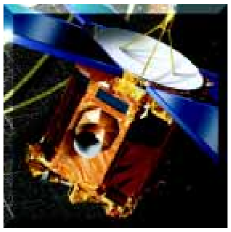

The Design and Testing of the NEAR Spacecraft Structure and Mechanisms

Theodore J. Hartka and David F. Persons

This article describes the primary structure and mechanisms of the Near Earth Asteroid Rendezvous (NEAR) spacecraft. Presented are design requirements as well as a description of the primary structure and mechanisms to meet those requirements. The test philosophy for this cost- and schedule-driven program is outlined along with a summary of the test flow and results. The structure and mechanisms were designed, assembled, and tested at APL, with most of the structure manufacturing subcontracted. Testing continued at Goddard Space Flight Center, and the final spin balance test was performed at Kennedy Space Center.
(Keywords: Design, Mechanisms, NEAR, Structure, Test.)

MECHANICAL DESIGN REQUIREMENTS

Spacecraft

The mission geometry for the Near Earth Asteroid Rendezvous (NEAR) spacecraft was designed to require a minimum of mechanisms with no gimbals. As the half angle from the spacecraft between the Earth and Sun was determined to be <30° for most of the mission, the high-gain antenna (HGA) could be mounted fixed and forward facing, and the fixed, forward-facing solar panels were sized for the 30° offset required to point the antenna toward Earth. Unfortunately, with the fixed antenna and array, the mission geometry dictated that the line of action of the large velocity adjustment (LVA) thruster be parallel to the plane of the solar panels and perpendicular to the axis of the antenna, that is, out the side of the spacecraft. Misalignment of the LVA thruster with the spacecraft's center of gravity would produce undesirable torques, which had to be minimized to conserve fuel. Therefore, precise tracking and control of the center of gravity became mission critical.

Weight also was of concern. Throughout the program, at least until the official flight weighing operation a few days before launch, the weight was always a few percentage points below what would be considered a comfortable margin.

The launch vehicle generated other spacecraft requirements. The interface with the Delta II 7925-8 launch vehicle is a 94-cm-dia. clampband. The spacecraft's center of gravity had to be located within about 0.1 cm of the geometric centerline of the launch vehicle because the Delta II third stage is spin stabilized, with no active thrust vector control while it is powered.

For the NEAR mission, the third stage was spun up to approximately 59 rpm before its release from the second stage. With the spacecraft attached to the third stage, any offset of its center of gravity would contribute to the third-stage injection error. In addition, the spacecraft's structure had to exhibit a minimum lateral stiffness of 15 Hz and minimum thrust stiffness of 35 Hz to avoid dynamic coupling with the launch vehicle or contact with the inside of the vehicle fairing.

McDonnell Douglas Aerospace (MDA) developed a relatively standard set of Delta II mechanical design environments for NEAR with appropriate modifications based on the spacecraft-to-vehicle coupled loads analysis. Summarized in the following discussion is the sequence of relevant launch events and the flight acceptance level requirements generated by each.

At *liftoff*, the spacecraft is subjected to quasi-static accelerations (Table 1) as well as acoustic noise reflected off the launch pad. Spacecraft flight acceptance acoustic vibration levels are 138.0 dB peak at 500 Hz and 144.6 dB overall for 5 s, relative to a sound pressure level of 2.0×10^{-5} N/m^2.

Table 1. Spacecraft center of gravity limit load factors.

Axis	Liftoff/ transonic (*g*)	Main engine cutoff (*g*)
Lateral	±3.0	±0.1
Thrust	–2.5/+0.2	–6.1/+0.6

The spin-stabilized third stage of the Delta II launch vehicle and the spacecraft are mounted on the second-stage spin table. Rotary motion is generated by firing four small tangentially mounted rockets. During *third-stage burn*, the resulting flight acceptance angular and centrifugal accelerations generated are significant (Table 2).

Table 2. Flight acceptance accelerations during third-stage burn.

Axis	Spin-up angular acceleration (rad/s^2)	Spin centrifugal acceleration (*g*)	Third-stage maximum (*g*)
Lateral/ radial	9.4	0–6	±0.1
Thrust	NA	NA	–8.5

Note: NA denotes not applicable.

