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EARTH RETURN MANEUVER STRATEGIES FOR GENESIS AND STARDUST

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As part of NASA's Discovery Program, Genesis and Stardust will be the first missions since the Apollo Program to return samples collected in deep space. To constrain costs of recovery, entry requirements must be much tighter than those imposed on Apollo. Spacecraft designs were also greatly simplified to limit costs, giving rise to a variety of operational limitations and constraints. In light of these considerations, approach to Earth presents a challenge in terms of both mission planning and navigation. This paper discusses strategies for trajectory correction during the Earth return phases of both missions.

INTRODUCTION

Genesis and Stardust are sample return missions selected as part of NASA's Discovery Program. For both missions, the samples will be delivered to the same specific recovery point on the Earth, the Utah Test and Training Range (UTTR), for subsequent analysis. In both cases, accuracy requirements for recovery of the samples are unprecedented and present a challenge in terms of both mission design and navigation. Overviews of the Genesis and Stardust trajectories are depicted in Ecliptic plane projection in Figures 1 and 2, respectively.

Genesis will collect solar wind samples for a period of approximately two and a half years around the first Earth-Sun Lagrange (L1) point. After collection activities are completed, Genesis will follow a free-return trajectory back to Earth, arriving in September 2004. The samples are fragile enough that mid-air capture of the Sample Return Capsule (SRC), following re-entry and deployment of a parafoil, will be required. The objective of Stardust is to collect interstellar and comet dust particles, the latter from a recent encounter with the comet Wild 2 in January 2004. A subsequent Deep Space Maneuver (DSM) in February 2004 has placed Stardust on an Earth return trajectory. After re-entry, a parachute landing and recovery on the ground is planned for the Stardust SRC in January 2006.

In many respects, the design of the two spacecraft could not be more different. In the case of the Genesis spacecraft, spin stabilization was chosen for attitude control. By contrast, the Stardust spacecraft utilizes three-axis attitude control. On the other hand, both spacecraft were designed with unbalanced thrusters, driven in part by the need to avoid contamination of the samples being collected. Specific design features will be discussed in subsequent sections.

This paper will discuss maneuver and calibration strategies employed to accommodate operational constraints, as well as results of Monte-Carlo analyses demonstrating that entry requirements can be met for Earth return legs for Genesis and, preliminarily, Stardust.

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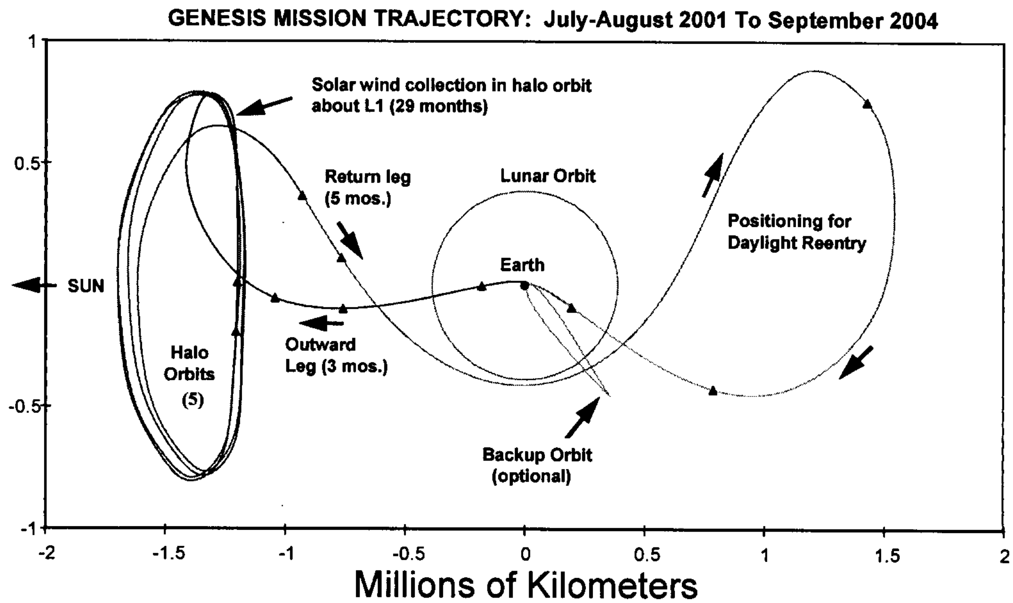

Figure 1. Genesis Mission Trajectory (Sun-Earth Rotating Frame)

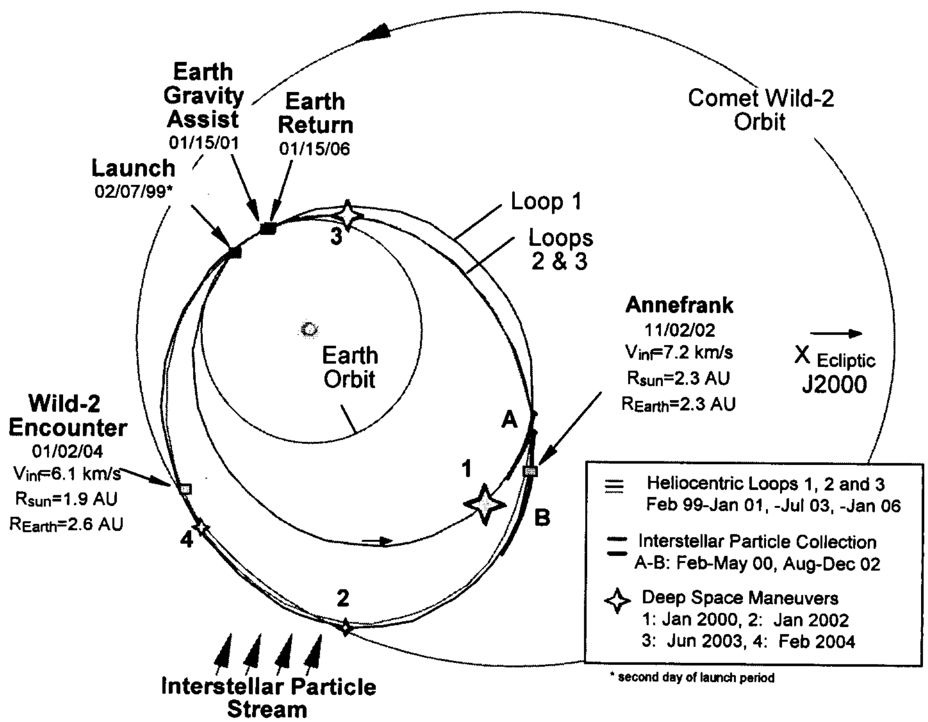

Figure 2. Stardust Mission Trajectory (Inertial Frame)

GENESIS SPACECRAFT DESIGN

Genesis is designed to spin about the spacecraft x-axis, as shown in Figure 3, with +x normally pointed to a near-Sun attitude for science collection and power maintenance purposes. The design includes the SRC, shown in the open or science configuration, and a supporting bus, which includes power, telecommunications, command and data handling, attitude control and propulsion subsystems. Mechanically, the bus consists of a single equipment deck with two solar arrays, hydrazine propellant tanks, nutation dampers and various components of the aforementioned subsystems. The SRC is attached to the sunward side of the deck, with the launch vehicle adapter ring and thrusters located on the anti-sunward side. For normal operations, the spacecraft spin axis is pointed toward the Sun for solar array power and to support solar wind sample collection. After sample collection is completed in April 2004, the spacecraft will be placed in cruise configuration with the SRC closed in preparation for Earth entry. The SRC is discussed in more detail in Smith et al.[1]

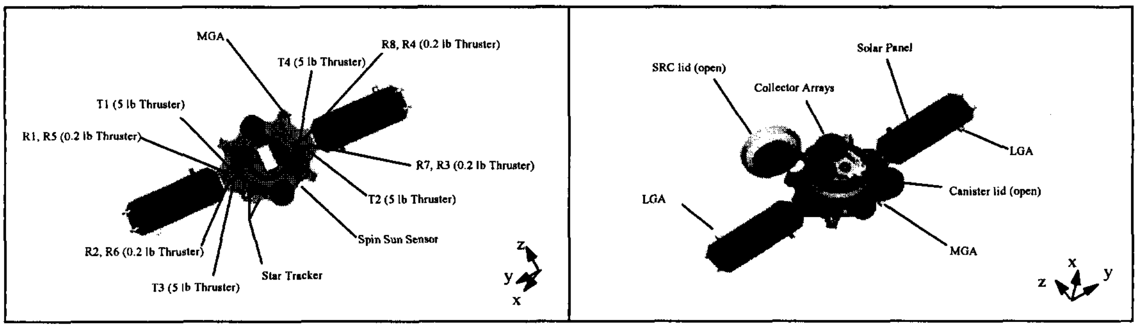

Figure 3. Genesis Spacecraft - Rear and Forward Deck Views

Power is supplied primarily by two solar arrays, populated with Silicon cells. In addition to solar arrays, the electrical power subsystem (EPS) also includes a power control assembly, pyrotechnics initiation unit and a 16 amp-hour Nickel Hydrogen rechargeable battery. To avoid unrecoverable battery power depletion, the x-axis of the spacecraft can be pointed more than about 30° off Sun for no more than 85 minutes at a time. The power control assembly performs load switching functions and controls battery discharge and recharge.

