Delta 269
(Delta III)

Investigation Report

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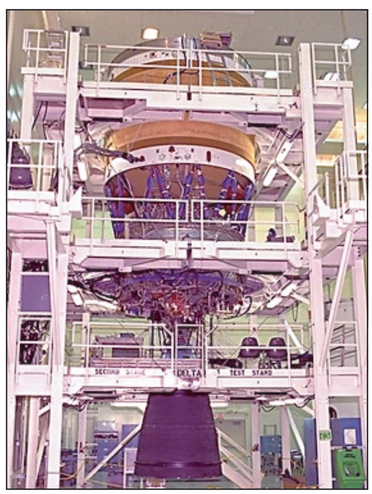


DELTA 269 (DELTA III) INVESTIGATION REPORT

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PREFACE

This report was prepared by The Boeing Company and is a record of the investigation into the 4 May 1999 failure of the Delta 269 vehicle carrying the Orion-3 communications satellite.

The Delta III launch vehicle lifted off from Space Launch Complex 17 (SLC-17) at Cape Canaveral Air Force Station (CCAFS) at 21:00 EDT on 4 May 1999. Without significant events in the launch preparations and countdown, it flew as expected through the boost phase, second-stage first burn, and coast period. The RL10B-2 engine abruptly shut down 3.4 sec into a 162-sec scheduled restart burn. The shutdown was accompanied by tumbling of the stage. Despite the engine anomaly, vehicle sequencing continued through the remainder of the mission, and the stage regained attitude control. Spacecraft separation occurred as scheduled, placing the Orion-3 satellite in a 745-nmi by 82.7-nmi orbit, short of the targeted 14,015-nmi by 100-nmi orbital requirement.

An investigation team headed by Dr. Russ Reck, director of Advanced Engineering at Boeing Expendable Launch Systems in Huntington Beach, California, was established immediately. The investigation team was composed of representatives from Boeing, Pratt & Whitney, Lockheed Martin, the National Aeronautics and Space Administration (NASA), the Aerospace Corporation, Space Systems/Loral, and Hughes Space and Communications, and observers from the Federal Aviation Administration (FAA) and the United States Air Force (USAF). The team was chartered to investigate and define the root cause of the failure and recommend corrective actions.

The failure investigation was concluded in September 1999, and an interim summary report was issued with findings, conclusions, and recommended actions (Ref. Boeing report MDC 99H0081 "Delta 269 Failure Investigation Interim Summary Report," included as Appendix K). Recommendations included actions for the Return-to-Flight program to pursue potential contributing factors to the failure. This report substantiates the results of the interim report. It also documents the results of the report's recommended actions as well as findings and further recommended actions from the Return-to-Flight activities.

The investigation determined that the Delta 269 second-stage engine stopped because of a combustion chamber breach at the structural jacket Strip 91. The investigation found the most probable cause of the combustion chamber breach was defective brazing as a result of poor manufacturing process controls. Contributing to the poor process controls was a braze inspection method (X-ray) that could not properly detect all voids or debonds. Also, the inspection results were not recognized as nonconforming because of improper translation of braze coverage design requirements to the acceptance criteria used by quality assurance procedures.

The combustion chamber manufacturing and process control deficiencies have been corrected, and Pratt & Whitney has been producing excellent chambers for the Delta, Atlas/Centaur, and Titan/Centaur programs since October 1999. Ultrasonic inspection has been added as a complement to X-ray to ensure detection of all voids and debonds. The design requirements for braze coverage have been translated properly to the quality assurance documents, and Pratt & Whitney added engineering reviews of all X-ray and ultrasonic test results.

The Return-to-Flight program conducted investigations into three other areas as possible causes of the chamber breach: vehicle-induced loads, engine start transient thermal damage to braze joints, and engine shutdown vent pressure damage. The investigations found that none of these by themselves could have possibly caused the failure; however, engine shutdown venting and nominal vehicle-induced loads, in the presence of defective braze, may have contributed to the progressive failure of the braze strip. Additional design modifications have been incorporated to reduce any potential adverse loads from these sources and to increase design margins.

ACKNOWLEDGEMENTS

The Boeing Company thanks the many individuals and organizations who contributed to this investigation. Boeing employees at Huntington Beach and Cape Canaveral Air Force Station (CCAFS) and Pratt & Whitney employees worked long hours researching possible causes of the failure, while performing analyses and tests to substantiate or refute various failure hypotheses. Boeing also acknowledges the outstanding efforts of the Aerospace Corporation and Boeing Rocketdyne Division for their support during the investigation and their contributions to the resolution of the cause of failure.

The investigation teams were composed of members appropriately qualified and experienced to define the root cause of the problem and provide wisdom and guidance for the design solutions. Boeing acknowledges and thanks the senior board members and personnel who supported the Delta 269 anomaly investigation and Return-to-Flight efforts. The key team members are listed in Appendix J. Within this list, three people deserve recognition for exceptional efforts and contributions: Dr. Russell Reck for his able leadership of the failure investigation, and Chuck Ordahl and Mort Markowitz for their contributions to both the investigation and Return-to-Flight efforts. Their dedication, high technical standards, and attention to detail ensured all issues were rigorously and completely addressed.

Boeing also sincerely thanks the many civilian and government men and women who comprise the Delta team, and without whose continuing contributions the success of the Delta programs could not be achieved.

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Section 1
EXECUTIVE SUMMARY

1.1 DELTA 269 FLIGHT

The Delta III launch vehicle lifted off from Space Launch Complex 17 (SLC-17) at Cape Canaveral Air Force Station (CCAFS) at 21:00 EDT on 4 May 1999. No significant incidents in the launch preparations and countdown occurred, and the vehicle flew as expected through the boost phase, second-stage first burn, and coast period. The RL10B-2 engine abruptly shut down 3.4 sec into a 162-sec scheduled restart burn. The shutdown was followed by tumbling of the stage. Despite the engine anomaly, the vehicle sequencing continued through the remainder of the mission and the stage regained attitude control. Spacecraft separation occurred as scheduled placing the Orion-3 satellite in a 745 nmi by 82.7 nmi orbit, which was short of the targeted 14,015 nmi by 100 nmi orbital requirement.

