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ABSTRACT

The Genesis is the fifth mission selected as part of NASA's Discovery Program. Genesis will collect solar wind samples for a period of approximately two years while in orbit about the Earth-Sun L1 point. This paper addresses the design of propulsive maneuvers for Genesis which achieve science objectives while minimizing cost in light of a number of spacecraft design challenges and constraints. Topics to be discussed include the spacecraft design, maneuver decomposition algorithm, as well as operational procedures and plans for calibrations to improve end-of-mission performance for accurate delivery and sample recovery.

MISSION OVERVIEW AND OBJECTIVES

The Genesis mission[1] is designed to deliver the spacecraft and its instruments into a series of four halo orbits about the Sun-Earth colinear libration point, L1, located between the Sun and Earth. The prime mission is scheduled for launch in January 2000, using a Delta II launch vehicle. Genesis will spend a minimum of 22 months collecting samples of the solar wind and taking science data, mostly during the halo orbits. At the end of the collection period, the spacecraft will return to Earth and be guided to a specific target that results in a parachute recovery at the Utah Test and Training Range (UTTR) outside Salt Lake City, Utah.

The Genesis trajectory is the first to be designed using modern dynamical systems theory[2]. The spacecraft will experience a low energy injection (maximum C_3 of -0.6 km^2/s^2) into a specially determined orbit transfer to L1 which requires 3–4 months, depending on launch date. One of the unusual features of the mission design is that it requires only one deterministic maneuver, which inserts the spacecraft into the Lissajous orbit at the end of the transfer. Following completion of the Lissajous orbit phase of the mission, however, the spacecraft automatically leaves the libration point region with no departure maneuver. The spacecraft is then on a free return past Earth, that loops around the L2 libration point (on the far side of the Earth from the Sun) before returning to the Earth's dayside in August 2003 for the prime mission. A single trajectory from libration point orbit insertion (LOI) through return and recovery may be used for all launch opportunities in January.

An overview of the Genesis trajectory is provided in Figure 1, which defines mission phases and indicates locations for expected trajectory correction maneuvers (TCMs), the first of which is likely to be 24 hours after launch to avoid unacceptably large Δv cost.

In addition to maneuvers indicated in Figure 1, there are 13 halo station keeping maneuvers (SKMs), three per halo orbit plus with an additional cleanup burn after LOI. Also, there are opportunities for special calibration maneuvers, mostly during the return after halo orbit departure, which will be discussed later in this paper. Except when maneuvers are required, the Genesis spacecraft will generally remain either sun pointed early in the transfer and during the return and recovery or in the prevailing solar wind direction over

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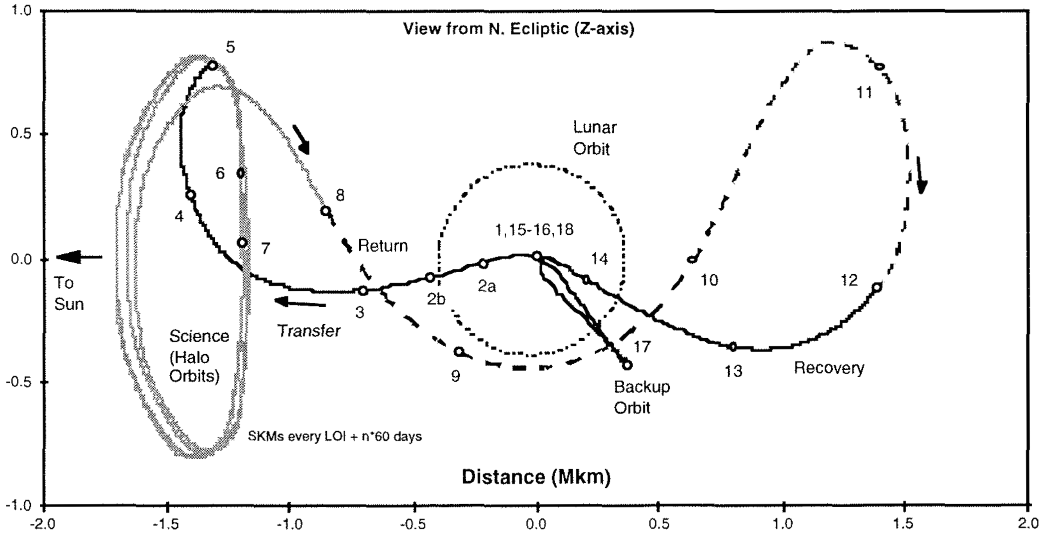
the science collection and checkout, the latter being 4.5±0.5° ahead of the sun near the ecliptic plane. Solar wind pointing requires that daily precession maneuvers of ~1° be performed during science collection and checkout to maintain this orientation to the tolerances required by the payload. Sun following precessions during non-collection periods can occur every seven days or so.



