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GENESIS BACKUP ORBIT CONTINGENCY ANALYSIS

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Abstract

The Genesis spacecraft returned its sample return capsule to Earth on Sept 8, 2004. In the event the capsule did not release as scheduled, a contingency plan was in place to offer an alternate, backup capsule-return-strategy subsequent to the first attempt.

INTRODUCTION

The nominal Genesis mission navigation is described in full elsewhere [Refs 1,2,3]. In preparation for the recovery of the source return capsule (SRC), Genesis planners also designed an alternate, delayed Earth-return trajectory as a contingent strategy in case of a possible SRC-release abort.

A contingent-return option existed because Genesis did not follow a hyperbolic trajectory with respect to Earth. In traveling to and from the Sun-Earth L_1 Lagrange Point, Genesis never fully escaped from Earth and, conceptually at least, the approach to Earth in August and September 2004 could be viewed as the inbound portion of a highly elliptical, Earth-centered orbit. Therefore because of the astrodynamics involved, in principle Genesis would be able to encounter Earth repeatedly on many subsequent opportunities (albeit with significant effort as we show below).

A backup orbit needed to be practical and feasible both in terms of risk and flight team workload. Such an orbit was discovered, but it pushed the bounds on what can be considered an acceptable workload [Ref 4]. Nevertheless it satisfied all other criteria for eligibility.

The criteria to be satisfied before a backup orbit could be accepted included the following requirements. First, the target and entry conditions needed to meet the constraints levied for the capsule return (most of the constraints were similar to the first SRC-return attempt). Second, the physics of returning to Earth had to be modelable. Third, navigating the backup orbit (operating the spacecraft) had to fit within Genesis' resources.

END CONDITIONS AND CONSTRAINTS

Two dominating end conditions were the time and location of the recovery. Targeting to a fixed site on Earth satisfied one of the end conditions (*i.e.* the landing needed to remain at the same location as the first attempt, namely UTTR near Salt Lake City, Utah). Targeting to a daytime entry satisfied the second end condition because the entry, descent, and recovery needed to occur in daylight for a mid-air capture. An operational constraint required that the propellant available to Genesis be sufficient to reliably navigate the orbiter to the backup entry. Another (softer) operational guideline limited the length of the backup orbit to less than one year.

The analysis did not, in reality, aim for the recovery point on the surface. The trajectory was designed to reach a target point at the top of Earth's atmosphere (the entry interface point, or EI). The recovery time at UTTR would be approximately twenty-three minutes after the entry time. See Table 1. The entry, descent, and landing portion of the mission (EDL), for the purposes of this analysis, were assumed to be equivalent to the nominal case, when given an equivalent state at the EI. (The EDL phase is not reported in this paper.)

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Table 1. Genesis Backup Orbit End Conditions
Entry Interface time: March 17, 2005 21:36:36.4 UTC

EI Target					Recovery Point		
Altitude* (km)	Latitude (deg)	Longitude (deg)	eFPA (deg)	Entry Speed (km/s)	Latitude (deg)	Longitude (deg)	Time (min wrt EI)
135.466	**44.29 N**	**122.02 W**	**-8.00**	**11.052**	**40.20 N**	**113.53 W**	**+23.5**

*Radius = 6503.140 km

TRAJECTORY DESIGN
The backup trajectory was designed using dynamical systems theory coupled with a two-level differential corrections process, subject to the entry-time constraint. Other references can provide details on the procedures employed to determine suitable trajectory solutions [Refs 5,6].

For the nominal mission, a divert maneuver was planned approximately 3.5 hours prior to SRC entry (after the SRC had separated from the spacecraft). The purpose of this maneuver was to avoid a collision between the spacecraft bus and Earth. The maneuver was sized to place the bus on an eccentric 58 day Earth-return trajectory, after which the spacecraft would escape from the Earth-Moon system and enter a heliocentric orbit at approximately 1 AU radius.