Spacecraft separation from the third stage introduces pyrotechnic shock into the spacecraft due to the ignition of the separation clampband release ordnance. *After separation*, the spacecraft is despun, and the solar arrays are released by the activation of a yo-yo despin mechanism. This mechanism imparts a 4.1-rad/s^2 angular deceleration to the spacecraft through a 50-kg maximum tension in each despin cable.

Propulsion Subsystem

GenCorp Aerojet built the custom spacecraft propulsion subsystem based on APL's mechanical design and dynamic test requirements. Owing to its large size, this component was allocated nearly all of the volume inside the spacecraft and was treated more as a separate spacecraft than a subsystem. Figure 1 shows the propulsion subsystem in its partially disassembled shipping fixture. Over the course of the program, the initial set of quasi-static design loads set by the Laboratory was reduced after the MDA coupled loads analysis was completed (Table 3).

Propulsion subsystem modes with significant mass participation were designed to be greater than 55 and 30 Hz in the thrust and lateral directions, respectively. This provided a comfortable frequency offset, which

Figure 1. The propulsion subsystem in its shipping frame before spacecraft integration. Note that the large velocity adjustment thruster is mounted horizontally. The fuel and oxidizer tanks are mounted symmetrically with respect to the thrust axis to maintain the thrust vector through the spacecraft's center of gravity during fuel depletion.

Table 3. Final quasi-static loads for the propulsion subsystem.

Axis	Liftoff (g)	Third-stage cutoff (g)
Lateral	±3.0	NA
Thrust	–2.5/+0.25	–7.5
Radial	NA	Centrifugal acceleration due to a 70-rpm spin

Note: NA denotes not applicable.

minimized any coupling between the spacecraft's outer structure and the subsystem during spacecraft-level swept sine testing.

The structural loads test for the propulsion subsystem was performed at GenCorp Aerojet using swept sine vibration to apply 1.25 times the quasi-static limit loads in Table 3, with the sweep continued to 100 Hz to apply a protoflight-level test representation of Delta II vehicle transients to the structure. Unlike the spacecraft-level swept sine test, which used the response at the spacecraft's outer structure's fundamental frequency to generate loads, analysis done by GenCorp Aerojet revealed that insufficient participating mass was available in the propulsion subsystem's fundamental modes below 100 Hz to allow use of the recommended MDA swept sine test. (Mechanical testing is detailed later in this article.) Therefore, a higher-level test was developed for structural qualification of the subsystem (Table 4).

The subsystem was exposed to base-excited protoflight random vibration before spacecraft integration. Levels (Table 5) were based on structure-borne random vibration levels recorded by MDA during past Delta II flights. With the integrated subsystem surrounded by spacecraft structure, analysis predicted that the spacecraft-level acoustic test would fail to excite the propulsion subsystem to these minimum levels.

APL accommodated uncertainties involved with LVA thruster alignment by requiring that the thruster mounting be designed to allow a ±2.5-cm vertical and horizontal adjustment. The final design from GenCorp Aerojet included provisions for shimming the three LVA thruster mounting bolts to align the thruster with the measured spacecraft center of gravity.

Table 4. Sine vibration levels for the propulsion subsystem.

Frequency (Hz)	Acceleration
Thrust axis	
5–22	1.0 cm (double amplitude)
22–23	9.5 g (zero to peak)
23–100	1.4 g (zero to peak)
Lateral axis	
5–11	1.6 cm (double amplitude)
11–12	3.8 g (zero to peak)
12–100	1.0 g (zero to peak)

Note: Sweep rate for both axes = 4 octaves/min.

Table 5. Propulsion subsystem random vibration levels.

Thrust- and lateral-axis frequency (Hz)	Power spectral density level
<100	0.001 g^2/Hz
100–300	+9.3 dB/octave
300–700	0.030 g^2/Hz
700–2000	–3.2 dB/octave
2000>	0.010 g^2/Hz

Note: Overall amplitude = 6.0 g rms, duration = 60 s.