Key Events (by Earliest Date/Mission Day)

1. Launch	07jan01/000	5. Start Science	20mar01/072	10. TCM-7	24apr03/837	15. Entry	16aug03/951
2a. TCM-1a	08jan01/001	6. TCM-4	20apr01/103	11. TCM-8	10jun03/884	or	
2b. TCM-1b	10jan01/003	7. LOI (TCM-5)	30apr01/113	12. TCM-9	17jul03/921	16. Divert	16aug03/951
3. TCM-2	14jan01/007	8. End Science	28mar03/810	13. TCM-10	06aug03/941	17. Apogee	26aug03/961
4. TCM-3	11feb01/035	9. TCM-6	10apr03/823	14. TCM-11	15aug03/950	18. B. Entry	04sep03/971

Figure 1. Genesis Mission Trajectory

Upon final return to Earth, the Sample Return Capsule (SRC) will be released by the spacecraft bus, directly enter the Earth's atmosphere, and descend over UTTR for mid-air retrieval by helicopter. A deboost maneuver is performed on the spacecraft bus after release of the SRC to enable descent over the Pacific Ocean and prevent entry over a populated area. In the event of adverse conditions at UTTR, SRC release and deboost will be aborted and the spacecraft will be diverted 12 hours prior to nominal entry to a 19-day backup orbit for later entry. Sample return to Earth from L1 has never been attempted before and presents a formidable challenge in terms of mission design and operations, particularly with regard to planning and execution of propulsive maneuvers.

SPACECRAFT DESIGN AND CONSTRAINTS

To achieve a level of cost-effectiveness consistent with a Discovery-class mission, the Genesis spacecraft design was adapted to the maximum extent possible from designs used on earlier missions, such as Mars Surveyor Program (MSP) 98 and Stardust, another sample collection mission. The spacecraft design for Genesis is shown from two perspectives in Figure 2.

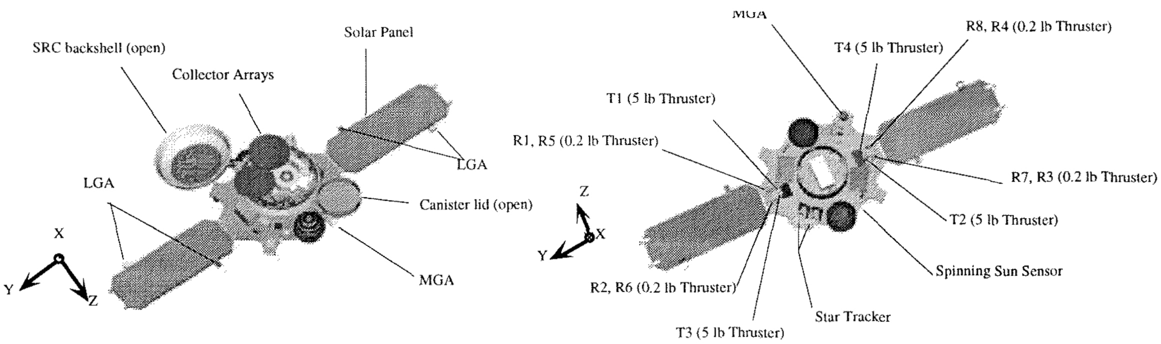


Figure 2. Forward Deck View (Normally Pointing Toward Sun) and Rear Deck View

The 646 kg spacecraft consists of the bus, SRC and two solar arrays deployed after separation from the launch vehicle. In addition to solar arrays, the electrical power subsystem (EPS) also includes a 16 amp-hour battery. To avoid battery power depletion, solar arrays are normally pointed to within 10° of the sun. For trajectory corrections, this imposes a Δv limit of about 110 m/s for each propulsive maneuver using the hydrazine fueled blowdown system, and a maneuver time limit of about 85 minutes when the spacecraft is more than 30° off sun. Total mission Δv is limited to 450 m/s (85% fuel tank capacity) and is allocated across various mission phases as shown in Table 1.