1.2 VEHICLE CONFIGURATION

The Delta III launch vehicle is an evolved Delta II capable of delivering 3.8 mT to geosynchronous transfer orbit. Several major design changes were implemented on the Delta III including high-performance strap-on solid boosters with thrust vector control, a 4-m cryogenic upper stage, and 4-m payload accommodations. Delta 269 was the second launch of the Delta III but was the first flight operation of the Delta III second stage. The first flight of the Delta III configuration (Delta 259) was terminated early in booster operation due to a roll control system instability, which prematurely depleted the hydraulic thrust vector control system oil supply.

The Delta III second stage is a high-performance liquid oxygen (LO_2)/liquid hydrogen (LH_2) system featuring newly designed propellant tanks, feedlines, pressurization, and thermal conditioning systems. The RL10B-2 engine descends from the same family of engines that have demonstrated exceptional reliability on the Atlas-Centaur and Titan-Centaur upper stages. Figure 1-1 shows the RL10B-2 engine, which incorporates a number of modifications aimed at achieving higher performance capabilities than in preceeding RL10 configurations. Key modifications include a slightly longer combustion chamber, an extendable carbon-carbon nozzle exit cone, and a dual exit manifold. The RL10B-2 operates at a slightly higher chamber pressure and mixture ratio than previous RL10 designs and utilizes a boost phase full flow hydrogen and pulsed oxygen prechill to minimize the amount of propellant consumed during pre-ignition thermal conditioning.

The RL10B-2 engine modifications and operational methods were qualified in a test program involving 188 tests on several engines for a total of 17,288 sec of accumulated test time. An upper stage evaluation program consisting of 13 tests and 860 sec of accumulated test time was also performed to demonstrate acceptable performance of the RL10B-2 engine and the Delta III systems and interfaces.

1.3 INVESTIGATION PROCESS

The investigation team adopted a method of fault isolation based on the systematic evaluation of a cause and effect diagram, called a fault tree. The fault tree method relates various causes to one or more failure modes (termed nodes) that could have resulted in the observed total system failure mode. Each node was then subjected to engineering analyses and testing where appropriate for traceability to the system failure. Each node was classified as either “not credible,” “unlikely,” or “credible” depending on evidence or deductive relationships to the primary anomalous events. Six hundred thirty fault tree nodes were addressed and dispositioned in the investigation. Probable cause was determined from the summary of findings and sequence of events (or scenario) that was consistent with the flight data and other test and analysis results.

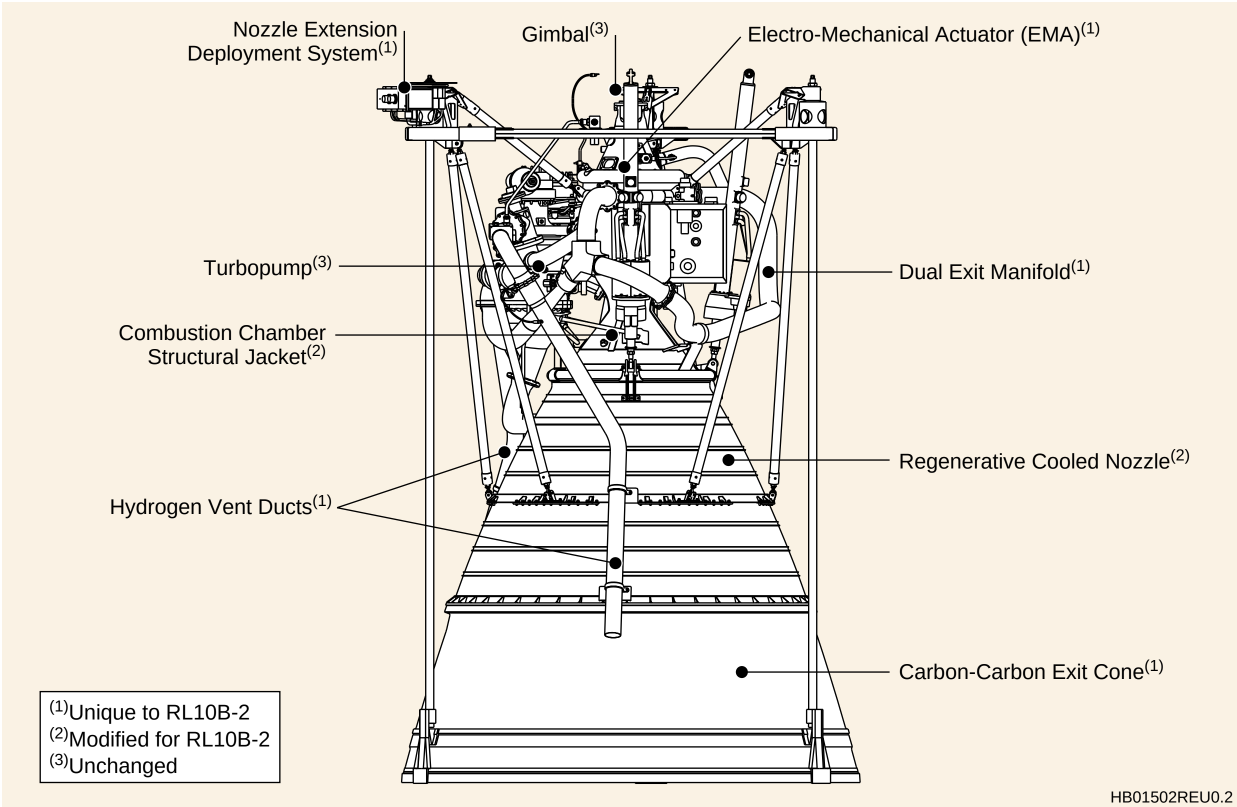

Figure 1-1. RL10B-2 Engine

Because the main anomalous findings were common to the second-stage system element, the investigation team rapidly narrowed the field of inquiry to the second-stage engine with an emphasis on the combustion chamber. Other major system elements were examined, but the absence of anomalous events pertaining to the first stage, payload adapter, and payload fairing quickly eliminated their direct involvement.