SPACECRAFT CONFIGURATION

Structure

The spacecraft is an octagon in cross section, with square forward and aft decks, standing about 280 cm tall and weighing about 806 kg. The forward and aft decks, eight side panels, and adapter comprise the spacecraft primary structure. It supports the main body of the propulsion subsystem and the six thruster modules and accommodates the LVA thruster, which protrudes through the side of the spacecraft. The stiffness of the aft deck dictates the thrust and lateral fundamental frequencies of the spacecraft structure.

The four removable side panels bolt to the edge of the forward and aft decks and provide access to the spacecraft's interior (Fig. 2). The four fixed side panels have integral magnesium edge members; horizontal flanges bolt into the underside of the forward deck and forward side of the aft deck. Adjacent side panels bolt together along their vertical inner edges through flanges formed by the magnesium vertical edge members in the fixed panels. The closing of the outside of the joint between side panels is accomplished by an angle section aluminum cover that runs the length of the joint vertically. The cover and side panel edge members form a column at each corner of the octagon that carries most of the loads from the forward deck to the aft deck (Fig. 3). The 68.6 × 144.8 cm aluminum honeycomb

Figure 2. The completed structural assembly with the four fixed side panels installed. The four removable side panels are not installed to allow access inside the spacecraft structure.

side panels weigh an average of only 1.1 kg each.

The 147-cm square aft deck mounts on an adapter that interfaces with the launch vehicle. The aluminum honeycomb aft deck, 5.7 cm thick with about 0.1-cm-thick face sheets, supports the main body of the propulsion subsystem through an internal ring that transfers launch loads from a 32-bolt interface ring in the subsystem into the top flange of the adapter. The adapter is an aluminum monocoque cylindrical shell machined from a single forging. The forward surface of the launch vehicle and the aft surface of the adapter form a 'V' section that is compressed together by a tensioned clampband. The clampband incorporates two pyrotechnically actuated bolt cutters. When the bolts are cut, the clampband releases the spacecraft from the third stage. Separation of the spacecraft from the vehicle is accomplished with four separation springs located at the separation plane. The spacecraft separates from the third stage with a relative velocity of 1.8 m/s.

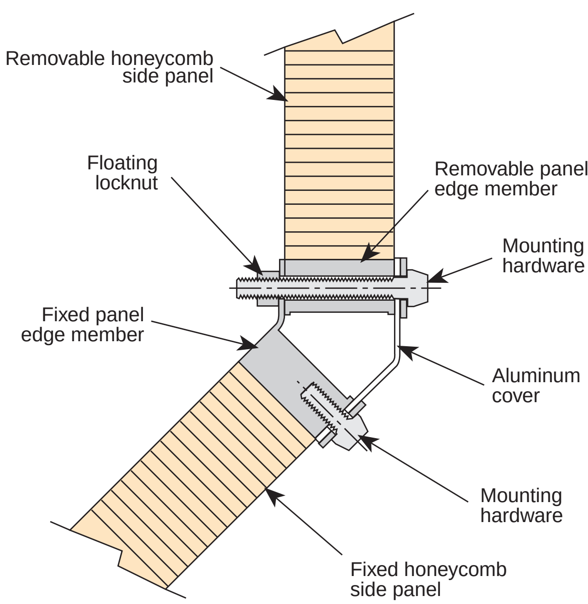

Figure 3. The cross section through the column formed by the side panel edge members and removable aluminum cover. The side panels are of aluminum honeycomb construction with magnesium inserts and edge members.

Packaging

Most of the electronics and instrument packages are mounted on the forward and aft decks, an arrangement dictated by the spacecraft's spin balance requirement and thermal considerations. The 150-cm-dia. HGA is mounted with the rest of the RF communications subsystem and four thruster modules on the forward side of the forward deck. The aft side of the forward deck supports the command and data handling subsystem and the terminal board. Most of the attitude determination and control subsystem is mounted on the forward side of the aft deck. The instruments, battery, and two thruster modules are mounted on the aft side of the aft deck. The propulsion subsystem and the spacecraft primary structure are joined mechanically by designing a single bolt circle interface between them, with four thruster pods mounted to the forward deck and two to the aft deck. This configuration allowed the propulsion subsystem to be tested independently since it carried no spacecraft loads. By not using the propulsion subsystem structure to help stiffen the spacecraft structure, the weight of the spacecraft primary structure did increase. However, in the interest of scheduling requirements, this weight penalty was deemed acceptable.