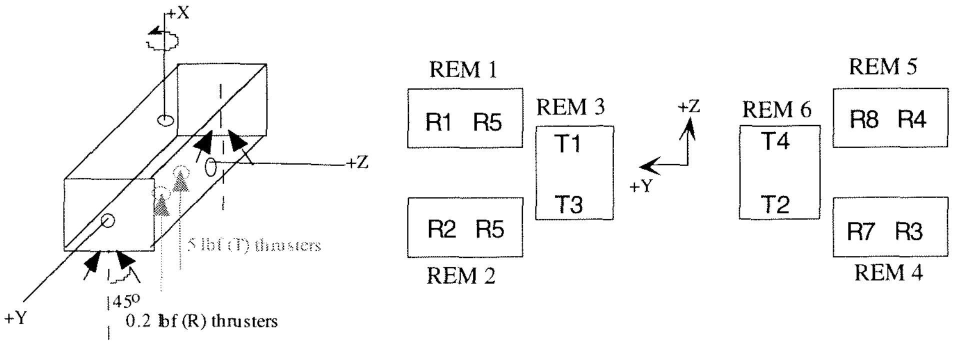
Budget Item	Mean (m/s)	Sigma (m/s)	Comments
Launch TCMs	53	43	TCM-1, 2, 3, target blocking, daily window
LOI	36	N/A	Deterministic, maximum across launch period
Halo Station keeping	10	15	Monte carlo statistics plus empirical sensitivities
Return/Entry	5	15	Monte carlo statistics plus empirical sensitivities
Deboost	20	N/A	Deterministic, s/c bus to Pacific Ocean post-release
Backup Entry	91	TBS	Deterministic, statistical analysis pending
ACS	64	20	Simulations and Statistical analysis
TOTAL	279	52	RSS (column entries, except N/A)
95% Mission Success		383	Mean + 2*RSS of sigmas
Margin		67	15% of total capability
Total Capability		450	85% tank capacity

TABLE 1. Δv Budget for Genesis Mission

After initial maneuvers and spacecraft checkouts are performed, the SRC backshell and collection canister lid will be open as shown in Figure 2. To minimize contamination risk to the samples collected for maneuvers other than LOI, all thrusters have been placed on the side opposite the sample collection canister, resulting in an unbalanced configuration. Moreover, cumulative fuel expenditure during collection periods is limited to a total of 30 kg (equivalent to Δv of over 100 m/s). However, for any large maneuvers, the SRC backshell and canister lids must be closed and latched, then reopened after the maneuver is completed. Nominally, this will only be required for the LOI maneuver. The SRC and collection canister lids will be closed and latched one final time when collection is finished at the end of the science collection.

Spin stabilization was chosen as a simple means of attitude control, in lieu of three-axis stabilization. No reaction wheels, gyros or accelerometers are included in the attitude control subsystem (ACS). The space vehicle normally spins at 1.6 revolutions per minute (rpm). Star trackers and sun sensors support attitude determination, but only the latter are effective when spinning at greater than 2 rpm. Thrusters are located on the opposite side of the space vehicle from science instruments to minimize contamination of samples over the course of solar wind collection. Any attitude changes, including spin changes and precessions, must be performed open loop with thrusters. Because the thrusters do not produce balanced torques, all attitude control maneuvers result in a translational Δv, affecting the spacecraft trajectory. Thruster activity, asymmetric mass properties and misalignments also induce oscillations in the spin vector, such as nutation and wobble.

The orientation and characteristics of thrusters needed for guidance and navigation are indicated in Figure 3 and Table 2. There are two separate thruster strings, a primary (1) and redundant (2). For large maneuvers ($\Delta v > 2.5$ m/s), two 5 lbf thrusters (I_{sp} of 210-234 sec) will be used for the burn or translational maneuver, with an initial precession to the burn attitude using alternating pairs of 0.2 lbf thrusters (I_{sp} of 198-222 sec), followed by spin up from 1.6 to 10 rpm with 0.2 lbf thrusters before the burn. The higher spin rate increases stability during large maneuvers and reduces execution errors. Following the burn itself, the space vehicle will precess back towards the initial near-sun pointing attitude. Spin down to 1.6 occurs after precession-induced nutation has damped out. For small maneuvers ($\Delta v < 2.5$ m/s), no spin changes are needed and four 0.2 lbf thrusters are used for the burn. 360° precession changes may be required to remove undesired components introduced by the precession itself. Another maneuver mode, which will be employed prior to recovery, involves use of spin changes in lieu of previously described translational maneuvers to achieve high delivery accuracy.