1.4 PRIMARY ANOMALOUS FLIGHT EVENTS/DATA

The key anomalous flight data used to isolate the failure cause is described as follows.

1.4.1 Premature Engine Shutdown

Engine parameters were nominal during the first burn of the second stage and during thermal conditioning preceding the restart burn. At restart ignition, the engine responded normally reaching 94% of peak chamber pressure before abruptly shutting down 3.4 sec after the ignition command.

1.4.2 Vehicle Shocks

Two unexpected shocks were recorded by second-stage accelerometers located on the RL10B-2 engine gimbal block and on the redundant inertial flight control assembly (RIFCA). The shock magnitudes exceeded 25 g's (saturation) at the gimbal block accelerometer. The first shock occurred at T+286 sec, 4.2 sec after the second-stage first-burn ignition and 2.3 sec after the engine achieved full power. The second shock event occurred at T+1317.196 sec, 3.4 sec after the second burn ignition while the engine was accelerating and coinciding with the premature end of the second burn. The second shock was at least an order of magnitude larger than the first shock.

1.4.3 Electromechanical Actuator (EMA) Loads

The pitch and yaw EMAs registered increased compression beginning 6-8 ms prior to the failure event and then momentarily exceeded full scale readings of ~5,000 lb within 2-3 ms after the shock.

1.4.4 Vehicle Disturbance Torques

The vehicle diverged in pitch and yaw after the second burn shock event, resulting in a tumbling motion that persisted throughout the scheduled second burn interval. Pitch and roll disturbance torques, which were present on the vehicle from the time of the failure event, gradually subsided at 94 sec into the restart burn period and disappeared at commanded shutdown.

1.4.5 Equipment Shelf Temperatures

Transient heating was recorded at six equipment shelf locations for a few seconds after the time of the failure event shock. During the remainder of the second burn interval, generalized cooling was recorded at the same equipment shelf locations.

1.4.6 Failed Engine Instrumentation

In addition to a sudden reduction in chamber pressure occurring at the time of the failure, several key engine transducers also failed within milliseconds after the restart burn shock event, including fuel venturi upstream pressure, fuel turbine inlet temperature, oxidizer pump inlet temperature, and oxidizer manifold pressure.

1.5 IDENTIFICATION OF FAILED COMPONENT

Through analysis of flight data in conjunction with special tests, the investigation was able to establish conclusively that a combustion chamber breach occurred at T+1317.196 sec coincident with the failure shock event.

Flight data from engine area thermocouples showed a rapid 4°F temperature rise in 3 sec immediately following the failure shock event. It was concluded that this indicated the presence of rapid convective heating. System modeling showed excellent correlation of this temperature rise using the release of the 5500°F combustion chamber gas following a pressure time history of the Delta 269 chamber pressure decay. The area of maximum heating was consistent with a chamber breach at one of the four strips that connect the four combustion chamber support segments (Strip 91). Subsequent to the noted temperature rise, generalized cooling was found to be consistent with cooling from propellant flow of the unlit engine.

The size and location of the chamber breach was determined from the pitch and yaw rates measured by the RIFCA. The disturbing vehicle torques indicated the majority of the impulse was applied in 40-80 ms, which is approximately the same timescale as the chamber pressure decay recorded on telemetry. The pressure decay measurements further indicated that the size of the breach was approximately 58 $in.^2$ (three times the throat area). The vehicle torque analysis showed that the rapid chamber venting occurred from the region of jacket Strip 91.

A venting and plume impingement analysis showed that a pitch impulse of -4990 ft-lb-sec, a yaw impulse of 3510 ft-lb-sec, and a resulting force vector at 35.1 deg would be obtained for a diamond-shaped, 67-$in.^2$ (3.4 times throat area) chamber breach centered on the chamber structural jacket Strip 91. This closely matched the results obtained from the RIFCA. The impingement analysis further established that the total force acting on the turbopump was about 25,000 lb, more than sufficient to create the collateral damage indicated by various telemetry measurements after the shock event.

1.6 COMBUSTION CHAMBER INVESTIGATION

Having determined that the combustion chamber failed, the investigation turned to a full evaluation of the design and manufacturing and the specific pedigree of the unit flown on Delta 269.

As shown in Figure 1-2, the combustion chamber is constructed from a series of 180 cooling tubes that are formed on a mandrel in the shape of the convergent/divergent nozzle and then brazed to a segmented structural jacket. A splice strip is installed every 90 deg over the four seams in the jacket. The entire assembly is furnace brazed. The chamber is X-ray inspected and

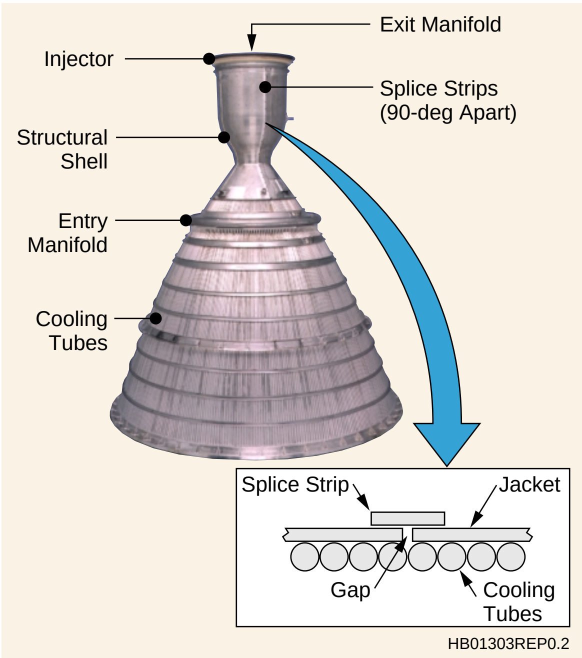

Figure 1-2. Combustion Chamber Construction

leak tested to ensure adequate tube-to-tube, tube-to-jacket, and splice strip-to-jacket braze coverage. The jacket carries more than 90% of the combustion chamber hoop load, so the splice strip braze structural integrity is of critical importance.

