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THE DESIGN AND NAVIGATION OF THE NEAR-SHOEMAKER LANDING ON EROS

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After a 4.5 hour controlled descent using five open-loop maneuvers on February 12, 2001, the Discovery-class NEAR-Shoemaker spacecraft successfully landed on the surface of Eros becoming the first spacecraft ever to touchdown on an asteroid. This landing was made extraordinary by the fact that the spacecraft was not designed for landing and it remained in telecommunications with NASA's Deep Space Network afterwards. The descent trajectory was designed primarily to acquire as many close range high-resolution images (< 1 km) as possible while providing optimal viewing geometries and secondarily to ensure the safety of the spacecraft by minimizing its impact velocity. Since the spherical harmonic representation of Eros' gravity diverges below the sphere circumscribing the asteroid (< 18 km), a polyhedral gravity field based on our Eros shape determination was used for integrating the trajectory below this limit. This paper discusses the design, navigation and the Monte Carlo error analyses that were critical to the design of this landing scenario. Also described is the reconstruction of the landing trajectory using radio metric, optical landmark and laser ranging tracking data, which determined the characteristics of the landing to be well within the error analyses.

INTRODUCTION

On February 12, 2001 at 20:02 Earth-Received-Time (ERT) UTC history was made when flight controllers at the Johns Hopkins University Applied Physics Laboratory (APL) and navigators at the California Institute of Technology Jet Propulsion Laboratory (JPL) softly landed NASA's Near Earth Asteroid Rendezvous (NEAR) Shoemaker spacecraft onto the surface of the 33 km x 17 km x 12 km, S-Type asteroid, 433 Eros. The NEAR spacecraft (S/C) was designed to orbit Eros, not to land on it, but it survived the landing and went on communicating with NASA's Deep Space Network (DSN) antennas and collecting science for over 2 weeks afterwards. Engineers at both APL and JPL gave a 1 in 10 chance that the S/C could survive the landing due the complexity of the irregular asteroid's gravity, and shape as well as the fact that NEAR had no landing apparatus.

After several frames of engineering telemetry were returned from NEAR, it appeared that the S/C had suffered no ill effects from the impact and all systems were determined to be completely healthy. In addition, through the attitude information returned, NEAR was found to be resting in a tripod configuration as planned with the tips from two of its solar arrays and the rear of the bus where the science instruments face outwards touching the asteroid. There was no expectation of gathering science after the touchdown while planning the landing, but although NEAR's Multi-Spectral Imager (MSI) was inoperable at such close range, the X-ray, Gamma-ray Spectrometer (XGRS) instrument was found to be in an incredibly fortunate position—to take unprecedented in-situ observations of the elemental constituents of an asteroid's regolith. The NEAR mission was originally scheduled to end on February 14, 2001, but because of this fortunate

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outcome NASA funded a short extension of NEAR's mission and allocated several tracking passes from the DSN's Goldstone, California 70 meter antenna to collect the valuable science observations. February 28, 2001 marked the last day of tracking NEAR when flight controllers at APL's Mission Operations Center (MOC) shut down its systems.

The NEAR-Shoemaker Mission

The primary objectives of the NEAR mission were to obtain unprecedented close-up physical and geological observations of a near-Earth asteroid by operating the spacecraft in orbit around it for one year. The NEAR mission was operated by APL, while navigation of the spacecraft was provided by JPL. Launched in February of 1996, NEAR was the first mission of NASA's Discovery Program which instituted the "cheaper-faster-better" philosophy for the future of deep space missions. NEAR became the first S/C to fly by a C-class asteroid (Mathilde) in late June 1997 en route to Eros. NEAR then performed a gravity assist by flying by Earth on January 23rd of 1998 which altered its orbital heliocentric inclination and placed it on course for a January 1999 rendezvous with Eros ($i \approx 10.8°$). For a general description of the NEAR mission and the design of its interplanetary trajectory, see Farquhar *et al.* [1]. Due to the early termination of the Rendezvous Maneuver (RND-1) which was designed to slow the S/C-Eros relative speed on December 20, 1998, the NEAR S/C flew past Eros at a distance of 3828 km. Another large maneuver had to be executed on January 3, 1999 to bring the S/C back to Eros within one year. Immediately after the aborted RND-1, the S/C lost control with its inertial guidance system and the rocket propulsion system fired thrusters frantically to regain a stable attitude before the fault protection finally put the S/C into a sun-safe mode. After being lost for nearly 30 hours, the DSN finally contacted NEAR and flight controllers were able to regain control of the S/C. During its lost of communications, it could not be determined with any certainty how many pulses the thrusters fired and thus how much fuel was left. Crude estimates based on tank pressures and indications that there may have been thousands of thruster pulses fired dramatically reduced the available fuel for the remainder of the mission. This meant a lower propellant margin for the orbit phase at Eros which had to be redesigned, see Helfrich, *et al.* [2]. Dunham, *et al.* [3] explain how contingency planning made it possible to recover the NEAR mission from this near disaster. An explanation of the design of the interplanetary trajectory to return to Eros is also given by Dunham, *et al.* [3].

Finally, orbital operations at Eros began on February 14, 2001, when the Orbit Insertion Maneuver (OIM) placed NEAR into a 300 x 356 km orbit. Through the first five months at Eros, NEAR progressively entered into lower circular orbits through a series of maneuvers. NEAR's orbital radius was incrementally decreased to 200 km, 100 km, 50 km and finally 35 km as knowledge of the mass, gravity distribution, pole direction, spin and shape of Eros were being steadily characterized. The navigation during this phase relied on a combination of DSN radio metric tracking data, on board optical imaging of landmarks on Eros and from the NEAR Laser Ranging (NLR) instrument. Williams, *et al.* [4] give a complete description of the navigational results during the Eros orbit phase.

Background

Although landing the NEAR-Shoemaker S/C on Eros was not part of the mission plans when NEAR launched, it was an idea that Mission Director, Robert Farquhar of APL, had thought about and discussed occasionally with the Navigation Team at JPL. Though it was not yet approved by NASA, planning for a possible landing on Eros slowly began in 1998 as NEAR was quickly approaching Eros for the first rendezvous in January 1999 [5]. These plans also included performing several close passes to the surface of Eros (< 5 km). Because of the lower propellant margin and the fact that the project had already sought funding for another year of operations, the idea of landing NEAR on Eros became closer to reality. Antreasian, *et al.* [6] further discussed plans for an 'End-of-Mission' scenario where several close flybys to Eros take place and lead to the eventually landing on Eros. Hovering was also presented as a possible technique for landing. In addition plans were discussed by Antreasian, *et al.* [6] to implement a simple on-board closed-loop control algorithm using the slant range to the asteroid surface information provided by the NLR instrument. These plans were abandoned when it became clear that the flight software would need to have several modules removed in order to fit a landing algorithm into the flight computer's memory. Since the flight software had undergone a few revisions prior to orbit insertion and it had been functioning nominally, there was no desire to make changes. Furthermore, there wasn't enough time to make the major changes to the flight code and thoroughly test it, so the landing design would have to be open loop.

