

NATIONAL TRANSPORTATION SAFETY BOARD
OFFICE OF AVIATION SAFETY
WASHINGTON, D.C. 20594

April 21, 2020

POWERPLANTS GROUP CHAIRMAN'S FACTUAL REPORT

NTSB ID No.: ENG17IA026

A: INCIDENT

Location: G344 airway northeast of CUTE intersection
Date: June 7, 2017
Time: 0950 universal coordinated time
Aircraft: Boeing 747-451, N668US, Delta Air Lines flight 276

B: POWERPLANTS GROUP

Investigator-in-Charge: Gordon J. Hookey
National Transportation Safety Board
Washington, D.C.

Member: David Gerlach
Federal Aviation Administration
Washington, D.C.

Member: Jo-Ann Theriault
Federal Aviation Administration
Washington, D.C.

Member: Joshua Migdal
Delta Air Lines
Atlanta, Georgia

Member: David Shinsky
Delta Air Lines
Atlanta, Georgia

Member:	Van J. Winters Boeing Seattle, Washington
Member:	Douglas Zabawa Pratt & Whitney East Hartford, Connecticut
Member:	Stephen Yee Pratt & Whitney East Hartford, Connecticut
Member:	Jeffrey Strausbaugh Pratt & Whitney East Hartford, Connecticut
Member:	Jeffrey Wait Air Line Pilots Association Herndon, Virginia

C: SUMMARY

On June 7, 2017, about 0950 universal coordinated time, Delta Air Lines flight 276, a Boeing 747-451 airplane, N668US, experienced a loss of power from the No. 1 engine, a Pratt & Whitney (P&W) PW4056, while in cruise flight over the Pacific Ocean northeast of Japan. The pilots reported that the airplane was level at FL 320¹ on the G344 airway between the CUTEE and CARTO waypoints when there was a sudden thump and the airplane yawed to the left. The pilots further reported that following the thump; the No. 1 engine's EPR² [engine pressure ratio] and N1³ were decreasing, the EGT [exhaust gas temperature] was increasing, and the oil quantity indicated about 9 quarts.⁴ The airplane was stabilized with the rudder and as the engine's EGT limit was reached, the engine's thrust lever was retarded to idle. The Engine Failure Checklist was accomplished and the VNAV drift down procedure was initiated while turning 45° off track to the right. The pilots declared an emergency and the airplane descended to FL 280. After consulting with Delta Air Lines' dispatch and maintenance at Atlanta, Georgia, it was determined that the airplane would divert back to Tokyo Narita International Airport (NRT), Tokyo, Japan. The pilots reported that about 30 minutes prior to landing, they jettisoned fuel to reduce the airplane's weight to below the maximum landing weight. The airplane landed at NRT without further incident. The airplane was operating on an instrument flight rules flight plan under the provisions of 14 *Code of Federal Regulations* Part 121 as a regularly scheduled international flight from NRT to Detroit Metropolitan Wayne County International Airport,

¹ Flight level (FL) 320 is 32,000 feet mean sea level based on an altimeter setting of 29.92 inches of mercury.

² EPR [engine pressure ratio] is a ratio of the total pressure of the gases in the exhaust pipe divided by the total pressure of the air entering the inlet duct.

³ N1 is the fan and low pressure rotor speed.

⁴ The PW4056 engine's oil capacity is 40 quarts.

Detroit Michigan. There were no injuries to the 4 pilots, 13 flight attendants and 309 passengers on board.

The examination of the No. 1 engine after the airplane landed at NRT revealed metal in the tailpipe. In addition, the low-pressure turbine (LPT) case had 360° separation directly forward of the case's rear flange. A subsequent borescope examination of the engine revealed damage to the 4th stage of the low-pressure compressor (LPC), which is the last stage of the LPC, and throughout the full length of the high-pressure compressor (HPC).

The engine was removed from the airplane and returned to Delta TechOps facility, Atlanta, Georgia for disassembly and examination. The engine was disassembled at the Delta TechOps facility in the presence of members of the Powerplants Group on July 10 – 14, 2017. The HPC was further examined by members of the Powerplants Group on July 17 – 19, 2018. The examination of the engine revealed extensive damage to the HPC, high-pressure turbine (HPT), and the LPT. There were multiple HPC blades between the 5th and 9th stages that had fatigue progressions with many blades having multiple fatigue progressions. Although a primary blade could not be identified, P&W advised that the damage was consistent with previous events that had been attributed to a 5th stage compressor blade separation. Both the HPT and LPT sustained extensive thermal damage. The separation of the LPT case appeared to be the result of turbine airfoil debris that became trapped by and dragged around by the 6th stage turbine blade tips and subsequently cut through the 6th stage blade outer airseal segments (BOAS) cutting through the LPT case.

