

# THE NEAR DISCOVERY MISSION: LESSONS LEARNED

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## Abstract

Under a contract from NASA The Johns Hopkins University Applied Physics Laboratory built and launched a spacecraft that will rendezvous and orbit the near earth asteroid 433 Eros. The Near Earth Asteroid Rendezvous (NEAR) spacecraft is the first under NASA's Discovery Program, which is a series of low cost solar system missions. While in orbit around Eros the spacecraft will measure the bulk, surface, and internal properties of the asteroid for 10 months. This paper describes the lessons learned from design, test, and fabrication that are appropriate to other programs in quick development, or of an interplanetary nature.

## I. Introduction

In any program there are important lessons learned that can benefit other programs. The NEAR mission is unique in its opportunity for lessons learned in that it was the first interplanetary mission for The Johns Hopkins University Applied Physics Laboratory (JHU/APL). In addition, because of the short 27 month development time, the length of the design iteration and optimization stage was short.

The building of the NEAR spacecraft caused the generation of 13 waivers and 227 Problem/Failure Reports. Waivers were created before any subcontracted subsystem was accepted as flight hardware if performance did not meet specification. Waivers also were created when the final spacecraft configuration, after completion of all testing, fell short of a performance goal.

The Problem/Failure Reporting (PFR) system requires all flight hardware anomalies

to be documented beginning with environmental acceptance testing at the subsystem or box level and continuing throughout build-up, integration and test of the complete spacecraft. The PFR system requires that the anomaly be documented by the test engineer and analyzed and corrected by the lead subsystem design engineer. Verification of the implementation of the correction(s) requires the signatures of a review board consisting of the subsystem design engineer, the spacecraft integration engineer, the spacecraft systems engineer and the program performance assurance engineer.

Much of the history of the NEAR spacecraft is contained in the records of the waivers and PFRs. Section II of this paper will give an overview of the NEAR design philosophy. Section III summarizes the parts procurement and subsystem fabrication activities. Section IV discusses the significant waivers for NEAR. Section V gives a detailed exposition of the notable technical problems that occurred in the execution of the subcontracts. Section VI summarizes all 227 PFRs by presenting the results of a statistical analysis. Section VII discusses design decisions. The summary and recommendations are found in Section VIII.

## II. NEAR Mission Design Philosophy

Reliability of the NEAR spacecraft and instruments was maximized at the outset by limiting the number of movable and deployable mechanical and electromechanical systems since such systems have proven to be frequent sources of failures in recent near-Earth and interplanetary missions. New technology was used when it was

necessary for the execution of the mission and not because it was neat to do.

The reliability philosophy was implemented specifically by:

- (1) not allowing any deployable instrument booms even for the magnetometer;
- (2) using fixed RF communications antennae;
- (3) using deployable but fixed solar arrays;
- (4) permanently fixing the instruments onto the spacecraft structure, allowing them movable filter wheels and requiring that deployable covers have viewing ports in them.

New technology was employed:

- (1) in the gallium arsenide solar arrays necessary for the proper energy/mass density in a photovoltaic system for a deep space mission;
- (2) in the IBM/LORAL 4MX4 LUNA-C DRAMs composing the solid state recorder taking advantage of the mission's very mild radiation environment and eliminating the reliability hazards inherent in tape recorders;
- (3) in the gamma ray spectrometer that is constructed by having the sodium iodide detector crystal inside a bismuth germanate crystalline shield for rejection of the interplanetary gamma ray background;
- (4) in the laser rangefinder which was a scaled down but more reliable version of one flown on Clementine; and
- (5) in the system of software autonomy rules developed to maintain the spacecraft and continue the mission between infrequent contacts with Earth through the Deep Space Network especially during the cruise phase.

### **III. Parts Procurement and Subsystem Fabrication**

The parts procurement process was streamlined for the NEAR program. Four major changes were implemented.

