magnitude of the maneuver is not affected significantly, the view angle from Earth now appeared to be most likely in the range of 65-80 deg such that the LOI burn would have a better chance of being visible in real time. Due to all of these factors, it was decided to cancel TCM-3.

In light of operational concerns, an additional sensitivity analysis was performed to determine the impact delaying the LOI maneuver without TCM-3. As indicated in Figure 8, the direction of LOI remains favorable, but the magnitude would grow considerably from around 25 to 43 m/s over the course of four weeks. More alarmingly, as indicated in Figure 9, the effect on the first LOI cleanup maneuver, designated SKM-1A, is quite significant. (SKM stands for Station Keeping Maneuver.) SKM-1A, scheduled nominally on December 12 or 26 days after LOI, could increase from less than 10 m/s to around 45 m/s, making it potentially larger than LOI itself.

To avoid these problems, re-optimization of the Genesis trajectory was needed. As shown in Figures 10 and 11, recomputing the trajectory limits the potential growth of delta-v costs as a consequence of LOI execution delay more than the baseline case. Such re-optimization also provided a better Earth view angle for LOI; in fact, the view angle actually improves with delays. The implications of this re-optimization for later SKM's will be discussed in a later section.

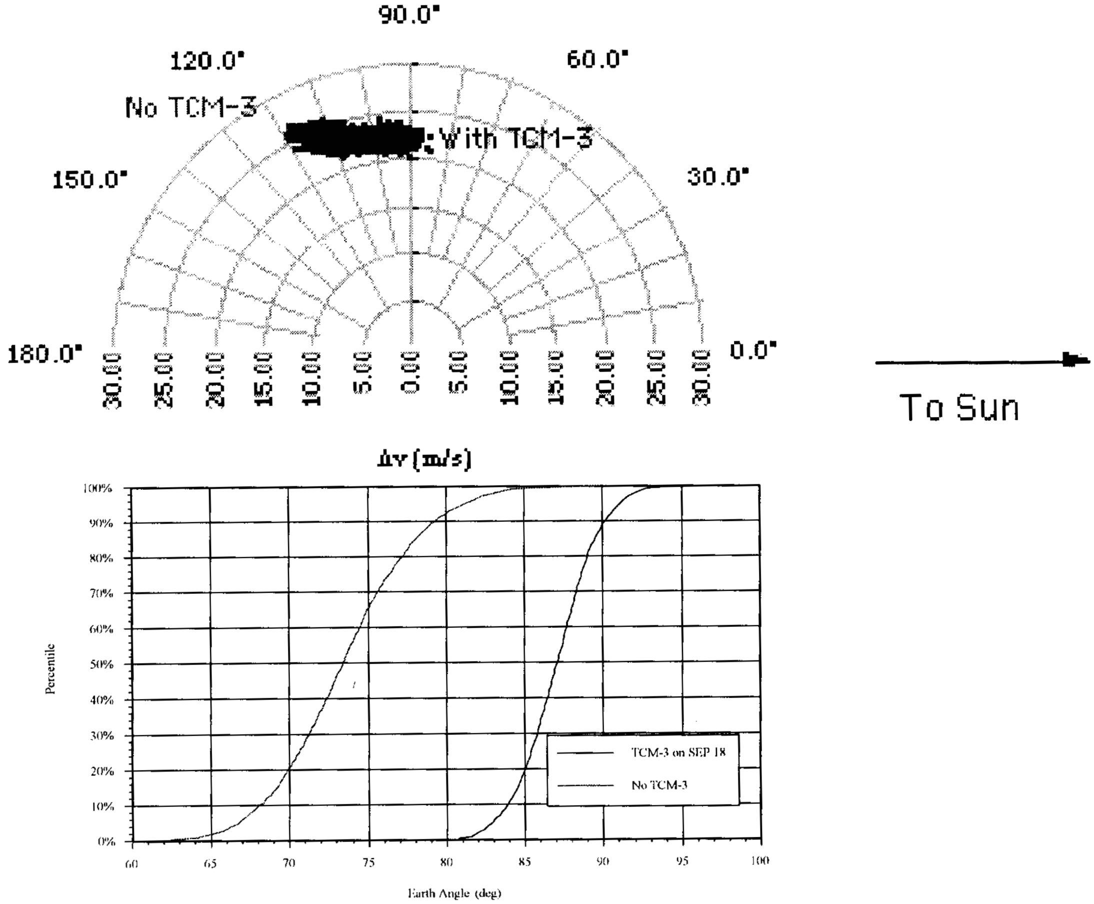


Figure 7. Impact of Excluding TCM-3 on LOI Magnitude and Direction

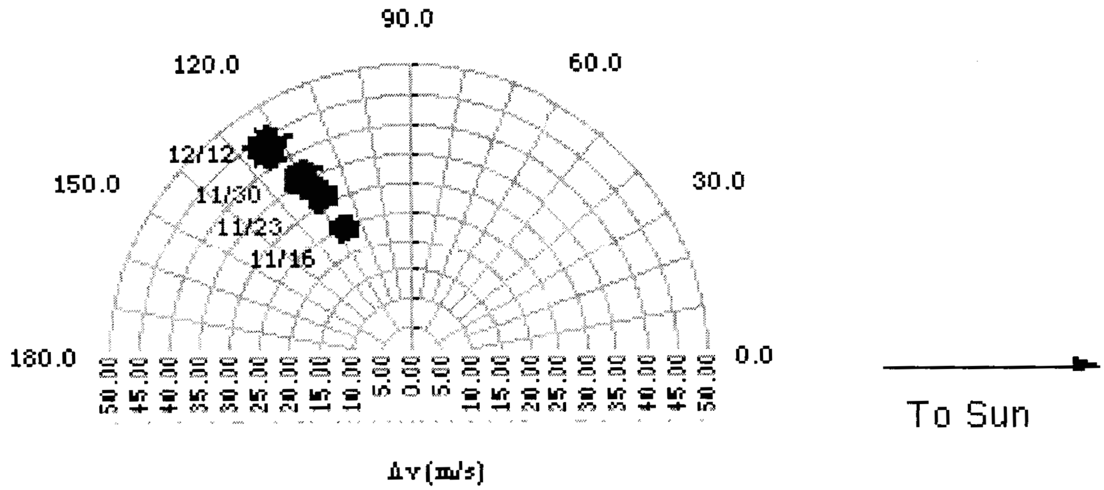


Figure 8. Sensitivity of LOI Magnitude and Direction to Delay

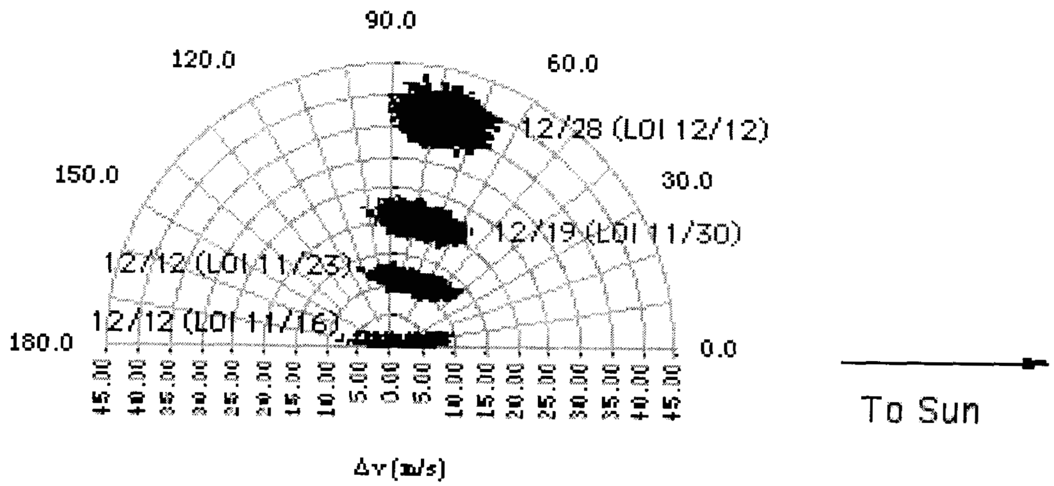


Figure 9. Sensitivity of SKM-1A Magnitude and Direction to Delays

Execution and Cleanup of LOI

LOI was executed as planned on November 16, 2001 with an overall magnitude of about 25 m/s and a –1.2% error (underburn) within 0.8 deg of the planned direction. After completion of this large maneuver, and with concerns about exacerbating the contamination problem alleviated, science collection could commence with opening of the SRC backshell. The resultant SKM-1A after LOI reconstruction was a mere 1.1 m/s at about 18 deg off Sun. This meant that SKM-1A would be small enough and in the right direction to allow for a single burn sequence. The primary benefit of this type of maneuver is that the SRC could be kept open with minimal interruption to solar wind collection and no risk of damaging the SRC backshell and concentrator with an additional close-open cycle. An assessment of SKM-1B, the first regular SKM, suggested that further re-biasing of SKM-1A would be beneficial for SKM-1B. Redirecting SKM-1A to about 25 deg off Sun would pull the direction of SKM-1B farther away from the Sun to maximize the prospects of a simple maneuver sequence. This minimizes the chances of a double maneuver driven by a sunward KOZ of 12.5 deg. It also avoids as much as possible going farther than 28 deg off Sun, which would degrade the accuracy of the maneuver by requiring intermediate turns to the burn attitude supported entirely via dead-reckoning on sun sensors. Uncertainties in SKM-1B magnitude and direction are shown in Figure 12, based on a post-LOI reconstruction with re-biasing of SKM-1A assumed. The boundaries shown reflect the effect on total delta-v of combining turns and spin adjustments with the burn itself.

SKM-1A was executed as scheduled on December 12 with a magnitude of only 1.115 m/s and a +2.30% error within about 0.5 deg of the planned direction. SKM-1B on January 16, 2002 ended up as a near-Sun single maneuver with a magnitude of 1.328 m/s with a 1.44% magnitude and 0.17 deg direction error.