Notes:
All thrusters are located on aft (-X) side, which usually face away from sun and toward s Earth.
T refers to the large 5 lbf thrusters (thrust axis toward +X)
R refers to the small 0.2 lbf thrusters, canted 45° off -X in the X, Z plane.
For precession maneuvers, use a single set, e.g., R1 and R2, once a spin cycle, or two sets, e.g., R1, R2 and R3, R4 in alternating half cycles for faster precessions, but with greater nutation at the end of precession.

Figure 3. Genesis Thruster Configuration

String 1 (Primary)	String 2 (Redundant)	Maneuver Usage
R1+R2 and R3+R4	R5+R6 and R7+R8	Precession (alternating pairs each half spin cycle)
R2+R4	R6+R8	+X spin up
R1+R3	R5+R7	+X spin down
R1+R2+R3+R4	R5+R6+R7+R8	$\Delta v < 2.5$ m/s
T1+T2	T3+T4	$\Delta v > 2.5$ m/s
T1 and T2	T3 and T4	Rapid Precession (alternating each half spin cycle)

TABLE 2. Thruster Combinations for Genesis

The telecommunications subsystem employs S-band uplink and downlink and includes low gain antennas (LGAs) directed both forward and aft plus a medium gain antenna (MGA) pointed in the aft direction. Navigation is performed with S-band Doppler and ranging data; this is sufficient to reconstruct any propulsive maneuver to within 15 mm/s (3σ) within 14 days of execution.

The spacecraft antenna coverage is limited to no more than 60-70° from either the +X or –X spacecraft axis, due to the effect of shadowing by the solar panels and other structures. While such a blackout zone is unavoidable for many maneuvers, much consideration has been given to minimizing the blackout time during a maneuver sequence, to reduce the risk of a potentially unrecoverable anomaly, particularly in light of off-sun power restrictions.

The Genesis spacecraft is also equipped with nutation dampers (not shown in Figure 2). Depending on the size and direction of the maneuver required, nutation could grow as large as 20°. As much as 6 hours could be required to allow damping to less than 0.5° of nutation, as well as full recharge of batteries. Therefore, a period of at least 12 hours for interruption of science collection will be reserved for execution of TCMs and SKMs.

MANEUVER PLANNING PROCESS

Operationally, propulsive maneuvers involving trajectory correction or maneuver calibration will be planned and executed via the process illustrated in Figure 4. Maneuver design will be performed jointly by the Navigation Team at JPL and the Spacecraft Team at LMA, in light of the aforementioned constraints. This process will normally take about 1–2 days before a propulsive maneuver. However, because of the time criticality of TCM-1, orbit determination and maneuver design must be completed within three hours after Doppler and ranging data cutoff.

An important function in this process is to decompose maneuvers into constituents to account for both the translational burn and spacecraft attitude changes required to carry out such a maneuver. Although the Spacecraft Team will have primary responsibility for this function, both the Navigation Team and Spacecraft Team will have software available which performs this function to support maneuver analyses, operational readiness tests (ORTs) prior to launch and for purposes of verification or operational backup. Attitude changes may include precessions, as well as changes to the spacecraft spin rate, such as those required to increase stability for large maneuvers. Maneuver performance and design account for spin rate, precession angle and burn time effects on Δv. A maneuver decomposition algorithm[3] has been developed for use by the Navigation Team. The Spacecraft Team will use a similar algorithm and other maneuver design models to support maneuver sequence planning.