- (1) The purchasing and receiving functions were consulted early in the program to obtain cooperation in expediting the procurement of flight parts. As a result orders for long lead items were placed about 12 weeks sooner than the normal processing had previously produced.
- (2) The number of specialized purchase instructions (PIs) was minimized so that the activity of writing the PIs would delay procurement only when absolutely necessary.
- (3) Once parts orders had been placed with manufacturers or distributors, periodic follow-ups were made to monitor and expedite the delivery of the order. In several cases orders had to be placed with a second vendor when it became apparent that the original vendor would not meet the quoted delivery date.
- (4) Incoming inspection and testing of flight parts at JHU/APL was minimized. Only activities that verified receipt of the correct devices and added value for flight quality were performed. Repeating tests or screening done by manufacturers was eliminated.

Generic parts issues were the cracking of ceramic capacitors in both active devices and filtered connectors and the lack of a standard foot print for D shell connectors (filtered and unfiltered) from different vendors.

The project leader from the fabrication department was a member of the engineering team and provided weekly status reports at the team meetings. These reports tracked such things as board designs, drawing sign-offs, board fabrication, assembly and delivery. A total of 118 flight boards were built. Tracking flight spares from the beginning is recommended since these items often become flight hardware. Tracking revised subsystem schedules should also be done since it makes the downstream schedule compression visible.

Separate, small shop focus meetings were held with the program office each week. Issues with individual subsystems were resolved in these smaller gatherings. A packaging engineer was assigned to each subsystem or box. At the end of the board/box design phase, but before fabrication, a Fabrication Feasibility Review was held to resolve major issues as early as possible. When flight fabrication was completed, photo documentation was necessary but it had to be expedited so that it did not delay test and integration.

Some specific fabrication issues of general interest will be mentioned in closing this section. Specifications on paint thickness and some mechanical tolerances should be relaxed since many configuration control discrepancy documents were created due to overly tight specs for which the resolution was "use as is." All cosmetic problems should not be ignored since they are sometimes symptomatic of poor workmanship; but overly conservative specifications do cost time, money and paper.

We found that laser cut co-therm gaskets could be used if cleaned properly of carbon deposit after cutting. We also avoided massive point-to-point wiring as much as possible.

#### **IV. Waivers**

Of the thirteen (13) waivers written for the NEAR mission, four involved the Inertial Management Unit (IMU). The first of these four waivers concerned the gyro operation and control when it was found that although four gyros were on board, it took an unacceptably long time to power and utilize all four simultaneously. NEAR uses only three gyros at a time.

The vibration sensitivity of the IMU required a second waiver which changed the operation to keep the IMU unpowered throughout launch; turning it on after the spacecraft separated from the Delta rocket to avoid any substantial drift. It was determined that the 450N thruster on board the spacecraft produced a vibration level that was too low to disturb the IMU.

The third waiver for the IMU system was because it did not meet specifications with respect to six of twenty performance parameters. The rate bias and rate bias stability deficiencies were compensated in the on-board guidance and control software. The other four parameters such as turn-on time were marginal versus the specifications and could be accepted as inconsequential.

In post vibration measurements two gyro alignment axes were determined to have shifted by 34 and 70 arcsec instead of within the specified 20 arcsec stability requirement. A fourth waiver on the IMU system was then written and the latest values for gyro axes alignment were inserted as updated information in the IMU EEPROM.

It should be noted that the constraints of the rapidly paced NEAR schedule meant that the IMU had to be integrated with the spacecraft in mid-summer 1995 allowing no time for fine tuning and mechanical stability adjustments.

The remaining significant waiver concerned the solar panel interconnects. The solar cells on the panels were interconnected by a web of silver which was advertised as a mesh but was really a rigid metal lattice. Silver was used for its low resistivity on the NEAR panels since the mission would be in sunlight most of its life and not experience the many light-dark or hot-cold cycles of a low Earth orbiter.

However, testing of the qualification panel in temperature cycling and thermal vacuum yielded breakage levels in the silver interconnects which revealed the panels to be limited life items. Therefore, the thermal vacuum temperature profile for qualification of the flight panels was descope to reduce stress. The range of the temperature profile was reduced by 10°C which still met standard workmanship test criteria but was not as robust as the cycle normally employed.