The telecommunications subsystem includes a near-Earth standard S-band transponder, one medium and two low gain antennas (MGA and LGA). The MGA, mounted on the anti-sunward side of the equipment deck, provides a 47.4 kilobit per second (kbps) downlink during science collection. LGAs, mounted on both the front and back of the solar arrays, provide 2-way communication with the ground at virtually all attitudes.

The attitude control subsystem (ACS) consists of a star tracker or scanner, as well as spinning sun sensors (SSS) and two-axis digital sun sensors (DSS) to provide sun-relative angle and spin rate. The nominal spin rate is 1.6 revolutions per minute (RPM), although for brief periods during

maneuvers and other parts of the mission the spin rate can reach as high as 15 RPM. During these periods, the star scanner does not function effectively, so that the SSS provides the sole source of attitude information. While on the SSS only, attitude changes must be performed by dead-reckoning from the last known three-axis attitude fix. To minimize prediction error, the dead-reckoned precession on SSS is generally directly away from the Sun with the subsequent precession directly back to a predetermined angle near the Sun. The two-axis sun sensors provide Sun location in two axes, but are limited to within 28° of the Sun.
In the original design, the star scanner was envisioned to provide accurate 3-axis attitude determination at virtually all attitudes while at 1.6 RPM. However, additional limitations arose from star scanner pre-launch performance tests. Such testing revealed that the star scanner could only reliably identify one star per spacecraft revolution. This necessitated a design workaround, where the ACS was modified to combine data from the star trackers and the DSS to obtain a three-axis attitude fix. This process, known as spin track, yields an attitude quaternion, but only while the Sun is within 28° of the spacecraft x-axis and the spin rate is less than 2 RPM. At attitudes farther off Sun, and/or when at a higher spin rate, star scanners cannot reliably identify even one star. Consequently, the SSS must provide attitude information under such conditions.

Moreover, because of the presence of wobble and nutation, which is exacerbated by any maneuvers, keep-out zones must be observed for spinning sun sensors at attitudes near the sunward and anti-sunward directions, to ensure that sun crossing times are accurately measured and spin rate knowledge is maintained. These zones are further augmented to allow for higher spin rates which are employed for propulsive maneuvers to guard against consequences of a failed thruster. As a consequence of these considerations, the best accuracy is obtained by constraining the direction of a maneuver within an annulus between 12.5° and 28° off Sun. As described in a later section, this performance will have a strong influence on the overall maneuver strategy for Genesis Earth return.

With no accelerometers or inertial measurement units, propulsive maneuvers must always be performed in an open-loop fashion. Genesis utilizes a straightforward, blowdown hydrazine propulsion subsystem design, which includes twelve thrusters, organized in two redundant strings. All thrusters are mounted on the anti-sunward side of the equipment deck to minimize sample contamination during collection. Two 22N thrusters provide axial velocity control for large trajectory correction maneuvers, and precession attitude control when the spacecraft spin rate is greater than 2 RPM. Four 0.9N thrusters provide axial velocity control for small maneuvers, precession attitude control when the spacecraft spin rate is less than 2 rpm, and spin rate change and control in all instances. Two tanks provide the hydrazine to support all thruster firings. Since thrusters so positioned do not produce balanced torques, all attitude control maneuvers contribute a translational delta-velocity (Δv) in addition to intended propulsive maneuvers. These must be accounted for orbit determination purposes and in terms of designing propulsive maneuvers.

For more information on the spacecraft and mission design, refer to Smith et al.[1]

GENESIS MANEUVER STRATEGY

The Ecliptic plane projection of terminal portion of the Genesis trajectory is shown in Figure 4. This includes the Return Phase, beginning in April 2004 after completion of science collection, followed by the Recovery Phase encompassing the final 30 days of the nominal mission. The strategy for Earth return, including calibration plans, maneuver biasing and other details, has evolved somewhat over the last four years from what was presented in earlier papers.[2,3] These changes were driven by modifications to the spacecraft design described previously, as well as changes to the mission plan itself to accommodate the launch of the Mars Odyssey spacecraft in early 2001, which led to a delay in the launch of Genesis from January-February 2001 until July-August 2001. In order to meet the original science requirement of 22 months of solar wind collection and arrive at UTTR in late summer when weather is optimal for SRC recovery, it was necessary to redesign the Genesis mission to return in September 2004, instead of the original target period of September 2003. The new mission profile included an extra halo loop, which allowed for six additional months of science collection. The lengthening of the science collection portion of the mission afforded the Mission Design and Navigation Team time to consider and analyze a number of options and contingencies which lead to the current plan described here.

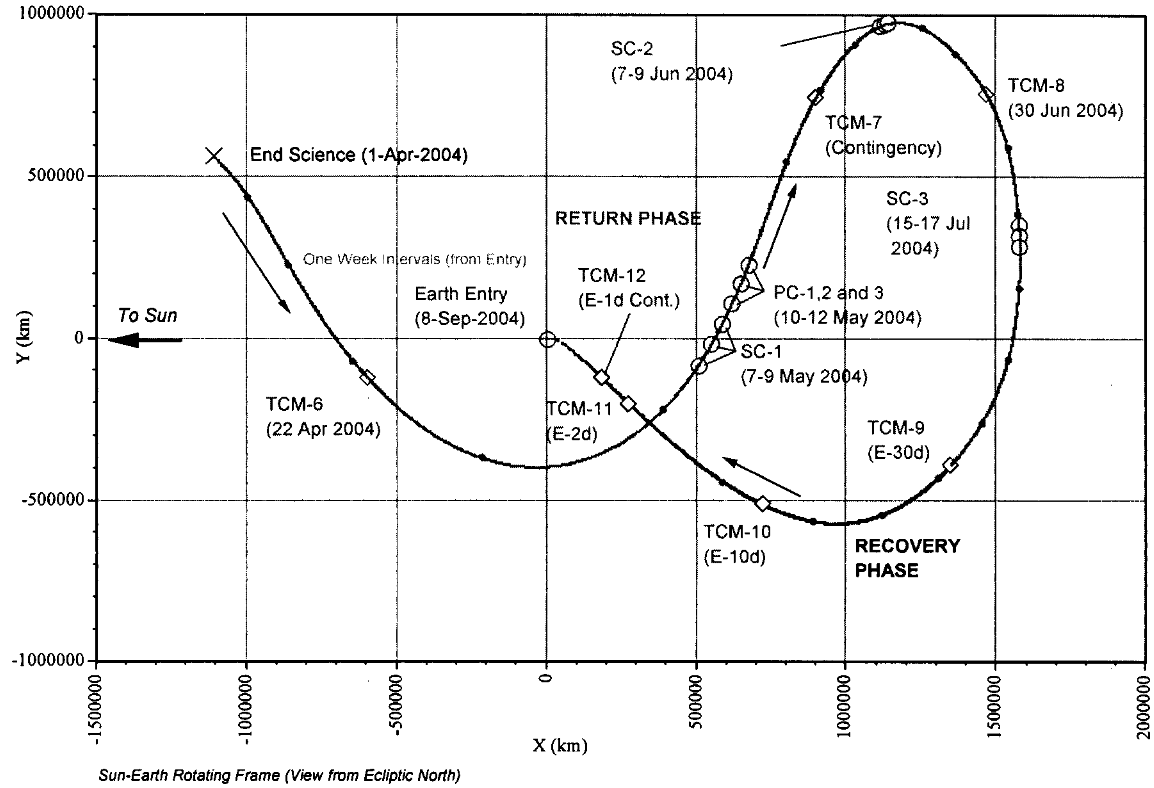

Figure 4. Trajectory for Return and Recovery Phases of Genesis Mission

Table 1 provides an overview of the recommended strategy for conducting maneuvers during Earth return, as of this writing (final strategy and plans are subject to further revision by the Genesis project). As indicated in Table 1, there a number of trajectory correction maneuvers (TCMs), needed to ensure a successful return to Earth. All TCMs are now biased (i.e.,a deterministic velocity change is included in the reference trajectory) in a direction near the Sun to support the best possible performance of the ACS and thereby produce minimal execution errors. The exact direction is consistent with an attitude maintenance plan which avoids placing the spacecraft x-axis closer than 3° from the Sun at any given time during Earth return to minimize potential problems with the SSS, the primary sensor by which the ACS determines the spin rate of the spacecraft.. The Sun-relative direction and magnitude of these deterministic biases are indicated in Table 1.