The combustion chamber S/N BXA804 flown on Delta 269 Engine P692803 was assembled and furnace brazed in January 1998. It was initially found to be leak free. During the course of its acceptance firing history, two leaks were discovered at Strip 91 as shown in Figure 1-3. The first was found after hot firing No. 9. This leak was repaired by hand brazing. After hot firing No. 12, another leak was discovered a short distance from the earlier leak. Because no additional ground firings were planned, this leak was repaired with RTV for moisture protection per standard procedure and the engine was then installed on the Delta 269 vehicle. Finding two leaks in the same vicinity on the same jacket strip after different hot fire tests is unique within the RL10 history.

A review of the radiographic inspection records of the combustion chamber BXA804 revealed jacket strip braze defects far in excess of the design intent. In some local areas, braze coverage was found to be as low as 20% per linear inch—a factor of four below design intent. The discrepancy has been attributed to an error in the translation of the braze coverage requirement. The engineering requirement was for a braze coverage requirement of 80% per linear inch. The drawing specified a 80% minimum coverage requirement, which was interpreted by the product inspectors as 80% coverage averaged over the entire length of the strip. The less stringent criterion was applied to chamber BXA804. In addition, the X-ray inspection techniques were shown to be inadequate. Ultrasonic evaluations of braze samples have detected the presence of braze voids not reported in X-ray records of the high-pressure, cylindrical section of the chamber. Given the X-ray inspection limitations, the actual condition of the BXA804 chamber strip braze is unknown; however, based on the ultrasonic findings, it was likely worse than recorded in the X-rays. In addition to the defective strip-to-jacket braze described above, X-rays also revealed that chamber BXA804 had poor tube-to-jacket braze in the throat area where the average braze coverage was 38% vs. a requirement of 60%.

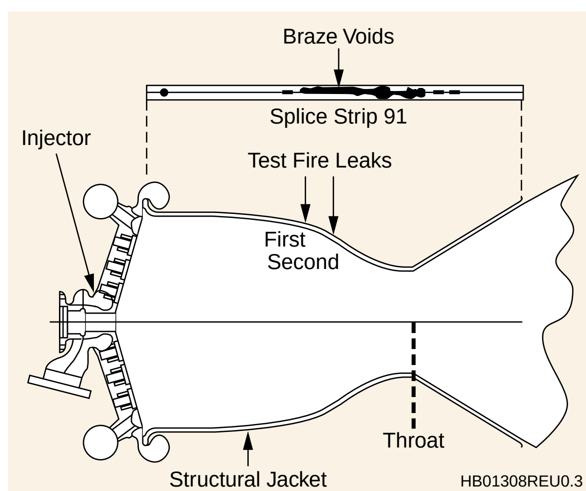

Figure 1-3. Combustion Chamber Test Leaks/Braze Voids

Examination of the brazing process revealed a number of factors that contributed to the defective braze. These included the cleaning solvent change made subsequent to engine qualification, fit up of the parts, tooling, and the amount of braze material. No single cause was identified, but these factors together caused large variations in braze quality.

An analysis of the splice strip strength capability was performed to evaluate ultimate strip margins under flight loads. The analysis results show marginal, nonrobust braze strength capability for all conditions where 20% braze coverage was present. The BXA804 chamber is predicted to have had marginal capability due to the global low braze coverage.

To help anchor the chamber structural analytical model and understand its reaction under load, a chamber with similar deficient braze was subjected to cyclical hydropressure tests. It should be noted that since BXA804 was only X-rayed, it may have had less braze coverage than the chamber used in these tests. The primary purpose of these tests was to evaluate the load redistribution around defective poorly brazed regions and the effects of pressurization cycles. The initial pressurization showed substantial plastic deformation on the first cycle with essentially no change in elastic deformation or additional damage on the subsequent 101 cycles. The chamber was further exposed to 20 cycles at 780 psia before being carried to burst at 816 psia. There was additional progressive joint overstress, deformation, and deterioration during the cycles to 780 psia.

The results indicated a substantial load redistribution around defective poorly brazed regions. Although the tests were limited because of the difference between hydropressure and the thermodynamics involved in full engine firings, they did indicate that a chamber with braze conditions far below minimum allowable has the capability to withstand cyclic hydropressures to 680 psi, which is above the nominal 640 psia operating pressure. The hydropressure test results could not be extrapolated further to the flight chamber because the true braze condition was unknown.

1.7 ADDITIONAL FLIGHT LOADING CONDITIONS

The defective combustion chamber braze that flew on Delta 269 was discovered fairly early in the investigation. However, it was not evident why the braze failed on its second in-flight start after having survived 12 acceptance ground tests and the entire first burn of the second stage. Additional weight was added to this question when tests of defective poorly brazed chambers showed adequate performance. Most of the investigation was spent evaluating numerous possible means by which normal in-flight loads to the chamber may have further incrementally damaged an already weak, defective chamber.

Figure 1-4 describes the failure scenario along with identification of potential vehicle and flight contributing factors that may have added stress to the engine and caused it to fail when it did. The investigation extensively pursued each of these possible contributing factor scenarios. This work was also intended to establish conclusively whether or not each of these conditions could be a possible failure cause. These scenarios included boost phase dynamic loads, propellant feedline loads, and EMA loads as well as the engine start thermal conditions and the possibility for engine shutdown pressure pulse and related hydrogen vent duct damage.

1.7.1 Vehicle Flight Loads Scenarios

In the case of boost phase dynamic loads, the investigation found that vehicle system-induced torsion loads had not been properly defined in the engine specification and that testing and analysis had not accounted for these loads. As a result, substantial additional testing and analysis were performed as a part of the investigation to determine the significance of this factor. The results showed that the engine fundamental torsional mode frequency is about 3.75 Hz, which is well outside the overall vehicle high gain torsional frequency range of 8-12 Hz. Coupled loads analyses were performed, and the engine specification was revised to include the correct torsion requirement. The previously defined axial and lateral environments were found to be acceptable.