Substantial thermal cycling of the qualification panel and test coupons indicated that

the solar panel interconnects could readily meet the NEAR mission four year requirement if the number of thermal cycles with temperature excursions exceeding  $T = 30^{\circ}\text{C}$  were limited to a number in the hundreds rather than in the thousands. Mission operations was informed and is limiting the number of spacecraft swings from Sun-pointing for optimum power to Earth pointing for optimum communications especially during cruise mode at times when the sun-spacecraft - Earth angle and, hence, the temperature cycle is largest.

## **V. Technical Problems Encountered on Subcontracts**

### **DC/DC Convertors**

Both performance and fabrication problems were experienced with the convertors. The excessive overshoot of one particular kind of convertor was solved by using a clamping diode circuit of JHU/APL design. Suspected damage to capacitors during the fabrication of the convertors was eliminated as an issue when the manufacturer at JHU/APL's direction mounted the capacitors on a separate daughter board and eliminated high thermal mechanical stress due to the soldering of the mother printed circuit board by mounting the capacitors in the last fabrication step before lidding the convertors, after all soldering had been completed.

### **4MX4 DRAMs**

During the manufacture of the NEAR Solid State Recorders the vendor notified us that in some lots 25-50% of the LUNA-C DD3 4MX4 DRAMs exhibited high standby currents. Currents with values as high as 20-25 mA were observed compared to the manufacturer's specified value of 1 mA. Questions about the end-of-life capability and performance of the Solid State Recorders led JHU/APL to conduct two life tests; one for 1000 hours at  $85^{\circ}\text{C}$  with a DRAM distribution representative of that used

for the memories in the NEAR recorder and a second for 1000 hours at  $125^{\circ}\text{C}$  with a DRAM distribution screened for high ISTBY samples.

Forty-three (43) devices successfully completed the  $85^{\circ}\text{C}$  life test. Twenty (20) of these had standby currents exceeding 1mA initially and twenty-three (23) did not. The distribution after 1000 hours was exactly the same. All other electrical functions and parameters were acceptable. The high current leakage while undesirable did not prove to be fatal.

Twenty-two (22) devices successfully completed the  $125^{\circ}\text{C}$  life test with one DRAM that was within specification initially having an ISTBY of 2 mA after 1000 hours. Again all other parameters and functions were acceptable.

The Solid State Recorders were accepted as fabricated by the vendor.

### **Oscillators**

Two oscillators, one in the Telemetry Control Unit (TCU) and another in the Command and Telemetry Processor (CTP), from the same vendor experienced stability problems. The TCU oscillator had excessive noise and instability over its required temperature range. After several interchanges with the vendor and an attempt at correcting the problem, we replaced the oscillator with a similar but not identical one from a second vendor. Fortunately, the new oscillator was in the flight stockroom as it had been purchased for the NEAR Guidance and Control system.

The CTP oscillator had an unterminated CMOS input as a result of a design error which created voltage instability and threatened to cause failure due to overheating. Samples were returned to the vendor, corrected and requalified in the CTP system at JHU/APL after a more aggressive approach was taken by the program with respect to design review and quality assurance support.

## **Magnetometer Electronics**

The NEAR Magnetometer was Government Furnished Equipment (GFE). However, shoddy fabrication procedures produced an electronics board on which many Field Effect Transistors (FETs) were shorted to board traces by flux contamination. JHU/APL replaced more than 20 of these 2N5434 FETs with those of a second manufacturer from existing flight stock and cleaned and requalified the board.

## **Infrared Spectrometer Pin Puller**

After more than ten (10) test firings the NIS (NEAR Infrared Spectrometer) pin puller that opens the NIS door was not operating correctly. Inspection by mechanical design engineers indicated that some gouging and galling damage was occurring in the pin puller channels due to design deficiencies. Mating surfaces were refurbished in the JHU/APL shop and the number of subsequent pin pull tests was limited to less than five to prevent a repeat of the problem.

## **High Gain Antenna Point**

Upon receipt of the High Gain Antenna, it was tested in the JHU/APL thermal vacuum (TV) test chamber as part of its space qualification. Following the TV test the paint was chipped or cracked in four areas. Two possibilities existed as causes for failure; either the mold release compound from the dish manufacture had not been completely removed or the primer had not been applied correctly.