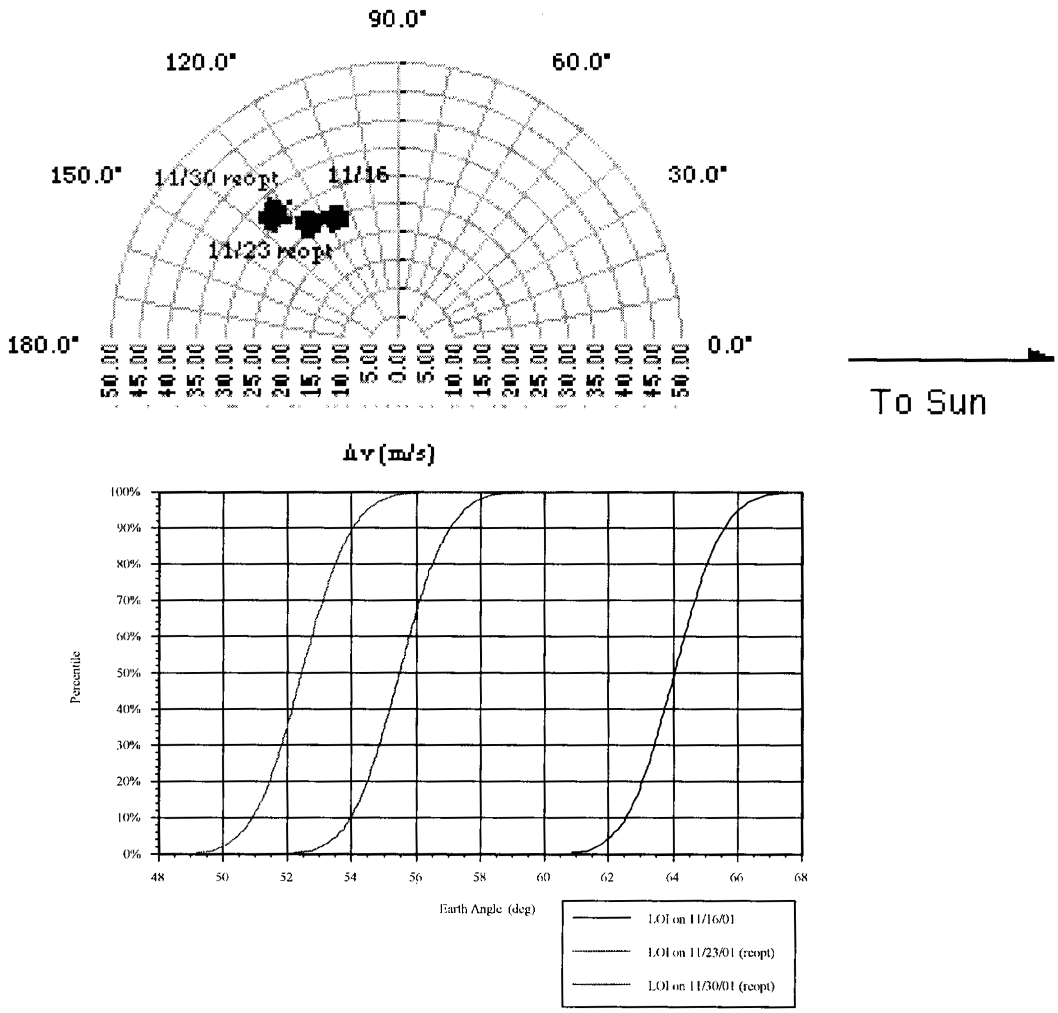


Figure 10. Implications of Trajectory Re-Optimization for Nominal and Delayed LOI

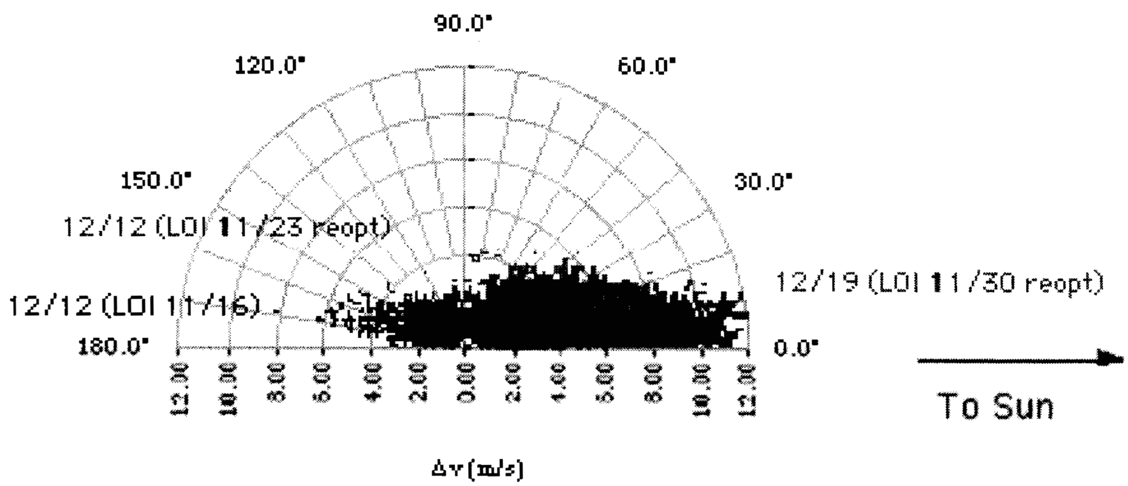


Figure 11. Implications of Trajectory Re-Optimization on SKM-1A for Nominal and Delayed LOI-SKM-1A Cases