Instruments

Four instruments—the NEAR Laser Rangefinder (NLR), Multispectral Imager (MSI), Near-Infrared Spectrometer (NIS), and the sensor heads of the X-Ray/Gamma-Ray Spectrometer (XRGS)—are co-aligned on the aft side of the aft deck. Two solar monitors, which are part of the XRGS, are mounted on the forward side of the forward deck so that they are exposed to direct Sun for most of the mission. The NLR, MSI, and NIS are optical instruments, which are packaged as far away from the two thruster modules on the aft deck as possible. In addition, the co-aligned fields of views are rotated 180° from the LVA thruster. These steps minimize the effect of contaminants from the thruster plumes on the sensitive optics.

A fifth instrument, the Magnetometer, is mounted above the feed of the HGA. Originally it was located at the end of one of the solar panels, but the additional weight required to stiffen the panel and the additional cost to wire it to minimize its magnetic field combined to make that location unacceptable. Special hardware and materials were required to minimize the local magnetic field of the HGA feed.

Weight

As noted earlier, schedule was the foremost design driver for the primary structure and mechanisms. However, wherever prudent, weight was minimized. Most of the honeycomb inserts and edge members are made of magnesium instead of aluminum. The NEAR HGA represents the first Space Department use of GrCE (graphite cloth, cyanate ester) material, resulting in a 150-cm-dia. antenna assembly that weighed only about 6.4 kg. To meet stiffness requirements, the propulsion subsystem is constructed completely of GrEp (graphite cloth, epoxy) honeycomb.

A large thermal radiator was required under the RF system DC/DC converters and power amplifiers. Rather than using aluminum for the radiator, a beryllium/aluminum alloy called AlBeMet was chosen for its high thermal conductivity and low density. The constant attention given to minimizing weight throughout the program enabled the spacecraft to measure 0.5 kg below the maximum liftoff weight.

Mechanisms

The only mechanisms required for the NEAR mission were the solar array deployment hinges and the spacecraft despin system. The saloon door–type hinges used on the solar arrays are a copy, with a few minor changes, of the hinges used on the APL Geosat satellite. A pair of hinges deploys each solar panel 90°, with a hard stop present at 135° so that the panel does not contact the HGA. Each hinge uses a steel torsion damper to stop the panel at 120°. The hinge is in its free condition when the panels are fully deployed. Each hinge pair provides ±29.8 N-cm of restoring torque to counter any torque on the panels induced by spacecraft maneuvers. All data received after launch indicated that all four panels deployed 90°.

The yo-yo despin mechanism performs a dual role: it despins the spacecraft and holds the solar panels in the stowed position during launch. The despin system design has been used successfully on dozens of APL satellites, most recently on Geosat. Two wire rope cables are each wrapped twice around the spacecraft (Fig. 4). A fitting on each cable attaches to a hook on one end of the despin cable. The despin cable is then wrapped around the spacecraft, engaging eight cable guides called "combs," two mounted on each solar panel. Engaging the cables to the solar panel combs restrains the panels in the launch configuration, thereby eliminating the need for separate panel release mechanisms. Attached to the other end of the cable is a despin weight, which is fastened to a bracket on the spacecraft and held in place with a cam on a spring-loaded plunger. The plunger is restrained by a common cable called the release cable, which is routed through a tube into a pyrotechnically actuated cable cutter.

During launch, the third stage of the Delta II launch vehicle and the spacecraft are spun to 59 rpm about the launch axis before third-stage ignition. After firing is complete, the third stage is separated from the spacecraft. Switches at the separation plane detect separation, and a signal is sent to cut the release cable. The spring-loaded plungers then release the despin weights. The spacecraft's angular momentum is transferred to the cables and weights as they unwrap from the spacecraft. When the despin cables reach the hooks, they slide off simultaneously. The weights are sized to despin the spacecraft to within 1 rpm of zero. Postlaunch telemetry indicated that the spacecraft was despun to a 0.8-rpm residual spin rate.

