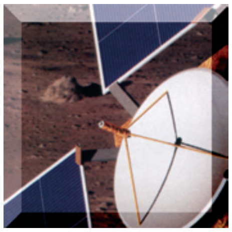

NEAR Spacecraft Flight System Performance

Andrew G. Santo

The Near Earth Asteroid Rendezvous (NEAR) spacecraft was built and launched in 29 months. After a 4-year cruise phase the spacecraft was in orbit about the asteroid Eros for 1 year, which enabled the science payload to return unprecedented scientific data. A summary of spacecraft in-flight performance, including a discussion of the December 1998 aborted orbit insertion burn, is provided. Several minor hardware failures that occurred during the last few years of operations are described. Lessons learned during the cruise phase led to new features being incorporated into several in-flight software uploads. The added innovative features included the capability for the spacecraft to autonomously choose a spacecraft attitude that simultaneously kept the medium-gain antennas pointed at Earth while using solar pressure to control system momentum and a capability to combine a propulsive momentum dump with a trajectory correction maneuver. The spacecraft proved flexible, reliable, and resilient over the 5-year mission.

INTRODUCTION

The NEAR spacecraft was designed by APL as a project under the NASA Discovery Program, a series of low-cost, quick-turnaround space projects. The spacecraft was launched in February 1996 by an unmanned Delta rocket. After a 3-year cruise phase, the project planned for a series of burns to slow the spacecraft's relative velocity with the asteroid to accommodate an orderly orbit insertion. However, the first rendezvous burn was aborted and, instead, the spacecraft flew by the asteroid at about 1 km/s. A recovery burn was quickly executed that greatly reduced the relative spacecraft-to-asteroid velocity, but added 13 months of extra cruise phase. Finally, in February 2000, the spacecraft achieved orbit around the asteroid Eros. One year of orbital operations was followed by a remarkable landing sequence.

SYSTEM DESCRIPTION

Figure 1 shows the spacecraft in the deployed flight configuration. A 1.5-m antenna and four solar panels were mounted on the outside of the forward deck. The solar panels were only deployable on the spacecraft. They were folded down along the sides of the spacecraft for launch and were deployed by a yo-yo de-spin system shortly after separation from the launch vehicle. Most of NEAR's electronics were mounted inside the spacecraft on the top and bottom decks. The science instruments were mounted on the external surface of the spacecraft. The magnetometer was mounted on the high-gain antenna feed, and the rest of the instruments were located on the bottom deck. The interior of the spacecraft contained the propulsion module. About 40% of the spacecraft mass at launch was fuel.

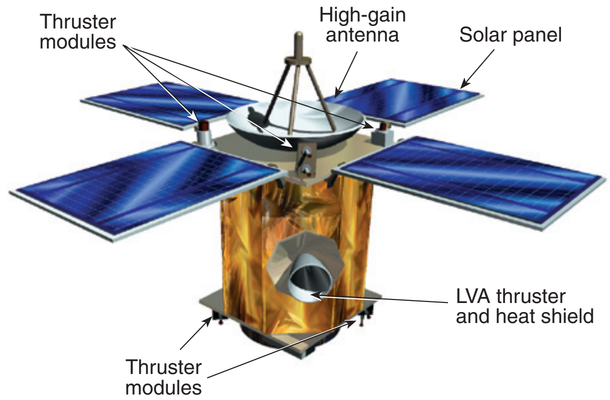

Figure 1. NEAR spacecraft flight configuration (LVA = large velocity adjust).

The system block diagram is shown in Fig. 2. The spacecraft avionics, which are highly redundant, were designed with a distributed architecture where subsystems do not share common hardware. This approach allowed subsystems to be developed in parallel so that the compressed integration schedule could be met. Within the spacecraft were seven processors: one in the command and data handling subsystem, two in the guidance and control subsystem, and four in the payload. With the exception of the processor in the command and data handling system, all the processors had software that could be modified in flight. The flight computer, which contained the main guidance and control software, was programmed in Ada. The attitude interface unit, which was the backup attitude processor, was programmed in C. The rest of the computers used the Forth programming language. The use of the MIL-STD-1553 bus as the main data interface between processors proved valuable as it provided redundancy and fault tolerance, reduced cable harness, and ensured compatibility with many commercially available components such as the star trackers and the inertial measurement unit (IMU). A detailed description of the spacecraft can be found in Refs. 1 and 2.