To prepare for the end-of-mission operations, a close flyby orbit was designed then executed on October 26, 2000 when the NEAR spacecraft safely flew within a distance of 5.5 km near the 0°longitude (long) end of Eros' elongated body (-21.52°Latitude, 328.8°E. Longitude). The timing and accuracy of the maneuver to initiate the flyby was critical to achieve the desired flyby location and the post-flyby orbital conditions. Several close flybys of the ends of Eros were then designed for the end of January 2001. The spacecraft was placed into an elliptical 36 km x 22 km equatorial orbit (retrograde) on January 24, 2001 for four consecutive days. Several close flybys of altitudes ranging from 4.4 to 7 km occurred at various regions close to the long ends of Eros. This low flyby phase came to a close on January 29 after a small maneuver lowered the S/C's periapsis to achieve the closest flyby of 2.7 km. The conceptual ideas set forth by Antreasian, *et al.* [5, 6] of performing low flybys of Eros towards the end of NEAR's mission had become a reality. An overview of the low flyby orbits are discussed by Williams, *et al.* [4], however, details of the actual design of these low flyby orbits will be the subject of a forthcoming paper.

Determination of Eros Physical Properties

It was important to have a precise knowledge of the dynamical environment encompassing Eros before attempting a landing on the asteroid. This dynamical environment includes Eros' mass, gravity distribution, shape, pole direction and spin. It was also imperative to have an understanding of how this environment influenced NEAR's orbit, especially at close range. Close to a distended body such as Eros as described by Scheeres[7] and Scheeres *et al.* [8], the orbital dynamics of the S/C are subject to strong perturbations from the gravity field, the major contribution coming from the 2nd degree and order gravity field, which can be reduced to the two terms C_{20} (oblateness) and C_{22} (ellipticity). At close altitudes, the strong perturbations from the irregular gravity field of Eros cause large changes to the S/C orbit characteristics. These effects can lead to unstable situations where the S/C is suddenly placed on either an escape or impact trajectory.

Preliminary estimates for Eros' mass, shape, pole position and spin rate were determined from the December 1998 flyby of Eros[9, 10]. Better estimates were obtained once NEAR achieved orbit around Eros [11]. After nearly 8 months of orbiting the irregularly shaped Eros at distances ranging from 367 to 34 km, these parameters have been well characterized through the orbit determination process using a combination of the radio metric, landmark and NLR data[12]. Miller *et al.* [12] describe the various procedures and analyses that went into the determination of the parameters presented in Table 1.

Table 1: Eros Physical Parameters

Parameter	Value
Gravitational parameter, μ	$(4.4631 \pm 0.0003) \times 10^{-4} \mathrm{km}^3/\mathrm{s}^2$
Pole Direction	
Right Ascension	11.369 ± 0.003 deg
Declination	17.227 ± 0.006 deg
Spin rate	1639.38922 ± 0.0002 deg/day
Prime Meridian	326.06 deg (J2000 epoch)
Principal Axis, x	9.29 deg East
Gravity Harmonics (normalized)	
C_{20}	-0.052478 ± 0.000051
C_{21}	0.0
S_{21}	0.0
C_{22}	0.82483 ± 0.000061
S_{22}	0.027909 ± 0.000035

Gravity Modeling

The successful landing was dependent upon a correct evaluation of the gravitational accelerations upon the S/C at close range. In order to prepare for the landing, a circular 35 km polar orbit was planned in July of 2000 to globally map the gravity of Eros at high resolution. The S/C's activities were reduced for 4 out of 10 days during this orbit to eliminate the possibilities of unmodeled perturbations on the S/C. Because the solar pressure which was the 2nd largest force acting on the S/C was well known when the S/C's body-fixed z-axis was placed coincident with the direction of the sun ($0°$incidence angle upon the solar arrays), the S/C remained in this attitude for this time. Side forces that developed when the S/C was turned off the sun for imaging or for high gain antenna contact with Earth were not modeled accurately due to partial shading on the bus from the solar panels and thermal emission from the instruments. This 35 km orbit enabled a fairly accurate determination of the spherical harmonic gravity field down to degree and order 10 [12].

Because of the asteroid's irregular shape, it has been known that the spherical harmonic representation of the gravity potential is deficient at locations closer than the largest triaxial dimension. These deficiencies are due to the divergence of the harmonic expansion series inside the circumscribing sphere about the asteroid centered at its center of mass[13]. Care must be given when numerically integrating the spacecraft orbit within these regions as the spherical harmonic gravity model could lead to erroneous results. To alleviate these problems, a polyhedron gravity model developed by Werner & Scheeres[13] had been incorporated into the integration of the trajectory through the close regions; however, the partial derivative derivations of the gravity potential with respect to the S/C state and dynamic parameters necessary for orbit determination have not been completed. The polyhedral gravity for Eros was based on one of the best determined shape models described below and assumed a constant density.

A comparison of the spherical harmonic expansion derived from the shape determination assuming homogeneity to that of the gravity field determined from the radio metric and optical data show an agreement to within 1% density variation on a large scale[12]. This agreement alleviated any worries of the constant density assumption using the polyhedral gravity model as including density variations within this model would have been a difficult task.

Another similar, but independent gravity modeling technique was developed to compute the gravity accelerations at close range. This model incorporated an integration of infinitesimal areas over the asteroid's shape using the spherical harmonic representation of the shape again assuming constant density. To account for density variations possibly due to distribution of regolith or small internal variations, a gravity sheet covering the asteroid could be estimated. Though these models were not finished in time to help in the design of the landing, they did help independently verify the accelerations produced from the polyhedral gravity model a few weeks before the landing.

Shape Models

Based on the fairly accurately determined Eros-relative S/C positions ($\sigma r \approx 20$ m) within the circular 35 to 50 km orbits, the NLR data was incorporated using an off-line method to compute the high fidelity shape model shown in Figure 1. A spherical harmonic representation of the asteroid shape was determined to degree and order 34. This model was then tessellated into 17788 small triangular surface elements each including three of the 8896 vertices found on the spherical harmonic surface. At each integration step below a predetermined orbital radius (set at 20 km), the evaluation of Eros' gravitational attraction on NEAR was computed by the summation of the elemental polyhedra formed by the three vertices and three sides of each surface element. This polygravity computation considerably added to the computer processing time to numerically integrate the S/C's orbit.