MECHANICAL TESTING

Two constraints were the most influential in determining the philosophy and final makeup of the spacecraft mechanical test program: (1) scheduling and logistical issues, as noted earlier, dictated independent development of the propulsion subsystem and the spacecraft structure, and (2) system-level mechanical testing could only take 4 weeks. Integration of the spacecraft could not begin until the propulsion subsystem was delivered to APL in the first week of May 1995. This hard date compressed the integration and test schedule severely. In addition, the cost and

Figure 4. The flight despin system. The two despin cables were each wrapped twice around the spacecraft, guided by the combs mounted on the solar panel edges. Foam pads were positioned between the cables and the solar cells to prevent cell damage during vibration.

scheduling impact of mass simulators and the extensive use of long lead aluminum honeycomb for the spacecraft structure resulted in the decision not to build an engineering model qualification structure. As a result, APL adopted a protoflight approach for the spacecraft test program based on extensive experience from other programs. This decision allowed simultaneous mechanical qualification of the spacecraft's primary and secondary structure, solar arrays, hinges, and despin system.

For protoflight testing, MDA recommends a spacecraft mechanical test program consisting of static load or centrifuge testing, acoustic testing, clampband-induced shock testing, and spin balance testing (for a vehicle with a third stage). Swept sine testing to generate protoflight-level structural loads is permitted in place of static or centrifuge testing. Resonant responses can be limited to produce load levels in the structure equal to those uncovered in the coupled loads analysis. Swept sine testing was chosen for the NEAR program in light of the extensive heritage this technique has at APL.

As in previous APL protoflight programs, the solar panels were not vibrated as components, but rather were qualified at the system level along with the spacecraft structure. This allowed the array to be vibrated using the actual boundary conditions for flight while simultaneously qualifying the array deployment hinges, ball and socket supports, and despin cable combs.

Mechanical testing began the last week of September and continued through the third week of October 1995. The Laboratory conducted sine vibration and pyrotechnic shock tests; Goddard conducted spin balance, mass properties, and acoustic vibration tests. A final spin balance test was successfully performed at Kennedy Space Center before launch.

Test Results at APL

Adapter Static Load Testing

Recall that the propulsion subsystem and spacecraft were designed to have sufficient separation between their fundamental natural frequencies in the lateral and thrust directions so as not to couple during system-level swept sine vibration testing. This separation required that the spacecraft adapter structure be qualified by performing a static load test.

Two load cases, liftoff and third-stage maximum acceleration, were applied to the upper flange of the spacecraft adapter. The loads were reacted through an MDA-supplied and -installed nonflight clampband and test payload attach fitting (PAF). With the load levels set at 1.25 times the maximum expected loads derived from the MDA coupled loads analysis, displacements, load, and strain were measured. The lateral load case was applied twice, with the lateral component rotated 90°. Each load was maintained for 5 min with no changes noted in the adapter. Hydraulic cylinders were used to apply the load into the adapter through a welded steel assembly that duplicated the aft deck interface with the adapter.

Previbration Mechanism Testing

Before integration with the spacecraft, a pair of solar array hinges was randomly chosen for a low-temperature deployment test in the APL vacuum chamber to (1) demonstrate that the design displayed adequate thermal clearances and (2) size the torsional damper so that the panel would halt before impacting the over-travel stop. An array mass simulator was designed to mimic the bending stiffness and inertia of the solar panel and was *g*-negated using an aluminum support frame. The simulator was successfully deployed and the flight torsional damper size established.

Before beginning spacecraft-level environmental testing, spacecraft commands were used to fire the ordnance for the NLR, NIS, and MSI instrument doors to demonstrate proper door operation and verify the operation of the electrical harness. A limit switch on the MSI door failed to function and was re-adjusted.

Demonstration of the proper deployment of the yo-yo despin system was necessary before environmental testing began. The despin cables were tensioned to 18.1 kg, which was the maximum predicted tension at the instant of deployment. After successful release of the despin weights, the cables were walked around the spacecraft and allowed to release from their hooks to verify the cable routing chosen for NEAR.

Spacecraft Sine Vibration Testing

Protoflight swept sine testing consisted of one sweep through the specified frequency range at a logarithmic sweep rate of 4 octaves/min. The protoflight levels recommended by MDA are listed in Table 6.