PRE-LAUNCH STATUS

The NEAR spacecraft was assembled and ground tested between June 1995 and February 1996. A 9-month period for integration is rather short; normal spacecraft integration periods are on the order of 14 months. Nevertheless, no corners were cut to reduce the quality of the test program. Over the 9-month test phase, 227 problem reports were opened by the spacecraft test conductors. All but two of those problems were understood and fixed

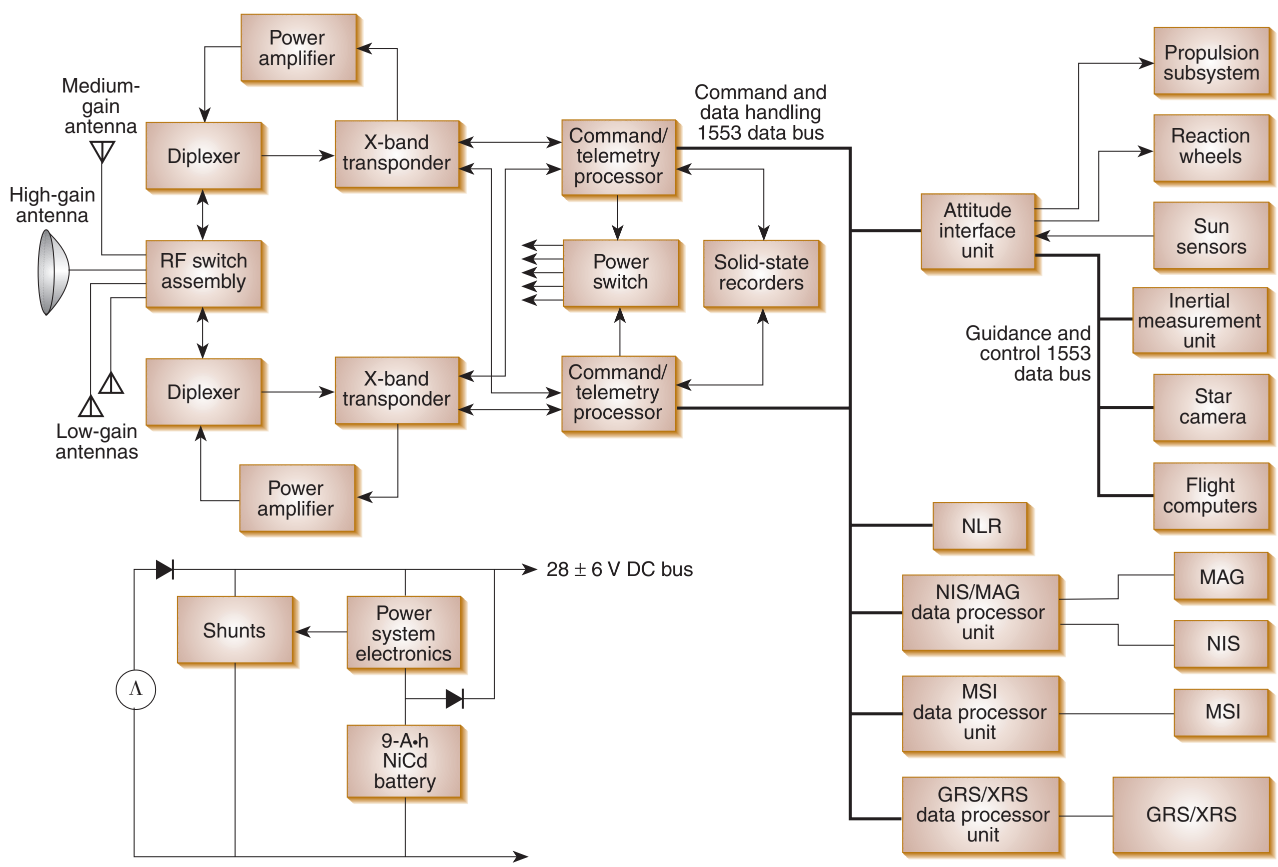

Figure 2. NEAR system block diagram (GRS = Gamma-Ray Spectrometer, MAG = Magnetometer, MSI = Multi-Spectral Imager, NIS = Near-Infrared Spectrograph, NLR = NEAR Laser Rangefinder, XRS = X-Ray Spectrometer).