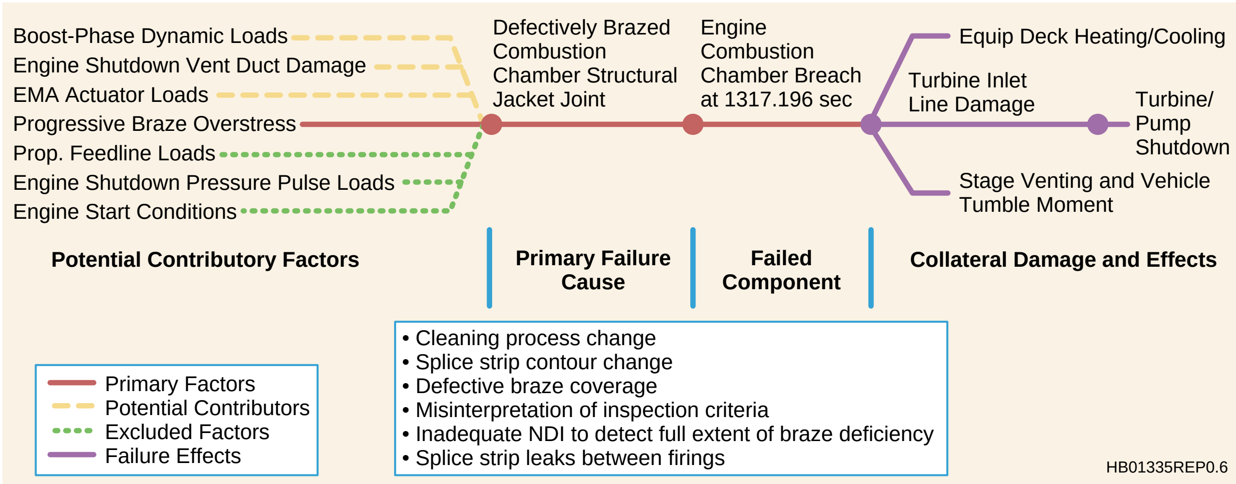

Figure 1-4. Delta 269 Failure

Qualification level torsion testing was then performed on an engine with a nozzle extension and LO_2 chilldown duct. Very low responses and strain levels were recorded. The updated correlated model showed an insignificant chamber torsional shear stress of 272 psi, which indicated that this load did not cause the failure. It was further determined that there are very high margins of safety in the chamber throat region for the LO_2 chilldown duct torsional induced loading and that the duct-to-chamber interface was not a failure cause.

Tests were performed to evaluate and requalify the propellant feedline loads to the engine shutoff valves and turbopump. These tests verified that no detrimental loading is caused by the feedlines.

Engine slew rate induced loads were evaluated through tests on an engine with a full nozzle extension and feedlines. The results showed low engine throat and feedline strains and indicated that the EMA loads at near the maximum slew rate are 1500 lb. The maximum Delta 269 measured EMA load in flight was 1560 lb. These values are less than half the qualified EMA load level of 3650 lb and show that gimbaling was not a failure cause. However, the review of this area found that the engine centering slew rate that produced the 1560-lb EMA load shortly after engine shutdown is unnecessarily high. This slew rate has been reduced to lower the EMA loads for this event.

It was discovered that the EMA monoball bearing and race were made of the same A286 stainless steel material and were not lubricated. In one case, this caused galling and seizure of an EMA in ground test. As a result, simulations with various combinations of seized monoballs were conducted. The conclusions were that these did not match flight data and that engine loads remained in an acceptable range even if a monoball was frozen. This condition could not have been a failure cause; however, a design change to the monoball has been made to prevent further recurrence.

In each of the above cases, the investigation determined that vehicle flight loads could not have been a failure cause although they all contributed some additional loading to an already weak, defective chamber and likely assisted its final failure. None of these avenues yielded conclusive evidence on how that additional damage occurred; however, such work was hampered because it was not possible to establish how poor the flight chamber braze actually was initially.

1.7.2 Engine Start Conditions Scenario

Concern existed that the efficiency of the oxidizer system boost phase pulse cooling may have caused the injector to be colder on the first start than on previous ground tests and that the exit cone heat soakback caused a hotter regenerative nozzle during the second start than previously tested.

Also, data from the flight indicated an unusually long time to first-burn ignition on the 13th spark and a long time to reach full acceleration during the second-burn start. These conditions were extensively investigated in a series of highly instrumented cold flow and engine firing tests. The objective of demonstrating thermal margins in excess of flight conditions was achieved with no detrimental effects on the engine. Nineteen full-power tests were completed for a duration of 1398 sec. It was established that the start conditions experienced on Delta 269 did not contribute to the failure. The engine used for this testing had extensive poor braze of the combustion chamber jacket splice strips. The braze of these strips did not significantly change over the total test period.

These tests helped demonstrate that the boost phase LO_2 pulse chilldown sequence was more than required for actual flight conditions. As a result, a reduced pulse sequence has been qualified and implemented.

1.7.3 First-Burn Shutdown Pressure Pulse Damage Scenarios

The RL10B-2 hydrogen vent system that routes prestart hydrogen cooling flow and vents hydrogen pressure at engine shutdown is substantially different than that on previous RL10 models. Two vent ducts are routed from the cooldown valves along the nozzle to exit a few inches below the nozzle regenerative section (see Figure 1-1). Both vent ducts are attached to the nozzle by clamps at the nozzle circumferential frames. In addition, several other engine component vent outlets are routed into these ducts. The complete vent system was not tested during engine firing because of facility limitations. This led to two concerns: (1) could the back pressure pulse at shutdown damage other components via the interconnected vent outlets, and (2) was the structural integrity of the vent duct system adequate for the shutdown loads?

Evaluation of each of the interconnected engine components showed that they were not adversely affected by the shutdown backpressure at their vent outlets and that this did not contribute to the failure. However, it was decided that this vent design exposed the components to an undesirable backpressure and, therefore, a change was implemented to revise this design to be similar to that used on earlier RL10 engines.