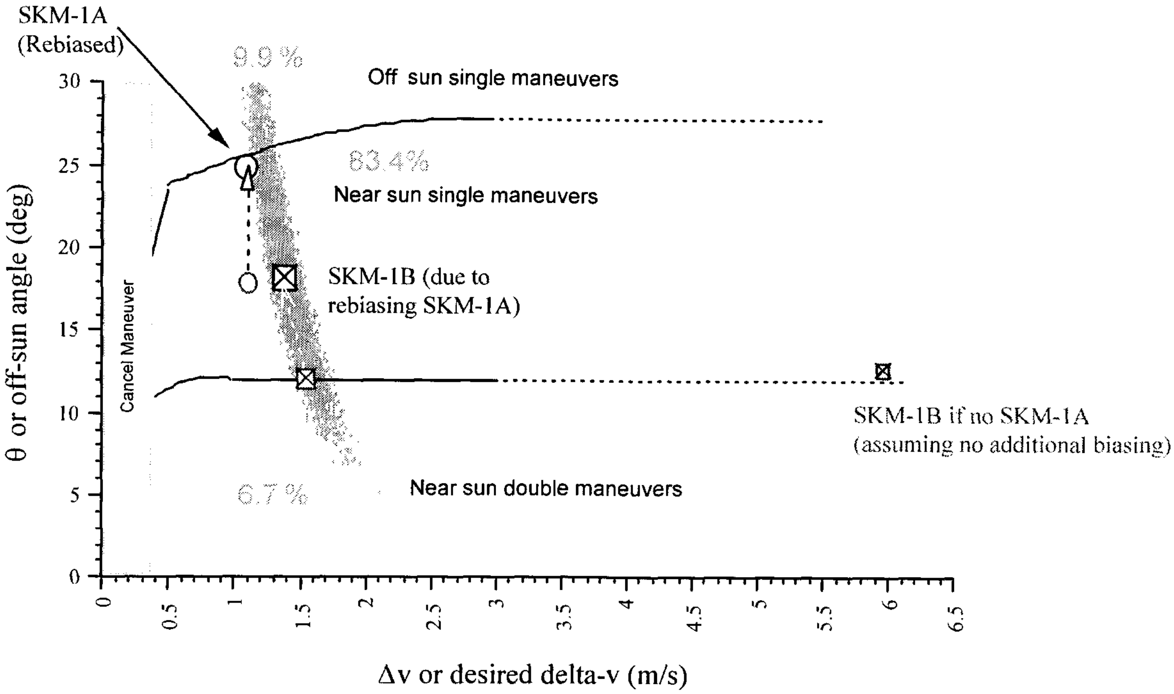


Figure 12. Re-Biasing Strategy for SKM-1A and 1B

STATION-KEEPING MANEUVERS

Initial Design

Due to modeling deficiencies inherent in any design and the sensitivity of trajectories in the three-body problem to small perturbations, it is necessary to periodically perform station-keeping maneuvers (SKM) to maintain the desired Lissajous trajectory. Although trajectories in the three-body problem are sensitive to perturbing forces, the time constant to instability is relatively long (on the order of weeks to months). Thus, statistically, the size of any given SKM is generally less than 0.5 m/s and is fairly constant over a few weeks period. Typical libration point missions perform SKM's once or twice per revolution (see references). This is acceptable since these missions typically do not need to be in a precise orbit; any orbit satisfying the mission constraints will suffice.

For Genesis, however, the Lissajous portion of the trajectory is critical to set up the "free-return" trajectory that brings the spacecraft and its samples back to Earth. In fact, the LOI maneuver at launch plus three months that placed the spacecraft into the Lissajous orbit is actually the maneuver that started Genesis on its path home. Pre-launch analyses performed at JPL and Purdue University suggested that, for a Genesis-type trajectory, SKM's should be performed every two months, or three times per revolution. Thus, the mission has planned 15 SKM's denoted by a number corresponding to the revolution and a letter A, B, or C. Hence, the order of the SKM's is 1A, 1B, 1C, 2A, 2B, 2C, etc. (Note that, as discussed, SKM-1A is actually the cleanup maneuver for LOI.) This station-keeping plan keeps the spacecraft on its intended path without allowing the magnitude of the maneuvers to grow beyond the typical 1 m/s or so.