For operational convenience and to ensure the best possible accuracy in the presence of such biases, maneuvers up through TCM-10, at ten days out from Earth entry, are targeted to the nominal Cartesian location at the subsequent maneuver epoch. However, TCM-10 and the final TCM (nominally TCM-11 at 2 days out from Earth) will be targeted directly to the atmospheric entry interface at 125 km geocentric altitude (about 6503.14 km distance from Earth center). The key parameters associated with this final target are inertial flight path angle (γ) and Earth-fixed latitude and longitude. Arrival time at the entry interface is not constrained, but of course must be correlated with the appropriate Greenwich hour angle and right ascension. Only the sample return capsule (SRC) will achieve the entry interface and ultimately return to UTTR. Therefore, targeting of terminal TCMs must not only account for

Table 1. Overview of Genesis Earth Return Maneuver Strategy

Maneuver	*Nominal Epoch**	*Deterministic Bias (m/s)*	*Sun Angle (deg)*	*Recommended Backup Maneuver*	*Backup Epoch**	*Notes*
TCM-6	22-Apr-04	1.47	20.4	TCM-7	25-May-04	Additional backup (TCM-6a) on 29-Apr-04 or one week after nominal TCM-6 also examined; TCM-7 may also be used to correct an extremely aberrant TCM-6.
TCM-8	30-Jun-04	1.45	22.4	TCM-8b	21-Jul-04	Additional backup (TCM-8a) on 7-Jul-04 or one week after nominal TCM-8 also examined; TCM-8a and 8b selected to bracket final set of spin calibrations (SC-3).
TCM-9	9-Aug-04 (~E-30d)	0.97	22.2	TCM-9a	16-Aug-04 (~E-23d)	Re-optimization of the Genesis trajectory likely to be needed if nominal TCM-9 not executed; 7 day delay for TCM-9a but skipping maneuver altogether also a possibility.
TCM-10	29-Aug-04 (~E-10d)	0.98	9.5	TCM-10a	3-Sep-04 (~E-5d)	Biasing turn to induce corrective delta-v for TCM-11/12 direction also a possibility 1-2 days after nominal TCM-10.
TCM-11	6-Sep-04 (E-52h)	0.96	15.2	TCM-12	7-Sep-04 (E-28h)	TCM-11 and 12 are currently envisioned as alternate final TCM opportunities; TCM-12 (1.992 m/s deterministic bias at 12.1 deg off Sun) is backup both to cover anomaly preventing TCM-11 execution and to provide alternative if TCM-11 direction not favorable for more accurate spin control implementation; however, use of TCM-12 to complete partial (aborted) TCM-12 also under study. Only spin control currently doable in 24-hour final development timeline for TCM-11/12 (must use 2 day or longer period if other maneuver type implemented, necessitating an earlier OD cutoff).

* Maneuvers through TCM-9 nominally at 19:00 UTC; final maneuvers nominally around 12:00 UTC.

the Δv biases associated with subsequent TCMs, but also those arising from activities associated with SRC release from the spacecraft bus. These release activities include two spin changes (1.6 to 10 RPM and 10 to 15 RPM), an intermediate precession to the release attitude and the actual mechanical separation of the SRC from the spacecraft itself.

Biasing of maneuvers means that such maneuvers must always be performed. All maneuvers for Earth return have at least one associated backup opportunity, to allow for recovery from spacecraft or ground anomalies which could arise at critical times. Recommended backup maneuvers and contingencies are outlined in Table 1. Great care must be taken in the design of such contingency or backup maneuvers. In principle, if a nominal maneuver is missed, then there will be an impact both in terms of changing the bias of the backup maneuver as well as the following maneuver, in order to return to the nominal trajectory. In practice, for operational simplicity, the return maneuver strategy allows for at least two pre-defined opportunities to execute each maneuver with a subsequent alternative maneuver to allow Genesis to achieve the target at entry interface.

Recovering from an anomaly becomes more and more problematic, the closer the spacecraft comes to the entry interface. Fortunately, there are multiple layers of defense, which can be exercised to avoid loss of the mission. For instance, there is the possibility of re-optimizing the trajectory (i.e., following a slightly different reference trajectory) to achieve the final target at Earth. Current backup maneuvers have been analyzed as single-failure events, but multiple failures or anomalies have not been studied in great detail. In the event of multiple anomalies, a full trajectory re-optimization might have to be employed. Also, the flight team must have the flexibility to disable SRC fault protection measures to reclaim entry performance margin if needed by Navigation. If all else fails, as a final measure, Genesis can be diverted into a backup orbit for later Earth entry, if nominal entry within requirements cannot be achieved. The backup orbit strategy under consideration, involving a six-month delay before Earth return, is described in another paper.[4]

GENESIS RETURN CALIBRATION ACTIVITIES

In addition to various anomalies which could delay maneuvers and other key events, there remains some uncertainty associated with both the nominal OD and spacecraft performance during Earth return and recovery. After science collection ends in April 2004, the science collection arrays and instruments will be stowed and the SRC backshell closed in preparation for Earth atmospheric entry and recovery. The mass properties and solar radiation pressure cross-section will be different than they have been for the majority of the mission. In order to properly characterize the behavior of the spacecraft in its post-science or cruise configuration, a number of calibration activities have been scheduled, as shown in Table 2. These activities include a quiet period immediately following the end of science collection and during which the influence of solar radiation pressure on the spacecraft will be characterized. Also, there are a number of maneuver calibrations which will attempt to characterize the spin and precession behavior of the spacecraft with the goal of reducing maneuver execution errors, indicated in Table 2.

Table 2. Genesis Earth Return Calibration Activities

Calibration	*Nominal Epoch(s)*	*Purpose/Description*
OD Quiet Period	5-21 Apr '04	Characterize solar radiation pressure cross-section while in cruise configuration (SRC backshell closed); reduce unmodeled non-gravs.
Spin Cal (SC)-1	7-9 May '04	Determine proportionality or function between spin rate change and delta-v while at Earth point (3 spin control maneuvers, approximately 0.5, 1 and 1.5 m/s).
PARL Cal	10-12 May '04	Verify delta-v model for precession to attitude using rhumb line, to be used during SRC release sequence; precess across Earth point 7.5 to 25 deg from Sun while at 10 RPM.
SC-2	7-9 Jun '04	Revisit previous 3 spin control maneuvers while off Earth point to determine orthogonal biases, if any.
SC-3	15-17 Jul '04	Determine proportionality or function between spin rate change and delta-v while at Earth point (3 spin control maneuvers, either verifying previous 0.5, 1 and 1.5 m/s behavior or testing three new alternative maneuvers).

Prior to launch, maneuver execution errors were estimated to be relatively large, as shown in Table 3. Flight experience thus far suggests that such errors have been significantly reduced, especially for near-Sun maneuvers. Nevertheless, it is hoped that such errors can be further reduced, as indicated in Table 3. The best strategy for reducing execution errors involves characterizing the mass properties and spin characteristics of the spacecraft as much as possible. Because Δv cannot be determined by on-board ACS software directly, the most accurate maneuver implementation possible for a spinning spacecraft makes use of the following relationship for spin rate changes $\Delta\omega_x$ along the spacecraft spin (+X) axis:

$$\Delta v \cong \frac{I_{xx} D w_x}{mr} = k\, D w_x \qquad (1)$$

Here I_{xx} is the moment of inertia, m is the mass, and r is the thruster moment arm. All of these quantities can be characterized as a single proportionality constant k which is determined via ground-based Doppler coupled with spin rate telemetry during calibration events near Sun-Earth line crossings, as shown in Figure 2. Subsequent quasi-closed-loop burns are then possible with the goal of achieving 3σ fixed errors ~3 mm/s and proportional errors ~1%. With such execution errors, it should be possible to easily meet the entry requirements outlined in the next section.