DESIGN of the LANDING

Goals

The primary goals for landing NEAR-Shoemaker on Eros was to acquire as many images as possible under 5 km and to acquire the last image as close as possible to the surface. Although very much desired, insuring the survival of the S/C upon the landing was a secondary goal. The requirements for the primary goals meant we needed to keep the HGA Earth-pointed during the entire decent to download the images as fast as possible and point the MSI imager as directly at the asteroid as possible. To meet the secondary goal required the impact speed to be minimized (< 3 m/s).

Geometry

The geometric locations of the Earth, Sun relative to Eros and its pole had to be taken into account for designing a landing trajectory that could achieve the primary goals while meeting the mission constraints discussed below. Figure 2 gives the relative locations of Eros and Earth during the landing phase in a north ecliptic view. The locations of the Earth and Eros during the time of the Eros orbit insertion (EOI) on February 14, 2000 are also indicated in this figure. The orientation of Eros' North pole is also shown in Figure 2. At the time of the landing, the distances of Eros to the Sun and Earth were approximately 1.44 and 2.11 A.U., respectively. The latitude of the Sun and Earth relative to Eros' equator are shown as a function of time in Figure 3 from EOI to February of 2002. As shown in Figure 3 during the day of the landing, the direction of the Sun from Eros was nearly aligned with the south pole of Eros with only 5.6°of separation (latitude $= -84.4°$). This meant that the southern latitudes of Eros were in constant sunlight while the northern regions were in constant darkness. The direction of Earth from Eros was approximately 18.5° off Eros' South pole. At this time, the Earth direction vector was approximately 23.8° from the Sun direction vector with respect to Eros. The implications of these geometries on the design for the close flyby and landing trajectories are discussed below.

Mission Constraints

All of the NEAR S/C subsystems and the on-board instruments are fixed mounted to the S/C bus. A complete description of the S/C is given by Santo *et al.* [14]. Two-way, X-Band Doppler and range tracking data were transponded over either NEAR's High Gain Antenna (HGA), fan-beam antenna (FBA) or low gain antenna (LGA). The HGA required Earth pointing within approximately 1° and was used mainly for the transmission of science telemetry. The 40° wedged radiation pattern of the FBA allowed the S/C to remain in Earth contact during most of the orbit phase and a portion of the landing phase while the S/C HGA could not be pointed directly toward the Earth. Science instruments point out the side of the S/C bus 90°to the solar array normal vector, so in order to obtain observations at non-terminator locations on the asteroid, the S/C's attitude must be adjusted accordingly while maintaining the requirements for solar lumination on the solar arrays and telecommunications stated below. The fixed mounting of the science instruments, solar array and HGA on NEAR had to be considered in the landing design in order achieve the best viewing geometries possible while maximizing the quantity of close observations returned to Earth during the descent. Optimal viewing geometry meant imaging the surface of Eros with the lowest emission angles possible. The maximum telecom downlink rate was achieved by continuously pointing the HGA towards Earth. This requirement constrained the imaging camera to point perpendicular to the Earth direction, although rolls about the Earth vector were permitted during the descent.

Because of the distance of Eros from the sun, to ensure adequate illumination of the solar arrays for power during most of the orbit phase, the S/C's attitude had to be oriented such that the normal of the solar arrays remained within 30° of the Sun direction. At the time of the landing phase, however, the Sun-Eros distance allowed the solar arrays to maintain adequate power margin with off-sun pointing attitudes up to 46°. Tracking also imposed a constraint that the orbit normal remain within 30 degrees of the Earth direction to ensure navigation data and science return. Relying on the S/C batteries was believed to be risky, so the S/C was constrained never to fly into the shadow of Eros.

Orbit Constraints

Another set of constraints are implied from the state of NEAR's orbit configuration toward the time of the landing. As mentioned beforehand, a series of low altitude flybys of the ends of the asteroid from 7 - 2.4 km were planned during Jan 24 - Jan 29, 2001. The landing design had to be developed in conjunction to the low flyby orbit plans in order to budget the fuel as well as to place the S/C into a desired orbit configuration to initiate the landing sequence. The post flyby orbit was similar to the orbit leading up to the low flybys: near circular 35 km retrograde equatorial orbit, nearly coincident with the sun plane-of-sky. Two Orbit Correction Maneuvers (OCM-24, 25) were planned between the close flyby orbits and the landing to target the inertial location of the landing initiation.

Tracking Requirements

In order to characterize Eros' physical parameters, and enable fairly accurate predictions of NEAR's orbit for planning near continuous X-Band Doppler coverage was required from the DSN's 34 m and 70 m antennas during the entire orbit phase as well as the landing phase. The 70 m antennas allow the highest data return. Optical landmark tracking of known crater locations using the spacecraft Multi spectral Imager (MSI) was also required during the orbit phase and during the time leading up to the landing. Owen *et al.* [15] describe the process of identifying landmarks and amassing a database of over 10,000. The Doppler data is sensitive to the dynamical interaction of Eros' gravitational attraction upon the S/C. The landmark tracking complimented the Doppler by fixing the orbit of NEAR relative to Eros' spin and principal axes. This was important for determining Eros' physical parameters. NEAR's laser ranging (NLR) data from the S/C to the surface of Eros was another important data type that wasn't used for real-time operations, but was used for shape modeling. Incorporation of the NLR data that was acquired during the landing may provide valuable information at reconstructing the landing trajectory.

The day of the landing was picked to be February 12, 2001. This date was picked two days before the scheduled end of mission on February 14 to allow time for contingencies. It was imperative to have redundant DSN 70 meter antennas tracking NEAR during the descent to reduce the dependency on one station in case there was a station problem. An overlap between Madrid and Goldstone was identified on February 12, 2001 from 16:00 – 20:06 UTC. The entire decent portion of the landing phase including touchdown was planned to occur during this overlap. Pad was also placed on the nominal time of impact to allow for $3-\sigma$ dispersions of off-nominal cases from a monte-carlo analysis that will be described below.

Early Designs

The planning of the landing phase had been performed in conjunction with designing the end-of-mission low flyby phase which would take place prior to the landing [4][6]. As the orbit phase was coming to an end, plans to perform low flyovers of various regions of the asteroid had begun. An aggressive plan was laid out to include several close flybys; some were designed to fly through the saddle regions with altitudes of 1 – 2 km. Based on tank pressures and blow-down curves, an lower limit of amount of fuel available for the end-of-mission plans was approximately equivalent to ΔV of 32 m/s. Depending on how the low flybys were designed, a large amount of fuel could be spent recovering from a flyby that increased the S/C's orbital energy.