Table 6. MDA-recommended prototype levels.

Frequency (Hz)	Acceleration
Thrust axis	
5.0–7.4	1.3 cm (double amplitude)
7.4–100	1.4 *g* (zero to peak)
Lateral axis	
5.0–6.2	1.3 cm (double amplitude)
6.2–100	1.0 *g* (zero to peak)

Note: Sweep rate = 4 octaves/min.

To support the structural qualification effort, an MSC-NASTRAN test finite-element model (FEM) was created using an updated version of the coupled loads FEM. The test levels shown in Table 6 were input in a forced response analysis to predict the response limiter levels required during swept sine testing. Responses at about 50 locations in the model were recovered and plotted for each axis. These locations corresponded exactly to the locations used in the coupled loads analysis performed by MDA.

The spacecraft was placed in full flight configuration except for the nonflight despin helper cables (loosely mounted), certain aft-deck/aft-side thermal blanket simulators in place of flight blankets, and a nonflight safing plug. The spacecraft was attached to the vibration table through a Delta II test PAF and test clampband. The propulsion system tanks were filled with deionized water to represent a fully loaded system at launch. The propellant tanks and the helium tank were pressurized to 0.96 and 11.70 MPa, respectively, with nitrogen, duplicating the flight mass while giving at least a 4:1 safety factor relative to burst pressure to allow for personnel access during testing. Testing, including electrical functional testing after each axis of vibration, proceeded smoothly, finishing in less than 4 days.

x- and y-axis tests. The results for both lateral-axis tests are presented together owing to their similarity for this essentially symmetric spacecraft. The NEAR FEM gave good agreement with the fundamental frequencies uncovered during the testing (Table 7).

Predictions for the spacecraft's outer structure and propulsion subsystem correlated well with test results. The FEM did not account for the nonlinearities associated with the solar panel hinges and the boundary condition imposed on the solar panels from the two double-wrapped tensioned despin cables that strap down each panel.

Listed in Table 8 are the protoflight accelerations (1.25 times maximum expected) as given by the NEAR coupled loads analysis to be achieved during lateral-axis structural testing. The protoflight sinusoidal test levels were successfully applied to the integrated spacecraft. Table 8 shows the accelerations generated at the fundamental spacecraft lateral resonance of 16.6 Hz. As shown, the acceleration levels needed to generate protoflight structural loads in the outer structure were achieved.

z-axis tests. The NEAR FEM again gave good agreement with the fundamental frequencies uncovered during thrust-axis testing, as seen in Table 9.

Table 7. FEM summary for *x* and *y* lateral axes.

Item	*x*-axis FEM prediction (Hz)	*x*-axis actual frequency (Hz)	*y*-axis FEM prediction (Hz)	*y*-axis actual frequency (Hz)
Spacecraft outer structure	17.1	16.7	17.1	16.5
Propulsion subsystem	29.3	32.4	29.3	32.3
Solar array				
1st mode	42	55	42	55
2nd mode	90	80	90	92

Table 8. Acceleration levels generated for *x* and *y* lateral axes.

Location	*x*-axis level to achieve (*g*)	*x*-axis level generated during testing (*g*)	*y*-axis level to achieve (*g*)	*y*-axis level generated during testing (*g*)
Forward deck	4.6	5.0	5.3	5.3
Spacecraft center of gravity	2.0	3.0	2.0	3.1
Aft deck	1.2	1.2	1.2	1.2
Propulsion subsystem center of gravity	2.7	2.5	2.7	2.7

Table 9. FEM summary for *z* thrust axis.

Location	*z*-axis FEM prediction (Hz)	*z*-axis actual frequency (Hz)
Spacecraft outer structure	31.5	32.6
Propulsion subsystem	72.4	70.4
Solar array center of gravity	19.0	24.5

The predictions once again correlated well with the test results. Although the panels proved to be slightly stiffer than forecast, the FEM correctly predicted the large *lateral* response of the solar panels to thrust-axis vibration, which was caused by the rotation of the rather long, offset panel hinge arm about the hinge axis.