With the help of Goddard Space Flight Center personnel, the antenna dish was stripped, cleaned and repainted. It was then requalified both with respect to its performance on the antenna range and in the TV chamber.

## **Laser Rangefinder**

The laser transmitter encountered three major problems in its fabrication and qualification at the vendor and JHU/APL.

The first was the approval, use and derating of individual transistors and integrated circuits. In particular, the inadequate derating of a power FET led to several problems with the inrush current limiter of the NEAR transmitter. The gate-source voltage for a JANS2N6849 device was derated only 10% instead of the standard 25%. This condition led to failure in the limiter due to overshoot and was resolved by both eliminating a potential shorting configuration and finding an individual power FET device that was robust enough to withstand the overshoots encountered in testing.

The second problem area had to do with the qualification of a non-hermetic hybrid used in the laser power supply (LPS) as a Q-switch driver. The original hybrid chosen for the LPS had a large substrate which was cracked in thermal shock screening. A second smaller hybrid suffered failures in thermal vacuum qualification tests. Failure analysis attributed these failures to corona problems in the uncoated non-hermetic hybrids due to inadequate bake-out, contamination, wire pigtailed and a redundant process which applied protruding ball bonds over stitch bonds. Solution to this problem involved cleaner hybrids with conformal coating and vent holes in the hybrid lid. In addition, flight units underwent an extensive bake-out before LPS fabrication and were allowed to outgas for 36 hours upon entrance into vacuum or repressurization before power was applied to the LPS.

The third problem area was the vibration qualification of the laser optics train. Two separate failures occurred, one when qualifying the laser transmitter itself and a second when qualifying the complete rangefinder, which were caused by the inadequate epoxy attachment of a corner cube and a Risley prism respectively. In both cases substantially more epoxy had to be applied to these optical components and their mounts so that they could successfully meet NEAR vibration requirements.

## VI. Problem/Failure Reports (PFRs)

The most significant results of the PFR activity were highlighted by the statistical analysis of means on Poisson distributed count data produced by a four way classification. The four factors used for classification were spacecraft subsystem, cause of the anomaly, hardware versus software and the time period before the field activity at Goddard Space Flight Center (GSFC) and Kennedy versus the time period after the field activity began.

With respect to the analysis of the main effects of the four factors:

- (1) the Guidance and Control subsystem had a significantly higher number of anomalies while the Propulsion system had less;
- (2) there were more software anomalies than hardware ones;
- (3) significantly more problems were caused by design with fewer caused by workmanship and parts;
- (4) as many anomalies occurred after going to the field as before.

As Figure 1, the NEAR PFR matrix, shows, there were 60 PFRs written against the Guidance

		DESIGN		WORK-MANSHIP		TEST OP./PROCEDURE		PARTS		
		PREGSFC	POSTGSFC	PRE	POST	PRE	POST	PRE	POST	
RF	H	4		5	2	2	1	4		18
	S		1			1	1			3
COMM/TELEM	H	2	1	3				1		7
	S	5				1	3			9
PROPULSION	H	2		1		2				5
	S									0
POWER	H	2		2	6	5	3	1		19
	S	2					2			4
GUID & CONT.	H	1		3	1		1			6
	S	6	33			1	14			54
INSTRUMENTS	H	13	1	6	2	4				26
	S	2	5				4			11
SPACECRAFT	H	5	1	4	2			2	1	15
	S	14	7			1	2			24
INT/GSE	H					1				1
	S	10	11			1	3			25
		68	60	24	13	19	34	8	1	227

H = Hardware  
S = Software

Figure 1. NEAR PFR Matrix

and Control subsystem and only five against the Propulsion system. There were 129 software anomalies versus 98 hardware problems. The causes for 128 anomalies were attributed to design with only 37 due to workmanship and 9 to parts. There was no significant difference between the number of PFRs generated before going to the field (119) and the number generated during testing at Goddard and Kennedy (108). The differences that were demonstrated are significant at a 95% confidence level.