The vent duct structural integrity concern related to the possibility of a failure that causes impact of the ducts on the chamber cooling tubes at first-burn shutdown. The damaged tubes would then affect the chamber cooling at second start, which could cause a chamber failure at that point. The most highly loaded vent duct is nearly inline with structural jacket Strip 91, the strip that ultimately ruptured. An additional concern is the possibility for damage to other connecting plumbing and components of the engine.

To evaluate the ducts structural integrity, tests were conducted in a rig assembly that included an accumulator to provide the pressure source to a flight configuration of the cooldown valves, vent ducts, and their attachment to a flight-like engine chamber. Nine cycle tests were performed at analyzed maximum operating pressure (MOP), with one test performed at 125% MOP. These tests showed substantial duct movement, localized buckling, and permanent deformation at the highest pressure, but no catastrophic-type failure occurred. Subsequently, additional analysis found that the pressure levels could be higher than that used in the tests because of faster operating valves and larger thermal effects. The investigation also found that during the original design phase, the full range of pressure and thermal operating conditions were not completely accommodated in the design with adequate margin. Further, it was discovered in a photograph that on the prior Delta III mission (Delta 259), the vent duct upper bracket was installed on the wrong side of the circumferential frame. Personnel involved with the installation believe that it was installed in the same manner on Delta 269, although no photographic record exists. This installation issue by itself could not have caused the failure.

The investigation found no evidence that the ducts were a failure cause. However, it was established that the design could be improved to increase its robustness and increase design margins. Design modifications and requalification testing were therefore implemented.

1.8 ANOMALOUS SHOCK DATA EVALUATION

The magnitude and frequency content of the T+286 sec and T+1317 sec anomalous shock events were evaluated. An extrapolation of the RIFCA accelerometer response was used for the 1317-sec shock, because the engine gimbal block accelerometer was fully saturated at that time. Analysis of all vehicle accelerometers indicated that both shocks emanated from the engine. The failure event shock (T+1317) was the much larger of the two. Its magnitude and spectra, as recorded at the gimbal block, was similar to the shock developed at the Delta III SureSep™ stage separation system location indicating it likely represented a similar high-energy metal fracturing event. The T+286 shock reached about 25 g's at the gimbal block but its magnitude was significantly less than the failure event shock by a factor of 10 or more. It contained unique higher level responses at 1100 and 1400 Hz. Survey tests showed that these modes were excited by tapping the engine in the vicinity of the aft edge of the combustion chamber jacket. An extensive investigation of potential reasons for the T+286 shock was conducted. The most probable cause was determined to be braze cracking in the structural jacket initiated by the thermal transient in the chamber shortly after the first-burn ignition. The results from this work were not conclusive but are consistent with the progressive braze failure scenario.

1.9 MOST PROBABLE FAILURE SCENARIO

The investigation determined that the most probable failure cause was the defective brazed chamber. It may have simply deteriorated and failed on the second in-flight start; however, it is more likely that it failed in conjunction with other nominal flight-loading conditions. The specific means by which the inflight braze deterioration took place could not be determined, but at the conclusion of the investigation, only one failure scenario credibly fits all the flight data and investigation results.

This most probable failure scenario, shown in Figure 1-5, begins with the initial deficient brazing at combustion chamber jacket Strip 91. The first anomalous performance of this joint was noted as a leak occurring during the leak test performed after engine acceptance hot run test No. 9. The leak location was braze repaired. Later, after hot run test No. 12, another leak on the same

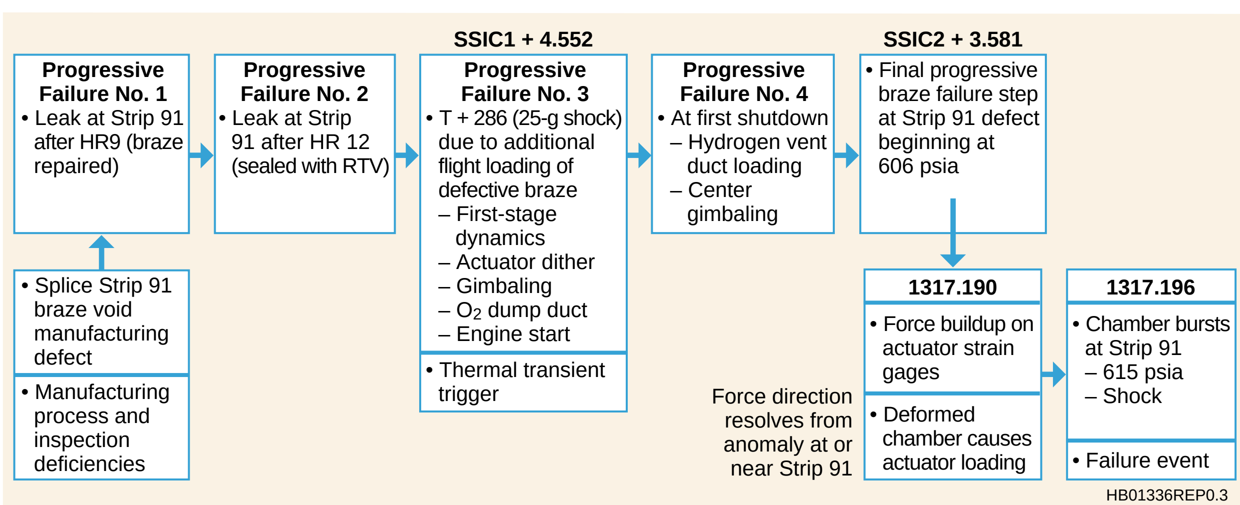

Figure 1-5. Most Probable Failure Scenario

strip was discovered. This was repaired with RTV. During the initial phases of flight, the braze joint may have suffered some additional deterioration due to a combination of nominal flight-loading conditions, which resulted in a partial braze fracture at 4.6 sec after the first-burn ignition when triggered by the thermal transient associated with engine start. The T+286 shock on vehicle accelerometers represents this event. At first-burn shutdown, the braze may have sustained further damage due to loadings from the hydrogen vent duct thrust or the rapid engine centering. Because of the progressive braze failure effects, the chamber no longer had the capability to hold the full pressure at the second start and failed during a period of rapid thermal and pressure changes. The initial indication of the failure onset was a force buildup in the pitch and yaw actuators 6-8 ms prior to the actual chamber rupture. This precursor anomaly was caused by a slight deformation in the chamber as the strip braze began its final failure, slightly redirecting a small portion of the thrust force. The final rupture caused all the remaining flight data responses.