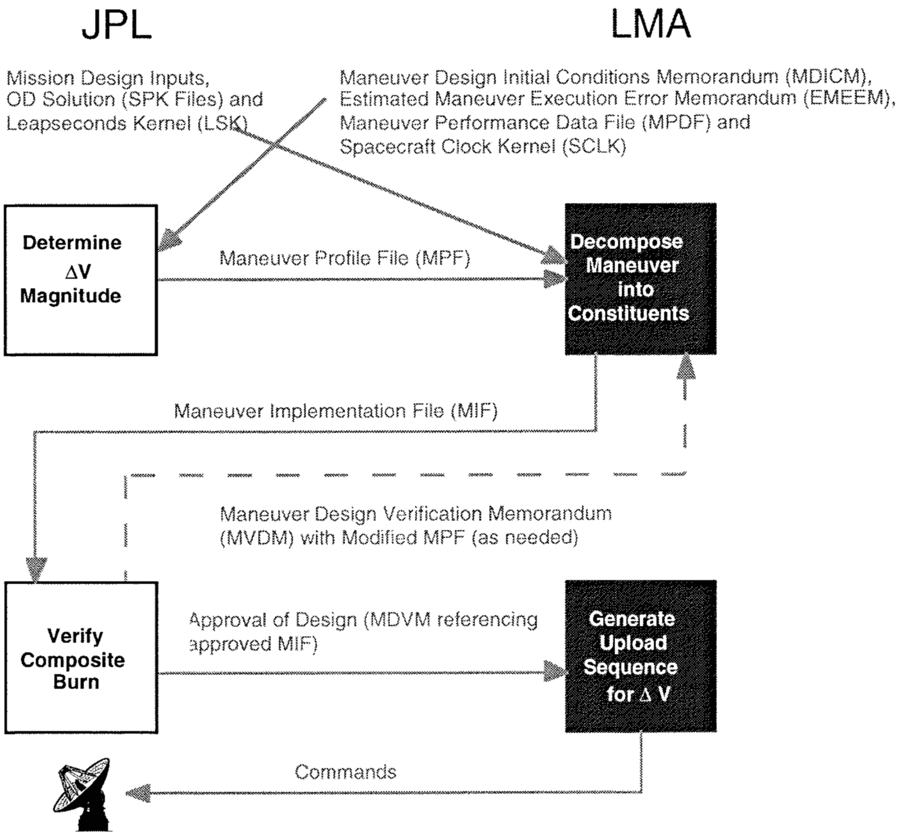


Figure 4. Genesis Maneuver Design Process

In the case of precessions, Δv is obtained from a series of pulses which occur each revolution or half-revolution. The cumulative effect of these pulses is to produce a Δv vector which lies along the chord of a turn circle in spacecraft centered Δv space (or along the +X body axis), as illustrated in Figure 5. The circumference of a turn circle traces out induced Δv resulting from a series of thruster pulses. Thus, a 360° turn in the absence of execution errors results in a net induced Δv of zero.

The diameter v_0 of the two-way turn circle shown in Figure 5 can be derived from mass properties and characteristics of the spacecraft as follows:

$$v_0 \cong \frac{4 I_{xx} \omega_x}{rm} \tag{1}$$

where I_{xx} is the spacecraft moment of inertia about the spin axis (x), ω_x is the component of angular velocity along the spin axis, r is the lever arm associated with either a 5 lbf thruster or 0.2-lbf thruster pair and m is the spacecraft mass[4]. Whenever the desired Δv falls inside the two-way turn circle (typically $v_0 <$ 0.5 m/s for Genesis), Δv resulting from spacecraft precession cannot be used effectively as a component of the desired Δv, and must therefore be removed by precessing at least one complete revolution around the turn circle. Most TCMs and SKMs are estimated to be small enough that 360° precessions might be required most of the time. However, biasing strategies are currently under investigation to attempt to reduce the number of such precessions that might otherwise be required.