The Guidance and Control subsystem and its software will be discussed below. Other software anomalies occurred with the insertion of autonomy rules into the spacecraft command system. The greater number of software versus hardware anomalies emphasizes the complexity and amount of code implemented for NEAR

including JHU/APL's responsibility for mission operations. Design emerges as the primary cause for anomalies due to the extensive software design activities that occurred after integration of the spacecraft. Figure 2 shows that while the number of hardware PFRs decreased from October 1995 to February 1996, the majority of software PFRs occurred after October 1, 1995. Thus, the total number of PFRs before and after going to the field were not significantly different. It should also be noted that because of the holidays and the launch activity, the amount of testing in December 1995 and January 1996 was limited.

The analysis of means on the two factor interactions showed that greater numbers of anomalies were produced by the appropriate combinations of:

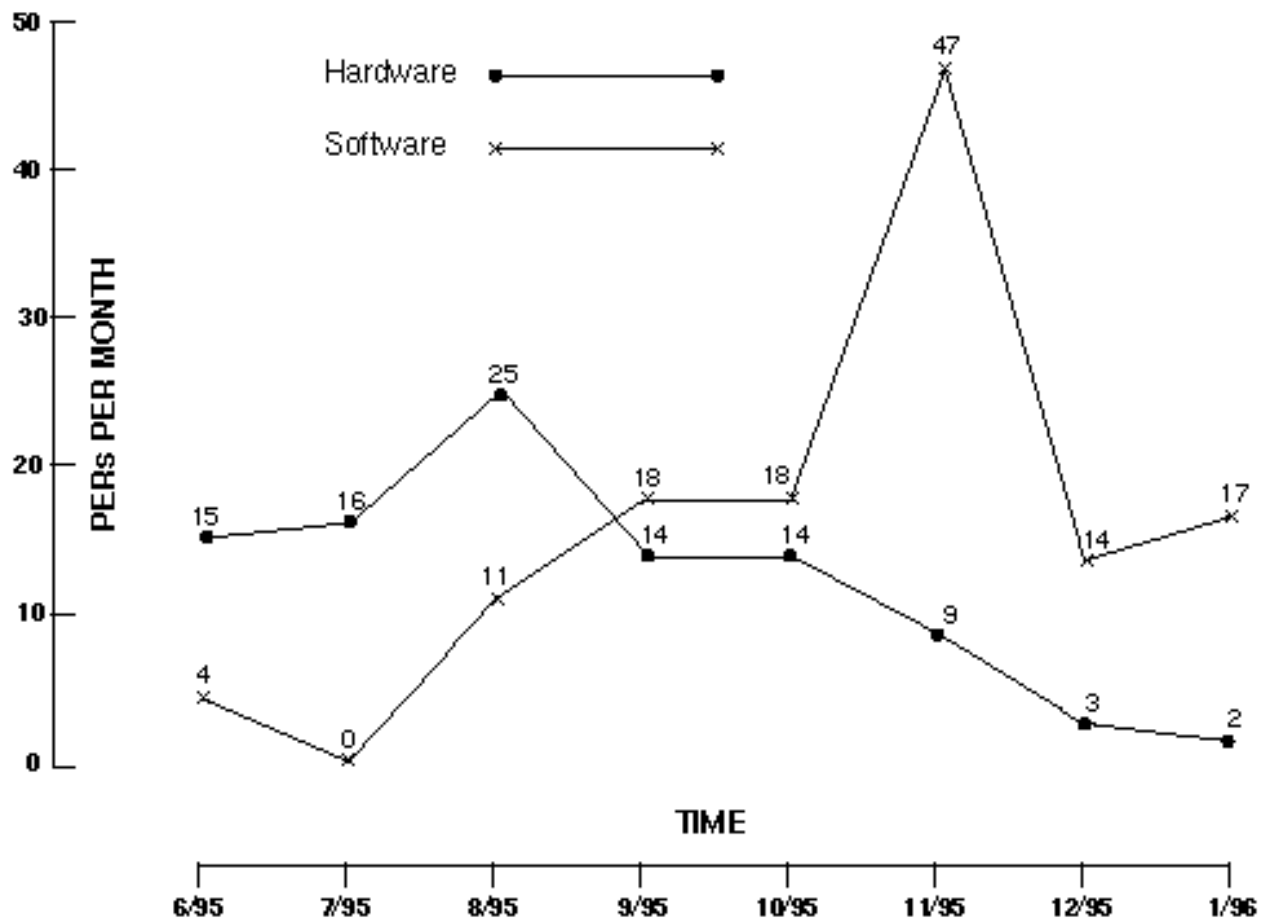


Figure 2. NEAR PFR Matrix

- (1) the design of the Guidance and Control subsystem software after going to the field due to the complexity of the G&C subsystem for an interplanetary mission and the fast paced NEAR schedule which required writing and debugging flight computer software in the field;
- (2) the design of the science instrument hardware before going to the field reflecting the less mature design of the instruments, particularly the x-ray/gamma ray spectrometer, and the design of doors and door opening mechanisms;
- (3) the basic spacecraft design before going to the field due to wiring, interface and connector problems (hardware) and the correct implementation of autonomy rules (software);
- (4) the ground support (test) equipment software design.

The low number of anomalies attributed to workmanship and parts after subsystem test and spacecraft integration can be directly linked to the excellent jobs done by the Technical Services Department with respect to configuration control of workmanship discrepancies during fabrication of the NEAR subsystems and by the Space Department's Reliability Group with respect to parts' procurement and screening (before the PFR system activities begin).

## **VII. Spacecraft Design**

In the course of working with a spacecraft design, shortcomings and disadvantages of various design decisions will inevitably be noticed. In a program with a short development cycle, improvements cannot always be implemented on the current program. During the NEAR development, several improvements for future designs were noted.

While the Command and Telemetry Processor (CTP) default autonomy rules are stored in EEPROM, it is not reprogrammable in flight.