Antreasian *et al.* [6] discuss a landing on the South pole of Eros. This original plan consisted of placing the S/C into a 35 km polar approach trajectory from a nominal retrograde 35 km equatorial orbit. The South Pole design was desirable from the standpoint that we could start the landing trajectory to the South pole from anywhere in the final equatorial orbits and the timing of the burns relative to the asteroid rotation was removed. In order to ease the planning, no hard constraints were given for the placement of the 1st landing maneuver which would change the orbit inclination from equatorial to polar. This meant that after the low flyby phase was finished, the landing sequence could begin from whatever subsequent orbit NEAR found itself in. Once the S/C flew directly above the South Pole (approximately 29 km from the surface), the first de-orbit maneuver would execute to cancel the S/C's orbital velocity and send it on a free-fall trajectory towards Eros then a series of braking maneuvers referred to as End-of-Mission Maneuvers (EMMs) were added to prolong the descent and allow more images to be taken. It was found that instead of starting the descent from the 35 km orbit, a 'shallow' approach orbit which would depart from the 35 km equatorial orbit and reach an altitude of 5 km over the surface at the South pole could save approximately 4 m/s ΔV. The location of the descent vector would coincide with that of the approach orbit's semi-latus rectum of 11 km since the South pole radial dimension was ~ 6 km.

One shallow approach would incorporate EMM maneuvers at 5 km, 1 km, 500 m, 200 m altitudes with ΔV magnitudes of 6.8, 5.1, 2 and 1.7 m/s to arrest the free-fall velocities and extend a 5 km free fall from 32 minutes without any braking maneuvers to 47 minutes. On the suggestion of Bob Farquhar, another shallow approach was designed to include 3 'bouncing' maneuvers of 4, 3 and 2 m/s at 500 meters altitude and a 2 m/s bouncing maneuver at 300 m to prolong the free-fall flight time to 1 hour and 29 minutes. The bouncing maneuvers were intended to propel the S/C upwards away from Eros momentarily before gravity would eventually return the S/C free falling once more.

Without regards to incorporating maneuver execution and trajectory position and timing errors, these ideas were beginning to look good on paper. Then we finally finished coding routines to enable a full-up Monte Carlo error analysis on this 'bouncing' trajectory. With the application of modest maneuver and initial state errors, it was found that the pumping of energy from the bouncing maneuvers into the trajectory was very problematic and could result in a escape trajectory rather landing. The altitude distributions for these Monte Carlo cases as a function of time are shown in Figure 5. To understand what is happening, Figure 6 depicts the escape velocities necessary for given orbital radii at Eros; circular velocities at Eros are also given for comparison.

As explained by Antreasian *et al.* [6], the South pole descent designs suffered from poor viewing geometry which were limited to very close range, and high emission angles. The primary goal could not be met with this first landing design. Since this would strictly limit the number of images that could be received before the spacecraft impacted the surface and additionally the closest that the last image could be downlinked to Earth was $\sim$ 2 km this plan was rejected.

Our navigation software had the ability to target maneuvers to change orbital conditions such as semi-major axis, eccentricity, e, semi-latus rectum, p, and inclination, i, in either the Earth-Mean-Equator of 2000 (EME), the Sun Plane-of-Sky (SPS) or Eros equatorial Inertial frame. It did not have the capability for targeting the trajectories to specific inertial or Eros body-fixed locations at specific times which was needed for better landing designs than what we had first planned above. If regions other than the South pole were desired for landing, it complicated the design. This required the knowledge of the effects from the morphology of the asteroid's shape, the non-uniform gravitational perturbations and timing of the maneuvers relative to asteroid rotation on the landing trajectory and the location of impact. The relative locations of the Earth and Sun to the landing were also important. The region of touchdown had to be in the sunlit portion of the asteroid which at the time of the landing amounted to the Eros' Southern hemisphere. Also, it was desired to keep the S/C in telecommunications with Earth after impact in case the S/C were to survive which meant that the S/C should not land behind the limb of the asteroid or behind a hill as seen from Earth. Several tools written in perl and matlab were quickly developed in the last 3–4 months before the landing to account for these issues and enable landing designs which were more favorable of achieving the primary goals.

The Eros South Pole-Earth Descent Plane

The first step to designing an orbit that could possibly meet the primary objectives involved the creation of a descent coordinate frame. The continuous durations of both the HGA pointing towards Earth and the MSI pointed towards Eros throughout the descent were the prerequisites for this system. By fixing the HGA pointed at Earth, the requirement to satisfy power could easily be satisfied because the incidence angle (angle between S/C - sun direction and the solar array normal) on the solar arrays as shown in Figure 4 was below 25°which was well under the 46°limit. This coordinate frame was centered at the center of mass of Eros and was formed by first including the Eros to Earth direction vector, $\hat{u}_{\oplus}$ as the X-axis:

$$\hat{u}_{D_X} = \hat{u}_{\oplus} \tag{1}$$

The Earth direction vector is then crossed into the South pole vector of Eros, $\hat{S}$, to compute the vector, $\hat{u}_{D_Z}$ that is normal to the plane that contains $\hat{u}_{\oplus}$ and $\hat{S}$:

$$\hat{u}_{D_Z} = \mid \hat{u}_{\oplus} \times \hat{S} \mid \tag{2}$$

Finally, the frame is completed by forming the remaining orthogonal axis, $\hat{u}_{D_Y}$,

$$\hat{u}_{D_Y} = \hat{u}_{D_Z} \times \hat{u}_{\oplus} \tag{3}$$

The Earth-pointing HGA and Eros-Nadir-point MSI constraints can be met by placing the descent trajectory in the 1st quadrant of the $\hat{u}_{\oplus} - \hat{u}_{D_Y}$ plane. The free-fall, descent trajectory was assumed to follow a linear path in this plane, so the optimum condition for meeting both the HGA and MSI pointing constraints would be satisfied if the S/C were to follow its descent along the $\hat{u}_{D_Y}$ axis. The HGA would be pointed directly at Earth while the camera would be in the position to image the asteroid at optimum emission angles during the entire descent. This would, however, represent a landing at approximately 18.5°South on the very limb of Eros if Eros was a sphere as seen from Earth. As much as 1 km of the last portion of the descent could be occulted from Earth if the actual morphology of Eros was taken into account.

It was desired to stay in view of Earth during the descent as much as possible, so conservatively picking a descent vector, $\hat{r}$, within this plane at lower Southern latitudes would only minimally affect the MSI emission angles. So $\hat{r}$ could be computed by picking a South latitude (ϕ) and preforming a negative rotation about $\hat{u}_{D_Z}$ from $\hat{u}_{D_Y}$:

$$\hat{r} = \sin(\phi - 18.5^\circ)\hat{u}_{D_X} + \cos(\phi - 18.5^\circ)\hat{u}_{D_Y} \tag{4}$$

The geometry of the descent portion of the landing is presented in Figure 4 relative to the South Pole of Eros ($\hat{S}$) and the Earth and sun directions.