Listed in Table 10 are the protoflight accelerations (1.25 times maximum expected) to be achieved during *z*-axis structural testing as provided by MDA for the NEAR spacecraft's thrust-axis third-stage cutoff event. Accelerations were generated at the fundamental spacecraft resonance of 32.6 Hz. The propulsion subsystem had previously been qualified to 10.6 *g* in thrust. An APL memorandum recommended $1.25 \times 2.48 = 3.1$ *g* (maximum expected from coupled loads analysis) as a reasonable level for the propulsion subsystem to achieve during spacecraft thrust-axis testing. The *z*-axis protoflight sinusoidal test levels were successfully applied to the integrated spacecraft.

Table 10. Acceleration levels generated for *z* thrust axis.

Location	*z*-axis level to achieve (*g*) (–1 *g* added by gravity)	*z*-axis level generated during testing (*g*)
Forward deck	+8.4/–9.4	12.4
Spacecraft center of gravity	+8.4/–9.4	8.7
Aft deck	+8.4/–9.4	6.4
Propulsion subsystem center of gravity	3.1	2.8

The thrust-axis mode shape of the spacecraft's outer structure at 32.6 Hz resulted in an uneven acceleration distribution thoughout the structure. Limit accelerometers were set on the basis of mass distribution within the structure to generate the proper structural loads at the top of the spacecraft adapter. When factored by mass, the outer structure saw an equivalent quasi-static acceleration of +8.9/–9.9 *g*, comfortably above the +8.4/–9.4 *g* needed.

Postvibration Mechanism and Shock Testing

The primary source of shock for the NEAR spacecraft is the clampband ordnance used to separate the spacecraft from the PAF. The pyrotechnic devices used to release the solar arrays and protective covers also produce a local source of shock. To expose the rest of the NEAR spacecraft to these shock sources, testing was conducted at the spacecraft level by firing the ordnance and allowing the release of the clampband, solar array, instrument covers, etc. These tests were performed twice to account for scatter in the shock spectrum.

The spacecraft separation shock test used a test PAF and clampband supplied by MDA that was assembled to the spacecraft adapter and separated using live ordnance. The maximum expected shock levels for an MDA 3712 PAF configuration peak at 4100 *g* between 1500 and 3000 Hz. The clampband was successfully separated from the spacecraft adapter and dropped approximately 46 cm onto foam.

Spacecraft commands were used to fire the ordnance for each instrument door to demonstrate proper door and electrical harness operation after vibration. All doors successfully deployed.

The despin mechanism configuration consisted of the flight despin weight assembly and cabling, with flight plunger lubrication, mounted on the spacecraft.

The despin cable tension was set to the flight tension (9.3 to 11.3 kg). Both pyros were fired using spacecraft commands, successfully deploying both weights with no problems noted.

Test Results at Goddard Space Flight Center

Acoustic Testing

To verify the ability to survive the liftoff acoustic environment and provide a workable acoustic test, MDA recommends a payload-level acoustic test in a reverberant sound pressure field. The maximum expected flight levels for the NEAR fairing were provided by MDA. For spacecraft acoustic testing, MDA requires increasing flight acceptance levels by +3 dB for protoflight-level tests. The protoflight duration is 60 s, identical to the flight acceptance duration, with a protoflight overall sound pressure level of 147.6 dB relative to a reference sound pressure level of 2.0×10^{-5} N/m^2, with a 141-dB peak at 500 Hz.

The test was conducted in Goddard's Reverberate Acoustical Test Facility. The spacecraft was placed in full flight configuration as it was during sine vibration testing, and was similarly attached to a handling dolly. The propulsion subsystem tanks were also filled with deionized water and pressurized as described previously.

An overall sound pressure level of 147.2 dB was achieved, fulfilling the requirement of 147.6 ±1 dB. No change or degradation of any part of the spacecraft was noted.

Mass Properties Testing

The objectives of this test were to determine the mass properties of the NEAR spacecraft with the integrated propulsion subsystem and to perform a spin balance of the system. To maintain third-stage orbit insertion accuracy, MDA requires that the NEAR spacecraft meet the following requirements: principal z axis (spin axis) within 0.13 cm of the geometric z axis, and principal z-axis misalignment of less than 0.25° with respect to the geometric z axis.