New rules can be uploaded to RAM, but they are not preserved across processor resets. Some reprogrammability of the CTP EEPROM could have greatly simplified the autonomy design. For example, the response to Low Bus Voltage (LVS) is tailored to the portions of the mission with the lowest power margin, which corresponds only to about six months out of the four year mission. To recover from LVS during this period, the Solid State Power Amplifier must be unpowered for 24 hours, thus extending the time needed to reacquire the spacecraft. If the response to this condition could be permanently reprogrammed, the amplifiers could be turned on immediately after LVS throughout most of the mission, when there is ample solar array output.

Other simple changes could have improved the autonomy design. For example, significant effort is expended to guard against an analog-to-digital (A/D) converter failure that causes multiple channels to saturate. Unless carefully handled, a single failure of this nature results in false indications of multiple failures, triggering autonomous actions that could themselves cripple the spacecraft. By creating a special A/D converter output with the sole purpose of checking the A/D converter health, autonomy could be disabled on the affected side, if a failure was observed. Another improvement to the autonomy design would allow rules to operate on computed values. For example, the autonomy rules that check for short circuits must operate on current readings that vary with bus voltage. It is difficult to detect an over-current unit, while avoiding false triggers due to low bus voltage. Truly, the rules should operate on power, calculated from the voltage and current readings.

The telemetry system outputs fixed-length, 8800-bit transfer frames. In emergency mode, each transfer frame takes approximately 15 minutes to transmit. Much of the data contained in the nominal mode transfer frame are not needed for recovery from an emergency. If a



shorter transfer frame were defined in emergency mode containing only important parameters, the frame synchronization time would be shortened, and critical telemetry parameters could be sampled at a higher rate.

Although the command system hardware supports eight uplink rates ranging from 7.8 bits per second (bps) to 1000 bps, only two rates (7.8 and 125 bps) were implemented. While this marginally simplified the command system, it greatly limited the ability to change the on-board reprogrammable software. For example, to upload one of the redundant sides of the Guidance and Control subsystem will require 12 hours. Three uplink rates should have been implemented: the 7.8 bps emergency rate, the 125 bps rate which can be used over the medium gain antenna throughout the mission, and the 1000 bps rate over the high gain antenna for large memory uploads. Another way to mitigate the time required to reprogram is to partition the flight software so that changes can be made without requiring the entire memory to be uploaded. This approach, however, complicates the software design, adding unwanted risk.