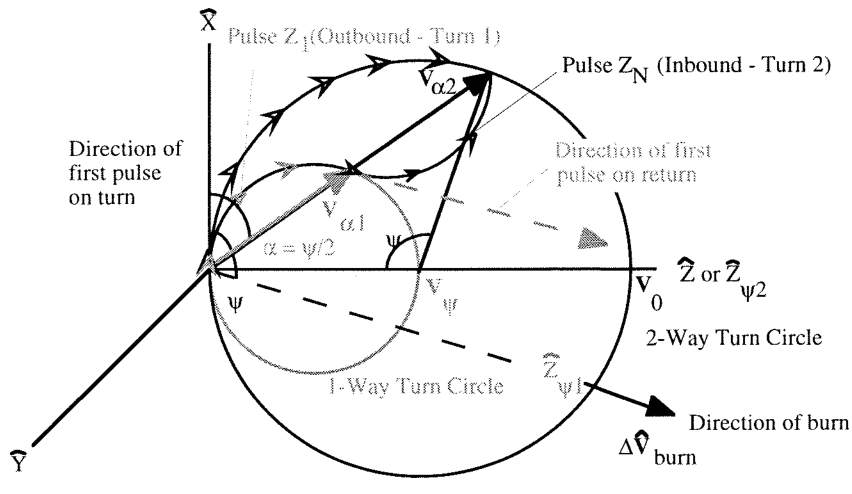


Figure 5. Turn Circle Geometry for Spinning Spacecraft

MANEUVER PERFORMANCE AND CALIBRATIONS

Maneuver calibrations are planned to update maneuver performance and design models, especially those associated with precession and spin rate changes. This will improve predictability and reduce execution errors for both propulsive and attitude control maneuvers. Maneuver errors in spin axis direction, spin rate and Δv are caused by thruster misalignments, knowledge (or sensor) errors, and other performance errors, such as mechanical failure or degradation. Uncalibrated execution errors for the baseline design shown in Table 3.

Such errors could be reduced over the course of the mission through observation and reconstruction of nominal maneuvers. However, many such maneuvers will not provide sufficient observability to derive the level of accuracy needed. Propulsive maneuver sequences merge a number of effects together, such as overlapping thruster combinations and mixing precession, spin change and translations. Routine attitude control maneuvers for satisfying sun or solar wind pointing constraints are generally too small or not sufficiently separated in time to permit accurate reconstruction of individual events via available radiometric tracking. Therefore, special calibration maneuvers are planned, especially during return in preparation for sample return and recovery. Note that conflicts with science activities are minimized as much as possible without placing navigation requirements and constraints at risk.

The most important goal of maneuver calibrations is to ensure sufficient robustness for meeting SRC entry requirements near the end of mission. These entail 3σ tolerances for entry angle of no less than ±0.08° (±0.05° of which is allocated to Navigation) as well as for placement within an 18 km (downrange) by 2 km (crossrange) target ellipse with a velocity uncertainty of ±13 m/s at the 125 km interface altitude. These tolerances are required to ensure sample return within the 84 km by 30 km recovery footprint at UTTR (all tolerances 3σ). The key error drivers are TCM-10 and TCM-11 at ten days and one day prior to entry, respectively, as well as SRC separation at entry minus four hours. TCM-10 and TCM-11 would employ a specific maneuver mode based on spin rate changes to produce a translational Δv. They would also use sunward biasing to avoid the use of a 360° precession, while keeping the spacecraft pointed to within 30° of the sun to ensure adequate power, but not pointed any closer than 5-15° from the sun to avoid sun sensor aliasing in the presence of nutation.

Maneuver Component	Error (Fixed)
Small Angle Precession ($\leq 10°$; error applies one way for pointing corrections)	≤ 0.0002 m/s (3σ) per axis per degree of precession
Large Angle Precession ($>10°$ and $\leq 180°$, error applies for entire two-way precession)	≤ 0.04 m/s $\times$ (1-way precess angle/180°) (3σ) along original spin pointing direction; ≤ 0.03 m/s $\times$ (1-way precess angle/180°) (3σ) per crosstrack axis with respect to original spin pointing direction
Spin Change (error applies to total of up and down spins)	≤ 0.06 m/s $\times$ (spin change/16 rpm) (3σ) along spin axis; ≤ 0.01 m/s $\times$ (spin change/16 rpm) (3σ) per crosstrack axis
Translational Δv Maneuver (after precession/spin-up and before precession/spin-down)	*Use values indicated in panel below.* Initial nutation angles are $\leq 10°$; for $\Delta v < 2.5$ m/s and SRC closed, assumes end of life performance in terms of mass properties and propulsion performance

Translational Δv Usage criteria	Thrusters	Magnitude proportionality %	Fixed magnitude m/s	Crosstrack proportionality % (per axis)	Fixed Crosstrack m/s (per axis)
$\Delta v > 2.5$ m/s; SRC closed	two 5 lbf	6.0	0	3.0	0
$0.05 < \Delta v < 2.5$ m/s; SRC open	four 0.2 lbf	6.0	0.01	4.0	0.005
$0.05 < \Delta v < 2.5$ m/s; SRC closed[†]	four 0.2 lbf	6.0	0.01	3.0	0.003