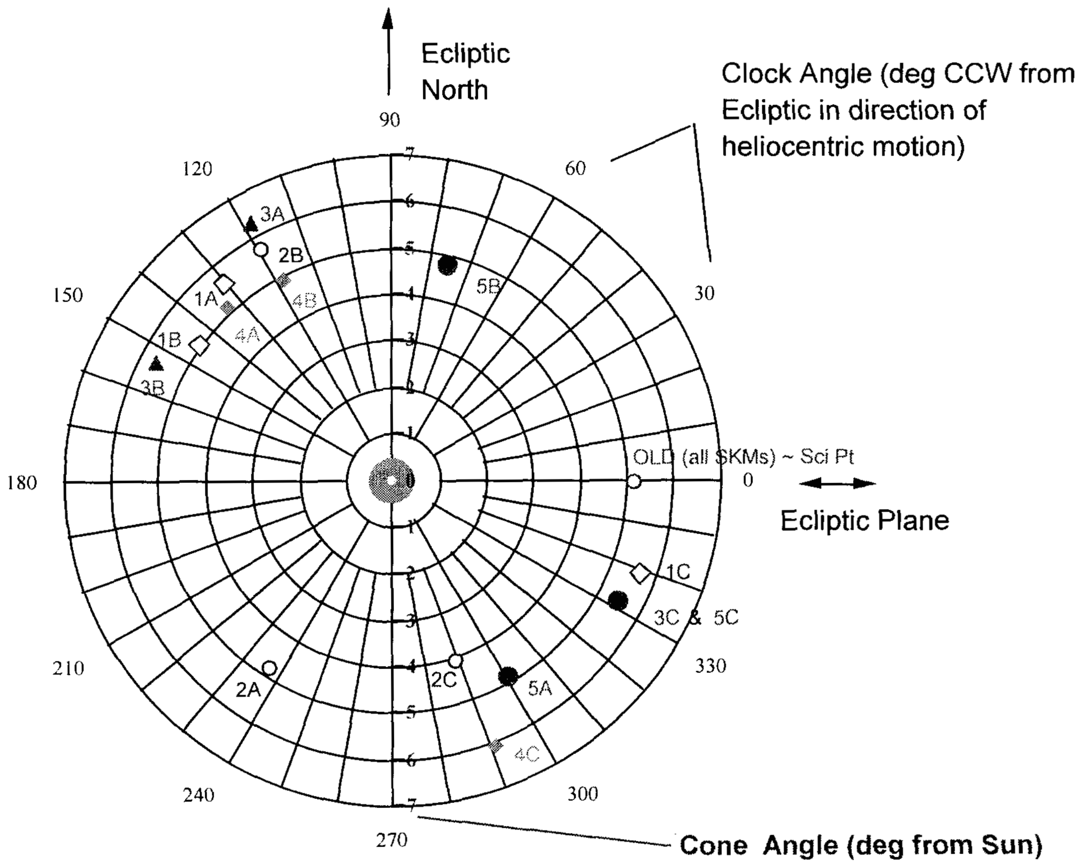
Statistically, any given SKM may be pointed in any direction with varying magnitudes. The Genesis spacecraft is capable of performing arbitrary maneuvers (with some limitations). However as has been pointed out earlier, the execution of certain maneuver types can become very operationally complex. For example, small (less than 0.5 m/s) anti-sunward pointing maneuvers are difficult to execute. To help mitigate the operational complexities associated with these maneuvers, it was decided pre-launch to bias each station-keeping maneuver in a regular manner. The initial bias chosen for each SKM was 1.5 m/s in a near-sunward pointing direction. When the statistical variations were overlaid with these biases, the resulting maneuvers were generally sunward pointing with magnitudes between 0.5 and 2.5 m/s. The upper magnitude coincides with a pre-launch guideline for maximum maneuver size on the smaller (0.2 lbf) thrusters. This is significant, since switching to larger (5 lbf) thrusters poses a contamination risk and would require an additional close-open cycle for the SRC backshell and concentrator cover.