The same divert maneuver was used to initiate the backup orbit. For the backup orbit, the SRC will *not* have separated from the rest of the spacecraft and the mass of the spacecraft will be larger. The higher mass yields a smaller divert maneuver and therefore the orbital period for the initial backup loop will be greater than 58 days. Ultimately, the spacecraft has to be re-positioned for a second attempt to re-enter the atmosphere. The SRC state at the time of EI must satisfy the requirements listed in Table 1.

To find the backup orbit, a design space was needed in order to parameterize and characterize all possible orbits. The design space selected consisted of the initial state and time, the EI state, and a propellant budget. The initial state and time were known from the state of the spacecraft at the time of the divert maneuver. The desired entry conditions were known from EI requirements, and the available propellant was known. But the entry time was unknown.

Iterating on the first four EI parameters in Table 1 determined the entry time. The first four EI entry parameters were initialized with *a priori* estimates. Then, starting with the *a priori*, a trajectory solution was found by iteratively adjusting third body interactions, ΔV impulses, and the end conditions until altitude, latitude, and flight path angle converged (tolerances were 0.001° and 1 m). Then right ascension at entry could be determined and a longitude derived. If the longitude, ΔV, or flight time were unacceptable, the calculation was seeded with a new arrival time and the search repeated until the longitude tolerance was met. The entry time was then known, *ipso facto*.

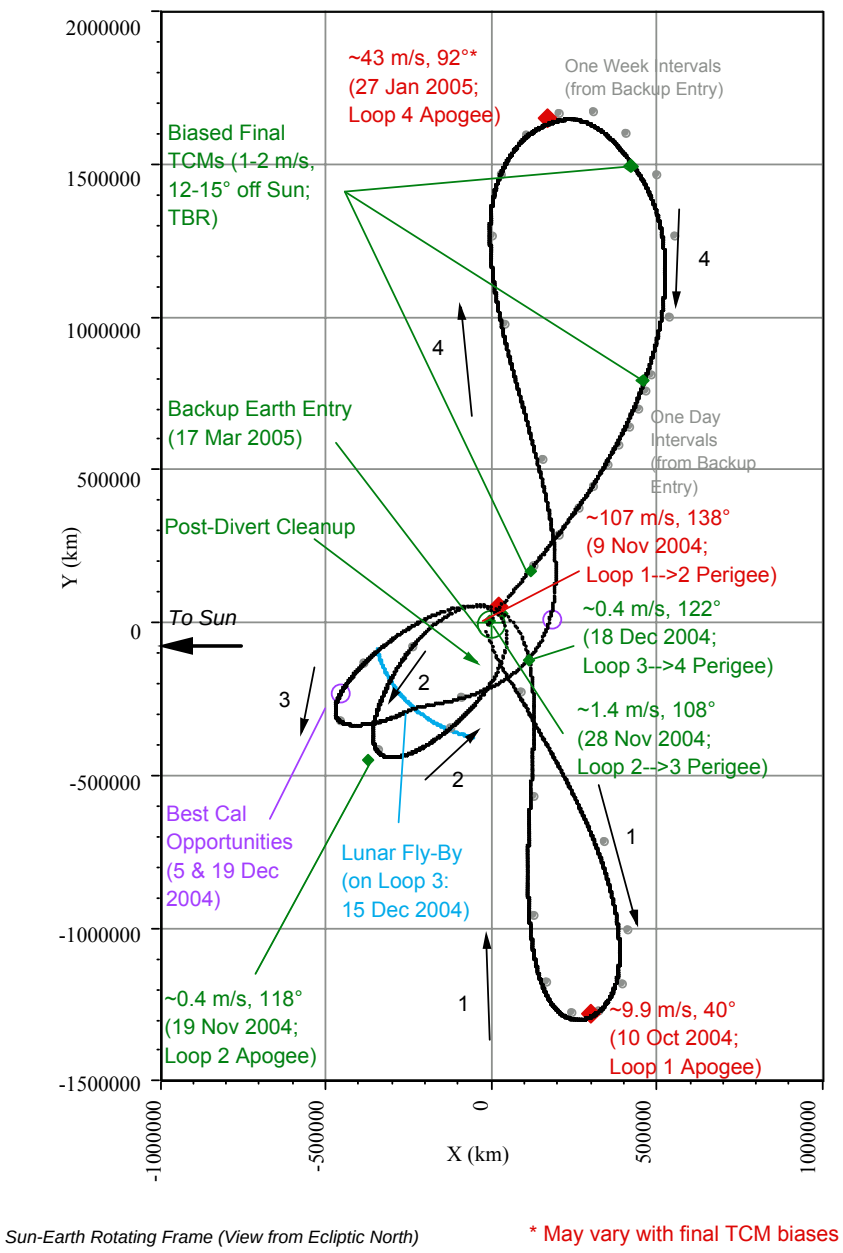
The LTool software set
was used for the design. LTool was developed at JPL and Purdue University specifically for the Genesis mission. It allowed broad explorations of the design space in relatively short intervals of time. Mission designers used LTool for the nominal mission as well as for the backup mission. See previous papers on Genesis mission design for details on the methodology [*e.g.* Refs 5,7].