Choosing the Landing Site

In order to choose the landing region, we projected the Eros shape model into the Earth Plane of Sky (EPS) during the last hour of the DSN overlap mention beforehand ~19:00 – 20:06 ERT-UTC and looked at regions that intersected the $\hat{u}_{\oplus} - \hat{u}_{D_Y}$ descent plane. To our surprise, the rim of a large depression known as Himeros along the smaller radius of Eros intersected the descent plane near the last few minutes of the DSN overlap. During simulations we noted that landing sites along the smaller axis of Eros had descent trajectories that were less sensitive to orbit determination timing errors as compared to those on the long axis, due to the shape of the gravitational potential around each of these sites. If a long axis end had been chosen for the landing site, much more stringent control of the trajectory approach before the de-orbit burn would have been required so that the spacecraft would arrive at a precise time over the end. If the spacecraft were early or late, the trajectory could be deflected to either side of the end due to the inverted saddle shape of the gravity equipotential surface over the ends of Eros, and we could not reliably predict the landing site. We were also interested in the Himeros area because of its smooth terrain, scarcity of large boulders which could further reduce the probability of damaging the spacecraft during touchdown. Furthermore, we knew that the Science Teams were very interested in the Himeros depression for the same reasons aside from the health of the S/C. We decided to pick this region (~ 45°latitude, ~ 80°E. Longitude) for landing despite the fact that we knew it was uncomfortably close to the end of the DSN window.

The first maneuver to initiate the landing sequence, End-of-Mission Maneuver 1 (EMM-1) was designed to change the plane of NEAR's orbit from retrograde equatorial ($i \approx 179^\circ$) and target the beginning of the descent trajectory at the 45°South latitude. The inertially fixed location to perform EMM-1 was found by finding the intersection of a vector in the 35 km equatorial orbit which was perpendicular to the descent plane as viewed in the EPS frame (true anomaly, $\nu = 90^\circ$). Then by designing a 2.6 m/s maneuver to achieve the -45°latitude, which would change the orbit inclination relative to the Eros equator from 179°to 145°, we found that the gravity perturbations from the irregular shape of Eros deflected the trajectory inclination by a sizable 7.5°at the intersection to the descent plane. This resulted in a latitude of -37.5°at the descent point. It appeared that a significant amount of ΔV would be needed to overcome the gravity perturbation on the inclination, so instead of reworking EMM-1 over and over to achieve the -45°latitude, we decided to adopt the -37.5°location for the first braking maneuver, EMM-2 which would take place 4 hours and 44 minutes after EMM-1. EMM-2 would then cancel NEAR's orbital velocity with a $\Delta V = 6.5$ m/s. Two more braking maneuvers, EMM-3 (3.5 m/s) and 4 (4.0 m/s) were added at 3 km and 1 km altitudes. Heeding warnings from the early bounce design, instead of bringing the S/C completely to rest at these altitudes, EMM-3 and 4, as well as EMM-2, were designed with a conservative 1 m/s downward bias. To improve the chances of survival on impact, EMM-4 was designed to remove the lateral body-fixed motion by adding the coriolis velocity, $\omega \times \overline{r}$ to the ΔV where ω is the Eros spin rate and $\overline{r}$ is the orbital radius of NEAR at EMM-4. It was hoped by removing the lateral velocity, the S/C would not cart wheel or flip onto its solar arrays, thereby rendering it inoperable subsequent to landing.

This plan (EOM-111900) resulted in a landing at -35.6°latitude, 82.1°E. Longitude after a descent time from EMM-2 of 45 minutes 21 seconds. NEAR had the capability of acquiring and downlinking images at the rate of 2 pictures per minute, so the number of nearly 90 images could possibly be received in this time. The impact velocity after EMM-4 would have been 3.3 m/s which was believed to be on the edge of survivability.

Another braking maneuver, EMM-5, was suggested by Gene Heyler (APL) and Robert Farquhar to be added to further reduce the impact speed. Before we were to add a 5th burn, we needed to complete the Monte Carlo analyses (described below) on this landing design.

The Monte Carlo analyses showed a non-gaussian distribution of landing times. A greater number of the cases were landing late by as much as 15 minutes while fewer cases were early by up to 5 minutes. To accommodate the late landing cases and keep the landing within the DSN Madrid–Goldstone overlap, we needed to shift the nominal landing time by 15 minutes. We were able to do this by fixing the entire landing trajectory from EMM-1 to EMM-4 with respect to Eros' body frame and rotating Eros 15 minutes earlier in inertial space. Then the EMM-1 inertial location and the braking maneuver ΔV's are rotated back into the EME-2000 system. The 15 minute shift would have minimal impact on the viewing geometries.

After the landing case was shifted by the 15 minutes which resulted in plan EOM-121900, The task was given at placing the EMM-5 burn into the design. EMM-5 was chosen to occur as late as possible after EMM-4 to have the maximum effect of minimizing the impact speed. Letting EMM-5 burn during impact was considered in its placement. It was believed that upon impact, the S/C would shut its thrusters down. EMM-5 was scheduled to occur approximately 5.5 minutes before impact. In addition, Gene Heyler suggested that we also add a Northward bias at the rate of 20 cm/s to EMM-5. The purpose of this maneuver was to insure that the S/C would touch down with momentum favoring a pitch about the touching tips of the solar array to the center of gravity to implant the the rear of NEAR into the regolith instead of the 'ostrich mode' whereby the S/C landed on the HGA and the solar arrays. By adding EMM-5 to the design, the nominal impact speed was reduced to 1.3 m/s.

Figure 7 details the final landing plan (EOM-020901) which incorporates all the elements mentioned beforehand leading back to the EOM-111900 plan. Figure 8 shows the landing trajectory relative to Eros in the EPS coordinate inertial frame. The altitude as a function of time for the descent is shown in Figure 9. The body-fixed velocity profile is shown in Figure 10.There were concerns over the ability of the spacecraft to perform all 5 preprogrammed burns in open loop. Contingency cases were integrated for the case where subsequent maneuvers failed to execute; the resulting altitude profiles are given in Figure 9. and the impact velocities are displayed in Figure 10.. As shown in Figure 7, the S/C was on an impacting trajectory after EMM-1 executed.