Maneuvers starting with TCM-9 will employ spin control to produce Δv in lieu of a normal burn using a combination of four thrusters, provided the direction is within 28° of the Sun. This angular constraint arises for limitations in ACS accuracy and power, alluded to previously, and accommodates the relatively longer duration of this maneuver activity. As an option, TCM-6 and/or TCM-8 may also utilize spin control in lieu of a nominal burn, to provide more accuracy and afford additional targets of opportunity for spin control calibration. Note that current operational timelines for TCM-11 and 12 allow only 24 hours to design, test and execute the maneuver, with the implicit assumption that such a maneuver must be executed via the spin control method.

Table 3. Genesis Maneuver Execution Errors

Proportional Execution Errors (3-sigma)

Case	*Magnitude*	*Direction*	*Comments*
Pre-Launch	6%	6.3%	Very conservative to cover possible worst case envisioned before actual flight experience.
Current Estimate (Off-Sun)	3%	3%	Upper limit projected for > 28 deg off Sun, based on current performance.
Current (Near-Sun)	2%	2%	Evident from halo station keeping maneuvers (SKMs) in science configuration; believed to be applicable to standard near-Sun maneuvers on small thrusters in cruise configuration.
Spin Control TCM Estimate	2%	2%	Upper limit applicable to TCM-6 when < 28 deg from Sun.
Spin Control TCM Estimate (Degraded)	1.50%	1.50%	Allows for unanticipated calibration result and/or possibility of swap to backup thruster string after calibration completed.
Spin Control TCM (Final Knowledge)	1.00%	1.00%	Applicable to final TCM; allows for improved knowledge from use on earlier TCMs or re-calibration after earlier TCMs if thruster swap occurred.

Dominant Fixed Execution Errors (m/s, 3-sigma)

Type	*Magnitude*	*Direction*	*Comments*
Precession	0.0003 + 0.04 • sin (ψ/2)	0.0003 + 0.04 • sin (ψ/2)	y is one-way precession in deg; 0.04 --> 0.05 on large thrusters; worst case for single precession based on 160 deg.
Spin Adjustment	0.0075 + 0.005625 * Dw	0.002 + 0.00125 * Dw	Dwis one-way spin change magnitude in RPM; applicable only for spin adjustments in conjunction with standard (non-spin-control) maneuvers (~1 RPM for SKMs).

GENESIS MANEUVER ANALYSES

To ensure the viability of the maneuver strategy outlined in Table 1, the performance of Genesis must be assessed relative to two driving requirements for Earth entry, both specified at 125 km geocentric altitude defined as the entry interface and compatible with delivery to UTTR where the SRC will be retrieved by helicopter in mid-air after deployment of a parafoil. The requirements may be paraphrased as follows:

- The Genesis SRC must pass through the entry interface at -8.0° ± 0.08° (3-sigma) flight path angle to achieve nominal aerodynamic and thermal conditions for atmospheric entry.

- The Genesis SRC must achieve a specified latitude and longitude within an elliptical "keyhole" with end-to-end dimensions of 33 km downrange and 10 km cross-range (both 99.7%) in order to achieve the prescribed trajectory for recovery at UTTR.

The latter requirement is capability driven and derived from analyses recently reported by Lockheed Martin Astronautics (LMA) and NASA-Langley. Such analyses are based on the Program for Optimization of Simulated Trajectory (POST) which models effects described in Desai et al.[8] All TCMs prior to Earth entry must ultimately meet these delivery requirements.

Studies have been performed to test the robustness of the aforementioned maneuver strategy and to ensure sufficient operational flexibility to meet any foreseen contingencies. Monte-Carlo studies were performed utilizing several simulations. These were supported by software simulations, including the Linear Analysis of Maneuvers with Bounds and Inequality Constraints (LAMBIC) and Sigma, which supplied orbit determination (OD) covariances to the former. Results from LAMBIC based on 5000 samples, with current or projected execution errors described in Table 3, were applied to obtain a preliminary estimate of the flight path entry angle and entry location uncertainties after the final Earth entry TCM and SRC release activities. A number of scenarios have been examined involving various TCM locations, different levels of assumed OD and maneuver execution error, and different OD data cutoff epochs have been examined. Several representative cases are discussed in the following subsections.

Alternative Final TCMs

Table 4 compares the entry performance for the two alternate final TCMs: TCM-11 at 52 hours prior to entry and TCM-12 at 28 hours before entry interface. Table 4 indicates the percentage of samples where spin control can be used, as well as the 3-sigma flight path angle (FPA) and entry ellipse (downrange by cross-range) achieved for the portion of samples where spin control could be used. However, the probability of success shown is the smaller of the percentages of samples meeting the ±0.08° FPA and 33 X 10 km entry ellipse requirements for all 5000 samples. For TCM-11, 3.7% of samples are not close enough to the Sun to use spin control, as illustrated by samples circled in red in Figure 5. Note that only the non-spin control samples fall outside the keyhole; however, for TCM-12, virtually all of the samples fall within the spin control region.. The alternatives for such a contingency include either waiting until TCM-12 to execute the final maneuver, or executing TCM-11 as a non-spin-control type. The latter alternative most likely precludes performing the final design in less than 24 hours, suggesting that it will either be necessary to defer to TCM-12 in any case or base the TCM-11 design on an OD data cutoff at an earlier epoch. Sensitivity to OD cutoff epoch is described further in the next subsection.

Table 4. Variation in Entry Performance for TCM-11 Versus TCM-12

Final TCM	Spin Control Feasible	3-Sigma FPA Error (deg) for Spin Control	99.7% Ellipse (km) for Spin Control (Entry interface)	Probability of Success* (All Maneuver Types)
TCM-11 (Nominal)	96.1%	0.049	20.3 X 1.1	99.1%
TCM-12 (Backup)	99.7%	0.054	23.0 X 1.1	100%

* Inside 33 X 10 km "keyhole" at 125 km geocentric altitude and within ±0.08 deg FPA error.

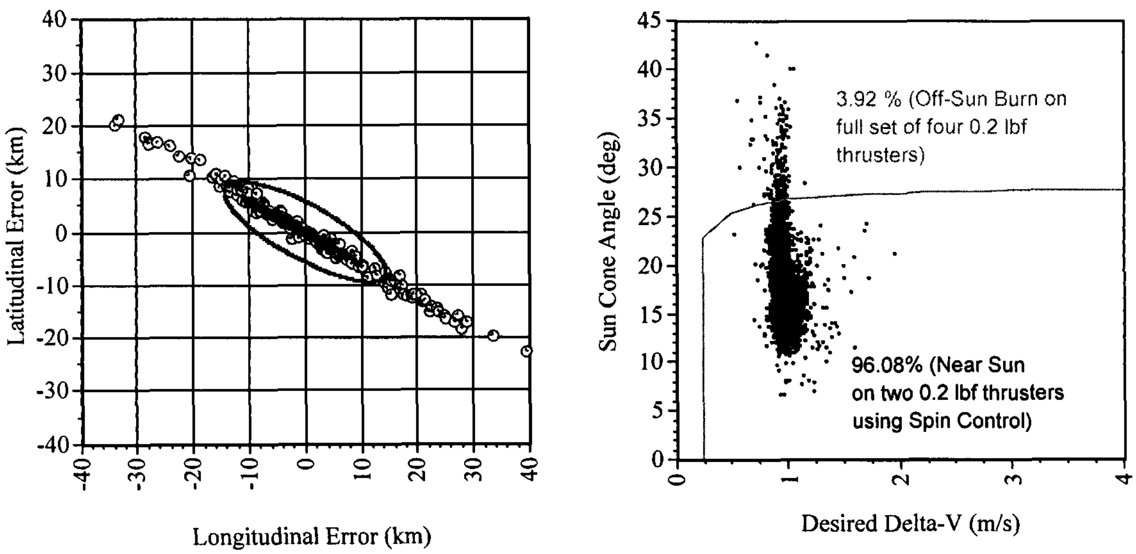

Figure 5. Illustration of Maneuver Performance and Type (Nominal TCM-11)

Variations in OD Cutoff and Quality and Maneuver Execution Error

For Genesis, the OD cutoff is defined as the last epoch for which radiometric data can be accepted for developing an OD solution in preparation for a TCM. Table 5 indicates the variation in entry performance as a function of various data cutoff epochs for TCM-11 and TCM-12, the alternative final TCMs prior to entry. Entry performance appears highly robust with regard to OD data availability. This implies that Genesis is relatively invulnerable to DSN station outages, which might otherwise severely hamper the final maneuver design.