Once a potential backup orbit had been determined using LTool, another JPL program called CATO optimized the solution and refined the propagation. The CATO trajectory was verified in turn using the JPL navigation software DPTRAJ. Other analyses also were performed to examine geometric situations such as eclipses or range variations. Finally, the feasibility of the solution was assessed by its fidelity in meeting the design requirements. The validated trajectory then served as the basis for flight operations.

The backup option selected was a six month trajectory with a lunar swing-by [Ref 4]. The orbits are shown in Figure 1. The trajectory is plotted in a rotating coordinate frame that keeps the X-axis coincident with the line from the Sun to Earth. The Z-axis is perpendicular to the X-axis in the direction of the normal to the orbit plane of Earth about

the Sun, and the Y-axis completes the orthonormal triad. As shown in the Figure, the divert maneuver on September 8 places the spacecraft into a 62 day orbit. (The divert maneuver is not shown in Figure 1 because of the scale of the figure.) A maneuver at the subsequent perigee reduces the orbital period and sets up for the lunar swing-by on December 15, 2005. The swing-by is necessary to establish the final phasing for returning the spacecraft to Earth with the desired entry conditions on March 17, 2005. Notice that the orbit petals are sequentially numbered, and arrows indicate the direction of spacecraft motion.

In summary, the trajectory is complex, but was selected by the Genesis project to be the best backup candidate in terms of maneuver cost and flight time.



Sun-Earth Rotating Frame (View from Ecliptic North)

* May vary with final TCM biases

Figure 1. Genesis Backup Trajectory.
Magnitude & direction of each maneuver is indicated on the plot.
(Magnitude equals deterministic component only; direction is with respect to the Sun-line.)

NAVIGATION

Navigating the backup mission was expected to be more difficult than the nominal because of its complicated trajectory and intense maneuver schedule (*i.e.* eleven maneuvers within six months compared with six maneuvers during the last six months for the nominal). Moreover, the magnitudes of several backup maneuvers were appreciably larger than any maneuver executed during the nominal mission. Our analysis also uncovered maneuver design issues that required changes in operations planning and trajectory re-design. We present some of these issues below.

Orbit Determination

Our spacecraft model was consistent with the nominal mission, as were the attitude correction impulses and station-keeping maneuvers [Refs 1,2]. The trajectory correction maneuver (TCM) sequence is provided in the Maneuver Design sub-section. All error analysis parameters, including SRC release activities important to navigation (spin-ups, turns, precession to attitude), are listed in Appendix A.

The tracking schedule assumed for the backup mission is shown in Table 2. This schedule is consistent with historical tracking during the nominal mission.