These biases were incorporated into the design of the Lissajous trajectory using the same techniques used to design the original un-biased solution. In the original design, the LOI maneuver was the only post-launch deterministic impulsive event. This design gave rise to the notion of the "free return" trajectory that has been mentioned earlier. By designing the biases at each SKM (and subsequently at each return TCM) into the trajectory, the solution is no longer a "free" return since each maneuver *must* be performed in order to return to Earth. This is the trade-off that was made to ensure that each SKM and TCM is pointed in a favorable direction with an acceptable magnitude.

SKM Redesign

After it was decided to cancel the rest of the transfer TCM's, an effort was undertaken to redesign the trajectory to achieve a more favorable LOI maneuver, as was discussed in the previous section. The goal of the redesign was to improve LOI without affecting the return portion of the trajectory. As part of this redesign, the pre-launch SKM biases were re-examined in light of the developing operational experience with the spacecraft.

The pre-launch Navigation Plan allowed for execution errors as large as 6% at the 3-sigma level. When such errors were modeled in monte-carlo simulation runs, large dispersions with respect to sun angle had been evident. Consequently, the 1.5 m/s biases were set close to the Sun at a 5 deg off-sun cone angle and a 0 deg clock angle, the prevailing attitude in the direction of expected maximum solar wind flux, as required for science collection. By allowing the biases to float in both magnitude and direction, a more optimal LOI was determined that placed the spacecraft into a slightly different Lissajous orbit than the pre-launch design. The resulting SKM biases are shown in Figure 13. Note that the cone angles range from 4 to 7 deg off sun with varying clock angles, and the magnitudes range from 1.509 to 1.524 m/s. The added flexibility in the biases aided in redesigning the trajectory to return on essentially the same path as the pre-launch solution, with no added cost after LOI. In retrospect, designing biases into the Lissajous orbit allows redesign flexibility that can only be achieved through additional maneuvers in an unbiased solution.



NOTE: Magnitudes are in range 1.509-1.524 m/s.

Figure 13. Bias Shift as Designed for First Post-Launch Trajectory Re-Optimization

SKM Re-Redesign

As further experience with the spacecraft and maneuver design process was gained during 2002 flight operations, it became apparent that a further change to the SKM biases would be beneficial to simplify maneuver operations. By shifting the off-sun cone angles to between 12.5 and 28 deg, all future SKM's after 1C could statistically be expected to be the most benign maneuver type, namely the single leg near-Sun variety. In this redesign, SKM-2A was allowed to vary in magnitude and direction (within reason) and in fact ended up being about 0.75 m/s at 22 deg off sun. The remaining 11 SKM's were then reoptimized to satisfy the new pointing requirements, while still targeting the same return trajectory back to Utah. The reoptimized SKM's are shown in Figure 14, where SKM's 2A through 3B are the actual maneuvers performed to date and 3C through 5C are the estimated maneuvers as determined by the redesign. Here again, the flexibility of the biases in the Lissajous for redesign of the trajectory is proven to be useful.