To control spacecraft torques during burns involving the 45.4-kg side-mounted thruster, the axis of the LVA thruster must intersect the spacecraft center of gravity in the orbital configuration within 0.64 cm. In addition, thrust-axis inertia must be precisely known to accurately size the yo-yo despin weights. To properly distribute the mass of the propulsion subsystem, the mass properties of the system must be measured with the pressurized propulsion subsystem fuel and oxidizer tanks filled with deionized water.

The spin rate during spin balance operations was predicted to vary from 25 to 69 rpm. For the NEAR spacecraft, the 3-sigma maximum spin rate of the third stage was predicted to be 69 rpm. As a qualification strength test, during spin balance testing, the spacecraft was scheduled to be exposed to this spin rate for 10 min.

The test article consisted of the NEAR spacecraft with the integrated propulsion subsystem. The spacecraft used the dummy attach fitting and dummy clampband for attachment to the mass properties mechanical ground support equipment.

Originally scheduled to be performed before vibration testing at APL, a failure in the Laboratory's mass properties machine (the day before testing was to have begun) caused the test to be rescheduled at Goddard. To streamline the test schedule there, mass properties testing of the tare items was performed before acoustic testing, with mass properties of the spacecraft measured after acoustics. The objective of the test was to generate a partial set of mass properties, in particular the spacecraft spin balance, thrust-axis inertia, and center of gravity along the thrust axis. Based on the measurements, the spacecraft mass properties spreadsheet was updated to duplicate the results; the spreadsheet, in turn, was used to generate the required items that were not measured. Confirmation of the accuracy of the value for thrust-axis inertia was obtained after launch when the NEAR yo-yo mechanism successfully deployed and despun the spacecraft to less than 0.8 rpm.

A proof test of the spacecraft's ability to withstand a 3-sigma-high (69 rpm) spin rate could not be accomplished because of the lack of horsepower available from the Goddard spin machine. This test was accomplished during Kennedy Space Center spin operations. The loads generated by the proof spin test were not design loads and were enveloped by the sine vibration testing done previously.

Spin balance testing was accomplished in three spin runs with 7.3 kg of weights required, whereas only 5.1 kg was predicted before testing. After a careful review of the final location of the center of gravity of the electrical harness (the least accurately known of the items in the sheet), the mass properties spreadsheet was used to shift slightly the locations of the various components of the harness, thus resulting in the proper prediction of the magnitude and location of the actual balance weights. Figure 5 illustrates the test setup for the horizontal mass properties measurements—center of gravity along the thrust axis and inertia about a lateral axis—using the turnover yoke. After thermal vacuum testing, the spacecraft was weighed again for the official preshipment measurement.

Test Results at Kennedy Space Center

Final spin balance operations were held in the NASA Payload Processing Facility with the fully flight-configured NEAR spacecraft including the spin weights added during spin testing at Goddard. Because of the

Figure 5. The spacecraft with solar panels removed and mounted on the turnover yoke for horizontal mass properties measurements at Goddard Space Flight Center.

excellent job done by the center's personnel to center the test ground support equipment on the spin table, a total of 1.0 kg of balance weights was *removed* from the spacecraft for balancing. The spacecraft's final tilt angle was calculated to be 0.10° (versus 0.25° maximum allowable), and the final static offset of the spacecraft center of gravity was calculated to be 0.01 cm, versus the 0.13-cm maximum allowable. Five runs were required.

SUMMARY

Building on the Laboratory's extensive protoflight spacecraft experience, the NEAR spacecraft mechanical design met all schedule, weight, and performance requirements. By adapting existing mechanism designs, scarce manpower and schedule resources were made available for other mechanical design tasks. Prudent application of weight-saving techniques enabled the NEAR spacecraft to weigh 0.5 kg under the maximum allowable launch weight without causing any adverse schedule impact. Early decisions to develop a separate propulsion subsystem structure with a simple interface to the spacecraft allowed parallel development of the two systems. This allowed integration to begin in May 1995, which maximized the time available for spacecraft integration and testing. An efficient test program mechanically qualified the integrated spacecraft and mechanisms in 1 month. The program enjoyed a successful launch on 17 February 1996 with the spacecraft on its way to 433 Eros.

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