Table 5. Variation in Entry Performance Based on OD Cutoff for Final TCM

TCM-11

OD Cutoff Relative to Entry	*Spin Control Feasible for TCM-11*	*3-Sigma FPA Error (deg) for Spin Control*	*99.7% Ellipse (km) for Spin Control (Entry interface)*	*Prob. Success* w/o TCM-12*
E-3d	96.1%	0.049	20.3 X 1.1	99.1%
E-4d	96.0%	0.053	22.8 X 1.1	98.9%
E-5d	95.8%	0.060	25.8 X 1.2	98.9%

TCM-12

OD Cutoff Relative to Entry	*Spin Control Feasible for TCM-12*	*3-Sigma FPA Error (deg) for Spin Control*	*99.7% Ellipse (km) for Spin Control (Entry interface)*	*Prob. Success* w/ TCM-12*
E-2d	99.7%	0.051	21.6 X 1.1	100%
E-3d†	99.7%	0.054	23.0 X 1.1	100%
E-4d	99.7%	0.059	25.2 X 1.1	100%
E-5d	99.7%	0.066	28.8 X 1.2	100%

* Inside 33 X 10 km "keyhole" at 125 km geocentric altitude and within ±0.08 deg FPA error.

† Baseline case for purposes of this analysis.

There is also a considerable amount of robustness evident with respect to execution errors. Table 6 indicates variations in maneuver execution error for two different OD variations with regard to TCM-11 and TCM-12. In the case of TCM-11, potential OD quality is varied for the approximate nominal Entry minus three day (E-3d) OD cutoff. In the case of TCM-12, OD quality is fixed at the nominal level and the cutoff epoch is varied (note that E-3d was used as the baseline OD cutoff epoch for both TCM-11 and 12 for purposes of this analysis; as of this writing, operational planning is leaning towards using E-2d as the OD cutoff for TCM-12). The potential OD quality improvement indicated here could result from the earlier quiet period solar radiation pressure calibration which would reduce the unmodeled non-gravitational (non-grav) acceleration to 2.5 x 10^{-11} km/sec^2 with a data weight of 0.1 mm/sec (in all other cases included here, the assumed non-grav acceleration is 5 x 10^{-11} km/sec^2 with data weight of 0.3 mm/sec, chosen to address the conceivable worst-case situation). Such improvement in OD quality improves TCM-11 performance such that the dependency on TCM-12 as a backup is reduced to 0.6% or less, even in light of relatively poor maneuver performance.

Table 6. Entry Performance Variations with Maneuver Execution Error and OD Quality

TCM-11 (Nominal OD and E-3d Cutoff)

Spin Control Execution Error (3-Sigma)	Spin Control Feasible for TCM-11	3-Sigma FPA Error (deg) for Spin Control	99.7% Ellipse (km) for Spin Control (Entry interface)	Prob. Success* w/o TCM-12
1.0%	99.7%	0.048	20.4 X 1.1	99.0%
1.5%	96.1%	0.052	22.3 X 1.2	98.9%
2.0%	95.5%	0.058	24.6 X 1.4	98.8%
2.5%	94.9%	0.068	28.5 X 1.6	98.7%
3.0%	94.3%	0.077	32.5 X 1.8	98.2%

TCM-11 (Optimistic OD and E-3d Cutoff)

Spin Control Execution Error (3-Sigma)	Spin Control Feasible for TCM-11	3-Sigma FPA Error (deg) for Spin Control	99.7% Ellipse (km) for Spin Control (Entry interface)	Prob. Success* w/o TCM-12
1.0%	99.4%	0.046	19.2 X 1.0	99.8%
1.5%	99.3%	0.050	20.9 X 1.2	99.8%
2.0%	99.3%	0.057	24.0 X 1.3	99.8%
2.5%	98.9%	0.067	27.6 X 1.5	99.7%
3.0%	98.3%	0.075	31.0 X 1.8	99.4%

TCM-12 (E-3d OD Cutoff)

Spin Control Execution Error (3-Sigma)	Spin Control Feasible for TCM-12	3-Sigma FPA Error (deg) for Spin Control	99.7% Ellipse (km) for Spin Control (Entry interface)	Prob. Success* w/ TCM-12
1.0%	99.7%	0.054	22.9 X 1.1	100%
1.5%	99.7%	0.063	26.6 X 1.3	99.9%
2.0%	99.6%	0.076	31.6 X 1.5	99.8%
2.5%	99.5%	0.091	37.7 X 1.8	99.2%
3.0%	99.5%	0.107	43.8 X 2.1	97.7%

TCM-12 (E-2d OD Cutoff)

Spin Control Execution Error (3-Sigma)	Spin Control Feasible for TCM-12	3-Sigma FPA Error (deg) for Spin Control	99.7% Ellipse (km) for Spin Control (Entry interface)	Prob. Success* w/ TCM-12
1.0%	99.7%	0.051	21.6 X 1.1	100%
1.5%	99.7%	0.061	25.4 X 1.3	99.9%
2.0%	99.6%	0.073	30.7 X 1.5	99.8%
2.5%	99.5%	0.089	36.9 X 1.8	99.3%
3.0%	99.5%	0.107	43.9 X 2.1	97.8%

* Inside 33 X 10 km "keyhole" at 125 km geocentric altitude and within ±0.08 deg FPA error.

TCM-9 Targeting Variations

Table 7 compares targeting variations for TCM-9. Variations shown include targeting the nominal TCM-9 at E-30d directly to entry, as well as targeting a delayed TCM-9 at E-20d to entry with TCM-10 deleted. TCM-10 must be skipped in the latter case because deterministic bias for TCM-10 is too small and in wrong direction if TCM-9 is delayed 10 days. These are contrasted with the nominal targeting to the next maneuver epoch, then switching to entry targeting starting at TCM-10 or 10 days out from entry. Targeting to entry before TCM-10, instead of to the TCM-10 location as an intermediate target, always results in worse performance, both in terms of maneuver size and entry accuracy; this is believed to arise in part from the biasing strategy (i.e., targeting through deterministic maneuvers) and in part from the characteristics of the three-body trajectory, both of which are departures from previous Earth return experience and traditional targeting strategy.

Table 7. Entry Performance Variations with Targeting Method for Nominal or Delayed TCM-9

TCM-9 Targeting Case	*Spin Control Feasible*	*3-Sigma FPA Error (deg) for Spin Control*	*99.7% Ellipse (km) for Spin Control (Entry interface)*	*Prob. Success* w/o TCM-12*
TCM-10 Location	96.1%	0.049	20.3 X 1.1	99.1%
Entry Interface	81.8%	0.05	21.0 X 1.0	95%
Entry Interface (10 Day Delay)†	80.2%	0.05	21.6 X 1.1	95%

* Inside 33 X 10 km "keyhole" at 125 km geocentric altitude and within ±0.08 deg FPA error.

† TCM-10 skipped but trajectory not re-optimized.

Other Maneuver Contingencies

In addition to the aforementioned cases, a number of other contingencies involving delayed or skipped maneuvers were examined, as shown in Table 8. Again, the robustness of the overall maneuver strategy is evident. These results tend to support the maneuver strategy outlined earlier in Table 1.

Note that TCM-10a at E-5d actually results in better entry performance than the nominal TCM-10 at E-10d. Nevertheless, it is better to preserve both the E-10d and E-5d options to guarantee operational robustness and flexibility. However, TCM-10a has one downside in that it can grow as large as 3.5 m/s at the 95% level, involving a longer, riskier execution sequence (see next subsection for Δv results).