prior to launch. Both of those open problems regarded guidance and control software that was operating within the flight computer. The first open problem, noted prior to thermal vacuum testing, was when the flight computer code was in boot mode and did not load and execute the flight application code when commanded. This problem happened only once during ground testing and was never experienced in flight. The source of the bug was uncovered some 2 years later, and at that time it was decided not to fix it as it was not a threat to the operational system. The second open problem, noted while the spacecraft was in thermal vacuum testing, was that the guidance and control software failed to execute a pointing command. This command-loss problem was experienced several times in flight. After much analysis, it was explained by a complex timing interaction between different parts of the guidance and control code. This software bug was never fixed. Instead, mission operations personnel devised a work-around, where the few pointing commands that were susceptible to the timing interaction were repeated within the command script enough times so that the probability of losing the command was less than 1 in 1000.

During ground testing, 13 performance waivers were granted; 3 of those waivers are of interest since they affected mission operations. The first waiver was that the solar panel interconnects (the wiring connecting adjacent solar cells) were fabricated with very little strain relief. Consequently, the interconnects on the qualification solar array were shown during testing to break after thermal cycling. Because the flight solar arrays were already built when this problem was discovered on the test panel, the flight arrays were accepted with the defect. As a result, mission operations were limited in how the spacecraft was pointed, so that the most severe thermal cycling of the solar panels was restricted. While in cruise phase and while the spacecraft was within 1.5 astronomical units (AU) of the Sun, spacecraft pointing was restricted so that the solar arrays were always pointed within 30° of the Sun. Outside the 1.5-AU Sun distance and during the orbital phase there were no solar array pointing restrictions. This pointing restriction proved beneficial as no solar panel degradation was observed.

The second waiver was that the performance of the IMU did not meet specifications. The Litton IMU used on NEAR was the first hemispherical resonator gyroscope flown in space. Because it was the first unit and its development slipped, the normal calibration cycle was cut short. As a result, the NEAR IMU delivered for flight was out of specification in many areas. In particular, the accelerometer bias error was measured at 100 μG when the specification limit was 20 μG. This error directly affected the accuracy of the onboard closed-loop trajectory correction maneuvers. Instead of meeting the system-level specification to execute propulsive maneuvers to a magnitude accuracy of 0.2%, only an accuracy of about 1% was achieved. The bottom line was that, even with the increased execution error, the navigation team was still able to predict orbits with sufficient accuracy to meet all of our science goals.

The third waiver of interest was that the Ball-provided star tracker had a thermal-electric cooler (TEC) that did not reliably turn on and cool the sensor when the sensor's temperature was greater than 10°C. Instead, in some test cases the TEC remained off until triggered by the next thermal cycle. The risk of a warm sensor was that the star tracker would not meet pointing specifications after the worst-case NEAR radiation environment (10-krad total dose) at the worst-case maximum operating temperature of 35°C. Because this failure was discovered late in the star tracker qualification program, the flight unit was accepted with the defect. As a work-around, the spacecraft thermal design was modified to prevent the TEC from being needed. Some thermal blanket area on the tracker was replaced with radiator area to keep the star tracker's sensor always colder than 10°C. Redundant spacecraft heaters were added to keep the tracker from being colder than –30°C.

Just prior to launch the spacecraft dry mass was measured at 475 kg, which was 5 kg under the maximum dry mass limit. As a result, 5 kg of extra hydrazine (25 m/s) was added. At launch, the spacecraft held a total of 1467 m/s of onboard ΔV (change in velocity), of which 138 m/s was booked as margin.

LAUNCH AND EARLY OPERATIONS

The spacecraft launched on 17 February 1996, the second day of the launch window. About an hour after launch the spacecraft separated from the launch vehicle while rotating at approximately 60 rpm. Two seconds after launch vehicle separation, the yo-yo de-spin system holding the solar arrays in a stowed configuration was released and the spacecraft slowed down to under 1 rpm. Shortly thereafter, the guidance and control system was powered and given spacecraft control authority. After a few short firings of the 4.4-N hydrazine thrusters, spacecraft momentum was reduced so that the four small reaction wheels could be used to control spacecraft pointing. Within a few minutes after separation, the spacecraft was safely pointing at the Sun and transmitting its health and safety status to the Earth. On the ground, however, things were not going quite as well. The Deep Space Network (DSN) Block V receiver had been set to the wrong signal level and had problems locking up on the NEAR signal. It took almost a half hour of debugging to find the problem in the ground station. It was an anxious mission operations team that found the spacecraft safe and well at the first contact.