All commands are decoded in the Command and Telemetry Processor software. The software is protected by two hardware watchdog timers: a 1 Hz timer and a 12.1 day timer. If the 1 Hz timer fails to catch a software problem, the command system will be unusable for 12.1 days. A hardware command decoder could have been implemented with a single additional high density programmable logic device to reset the processor from the ground.

The redundant command receivers are wired through unswitched power. This conservative approach is traditional for low Earth orbit spacecraft with omni-directional antenna coverage. The spacecraft is protected from a short circuit through fuses. This design is vulnerable, since a soft short circuit failure may not blow the fuse, but drains the spacecraft power system. Receiver reliability would also be

increased by unpowering a receiver when not needed. Finally, the power saving resulting from unpowering one receiver would boost the power margin by 3% and greatly simplify the low voltage emergency recovery procedures. If the command receivers were on switched power, such that both receivers could not be turned off at the same time, the effect of the conservative design would be achieved, since autonomy could be used to switch receivers in case of a failure.

Throughout the course of integration and test the guidance and control system required several software loads. Unfortunately, the guidance processor did not have a separate test port. Consequently, to avoid disassembly of the flight cabling interfaces the software loads were done via the flight command system at 125 bps. This caused each load to take several hours and created bottlenecks in the processing schedule. Test ports should be required on all units that contain significant amounts of reprogrammable software.

“Test it as you fly it” is the common sense theme of many design reviews. However, because of the lack of integration between the ground system computers testing in flight-like conditions with the safing software enabled was cumbersome. The quality of the test program would have improved if more of the system level testing was done in flight-like conditions. Similarly, the use of mission operations test time would have been improved if more testing was done in a realistic operations environment, i.e., low downlink data rates and significant round trip light time delays.

This spacecraft was JHU/APL’s first use of the 1553 data bus interface for a flight application. Use of the 1553 data bus proved extremely beneficial as it enabled off-the-shelf procurements, off-the-shelf test equipment, and a standard well defined interface that saved weeks of integration and test time. In addition, the bus provided a simple way for the ground based attitude control and determination

simulator to insert sensor stimulus and read actuator commands from the flight system without the use of expensive test ports.

With a streamlined documentation approach the subsystem and component level design reviews are the cornerstone for program communication and design verification. Of exceptional benefit to the program was the participation of experts from JPL and NRL in design reviews where JHU/APL had little experience in the context of a deep space mission. In particular, the outside reviewers greatly improved the robustness of the telecommunication and safing designs. A good example is in the area of data rate selection. A design review suggestion to lower the “nominal” uplink and downlink data rates simplified operations and increased system reliability by enabling the spacecraft to maintain a conservative Sun pointing attitude over the whole mission.

While for hardware the development process is well understood, there existed a non-uniform approach to the software development process. As would be expected, the quality of the software, both in-house and subcontracted,

varied greatly. Of most interest is the ability to predict the rate of flight software problems by the rate of ground test problems. The software that had a “low and flat” error rate during the last few months of ground test has been flawless in flight; the software that had a “moderate and decreasing” error rate prior to launch has shown problems during flight at a rate comparable to the pre-launch.

For low cost programs with complex software there are a limited number of people that will have the knowledge to adequately test the system. Testing 24 hours a day is of little benefit in a case where only one shift of skilled test personnel are available. For each program there will be a solution where cost is optimized by selecting a balance between team size and program duration. The NEAR program erred on the side of having too few test personnel. This created a problem that was most evident during the final month of testing, when the number of shifts per day, not the available number of hours, limited the amount of testing.