The Approach to the Landing Orientation Strategy

A series of low flyby orbits were designed from January 22 - 29. These orbits consisted of placing the S/C into a tight elliptical orbit with periapsis around 22 km and apoapsis at 35 km. During each periapsis passage 1 or 2 close flybys under 8 km of possibly both ends would result. On January 28, 2001 a maneuver was performed to lower the periapsis even more, this resulted in NEAR's lowest flyby of 2.7 km. After this low flyby phase, the S/C was placed into a fairly benign 35 km equatorial orbit. All Science Teams' requests to point the S/C for observations except for landmark tracking imaging were denied during this period so that the Navigation Team could get a good handle on the orbit determination and prediction of the S/C orbit at the time of the landing. Since the problem of targeting a inertially fixed location at a fixed time and targeting EMM-1 to a fixed location below the spherical harmonic limit was beyond the capabilities of the our targeting software, we had to quickly develop methods to achieve the designed conditions within allowable tolerances. Now that the orientation of the landing geometry was fixed in inertial as well as in Eros body-fixed space we needed to come up with a way to target the orbit to achieve the inertial position vector for the time of EMM-1. OCM-24 on February 2, 2001 and OCM-25 on February 7, 2001 were scheduled to target EMM-1 by performing a small maneuver near the desired inertial location of EMM-1 using the following equation,

$$\Delta V = \frac{\mu}{3VaT}dt \tag{5}$$

where μ is the gravitational constant for Eros, V is the S/C inertial velocity, a is the semimajor axis, T is the orbital period and dT is the change in period needed. This targeting amounted to a slight adjustment of the orbital period which had to be added up over the number of days from OCM-24 or 25 to the time of the EMM-1.To correct EMM-1 for conditions where the achieved inertial position was up to 5 km off the designed EMM-1 position vector, a simple targeting scheme was developed to realign the post-EMM-1 trajectory with the designed and bring NEAR to the correct descent point (12 km radius) at the time of EMM-2. This technique involved creating mapping matrices from partials of the EMM-2 position sensitivity to the EMM-1 ΔV errors by perturbing the post-EMM-1 state (± 1 mm/s per axis) and integrating the trajectory to the time of EMM-2. The correction to be added to the EMM-1 burn was computed as

$$\Delta \overline{V} = \frac{\partial \Delta \overline{V}_{EMM-1}}{\partial \overline{X}_{EMM-2}} \Delta \overline{r} \tag{6}$$

where $\Delta \overline{V}_{EMM-1}$ is the 1 mm/s vector perturbation to the EMM-1 velocity, $\overline{X}_{EMM-2}$ is the change to the inertial state vector at the time of EMM-2, and $\Delta \overline{r}$ is the vector difference in the state at EMM-1.

NEAR was capable of performing vectorized maneuvers in the 'fancy burn' mode which involved the simultaneous thrusting of orthogonal sets of thrusters while managing angular momentum desaturation of the reaction wheels during the burn. The imparted ΔV This burn implementation method, though simplified operations, caused problems for navigation. Often the burns under performed at the 1 to 2% level or had relatively large pointing errors, but these errors were tolerable for most of the orbit phase. Our OD software were incapable of modeling the simultaneous execution of two maneuvers of different thrust levels and durations, so these types of burns took longer spans of post-maneuver tracking data to reconstruct its ΔV than single component maneuvers. It was determined that the most accurate maneuvers were performed on the +xA thrusters due a well calibrated accelerometer on the x-axis.The Navigation Team preferred to use these thrusters for all critical maneuvers if possible. It was up Dave Dunham of APL to implement the maneuver under the constraints of having enough power from the solar cells, telecom and guidance stars. As result of the lost of the Mars Climate Orbiter, Mars Polar Lander and Mars Observer, telecom during maneuvers had become a mandated rule for operations. We knew that we had to perform EMM-1 as accurately as possible because of the sensitivity of the series of descent maneuvers effect on the landing the to radial distance at the time of execution, so we needed to perform EMM-1 with the +xA thrusters. Dave determined that we could use the +xA thrusters, however, we wouldn't have telecom at the burn attitude. By deciding to perform this maneuver in the 'blind', we were departing from nominal operations, but we believed that we were trading a small risk of losing the S/C for a better chance of a successful landing. As the Navigation and MOC Teams collaborated to build a sequence for the landing, it became clear that the S/C's fault protection, safing commands had to be disabled prior to the EMM-1 execution. This of course meant that if for whatever reason EMM-1 aborted, no signal from NEAR may never be detected. The S/C had to spend approximately one hour in the blind before performing the 30 second, 3 m/s maneuver–this only added to the anxiety and suspense of receiving the first signal after the burn.

SUMMARY

Several design elements presented within this report were brought together to produce the first ever asteroid landing. The landing trajectory's success was dependent on the accurate characterization of Eros' mass, gravity distribution, spin, pole and shape determined in the special 35 km gravity mapping orbit the Navigation Team requested. Since the spherical harmonic representation diverges below the asteroid's circumscribing radius other gravity modeling methods were incorporated to integrate the S/C's trajectory through this region. Because of the low gravity, care must be taken in the open-loop landing design to reduce the risk of a runaway control condition. The Monte Carlo analyses were useful in identifying vulnerable design conditions. Using these results, the nominal landing was adjusted to bring about a successful landing. Also, constant communication and cooperation between the MOC Team at APL, the Navigation Team at JPL and the Sequencing Team at Cornell University and the diligent final reviews of the landing time line were critical to the successful landing.

As of April 2001, the region where NEAR landed on Eros began to have part of its rotation in darkness. As of August 2001, the sun would have set until December of 2001.

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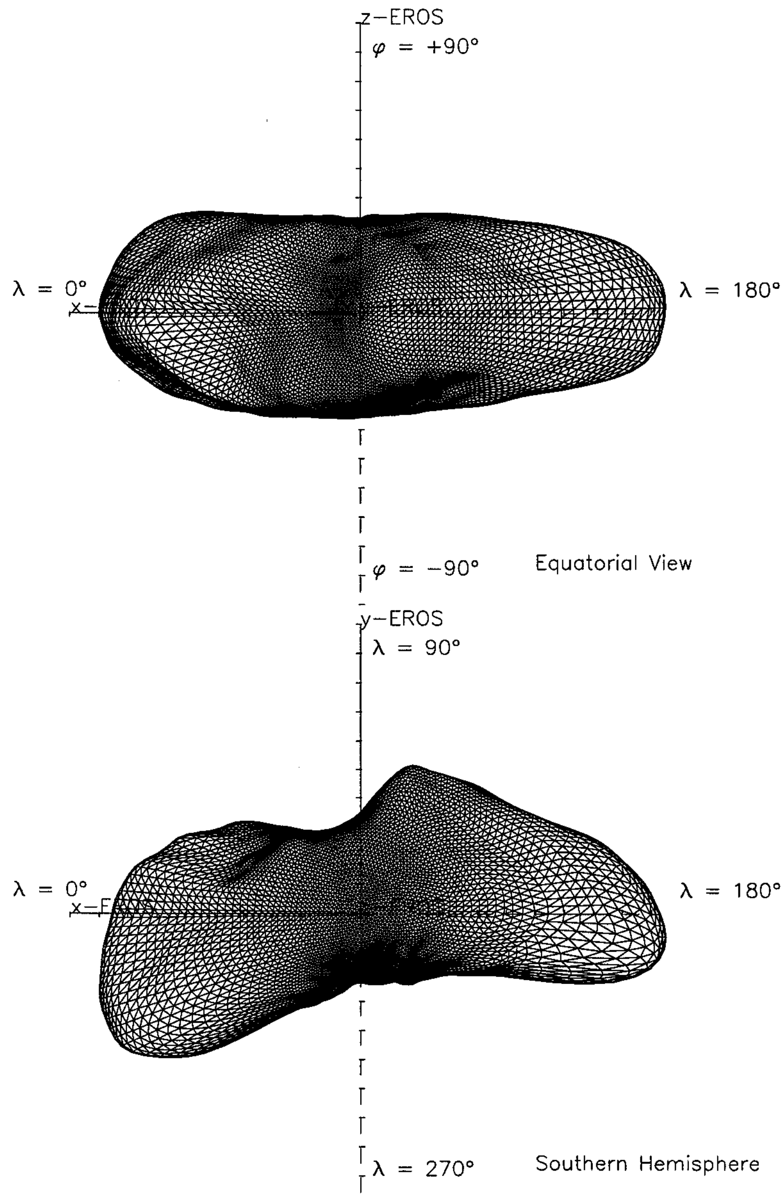

Figure 1: The 34 x 34 shape model used for the polygravity computations.