D: DETAILS OF INVESTIGATION

1.0 Engine information

1.1 Engine description

The P&W PW4056 engine is a dual-spool, axial-flow high bypass turbofan engine that features a 1-stage fan, 4-stage LPC, 11-stage HPC, annular combustor, 2-stage HPT that drives the HPC, and a 4-stage LPT that drives the fan and LPC. The engine is 153.6-inches long, 97.5-inches in diameter, and weighs 9,420 pounds. According to the Federal Aviation Administration's Type Certificate Data Sheet No. E24NE, the PW4056 engine has a take off thrust rating of 56,750 pounds flat-rated to 92°F (33°C).⁵ When installed on the Boeing 747, the PW4056 engine has a maximum continuous thrust rating of 47,970 pounds flat-rated to 86°F (30°C).⁶ (Figure 1)

⁵ Flat-rated to a specific temperature indicates that the engine will be capable of attaining the rated thrust level up to the specified inlet temperature.

⁶ The PW4056 engine when installed on a Boeing 767 airplane has a maximum continuous thrust rating of 49,530 pounds, flat-rated to 77°F (25°C).

PW4000 94-INCH FAN ENGINE

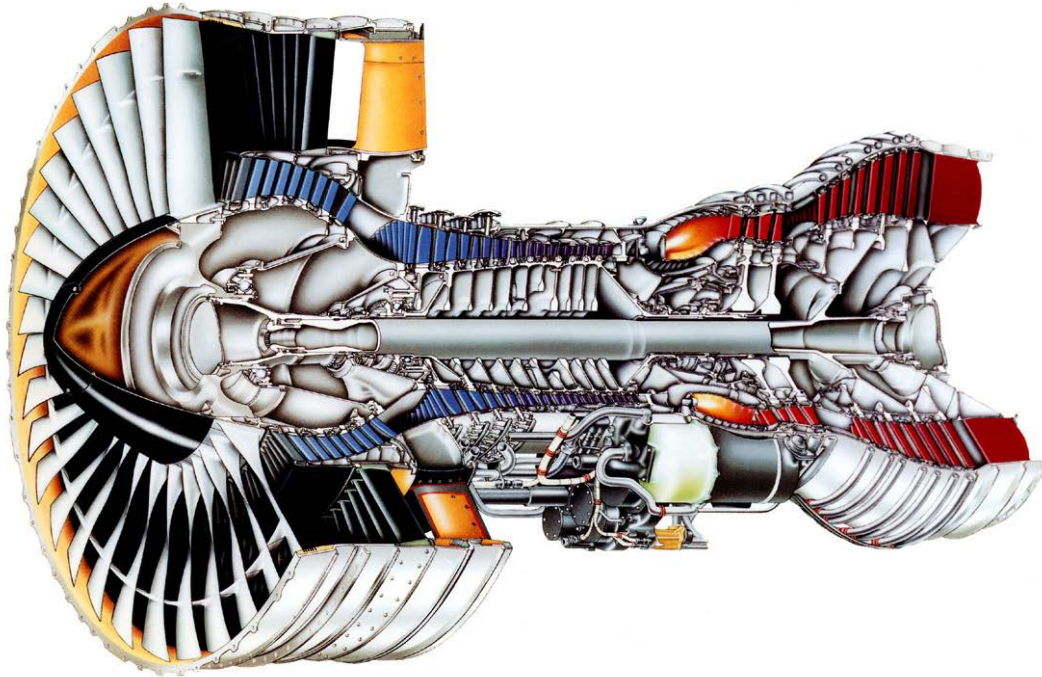


Figure 1: Cross section of a PW4056 engine.

1.2 Engine history

The No. 1 engine on the airplane was a PW4056, serial number (SN) P717513. According to Delta Air Lines' maintenance records, the engine had accumulated 82,303 hours and 10,645 cycles since new and 5,446 hours and 600 cycles since the last overhaul. The engine's last overhaul was accomplished in May 2014 at Delta's TechOps facility. The engine was overhauled following a 3rd stage turbine vane fracture.

Delta's records show that the engine had been installed on the airplane on November 30, 2016. Delta's records also show that the engine had been previously installed on and removed from two other airplanes that had been retired from service.

2.0 5th, 6th, 7th, 8th and 9th stage compressor blades

2.1 Visual examination

The 5th, 6th, 7th, and 8th stage compressor blades were all in place in their respective blade slots and all were fractured transversely across the airfoil adjacent to the blade root platform.