Table 2. Doppler & Range Tracking Measurements

Activity	Start	End	Radiometric Coverage
Nominal	E-190days	E-0	3 tracks/week
Maneuver Delivery	DCO-7	DCO*	1 track /day
Maneuver	M-1	M+1	3 tracks/day

*DCO =data cut-off. DCO occurs 7 days before a maneuver (M).

The spacecraft's position uncertainty as a function of time is shown in Figure 2. The uncertainty is represented by position eignenvalues, with maximum and minimum eigenvalues plotted versus days from Earth-entry. As shown in the figure, the spacecraft's position is determined to a high degree of certainty at all times.

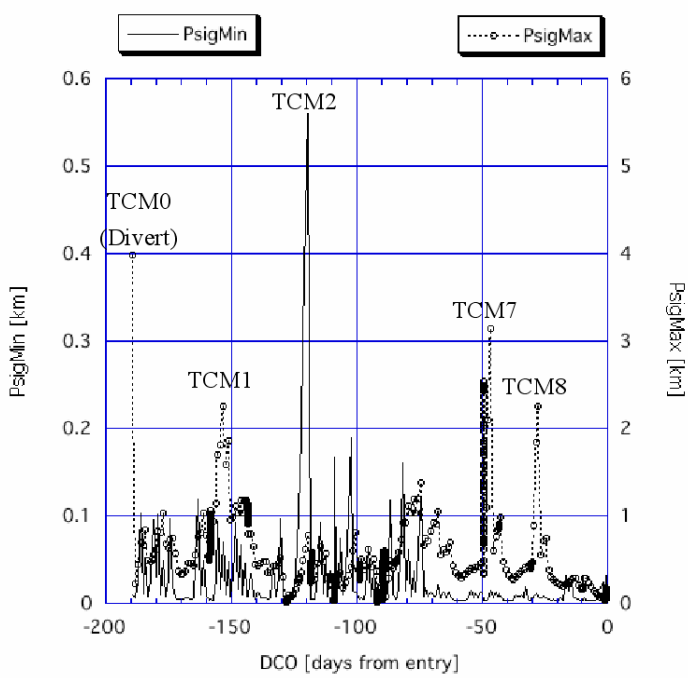


Figure 2. Current-time Position Covariance Eigenvalues (1σ).
PsigMin represents half-width of the shortest error ellipsoid axis.
PsigMax represents half-width of the longest error ellipsoid axis.

Maneuver Design

The primary objective of this analysis was to determine a propellant budget for the backup orbit. Another objective was to identify infeasible maneuvers as well as any high-risk implementation issues.

A propellant budget was estimated by combining spacecraft state uncertainties (as determined from a time series of tracking data) with anticipated maneuver execution uncertainties (caused by performance limitations of the engine), thereby yielding a total state error covariance for the maneuver. The state uncertainty at a future date (*e.g.* at an upcoming encounter) was determined by mapping the covariance to that date. Monte carlo methods computed statistical components of each trajectory correction maneuver. The propellant budget equaled the total ΔV from all TCMs due to both statistical and deterministic components.

Table 3 (and Figure 1) summarizes the deterministic trajectory characteristics. The backup trajectory had more maneuvers than the nominal mission over an equal period of time, and the ΔV magnitudes were significantly larger [Ref 2]. (This applied to the statistical ΔV components also.)

Through these analyses it became clear that a departure from the standard methodology for operating Genesis was necessary. As discussed in Reference 2, the navigation strategy for the nominal mission implemented a straightforward methodology to maintain the trajectory close to the baseline reference trajectory. In the nominal mission, trajectory deviations resulting from imperfect knowledge of the orbit were removed at intermediate waypoints along the trajectory by corrective maneuvers (thereby returning the spacecraft to the reference trajectory). For the backup orbit, this strategy was too costly in terms of propellant. Therefore, in the strategy implemented for the backup orbit, intermediate waypoints were ignored and the focus was on achieving the ultimate entry conditions at the entry interface point. Thus a single 'reference trajectory' did not exist for the backup missions because the reference trajectory was re-computed and updated after each swing-by. This optimization strategy is common in multiple swing-by satellite tours (*e.g.* Galileo) but had been an unnecessary complication for the primary Genesis mission.