1.10 PRATT & WHITNEY REVIEWS

As a means to gain additional assurance for the Delta III Return-to-Flight program, Pratt & Whitney senior management conducted several additional reviews of technical adequacy.

An independent team reviewed the changes made to convert the Centaur RL10A-4-1 design to the RL10B-2 and the methods used to qualify the RL10B-2 engine for use on the Delta III. The review identified and evaluated all untested flight conditions and the qualification to each of the requirements identified in the engine specification. In addition, an engine mission requirements qualification matrix was independently developed and compared to the actual tests and analyses performed. Differences were evaluated and, where necessary, additional tests or analyses were completed. The failure modes and effects analyses were updated for all RL10B-2 engine components. An improved pedigree review process was developed. An independent assessment of the failure scenarios with special emphasis on multiple interacting conditions was conducted, and a new formalized risk assessment process was introduced. All immediate actions identified in these reviews were addressed and satisfactorily completed in the Return-to-Flight program.

1.11 SUMMARY

There is clear evidence that the combustion chamber was originally built defectively with a very weak structural jacket splice braze in the area that finally failed. The poorly brazed chamber was the only defect found that was directly traceable to the failure. It is possible that the defective weak joint simply failed on the second in-flight start. However, it is more probable that the failure occurred at that point due to the contribution of additional nominal flight-loading conditions. The investigation pursued extensive testing and analyses of all known flight conditions. This work was comprehensive and conclusive in demonstrating that a properly brazed chamber will not be overstressed by flight loads or engine flight operating conditions. In the course of this activity, a number of areas were uncovered where design changes would improve system robustness, and these changes have been made. The testing and analysis adds greatly to the qualification baseline and the substantiation of design margins.

The most probable failure scenario describes the only credible means to match the known evidence, investigation results, and flight data. Redesigns and requalifications have been accomplished to assure that this possibility will not recur.

Section 2
FINDINGS/CONCLUSIONS

The investigation determined that the second stage engine stopped because of a combustion chamber breach at the structural jacket Strip 91. The most probable cause of the combustion chamber breach is defective brazing as a result of poor manufacturing process controls. Contributing to the poor process was a braze inspection method (X-ray) that could not detect all potential voids or debonds. Additionally, the defects were not recognized as nonconforming because of improper translation of braze coverage design requirements to the screening criteria used by Quality personnel. Extensive investigations were conducted into three other areas as possible causes of the failure: vehicle induced loads, engine start transient damage, and engine shutdown pressure vent damage. These investigations found none of these to be the possible cause of the failure; however, nominal vehicle-induced loads and engine shutdown venting may have contributed to the progressive failure of the braze strip.

Other specific findings are as follows:

- The combustion chamber design margin of safety was not fully established by analysis and test for all loading conditions.
- The hydrogen discharge vent duct design should be improved to provide adequate margin for the loads imposed at engine shutdown.
- The hydrogen vent system subjects the turbopump seals to a high pressure transient at engine shutdown.
- The vehicle-induced torsional dynamic loads applied to the engine interface during vehicle boost phase were not specified in the engine specification, and the engine had not been tested for these conditions.
- The engine ground test shutdown loads are quite high, leading to the need for improved monitoring and load limits.
- The nonoperating engine return-to-null slew rate caused the highest flight actuator gimbal loading. Although this load was well within qualification levels, it is unnecessarily high.
- The boost phase oxidizer system chilldown is substantially more than required.
- The early turnoff of the engine helium purge provides more opportunity for adverse backflow of the main fuel shutoff valve.
- Under certain thermal conditions, the fuel pump inlet pressure may reach 33 psig, which is above the qualified value.
- A helium supply line is located too close to the electromechanical actuator (EMA).
- The chamber structural jacket leak test port is uncapped from the time the stage leaves DMCO until launch, offering opportunity for moisture intrusion.
- The EMA monoball bearing and race material can cause galling and binding.
- The redundant inertial flight control assembly (RIFCA) propellant depletion algorithm design is not sufficiently robust in the presence of the 23 Hz rocking mode oscillations.

Section 3
RECOMMENDATIONS AND CORRECTIVE ACTIONS

Figures 3-1 through 3-4 list the recommendations and corrective actions that resulted from the failure investigation, discoveries during Return-to-Flight testing, and vehicle preparations for launch. Implementation status is also provided.

Corrective Actions	Completion Date
1. Develop an improved braze process and NDE to verify design requirements are satisfied	October 1999
2. Modify combustion chamber manufacturing process to ensure units meet design requirements	October 1999
3. Implement combined ultrasonic and radiographic braze inspection techniques and acceptance criteria	October 1999
4. Perform post-firing ultrasonic inspection of chamber structural jacket brazed joints	December 1999
5. Implement combustion chamber manufacturing statistical process controls	December 1999

002280.5

Figure 3-1. All Corrective Actions Completed to Fix Cause of Delta 269 Failure and Prevent Recurrence