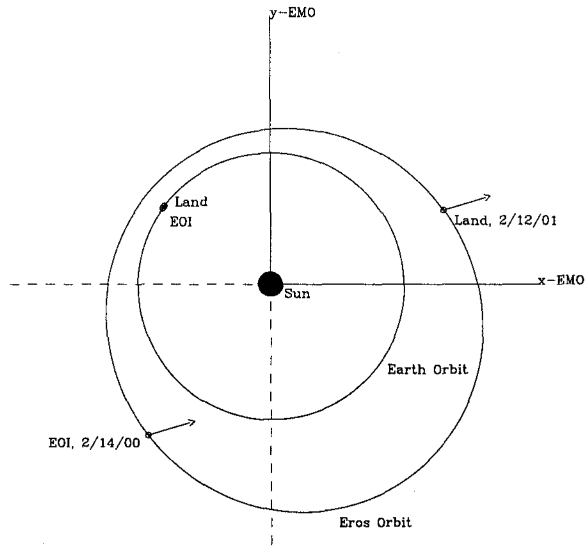

Figure 2: The relative locations of Eros and Earth at the times of EOI and the landing in a North ecliptic view. The direction of Eros' North pole is indicated by the arrow.

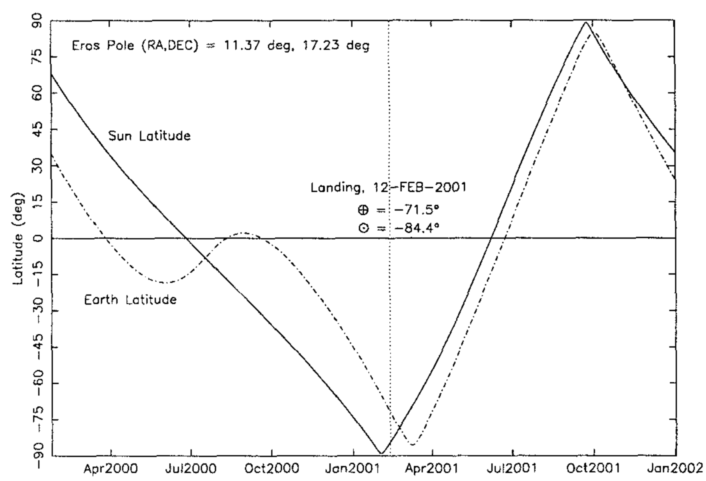

Figure 3: The latitude of the Sun and Earth relative to Eros' equator during the orbit and landing phases.

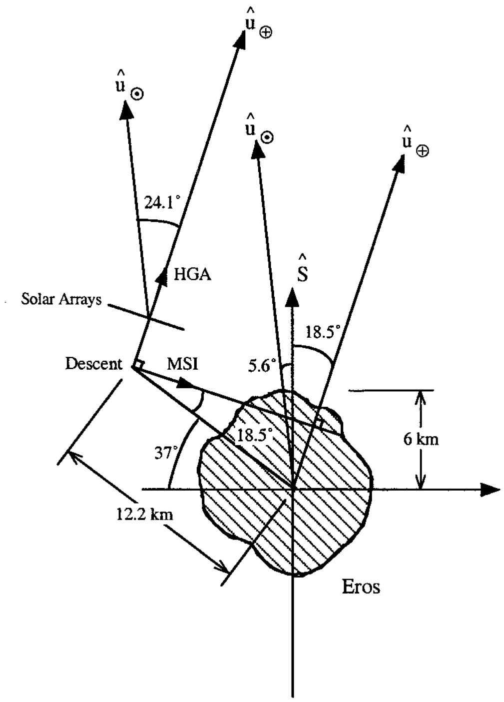

Figure 4: The geometry of the descent.

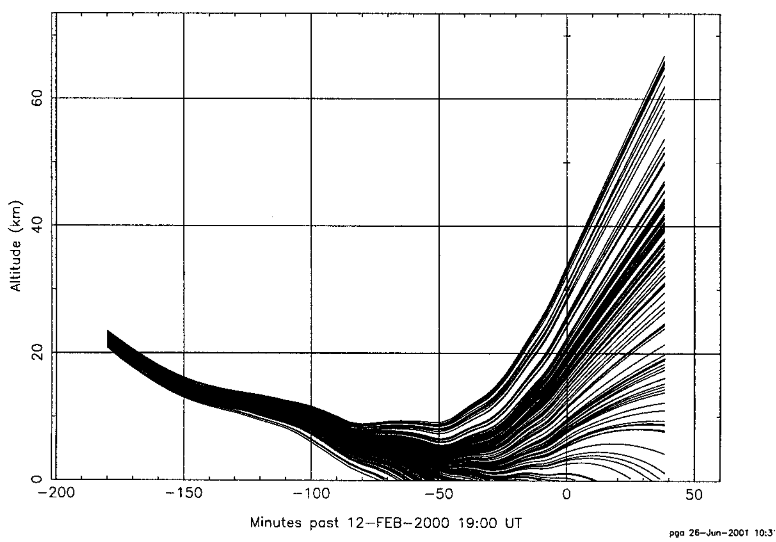

Figure 5: The Monte Carlo dispersions on altitude for the bouncing design

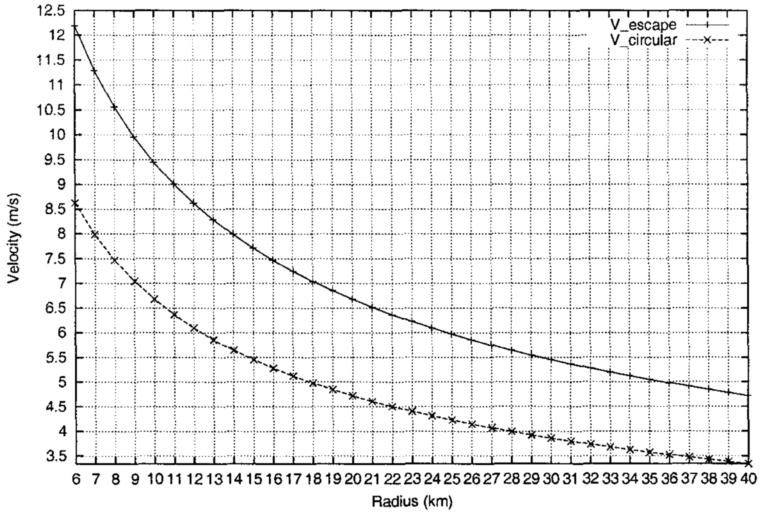

Figure 6: Eros escape and circular orbit velocities.