During the first day of checkout it was discovered that the star catalog within the guidance and control code of the flight computer had an error for all stars in the southern sky. Within a few days a new star catalog was generated and loaded to the flight computer. A second unrelated problem was discovered with the star tracker. It would, for an unexplained reason, periodically drop one or more of the stars it was tracking. This did not seem to affect star tracker performance since the APL-provided star tracker software within the flight computer would automatically detect the problem and command the tracker to look in the correct position for the lost star. Over the next several months diagnostic data were collected and provided to the star tracker vendor. The vendor offered to provide a software fix but it was refused since the performance was acceptable.

Within 2 weeks after launch the first trajectory correction maneuver (TCM) was executed to correct the launch-injection errors. The navigation team measured burn accuracy by tracking the spacecraft for a week after the burn, and a surprising 10% magnitude error was discovered.

Examination of the wheel torque commands revealed excessively large values. This was traced to larger-than-expected noise from the IMU. A software change was designed to filter the IMU data and included in the next flight processor software load. When implemented, this quieted the wheel torques considerably and pointed jitter was greatly improved.

Diagnostic burn data from TCM-1 were analyzed, and two problems were uncovered. First, the IMU was performing much worse than expected. Software additions to the flight computer would be needed to filter the IMU noise and reduce the accelerometer bias error. More seriously, the thruster control code did not correctly model the force profile of the turn-on and turn-off characteristics of a thruster pulse. The guidance lead decided to rewrite the thruster control section of code. Until the next upload of flight computer code, the team was advised not to use the thrusters. Because the software change and review cycle was expected to take many months, the mission operations team was given the task of maintaining system momentum so that momentum dumps were not needed. To control system momentum, the operations team would control spacecraft pointing so that a desired solar torque was generated by off-pointing the spacecraft slightly away from the Sun. The pointing commands were manually calculated on the ground and uplinked to the spacecraft about once per week. This procedure worked well since no propulsive momentum dumps were ever required throughout the entire cruise phase. Later, this function was added to the flight code so that the spacecraft would autonomously calculate the off-pointing attitude to control system momentum.

During the first few weeks of the mission it was noticed that the receiver frequency of transponder-B drifted some 10 kHz. The frequency was monitored carefully over the rest of the mission, and the drift did not affect operations. Within the first 30 days, instrument checkouts were preformed; the only problem noticed was that two of the X-ray connectors were swapped. Fortunately there was no damage, and the improper harness connections could be fixed with a new software load to the X-ray instrument. After documentation review it was discovered that the cable mix-up happened after a late hardware modification was made to the X-ray instrument when it was noticed that the instrument was sensitive to the mechanical noise from the reaction wheels. The instrument was removed from the spacecraft and modified to filter out the noise in the frequencies of concern, and the mistake was made upon re-integration on the spacecraft.

CRUISE PHASE OPERATIONS

In the fall of 1996 the first software upload, version 1.09, was provided to flight computer-B. The new software upload consisted of 14 code changes. The major changes were the new thruster control code and the addition of a pointing scenario to automatically point the spacecraft with respect to the Sun to control system momentum while maintaining the medium-gain antenna pointed at Earth.

TCM-2 was executed on 13 September 1996, shortly after the new code upload. Burn performance of TCM-2, which was reported by the navigation team, was better than it had been, with a 2% magnitude error, but it was still well out of specification. After analysis of detailed diagnostic burn data it was discovered that reading the 100-Hz IMU buffer at 25 Hz would sometimes cause a measurement to be missed as the IMU clock slowly drifted with respect to the flight computer clock. This dropped measurement would be treated as a value of 0 by the software and would cause an error in the acceleration calculation. The navigation team expressed concern about the safety of conducting the orbital mission with such large execution errors. Therefore, a new software upload to improve maneuver execution accuracy was undertaken.