Deletion or delay of TCM-9 can have a more profound impact on entry targeting than other maneuvers, either before or after TCM-9. In either case, trajectory re-optimization is recommended to ensure best possible directions for TCM-10, 11 and 12 to achieve minimal execution errors for optimal entry targeting performance. If TCM-9 is deleted altogether, the performance is nearly as good as TCM-9a (7-day delay). However, this causes a very large TCM-10 (over 8 m/s, 95%). As noted previously, the 10-day delay case has several drawbacks as well, so it appears that the 7-day delay with re-optimization is the preferred backup strategy for TCM-9. Further study is warranted to ensure successful recovery from an aborted or delayed TCM-9.

Table 8. Some Additional Entry Performance Variations based on Other Contingencies

Contingency Case	Spin Control Feasible	3-Sigma FPA Error (deg) for Spin Control	99.7% Ellipse (km) for Spin Control (Entry interface)	Prob. Success* w/o TCM-12
TCM-10 5 Days Late	99.8%	0.051	21.7 X 1.0	100%
TCM-9 7 Days Late (No Re-optimization)	76.7%	0.052	22.0 X 1.1	93.1%
TCM-9 7 Days Late (Re-optimization)	91.9%	0.049	21.0 X 1.1	97.4%
TCM-9 Skipped (No Re-optimization)†	82.8%	0.051	21.8 X 1.1	95.5%
TCM-9 Skipped (Re-optimization)	90.6%	0.052	22.4 X 1.1	97.2%
TCM-8 7-Day Delay	94.6%	0.051	21.7 X 1.1	98.6%
TCM-8 21-Day Delay	94.3%	0.051	21.2 X 1.1	98.6%
TCM-8 Skipped (No Re-optimization)†	70.9%	0.056	23.8 X 1.1	91.2%
TCM-6 7-Day Delay	94.6%	0.051	21.8 X 1.1	98.5%
TCM-7	95.3%	0.050	21.3 X 1.1	99.0%

* Inside 33 X 10 km "keyhole" at 125 km geocentric altitude and within ±0.08 deg FPA error.

† OD covariances after skipped maneuver very optimistic.

Earlier TCMs can more easily accommodate delays with less impact on final entry targeting. TCM-8b, the 21-day delay case, is comparable to TCM-8a, the 7-day delay case, in terms of entry targeting performance. TCM-8b seems preferable, as it allows more time to diagnose potential problems arising from a delayed or aborted TCM-8, although it can result in a larger TCM-9 (nearly 3 m/s, 95%). Note that skipping TCM-8 altogether without re-optimization appears undesirable, and would engender a very large TCM-9 (over 18 m/s, 95%) and TCM-10 (nearly 6 m/s, 95%).

For TCM-6, the 33-day delay (TCM-7) appears just as good as 7-day delay case (TCM-6a), and perhaps even slightly better in terms of entry targeting; however, it may require nearly 9 m/s instead of 2.3 m/s, although TCM-8 is potentially reduced from 4 to only 1.7 m/s (all 95% values).

All of the aforementioned contingencies will be examined further in the months leading up to Earth entry, and many of these will be rehearsed during upcoming Operation Readiness Tests (ORTs).

Δv Estimates for Representative Cases

Δv estimates for most of the aforementioned cases are indicated in Table 9. Here, Δv or fuel requirements are a secondary concern, since Genesis has a relatively large reserve of fuel originally provided to guard against launch and early mission contingencies which never arose. Nevertheless, possible implications on entry targeting remain (e.g., larger maneuvers near entry are less desirable than smaller maneuvers due to impact of proportional execution errors on delivery accuracy).

Table 9. Δv Estimates Associated with Various Genesis Maneuver Analysis Cases

Case	Mean Delta-V						95% Delta-V					
	TCM-6/7	TCM-8	TCM-9	TCM-10	TCM-11(12)	TOTAL*	TCM-6/7	TCM-8	TCM-9	TCM-10	TCM-11(12)	TOTAL*
Baseline	1.49	1.54	1.00	1.00	0.97 (2.01)	22.9 (24.0)	1.87	2.63	1.39	1.30	1.08 (2.44)	24.0 (25.1)
Baseline with TCM-9 Targeted to Entry	1.49	1.54	1.39	1.46	1.00	23.8	1.87	2.63	2.40	2.57	1.40	26.1
Optimistic OD (TCM-10 Onward)	1.49	1.54	1.00	1.00	0.96	22.9	1.87	2.63	1.39	1.30	1.05	24.0
TCM-10 5 Days Late	1.49	1.54	1.00	2.35	0.96	24.3	1.87	2.63	1.39	3.22	1.02	25.6
TCM-9 10 Days Late (TCM-10 Skipped)	1.49	1.54	2.16	---	0.99	23.1	1.87	2.63	2.19	---	1.40	24.2
TCM-9 7 Days Late (no reopt)	1.49	1.54	1.67	0.53	1.00	23.2	1.87	2.63	2.41	0.83	1.28	24.2
TCM-9 7 Days Late (reopt)	1.49	1.54	1.64	0.55	0.97	23.1	1.87	2.63	2.41	0.83	1.28	24.1
TCM-9 Skipped (no reopt)	1.49	1.54	---	5.60	0.99	26.6	1.87	2.63	---	8.15	1.35	28.7
TCM-9 Skipped (reopt)	1.49	1.54	---	5.61	0.96	26.5	1.87	2.63	---	8.14	1.31	28.7
TCM-8 7-Day Delay	1.49	2.00	1.07	1.00	0.97	23.5	1.87	3.64	1.41	1.42	1.09	25.2
TCM-8 21-Day Delay	1.49	3.92	1.82	1.00	0.97	26.2	1.87	7.57	2.94	1.45	1.10	34.3
TCM-8 Skipped (no reopt)	1.49	---	9.01	3.39	1.04	31.9	1.87	---	18.13	5.99	1.62	43.4
TCM-6 7-Day Delay	2.01	2.03	1.08	1.01	0.97	24.0	2.34	4.00	1.59	1.40	1.09	26.2
TCM-7 (TCM-6 33-Day Delay)	5.67	1.18	1.00	1.00	0.97	26.7	8.67	1.72	1.47	1.40	1.09	30.0

* Includes all ACS maneuvers during Return/Recovery, in addition to TCMs.

STARDUST SPACECRAFT DESIGN AND PERFORMANCE

The Stardust spacecraft, as shown in Figure 6, is a three-axis stabilized spacecraft which differs in many ways from Genesis, but also possesses many similarities with regard to operational capabilities and limitations, especially for Earth entry. Stardust has a star tracker with analog sun sensors as backup, but also provides an inertial measurement unit (IMU) with gyros and accelerometers allowing for some closed-loop control of propulsive maneuvers. As in the case of Genesis, thrusters are located on the opposite side of the space vehicle from sample collectors to minimize contamination of samples. These include two strings (prime and backup) of four main thrusters (1 lbf each) used for TCMs and four reaction control subsystem (RCS) thrusters (0.2 lbf each) supporting attitude control and turns before and after the main burn. Again, thrusters so positioned do not produce balanced torques, so that all attitude control maneuvers contribute a translational Δv in addition to intended propulsive maneuvers. These small forces must be accounted for orbit determination purposes and in terms of designing propulsive maneuvers. As before, power is provided by solar arrays with a battery in reserve, limiting time at which either spacecraft can point far off Sun.

Most TCMs performed by Stardust thus far have proven difficult to predict accurately. In particular, fixed errors, originally estimated before launch to be only 2 mm/sec, 1-sigma[5], have grown as large as 5 to 7.5 cm/sec, after reconstruction of TCMs. While this level of error is acceptable for much of the mission, including the recent approach and encounter with comet Wild 2, such error is unacceptable in terms of successful delivery of the Sample Return Capsule (SRC) at Earth entry in January 2006. The larger execution error arises from "bang-bang" controlled slews and settling associated with clamping and other components of the TCM sequence itself, effects which are difficult to predict. For slews, the spacecraft changes orientation by accelerating to a maximum turn rate near the initial attitude with corresponding deceleration and settling near the target attitude. This is accomplished exclusively via RCS thrusting. These slews are used to turn to the TCM attitude, then back to Earth pointing for communications purposes. Settling Δv in particular has been difficult to model accurately, perhaps because of sloshing of fuel inside tanks and flexing of various structural components.