Recommended Actions	Completion Date
1. Perform gimbal loads test with EMAs and feedline pressure loads present. Instrument feedlines to measure stresses, deflections, and clearance to adjacent structure. Measure EMA response to feedline loads	October 1999
2. Perform engine modal survey to verify engine modes and effects of boost phase torsion loading. Update product spec to include torsion load requirements	January 2000
3. Demonstrate by test adequate engine structural margins for boost phase torsion loads	January 2000
4. Measure transient and steady-state chamber structural jacket loads under engine operating conditions (ground test firings)	March 2000
5. Measure torsional stiffness of the LO_2 dump duct assembly with its first stage attachments	September 1999
6. Ensure LO_2 dump duct has correct torsional compliance with engine chamber to prevent amplification of boost phase loads	November 1999
7. Perform testing to validate chamber structural model and to demonstrate acceptable chamber capability	March 2000
8. Perform testing to revalidate proper coolant distribution around combustion chamber coolant tubes	March 2000
9. Perform testing to measure the effect of pulse chilldown on engine start conditions. Demonstrate margin to worst case conditions	March 2000
10. Perform testing to measure the effect on regen coolant of nozzle extension heat soakback. Demonstrate margins to worst case conditions	March 2000
11. Develop thermodynamic simulations of start condition to predict max coolant temperatures and tube temp margins. Validate by test	March 2000
12. Review all engine design changes since qual to ensure qual baseline has been maintained	April 2000
13. Review all RL10B-2 process changes implemented since 1996 to ensure process certification	April 2000
14. Conduct in-depth review of RL10-B2 qualification testing to certify engine is qualified to Delta III operating conditions and environments. Include comparative analysis to Centaur operating conditions	April 2000

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Figure 3-2. All Tests and Analyses Completed to Investigate Potential Contributory Causes to the Failure

Recommendations and Corrective Actions	Status
1. Re-route helium tube to solenoid control valves to remove potential for EMA interference	Complete
2. Review the acceptability of the engine post-shutdown return-to-null slew rate	■ Complete ■ Slew rate reduced to 1 deg/sec from 10 deg/sec
3. Redesign vent plumbing to prevent turbopump and thrust control valve back pressure	Complete
4. Modify stage purge system to be compatible with engine vent modifications	Complete
5. Redesign LH_2 vent ducts to reduce loads at shutdown	Complete
6. Improve procedure that aligns engine to stage centerline to compensate for cryoshrinkage of combustion chamber (0.3 deg)	Complete
7. Implement S/W modification to improve design robustness of the RIFCA propellant depletion algorithm	Complete
8. Revalidate engine operation at 33 psig fuel pump inlet due to worst case thermal analysis of ullage temperature	Complete
9. Implement combustion chamber leak tests at DMCO	Complete
10. Implement improved load measuring struts and load limits to ensure acceptability of test stand shutdown loads	ECD Oct 2000
11. Perform EMA-to-vehicle clearance checks at maximum slew angles to verify acceptable as-built clearances	Complete
12. Implement structural jacket leak test port moisture protection equivalent to other RL10 configurations	Complete

002282.3

Figure 3-3. Recommended Design and Operational Changes Not Related to Delta 269 Failure, but to Improve Robustness and Increase Design Margins of the Stage and Engine Systems

Recommendations and Corrective Actions	Status
1. Modify EMA rod end monoball materials to eliminate galling and improve lubrication	Complete
2. Improve EMA-controller CCA component mounting and conformal coating process to reduce susceptibility to thermal induced component failures. Also for TVC E-pack	Complete

002283.1

Figure 3-4. Recommended Design Changes to Correct Design Errors Uncovered During Return-to-Flight Process

Appendix
SENIOR BOARD MEMBERS AND KEY PERSONNEL

The following list includes the failure investigation senior board members.

Dr. Reck extends his sincere appreciation to the senior board members and personnel who supported the Delta 269 anomaly investigation.

- R. J. (Rick) Arvesen, Boeing director, Engineering, Delta II/III/Titan programs.
- Kristie Bateman, Second Lieutenant, United States Air Force (USAF), Expendable Launch Vehicle program, Special Systems Project Office.
- Paul Beyersdorf, Boeing senior manager, Cape Canaveral Air Force Station (CCAFS), Delta II/III programs.
- Daniel J. Collins, Boeing director, Product Integration, Delta IV program.
- Bill Ezell, consultant, former Rocketdyne vice president, Engineering.
- Michael P. Hallet, manager-West Coast Office, NASA Kennedy Space Center.
- Larry Joy, Boeing senior manager of Liquid Propulsion, Delta III/IV programs.
- Tom Kmiec, RL-10 chief engineer, Pratt & Whitney.
- Donald Kutyna, vice president Space Technology, Loral Space and Communications.
- Tom Legerton, Boeing senior manager, Upper Stage integrated product team (IPT), Delta III/IV programs.
- Don Magill, former McDonnell Douglas Corporation vice president, Space Products.
- Mort Markowitz, Boeing senior specialist, Structural Dynamics and Loads, Delta II/III programs.
- Patrick Martin, aerospace engineer, FAA Commercial Space Transportation.
- Tom McGunigal, Orion-3 program manager, Loral Orion.
- Royce Mitchell, consultant and former Boeing director.
- Jack Murphy, chairman Independent Review Team, Hughes Space and Communications Company.
- Chuck Ordahl, former Boeing vice president, Advanced Space Systems.
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- Alben Sakaguchi, Boeing senior manager, Flight Systems, Delta II/III programs.
- John Silverstein, former General Dynamics program director, Titan Centaur program.
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- Ted Smith, Launch Vehicle consultant, Hughes Launch Services.
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- Darryl Van Dorn, Boeing director, Commercial/NASA programs.
- William Wang, senior engineering specialist, The Aerospace Corporation.
- Dick Weiss, consultant.
- Mark Wilkins, chief engineer, Delta IV.
- Walt Wilson, former Delta II Air Force program manager.
- Eric Zimmerman, Boeing director, Upper Stage IPT, Delta III/IV Programs.

The following list includes the Return-to-Flight program senior board members.

Dave Crosse expresses sincere thanks to the Boeing and Pratt & Whitney senior board and key personnel who supported the Return-to-Flight program.

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- Walt Wilson, former Delta II Air Force program manager.
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The following people are Pratt & Whitney senior board and key team members.

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- Troy King, Pratt & Whitney, manager, Structural Analysis, and lead, Structures Working Group.
- Tom Kmiec, Pratt & Whitney RL-10 chief engineer.
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