TABLE 3. Uncalibrated Maneuver Execution Errors

Spin change can be characterized extremely well, because a spin change of magnitude $\Delta\omega_x$ produces a Δv_{spin}, along the +x axis and inertially fixed, according to the following relationship:

$$\Delta v_{spin} \cong \frac{I_{xx}\Delta\omega_x}{rm} \tag{2}$$

The thruster lever arm r is well defined beginning before launch. Mass properties are also fairly well defined at launch, but will change over the course of the mission. Because there are no accelerometers on Genesis, Δv_{spin} cannot be determined by the on-board ACS software. However, through observation of a spin change event from the ground coupled with spin rate estimates available from telemetry throughout the event, the spacecraft mass property I_{xx}/m can be established very accurately. In particular, studies[5] have demonstrated that reduction of all fixed errors to 3 mm/s and proportional errors to 1% would permit all of the aforementioned entry requirements to be met with reasonable performance margin.

To establish a high degree of accuracy in characterizing Δv_{spin}, the spacecraft spin axis should be along the line of sight (LOS) to the Earth. This geometry affords direct observation of the event on the ground through radiometric tracking. Doppler measurements with 1-3 mm/sec accuracy are achievable in S-Band if the spacecraft spin axis is 1-2° of the Earth LOS for a tracking duration of 2-4 hours. Power and telecommunication constraints allow favorable alignment only during certain portions of the mission when the Earth-spacecraft-sun geometry is favorable. Although not useful in preparation for entry per se, the first opportunity (18-19 days after launch) could be used as a proof-of-concept test and applied to the design of LOI and various SKMs, dependent on other mission priorities. Two opportunities during return (one week periods beginning around mission days 840 and 910) would allow spin up and down to be tested for both thruster strings in case of a thruster failure. Induced Δv from each of the calibration periods is expected to be no more than 1 m/s, but must nevertheless be removed by subsequent TCMs.

A secondary objective of calibrations is to better characterize precession maneuvers to improve pointing performance for science and for other TCMs. Depending on fuel usage early in the mission, such calibrations could also prove crucial to maintaining the Δv or fuel budget for the mission and avoiding costly mission redesign. Such maneuvers are part of most propulsive maneuvers, but are also required periodically to maintain sun-pointing for power or daily during solar-wind pointing during collection periods. Although relatively small, such maneuvers will have a significant cumulative impact on orbit determination, particularly in the halo portion of the mission. The effect of such maneuvers can be deduced indirectly via two-way ranging, trending and trajectory reconstruction over periods of days to weeks to an error of no greater than 15 mm/sec with some improvement (< 50%) in predictability and execution error possible. Of particular importance, precessions of about 10-30° will need to be performed in conjunction with the pre-entry TCMs mentioned previously. Characterization of Δv resulting from precessions of this magnitude is critical to ensure that execution errors are still kept to a minimum. Although use of 0.2 lbf thrusters is most likely, high fuel usage could cause tank pressures to drop significantly towards the end of mission, possibly requiring exclusive use of the larger 0.5 lbf thrusters. Such maneuver calibrations can be superimposed on precessions required for sun-pointing or telecommunications maintenance, such as those on mission days 812, 826, 829, 836, 848, 864 and 894. Ideally, each event would be preceded by two radiometric tracks, concurrent with an additional DSN pass and followed by two more, each on separate days and four hours duration, with no other maneuvers of any kind occurring over the tracking arc. However, usage of ground-based resources is likely to be restricted. Reconstruction accuracy of 3-10 mm/sec should be possible, enabling >50% improvement in execution errors associated with such precessions.