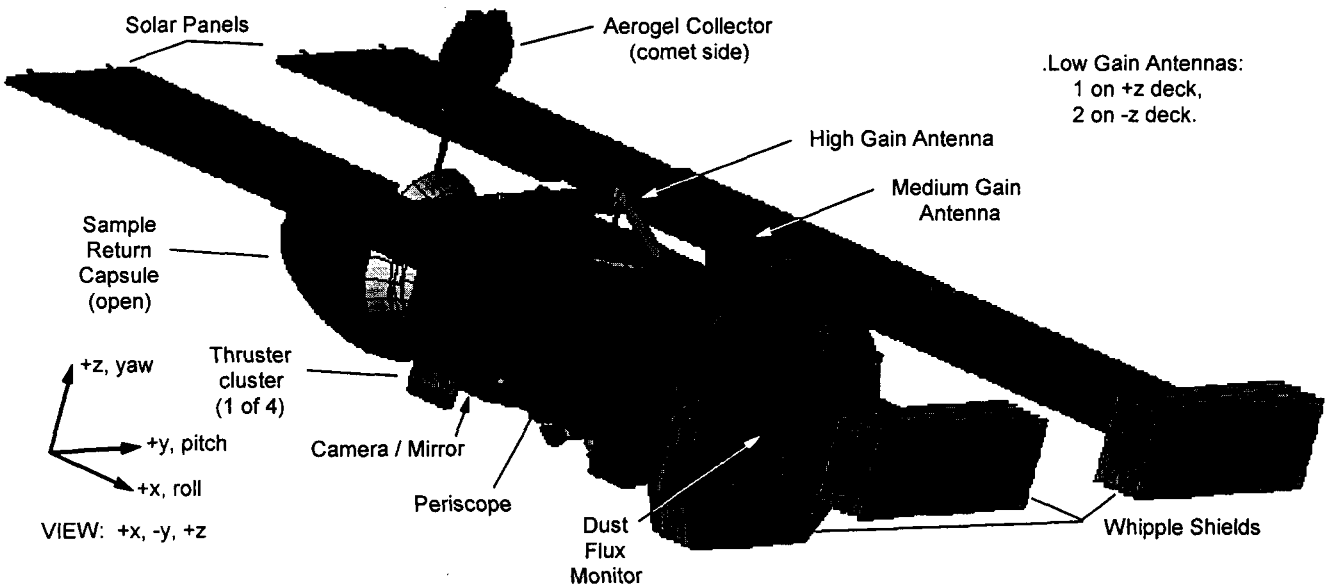

Figure 6. Stardust Spacecraft (+z Axis Normally Pointing Toward Sun or Earth)

As an alternative to "bang-bang" slews, it is possible to turn the spacecraft by slowly adjusting the deadband box to move towards a target attitude. Such deadband walks involve a maximum turn rate of about 1-1.5 °/min and are therefore practical only for attitudes close enough to the Sun such that a healthy power state can be assured over substantial time periods. With deadband walks, fixed errors comparable to the 2 mm/sec, 1-sigma, pre-launch estimates for TCMs are considered an achievable goal. Stardust closed-loop burns have proven to be fairly accurate (<1% or so, 1σ). However, errors associated with post-burn settling and slews to and from the burn attitude, as well as small forces arising from deadband limit cycling for attitude maintenance, have proven to be much larger than anticipated prior to launch.

STARDUST MANEUVER STRATEGY

The large uncertainties associated with small forces led to a redesign of the maneuver return strategy in order to return the SRC safely to Earth. A new baseline plan for Earth return, involving biased TCMs and including the introduction of a new TCM-20 at a fixed attitude compatible with conditions for SRC release, is outlined in Table 10. Further refinements to this plan are possible, once the errors are better understood, to achieve Earth entry requirements.

Because of the poor predictability of slews, it is evident that Stardust must utilize deadband walks as a primary means of reducing execution errors near earth entry. Originally, maneuvers associated with the earth entry phase of Stardust were entirely statistical.[5] However, purely statistical maneuvers are more likely to occur at attitudes far off Sun and for practical purposes are not achievable with deadband walks. As a primary means of avoiding excessive turns away from the sun, final trajectory correction maneuvers during Earth return must be biased near the Sun in a manner similar to Genesis, as indicated in Table 10. Such biasing near the Sun allows turns for final maneuvers leading up to Earth entry to be limited to deadband walks, instead of the less accurate "bang-bang" slews.

For periods between maneuvers, limit cycling or deadband control must be employed to maintain attitude within deadbands of various sizes. For the period prior to Earth return, use of ±0.25° deadbands (requiring the IMU to be activated) is planned to provide accurate attitude control for release of the Sample Return Capsule (SRC). During the bulk of the mission, larger deadbands (2°, 6° and 15°) have been preferred to limit use of the IMU within a limited operational lifetime. However, larger deadbands can often be one-sided due to solar torque effects. The RCS response to this one-sided deadbanding, and to clamping from larger deadbands down to the tighter 0.25°, is difficult to predict accurately, in part because of the randomness of the starting point within the deadband box. Maintaining a tighter deadband over a long period has the benefit of maximizing the predictability of acceleration achieved over time from RCS thruster firings, as long as the IMU lifetime is not exceeded prior to Earth entry.

Table 10. New Baseline Plan Recommended for Earth Return

Event	Epoch (ET)	Approx. Time from Earth Entry	Δv Bias (m/s)	Notes	Previously
TCM-18	02-Jan-2006 18:01:04	-13 days	1	Sunward bias direction	Unbiased (statistical only)
TCM-19	14-Jan-2006 09:58:11	-1 day	1	+z direction at SRC release (~26° off Sun), Sun in xz plane	Unbiased (statistical only)
TCM-20	14-Jan-2006 21:58:11	-12 hours	0.25-0.5	Fixed aimpoint, rolled 18° from TCM-19 bias attitude	N/A
SRC Release	15-Jan-2006 05:58:11	-4 hours	0.3408	Along Earth-based SRC velocity at 100 km altitude (effect on SRC, not S/C bus)	—
Entry	15-Jan-2006 09:58:11	—	n/a	FPA = -8.2°, Alt = 125 km, RA = 139.924°, Dec = 41.823° (latter two placeholders - TBR pending further end-to-end studies)	—

STARDUST 1-AU CALIBRATION ACTIVITIES

In the period June-July 2003 between solar conjunctions, Stardust was at a solar range of about 1 Astronomical Unit (AU) with solar radiation conditions similar to the later Earth return period. This provided an opportunity to calibrate the behavior of the spacecraft during turns and while limit cycling. Deadband walks of up to 40° as well as limit cycling at sun-pointed and possible SRC release attitudes relative to the Sun were assessed. Test TCMs known as Entry Maneuver Demonstrations (EMDs) were also performed along the Earth line of sight as a means of observing the performance of a burn itself based on the existing ACS controller.

Based on different assumptions about TCM sequence design, associated ACS controller and how well observed uncertainties might be characterized, four different capability cases have been postulated:

- **Current** – Current performance with removal of mean systematic error (may be overly optimistic based on only four samples).

- **Current Worst** – More realistic case where only minimal systematic error is assumed.
- **Improved** – An improved sequence with attitude commanded back to the original target attitude prior to initiation of the burn, minimizing settling error.
- **Improved Best** – Same as improved case, but with additional modifications to ACS controller to reduce error.

Execution errors estimated for these cases are shown in Table 11. Note that the improvements suggested for the TCM sequence are extrapolated from the 1-AU calibration results and will need to be implemented and verified as part of proposed new calibration activities in 2005 prior to Earth return. Additional details are described in the associated references. [6,7]

Table 11. Execution Errors Expected for Stardust Near Earth from 1-AU Calibrations

Case	Fixed Magnitude (m/s , 1σ)	Fixed Direction (m/s, 1σ)	Prop Magnitude (%, 1s)	Prop Direction (%, 1s)
Current	0.0035	0.0074	0.14%	1.31%
Current Worst	0.0160	0.0074	0.14%	1.31%
Improved	0.0030	0.0029	0.14%	0.14%
Improved Best	0.0030	0.0017	0.14%	0.14%

STARDUST MANEUVER ANALYSES

To assess the impact on Earth entry of observed performance during the 1-AU tests and to ensure the viability of the maneuver plan outlined in Table 10, the performance of Stardust must be assessed relative to two driving requirements for Earth entry, both specified at 125 km geocentric altitude defined as the entry interface and compatible with delivery to UTTR where the SRC will be retrieved by helicopter after deployment of a parachute and ground landing. The requirements may be paraphrased as follows:

- The Stardust SRC must pass through the entry interface at -8.2° ± 0.08° (3-sigma) flight path angle to achieve nominal aerodynamic and thermal conditions for atmospheric entry.