Table 3. Backup Trajectory TCM Placement and Deterministic ΔV

This strategy allowed the backup orbit's total maneuver expenditure to fit within the propellant remaining on board Genesis. The plan, however, complicated operations and required revised targets for each upcoming encounter after each swing-by.

Using the above strategy, Table 4 summarizes the total ΔV (deterministic and statistical) needed to navigate the backup trajectory. The primary contributors to the statistical ΔV are orbit determination errors and the ability of the Genesis flight system to accurately execute commanded ΔVs. Assumptions made in both of these areas are consistent with the practices of the nominal flight operations.

The deterministic mean ΔV total equals 166 m/s. But statistically, the mean ΔV equals 178±9 m/s. This total climbs to 214 m/s at the 99th centile level of confidence.

The ΔV magnitude for TCM2 is significant -- on the order of 108 m/s (99th-centile). It was required to shape the trajectory and occured at the perigee following the divert maneuver (November 9, 2004). In the constrained maneuver-implementation space of the Genesis spacecraft, this maneuver would be difficult to perform. (Maneuver-implementation space is a function of the direction of the ΔV as well as its magnitude.)

A risk mitigation measure was investigated whereby TCM2 was split into two manageable maneuvers separated by 24 hours (*i.e.* E2 perigee ±12 hours). The total mission ΔV would increase by approximately 40 m/s to accommodate this divide-and-conquer strategy (*i.e.* to ~255 m/s). Although no decision was made to split TCM2 (nor was one necessary prior to the nominal SRC entry time) one could have chosen to use more of the ΔV margin to spread the implementation of TCM2 into more than two, and therefore even much easier-implemented TCMs.

Table 4. Backup Trajectory TCM ΔV Statistics

- Stress case: TCM-10 ΔV implemented at TCM-11 (contingency) epoch.

RESULTS

The altitude uncertainty prediction at the time of the last maneuver delivery preceding each swing-by is shown in Table 5. The error is insignificant given the distance of each swing-by. The last Earth swing-by (Earth 5) is also the backup probe delivery; a divert maneuver for the spacecraft bus also occurs at Earth 5.

Table 5. Swing-bys

Encounter	Date (UTC)	Altitude 1σ	Approach Maneuver
Earth 1	Sept 8, 15:56	206 ± ~5	E1 – 3.5 hours ΔV =30 m/s
Earth 2	E1 + 62 days	57,624 ± 475	E2 – 12 hours* ΔV =52 m/s
Earth 3	E2 + 19 days	41,028 ± 346	E3 – 9 days ΔV = 3.6 m/s
Moon 1	E3 + 17 days	10,954 ± 25	M1 – 7 days ΔV = 3.3 m/s
Earth 4	E3 + 20 days	165,530 ± 173	M1 – 7 days ΔV = 3.3 m/s
Earth 5	E4 + 89 days	206 ± ~5	E5 – 3.5 hours ΔV =40 m/s

*Assumes TCM2 has been split into two maneuvers 24 hours apart.

The backup trajectory propellant budget of 214 m/s was accommodated within the backup mission Genesis propellant margin of ~500 m/s.

The prediction of the entry flight path angle (eFPA) uncertainty is plotted in Figure 3 (as well as a close-up of the last 5 days in Figure 4). Figure 3 begins subsequent to TCM7 (the last apogee maneuver, ΔV = 43 m/s). Included in the error model used to compute Figure 3 were TCMs 8, 9, & 11, all attitude control thrusting events, and all events associated with the SRC release (spin-ups, turn, separation). The final SRC targeting maneuver was scheduled one day before entry (with a data-cutoff at entry – 2 days). This delivery yielded an entry flight path angle uncertainty of eFPA = ±0.04 (3σ). The requirement was ±0.08 (3σ).