Priority	**Maneuver**	**Thruster Type/String**	**SRC Lid (ω in rpm)**	**Mission Application**
P1	Spin rate change	R/1	Closed (1.6↔8.5)	Biased final TCM
P2	Spin rate change	R/2	Closed (1.6↔8.5)	Biased final TCM(redund.)
P3	Small (~ 10°) precession	R/1	Closed (1.6)	Biased final TCM, SRC release
P4	360° precession	R/1	Closed (1.6)	Backup final TCM, SRC release
S1	Small (~ 10°) precession	R/1	Open (1.6)	Biased SKM, daily precessions
S2	Small (~ 10°) precession	T/1	Closed (1.6)	Biased final TCM, high fuel usage
S3	Small (~ 10°) precession	R/2	Closed (1.6)	Biased final TCM(redund.)
S4	Small (~ 10°) precession	R/2	Open (1.6)	Biased SKM, process (redundant)
S5	360° precession	T/1	Closed (1.6)	Backup final TCM, high fuel usage
S6	360° precession	R/2	Closed (1.6)	Backup final TCM(redund.)
S7	360° precession	R/1	Open (1.6)	Unbiased SKM
S8	360° precession	R/2	Open (1.6)	Unbiased SKM (redundant)
S9	Translational Δv < 2.5 m/s	R/1	Closed (1.6)	Backup final TCM, small TCMs
S10	Translational Δv < 2.5 m/s	R/1	Open (1.6)	SKM
S11	Translational Δv > 2.5 m/s	T/1	Closed (1.6)	Large TCMs
T1	Small (~ 10°) precession	T/2	Closed (1.6)	Biased final TCM, high fuel (redundant)
T2	Translational Δv < 2.5 m/s	R/2	Open (1.6)	SKM (redundant)
T3	360° precession	T/2	Closed (1.6)	Backup final TCM, high fuel (redundant)

TABLE 4. Prioritized Calibration Events

The calibration methodology requires both predicted and observational data. Prior trending and performance assessment data, where relevant, provide some validation and context for calibration results. Predicted data encompass initial and final conditions and include thruster selection and planned usage, pointing and spin rate knowledge and control, spin rate change(s), precession angle(s), burn times, past performance (thrust, Isp, etc.), duty cycle, trajectory state, Δv (induced and planned) and mass properties with fuel state. Observational data can encompass pre-maneuver, during maneuver and post-maneuver. These include spacecraft telemetry in the form of propulsion information (tank temperatures/ pressures and engine pulse timing) and ACS data, including principal axis misalignment, sun sensor measurements (spin rate/direction and off-sun angle) and star tracker data (attitude quaternions and angular momentum). Observed data from tracking and navigation include tracking data (range and Doppler), and actual Δv and orbit determination results based on direct Doppler and/or Δv reconstruction (short and long arc solutions with optimal weightings). Navigation data is used with predicted performance and telemetry data to update maneuver performance and design models, including an estimate of confidence or uncertainty level.

Priorities for various calibration events proposed to date are shown in Table 4. Columns indicate priority (P = primary, S = secondary, T = tertiary); desired calibration maneuver conditions, thruster type and string, spacecraft configuration (SRC closed or open with and canister open with collector arrays deployed) and spin rate, and finally where in the mission the calibrated maneuver data is used. The calibration configuration is designed to be as close to the target mission application as possible, including the fuel state. Ideally, calibration maneuvers would exercise as many thruster combinations as possible. Although 360° precessions will be avoided as much as possible, calibrations for these might be performed to improve the accuracy of smaller propulsive maneuvers which may require such precessions as a backup mode in the absence of biasing. In addition, although not completely isolated from other events, TCM-1 and LOI will provide an opportunity to observe the effects of the 5 lbf thrusters. In particular, Earth LOS might be favorable for direct observation of separate components of TCM-1.

CONCLUSION

For the Genesis mission, an approach to maneuver design and calibration has been developed which meets spacecraft design challenges and constraints to date. Additional studies are underway to address operational contingencies and further enhance the robustness of the overall design to ensure mission success.

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REFERENCES

[1] Martin W. Lo, et.al "Genesis Mission Design," AIAA 98-4468, AIAA/AAS Astrodynamics Conference, Boston Massachusetts, August 10-12, 1998.

[2] Julia L. Bell, Martin W. Lo, Roby S. Wilson, "Genesis Trajectory Design," AAS 99-398, AAS/AIAA Astrodynamics Specialist Conference, Girdwood AK, August 16-19, 1999.

[3] Kenneth E. Williams, "Maneuver Decomposition Algorithm," JPL Interoffice Memorandum 312.H-99-011, July 26, 1999 *(JPL Internal Document).*

[4] M. H. Bentell, Jr., "Control Mode Synthesis for Galileo Spacecraft Maneuvers," AIAA, 1980.

[5] Document GN-61000-200, "Genesis Planning, Control, Analysis and Recovery Reference Document," Volume 19 (third release), July 23, 1999 *(JPL Internal Document).*