Throughout the entire program mass was the main spacecraft driver. Figure 3 shows the dry

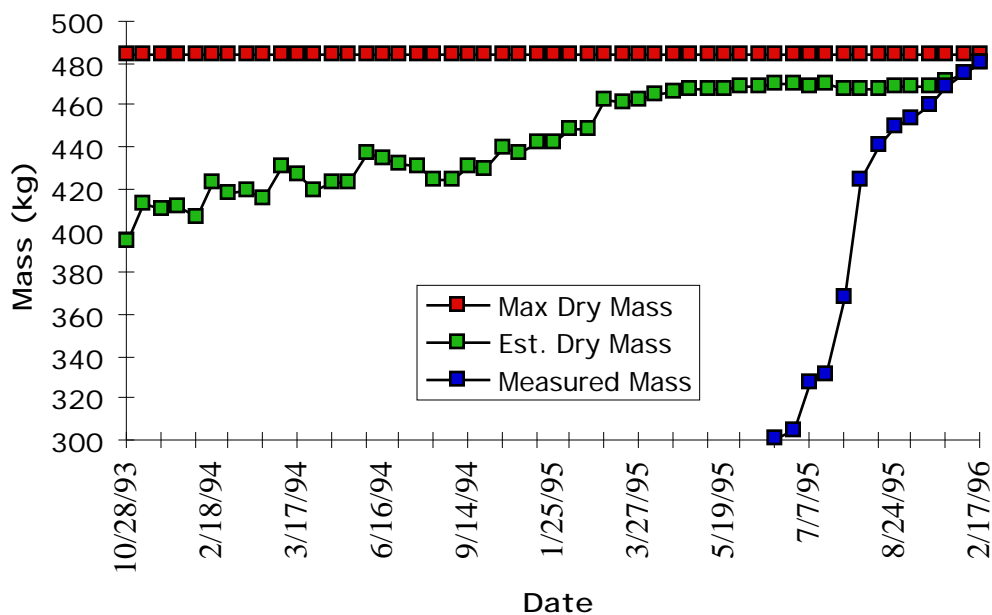


Figure 3. Dry Mass History

mass history. The program started with 17% margin and launched 5 kg extra fuel, (1% margin). It can be seen that the dry mass had a steady upward trend of about 1% per month during the design and fabrication phase (11/93 till 3/95), was relatively flat during the integration and test phase (4/95 till 2/96), and had a small jump near the end of the program. The increase at the end was due to the thermal blankets and the spin balance mass, both of which were carried as estimates until they were installed at the launch site.

Finally, for every processor that has reprogrammable software there was supposed to exist a functionally equivalent breadboard model and appropriate simulators that could be used for both pre-launch and post-launch testing. For the safing software this proved to be difficult as a complete spacecraft simulator was originally seen as not necessary and otherwise cost prohibitive. Consequently, all of the safing software was debugged and acceptance tested using the spacecraft. This approach is not recommended because:

- (1) flight hardware is susceptible to damage during safing software checkout,
- (2) the amount of safing testing is limited by the use of available spacecraft test time,
- (3) checkout of the safing software uses spacecraft test time at the expense of other systems, and
- (4) after the spacecraft launches it is impossible to modify the safing software or test the interactions of safing with spacecraft operations.

In retrospect, the importance of a safing software simulator was underestimated. Work is now underway to assemble a complete spacecraft simulator.

## **VIII. Summary and Recommendations**

Much of the success of the NEAR mission was due to ability of the program to follow the

original design philosophy that limited the number of deployable and movable hardware items. Aggressive monitoring of the various subcontracts resulted in most subsystems being delivered early or on time. Helping the vendors solve problems quickly often meant numerous phone calls and frequent travel on short notice. JHU/APL's willingness to monitor, run or repeat tests and evaluations often filled knowledge gaps or resolved uncertainties.

With several subcontracts, particularly the science instruments, we decided that JHU/APL could have done a more efficient job in-house especially with respect to spaceflight quality of design. We only lacked the necessary confidence at the beginning usually because of the fast paced schedule. In contrast, we cannot think of a case in which a task was performed at JHU/APL that we subsequently decided that we should have executed a subcontract for that task.

The analysis of the Problem/Failure Reports demonstrates the need for

- (1) better communication between lead engineers to minimize interface problems and
- (2) a more disciplined approach to software development and quality control.

In addition, consideration must be given to use flight computer systems that are user friendly, offer readily available development tools and provide platforms for accurate simulation of integrated flight tasks in the laboratory concurrent with the hardware fabrication.