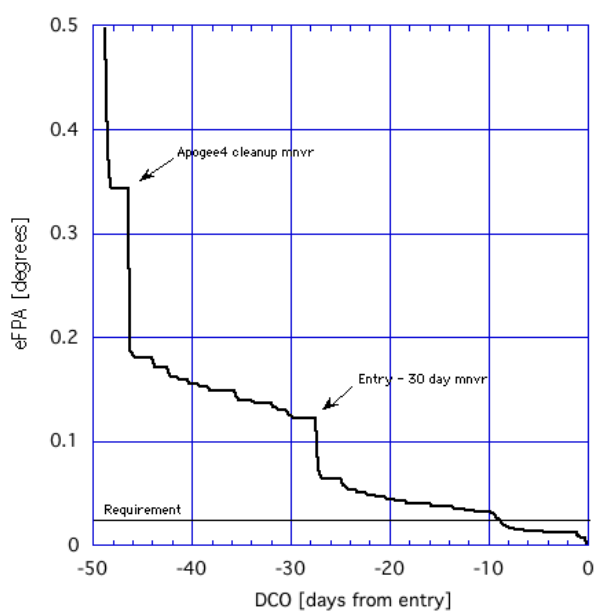


Figure 3. Entry Flight Path Uncertainty (1σ)

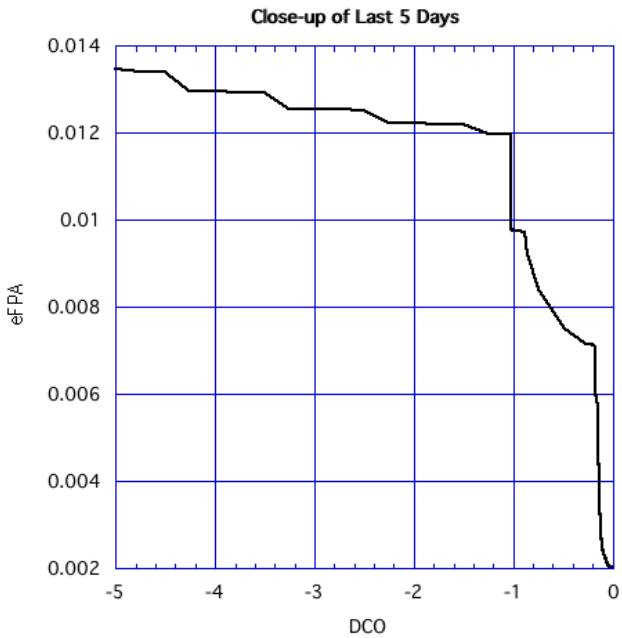


Figure 4. Entry Flight Path Uncertainty (1σ). Last 5 days.

SUMMARY

The delivery to the backup mission's entry interface point satisfied all requirements. Indeed the backup orbit was demonstrated to be completely feasible, although demanding of the flight-team. The entry occurred in daylight hours with a landing at UTTR within six months of the nominal entry time. Genesis easily accommodated the 214 m/s propellant budget computed for the backup orbit. The entry flight path angle uncertainty at the final E-2 day delivery equaled ±0.04 degrees (3σ), a factor of two beneath the requirement of ±0.08, but the requirement could be met anytime after E-9.5 days. Therefore the entry corridor was a readily achievable target, proving the validity of the backup orbit concept.

The single item of concern to the Genesis project would have to be the flight-team size necessary to operate the backup trajectory. It is likely that, because of the maneuver schedule, the backup mission navigation team would have needed to be approximately twice the size of the nominal mission navigation team.

ACKNOWLEDGEMENTS

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APPENDIX A: BACKUP-MISSION *A PRIORI* NAVIGATION MODEL

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