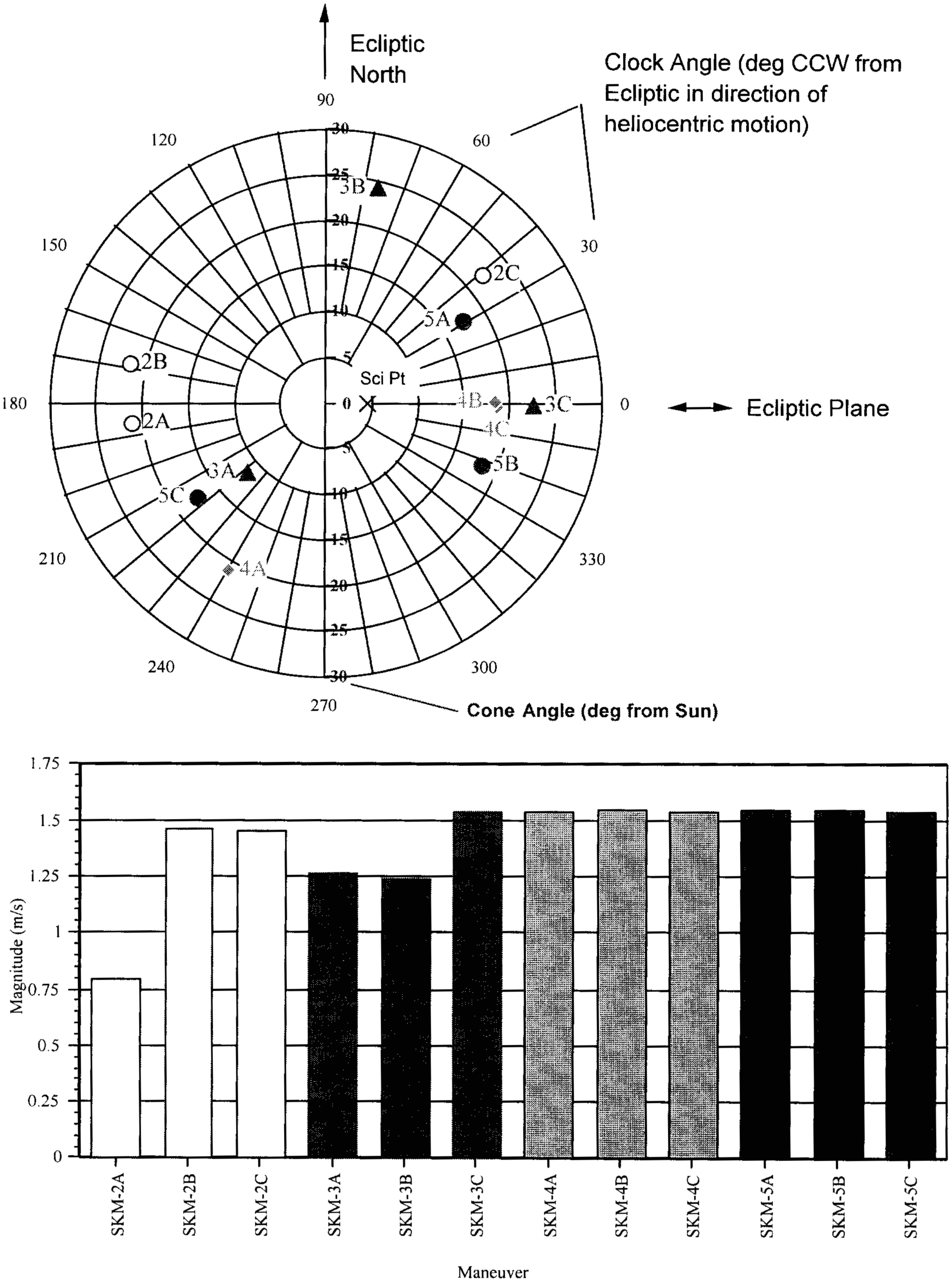


Figure 14. Actual Biases to Date for SKM-2A through SKM-3A with Revised Estimates for SKM-3B through SKM-5C

CONCLUSIONS

The Genesis spacecraft has proven to be a very reliable platform and the operational experience to date has been nominal. The flexibility of the Lissajous portion of the trajectory to redesign efforts has been shown to be quite beneficial to simplifying the operations, while not compromising the return of the samples. Future papers will discuss the operations through the remainder of the Lissajous and during the return to Earth and subsequent entry leading up to the mid-air recovery in Utah.

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