Version 1.10 was uploaded to flight computer-A in December 1996. It consisted of two major software changes. The first change fixed the problem of the dropped IMU parameter. Instead of treating a missing IMU sample as 0, the software repeated the last valid reading. This change did improve burn performance. After some experience the average burn execution error was reduced to around 1%, which is outside of the performance specification but good enough for navigation. The second change was to add a new pointing definition for the asteroid Mathilde flyby. The new pointing scenario allowed a special roll orientation pointing definition needed for the unique geometry of the flyby. The special pointing scenario proved useful, and the main-belt asteroid Mathilde flyby in June 1997 was a great success.[3]

Shortly after the flyby, the July 1997 deep space maneuver was executed to target the spacecraft for the Earth flyby. This burn was the first use of the 450-N bipropellant engine. The burn was a complete success. A transient lateral acceleration was noted on the start-up of the large thruster. This transient pulse, however, was under the safing limit and was ignored by the flight code. Without a rational explanation for the cause of the acceleration spike, the data were thought to be measurement errors, and no further action was taken. During the insertion burn the same transient spike appeared again, but that time it caused the burn to abort.

There were three other items to note in 1997. The first was that while at a distance of about 2.0 AU from the Sun, the Adcole-provided Sun sensor once reported an invalid, no-Sun indication to the flight computer. This no-Sun error condition was flagged by the flight software, and it sent the spacecraft into safe mode. Upon entry into safe mode, the spacecraft changed its attitude slightly and the Sun was immediately reacquired. Analysis by Adcole showed that at low intensity the Sun sensor could be fooled to report a no-Sun condition if multiple heads reported the same intensity. This would not be a problem over the rest of the mission, as the Sun-distance maximum would always be less than 1.75 AU. However, this error caused a re-examination of the fault protection code in the attitude interface unit (AIU). The AIU performs the function of checking the flight computer software and acts as a backup guidance and control computer. It was found that a consistent no-Sun condition from the Sun sensor while everything else was operating would cause unnecessary switching of flight computers. This bug would be fixed in the next software upload.

The second item to note happened in February 1997 as the spacecraft passed behind the Sun for a solar conjunction. The spacecraft successfully operated without ground commands for a 10-day period while the Sun-probe-Earth angle was under 2° and communication was unreliable. During this period, special DSN equipment was trained on NEAR and important X-band communication data were collected.[4]

The spacecraft executed a very successful Earth flyby in January 1998. Prior to the Eros encounter we updated the guidance and control software in the flight computers and the AIU to fix the most troublesome bugs and add extra operational capability. In June 1998 the software was updated in both AIUs.

For this update two major changes were made. The first change was to fix the safing response under a no-Sun condition if everything else looked fine. The second change was to rewrite the thruster control logic as was done in the previous flight computer loads.

In September 1998, software version 1.11 was uploaded to flight computer-A. There were two major changes to this version of software. First, a new pointing scenario was added for simplifying imager scans. This special scan was requested by the imager team and proved to be very useful throughout orbital operations. Second, a capability was added to combine a momentum dump with a TCM. This capability was vital for orbital operations since the off-Sun pointing technique of passive momentum dumping used during the cruise phase could not be used and weekly propulsive momentum dumps would be needed. This "combined-burn" capability effectively cut the required number of burns in half since, throughout the orbital phase, a TCM was planned about once per week and a momentum dump was needed about once per week. Because execution of the combined burn proved to be just as accurate as a traditional TCM, combined burns were routinely used throughout the orbital phase.

ABORTED RENDEZVOUS BURN AND EXTENDED CRUISE PHASE

On 20 December 1998, NEAR was to begin the rendezvous sequence with the first of a four-burn sequence that was designed to slow the spacecraft down to enable a slow flyby of the asteroid followed by a small, final mono-propellant rendezvous burn. However, almost immediately after the start of the bi-propellant burn the onboard software terminated the burn and demoted the spacecraft into safe mode. The reason for the burn abort was that the lateral acceleration safety limit was violated. The transient pulse that is now known to occur each time the bi-propellant engine is started triggered this limit violation. It was merely luck that the first time the bi-propellant engine was fired, in 1997, when the spacecraft was 10% heavier, this violation did not cause NEAR to go into safe mode.

After demotion into the safe mode the spacecraft executed a pre-loaded command sequence to reset its configuration. However, a missing command to re-enable use of the reaction wheels in the command sequence script unfortunately put the spacecraft into a unstable configuration. This unstable configuration resulted in an initial loss of attitude control. Onboard fault protection software correctly identified the configuration problem and took the designed, programmed action of switching flight computers.