- The Stardust SRC must achieve a specified latitude and longitude within an elliptical keyhole with end-to-end dimensions of 33 km downrange and 10 km cross-range (both 99.7%) in order to achieve the prescribed trajectory for recovery at UTTR.

The latter requirement is a current placeholder derived from previous analyses for Genesis by LMA and NASA-Langley and is subject to further revision. All TCMs prior to Earth entry must ultimately meet such delivery requirements.

Monte-Carlo studies were performed utilizing several simulations, including both LAMBIC and Sigma. As with the Genesis analyses described previously, results from LAMBIC based on 5000 samples were applied to obtain a preliminary estimate of the flight path entry angle and entry location uncertainties after the final Earth entry TCM and SRC release activities. Assumptions include execution errors based on assessment from 1-AU tests for specific capability cases as characterized in Table 11. In each case, much larger fixed errors (7.5 cm/sec, 1-sigma) were assumed for "bang-bang" controlled slews, when the target attitude was outside a specific off-Sun angle threshold (45° and 60° off Sun were mainly considered). Only inside such a threshold could a healthy power state be assured to support slow-duration deadband walks. Larger proportional errors (2.5% in magnitude and 18 mrad in pointing, both 3-sigma) were also assumed for samples beyond the off-Sun threshold.

Table 12 provides a summary of key results for a number of cases examined. Note that only improved capability cases which use deadband walks as far as 60° off Sun are able to approach the required entry performance. In this situation, 99.1% of samples did achieve ±0.08° or smaller flight path angle error; although this does not technically meet the requirement of 3-sigma or 99.7%, it does come quite close. However, entry requirements are easily met nearly 100% of the time for a case where the off-Sun limit for deadband walks is extended to 75°, although such a condition may not be achievable due to spacecraft power constraints (45 min is the maximum off-Sun period experienced thus far).

Table 12. Key Entry Results from Selected Stardust Cases

Capability Case	*Maximum Sun Angle for Deadband Walks (deg)*	*3-Sigma FPA Error (deg)*	*Prob. Inside 0.08 deg FPA Error*	*Prob. Inside 33 X 10 km Ellipse at Entry Interface*
Current Worst	45	0.142	91.5%	89.1%
	60	0.127	93.4%	95.7%
Current	45	0.125	94.7%	90.6%
	60	0.113	97.4%	97.4%
Improved	45	0.116	96.7%	92.2%
	60	0.098	99.1%	98.2%
	75	0.053	99.9%	99.8%
Improved Best	45	0.116	96.6%	92.2%
	60	0.098	99.1%	98.2%

For the improved cases, a small number of samples (about 3%) require slews instead of deadband walks for both TCMs 18 and 19 whenever the TCM direction exceeds 60° from the Sun, while another 10% or so require slews for at least one of these TCMs. Many of these samples have a considerable cross-range error due to the larger execution errors associated with "bang-bang" slews. Based on a characterization of samples shown in Figure 7 and similar data for other cases, it is evident that samples not meeting entry requirements generally involve use of slews instead of deadband walks for at least one of TCM-18 and 19 or both. Again, use of slews is required when the burn attitude is significantly off Sun. Such samples arise when accumulated errors from earlier TCMs and small forces are significant enough to require a direction, which deviates significantly from the nominal bias direction. It may be possible to improve entry performance by biasing TCM-17 (60 days prior to entry) or introducing additional biased maneuvers prior to TCM-18 to further reduce off-Sun probability for TCM-18 and/or TCM-19.

Based on these results, it appears that Stardust is capable of achieving Earth entry conditions about 99% of the time, provided that:

1) The improved EMD sequence is implemented.

2) Deadband walks are allowed at +z-pointing angles up to 60° off Sun.

3) The UTTR ground recovery footprint is expanded to achieve the largest possible keyhole at the entry interface.

Nevertheless, it should be noted that, technically, a very slight improvement (1% or so) is still needed between now and the end of the mission in order to meet entry requirements at a 3-sigma (99.7%) level of confidence. This improvement could be realized by extending the off-Sun angle for deadband walks beyond 60°; however, it is not certain that this is feasible. Further measures have been recommended to the Stardust project to address remaining performance shortfalls.[7]

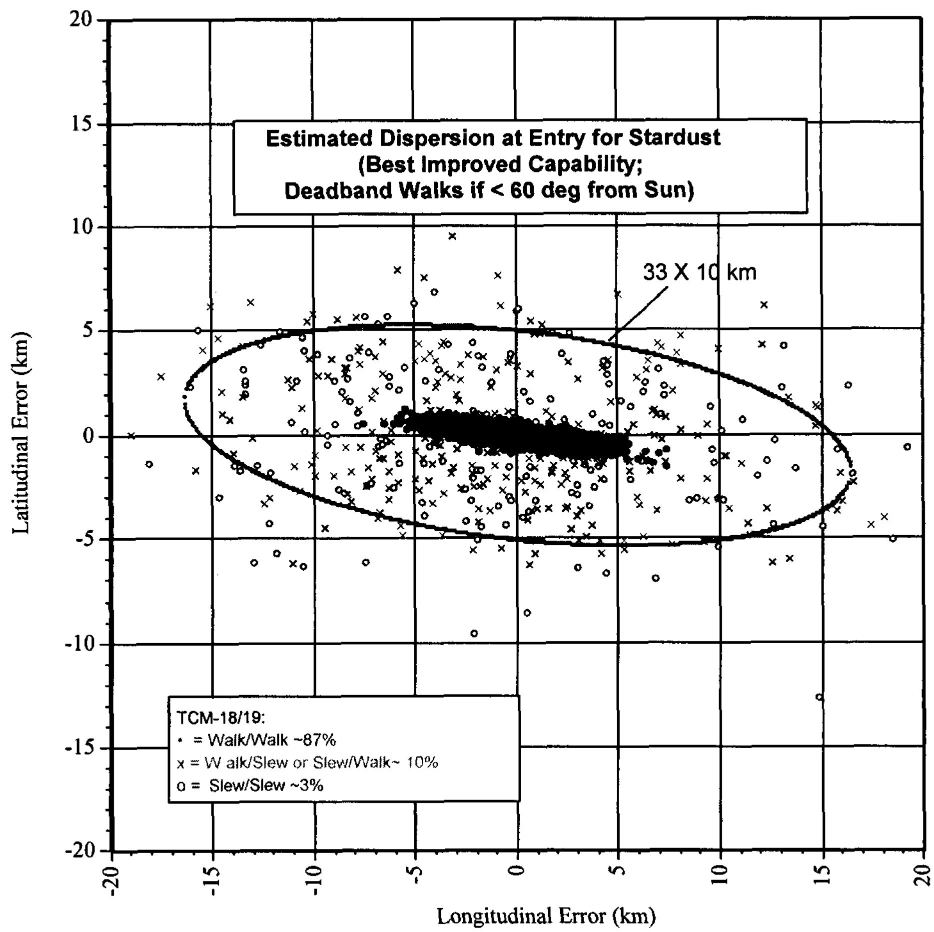

Figure 7. Monte-Carlo Results Showing Influence of Deadband Walks on Entry Targeting

CONCLUDING REMARKS

The Genesis maneuver strategy for Earth return is highly robust. Efforts over the remaining months prior to Earth return in September, 2004 will focus on fleshing out contingency plans to enhance robustness even further, as well as training and testing of an expanded flight operations team leading up to this critical period. A key element for successful completion of the Genesis mission is efficient coordination between the Mission Design and Navigation Team at JPL and the

Recovery Team at LMA and Langley. The Genesis project is highly confident of achieving a successful Earth return later this year.

The corresponding Stardust maneuver strategy for Earth return in January 2006 is currently somewhat less mature and robust than Genesis. Nevertheless, a roadmap has been established for further development of operational capabilities and plans in 2005. In particular, additional calibration activities are recommended to ensure that spacecraft performance can be predicted to a level which satisfies Earth entry requirements. With such efforts in the not-too-distant future, as well as the benefit of lessons learned from the Genesis experience, a successful return of Stardust samples to Earth appears attainable.

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