EOM Plan 02/05/01

Landing Phase

Phase	EMM[†]	Date Time[§] (SCET-UTC)	Date Time[§] (SCET-ET)	DOY	Orbit Radius (km x km)	Altitude (km)	Length (hh:mm:ss)	Inclination (deg) ATE	Inclination (deg) SPS	ΔV (m/s)	Burn Duration* (min)
Shallow Approach to -35 deg lat											
Start Descent Start EMM-1[‡]	1	2/12/01 15:13:56	2/12/01 15:15:00	43.6	35 x 7.5	25.6 – 5.1	3:44:39	135.2	129.4	2.567	0.243
End EMM-1		2/12/01 15:14:10	2/12/01 15:15:15								
Free Fall from 5 km Start EMM-2[‡‡]	2	2/12/01 18:58:35	2/12/01 18:59:39	43.8	12.2	5.1	0:15:21	50	55	6.479	2.545
End EMM-2		2/12/01 19:01:08	2/12/01 19:02:12		11.9	5.0					
Free Fall from 3 km Start EMM-3	3	2/12/01 19:13:56	2/12/01 19:15:00	43.8	10.2	3.1	0:16:00	46	51	3.473	5.134
End EMM-3		2/12/01 19:19:04	2/12/01 19:20:08		9.5	2.6					
Free Fall from 1 km Start EMM-4	4	2/12/01 19:29:56	2/12/01 19:31:00	43.8	8.1	1.5	0:11:00	36	42	4.024	6.240
End EMM-4		2/12/01 19:36:10	2/12/01 19:37:14		7.3	0.9					
Free Fall from 0.5 km Start EMM-5	5	2/12/01 19:40:56	2/12/01 19:42:00	43.8	6.76	0.48	0:05:30	36	42	2.761	4.456
End EMM-5		2/12/01 19:45:23	2/12/01 19:46:27		6.30	0.05					
Nominal Contact w/ Eros Surface		2/12/01 19:46:26	2/12/01 19:47:30	43.8	6.2	0.0			Impact Speed	1.3	
		DSS-63 sets at 20:06 UTC				Total Land Time	4:32:30		Landing ΔV	23.22	
						Descent Time from 5 km	0:47:51	Total Descent Burn Duration from 5 km not including EMM-1 transfer burn			18.4

*Maneuver Duration is based on: Dave Dunham's latest designs

†EMM = End Mission Maneuver

‡EMM-1 is to be performed on the +xA thrusters (20 sec burn time)

‡‡EMM-2 is to be performed in components with majority of ΔV on +xA thrusters

§One Way Light Time = 17:34.5

Figure 7: Final Landing Design Plan

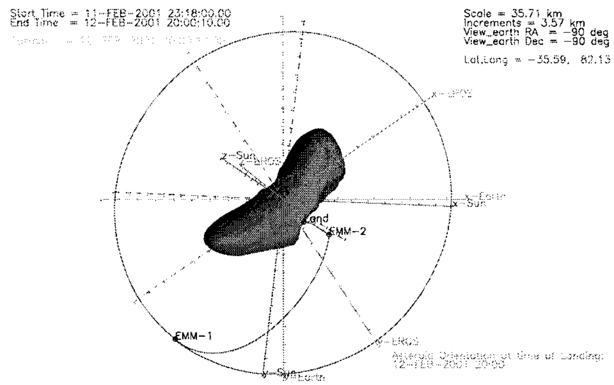

Figure 8: Final landing orbit design as viewed from the Earth Plane-of-sky coordinate frame

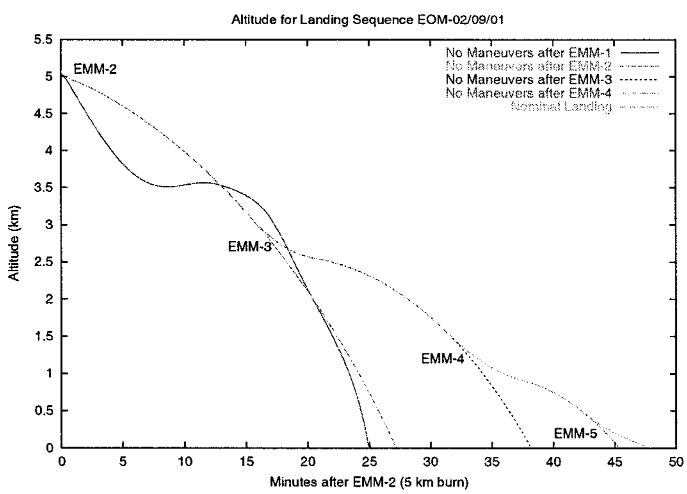

Figure 9: The altitude profile during the descent following EMM-2 through EMM-5. Trajectories are also shown for contingency cases where subsequent maneuvers failed to execute.

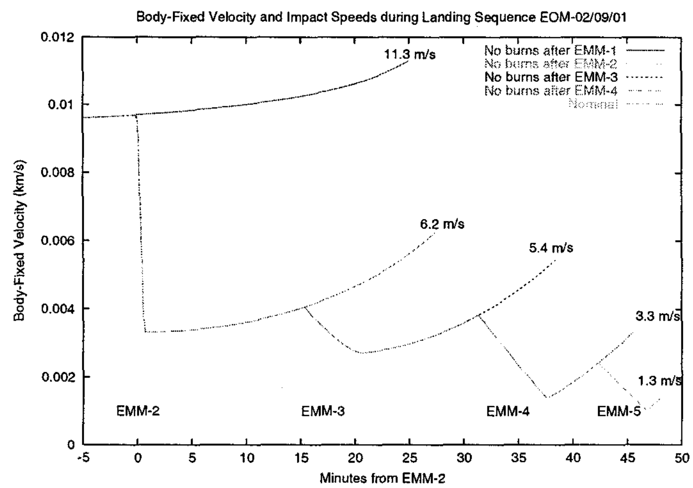

Figure 10: NEAR's body-fixed velocity during the descent following EMM-2 through EMM-5. Impact speeds are given for the nominal landing case and for contingency cases where subsequent maneuvers failed to execute.