Switching flight computers should have fixed the problem; however, for some unknown reason, attitude control was not regained. For the next 8 h the spacecraft continued to have attitude control problems and the safety software continued to periodically swap flight computers each time it flagged an "excessive thruster use" violation. At some unknown time during the event the back-up attitude processor, the AIU, took over and regained control. During the 8 h when the spacecraft was out of control, 29 kg (96 m/s) of hydrazine was used. The reduced solar array output during the period of uncontrolled attitude ultimately led to a low-voltage shutdown in which the downlink transmitter was powered off for 24 h and the solid state recorder was powered off to save energy. As soon as the recorder was powered off, all of its data were lost.

Communication with NEAR was re-established 27 h after the anomaly. A contingency plan was quickly prepared, and the spacecraft successfully captured flyby images of Eros on 23 December 1998. Mission operations executed a make-up rendezvous burn on 3 January 1999, which put the spacecraft on a trajectory to catch up with Eros on 14 February 2000, 13 months later than originally planned and with little fuel margin.

A NEAR anomaly review board was formed to determine the reason for the burn abort anomaly and to make recommendations for NEAR and for future programs. The board's investigation included a painstaking reconstruction of the post-abort timeline from the small amount of data recovered. Many simulations were run on a spacecraft real-time simulator that accurately represented the actual flight hardware configuration. These simulations all showed that the fault protection actions should have ended the attitude anomaly quickly. No satisfactory explanation of the abort events was ever obtained by the review board.[5] Obviously the hardware simulator was deficient in some respect since the cause of the anomaly could never be found.

Aside from the propellant loss, two hardware problems were initiated by the burn abort. First, there was noticeable contamination of optics of the Multi-Spectral Imager. During the orbital phase, frequent in-flight calibrations and ground processing were used to partially offset the contamination effect. Second, the efficiency of the 4.4-N thrusters used during the fuel loss was noticeably degraded throughout the orbital phase. Fortunately, the amount of fuel used by these thrusters over the rest of the mission was small and the effect on the fuel budget was not significant. The reduced thruster efficiency was believed to be caused by a number of thruster cold starts that occurred during periods when the spacecraft was tumbling and out of control.

Throughout the rest of the extended 13-month cruise phase four minor problems were observed. First, on 23 February 1999, the flight computer re-set and caused the spacecraft to enter into safe mode. This re-set was never explained. A likely explanation was that the re-set was caused by a single-event upset of the processor RAM. Second, the high-pressure latch valve telltale failed to give a reliable indication. Fortunately, the flight software did not use this reading and the valve was never planned to be exercised again, so this failure had no effect on mission operations. Third, transponder-B was noticed to drop about 10% of all commands. In-flight experiments were done to characterize this software failure. It was observed that the stronger the uplink signal strength and the shorter the command reception time, the more likely the command was to be properly received. It was agreed that this backup receiver could successfully support mission operations if needed by connecting it to the high-gain antenna and changing the ground command creation software to form small command sequences. Fortunately, the prime receiver worked during the entire mission and these contingency plans never had to be executed. Finally, on 7 February 2000, just weeks before the second rendezvous attempt while power-cycling the IMU to turn on the accelerometers in preparation for TCM-22, a set of transient IMU gyro readings triggered an autonomous momentum dump and sent the spacecraft into a safe mode. Recovery from the autonomous dump was quick and the rendezvous burn was executed a week later as planned. Why the IMU output invalid rates on start-up this one time remained unknown. To prevent this fault from happening again it was decided to keep the accelerometers powered for the remainder of the mission.

ORBITAL OPERATIONS

The spacecraft performed almost flawlessly during orbital operations. The new software pointing scenarios and the ability to combine a TCM with a propulsive momentum dump proved extremely valuable for mission operations.[6,7] The long cruise phase provided mission operations with plenty of opportunities to fine-tune their scripts and learn how to work around known software bugs. Over the entire orbit phase only one new problem resurfaced: on 13 May 2000, the Near-Infrared Spectrograph (NIS) failed to return useful telemetry. After much analysis and a few onboard experiments failed to fix the problem, the NIS was turned off for the remainder of the mission. The cause of the failure was never fully understood but was believed to involve the power distribution circuitry.

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