All of the 5th stage compressor blades were fractured through the airfoil between the blade root platform and about 0.85-inches above the blade root platform. Several of the 5th stage compressor blades were displaced aft of their normal installed position. There was evidence of the orange RTV [room temperature vulcanizing] rubber on the blade locks that corresponded to the orange RTV noted in the 5th stage compressor disk's blades slots. The front and rear faces of the blade root platforms as well as the front and rear faces of the blade roots had circumferential rubbing damage. There was one 5th stage compressor blade, part number (PN) 58H305, that had a flat, planar fracture surface with an elliptical-shaped area radiating from the convex side of the root max thickness area of the blade.⁷ (Photo No. 1)



Photo No. 1: Close up view of 5th stage compressor blade fracture surface showing an elliptical shaped area radiating from the convex side of the airfoil. (P&W)

All of the 6th and 7th stage compressor blades were fractured through the airfoil between the blade root platform and about 0.28-inches above the blade root platform. Over half of the 6th and 7th stage compressor blades had fracture surfaces with elliptical-shaped features on the convex side of the airfoil in the mid-chord area. (Photos Nos. 2, 3, and 4)

⁷ The 5th stage compressor blade with the elliptical-shaped, flat planar fracture surface was lost while being shipped from Delta's TechOps facility to P&W's lab facility in East Hartford, Connecticut.



Photo No. 2: 6th, 7th, and 8th stage compressor blades fractured transversely across the airfoil adjacent to the blade root platform. (P&W)



Photo No. 3: Close up view of a 6th stage compressor blade showing two elliptical-shaped areas on the convex side of the fracture surface. (P&W)



Photo No. 4: Close up view of a 7th stage compressor blade showing elliptical-shaped area on the convex side of the fracture surface. (P&W)

All of the 8th stage compressor blades were fractured through the airfoil between the blade root platform and about 0.2 inches above the blade root platform. There were a number of 8th stage compressor blades that had elliptical-shaped areas on the fracture surfaces on both the convex and concave surfaces in the mid-chord area. (Photo No. 5)

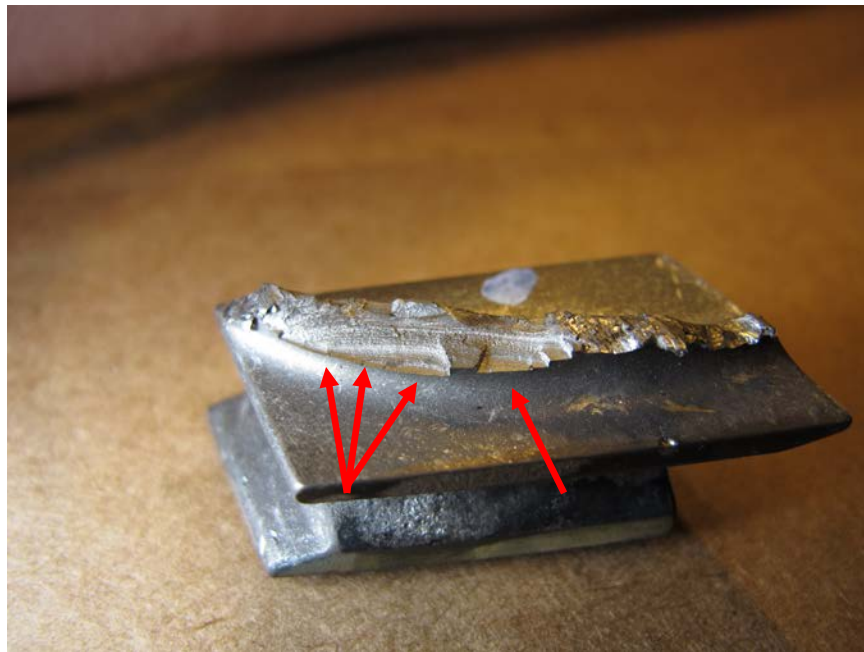


Photo No. 5: Close up view of an 8th stage compressor blade showing several elliptical-shaped areas on the convex side of the fracture surface. (P&W)

All of the 9th stage compressor blades were fractured through the airfoil between the blade root platform and about 1.5-inches above the blade root platform. (Photo No. 6) There was one 9th stage compressor blade that had elliptical-shaped areas on the fracture surfaces on both on the convex and concave surfaces in the mid-chord area. The longer blades were bent opposite the direction of rotation. (Photo No. 7)



Photo No. 6: View of HPC stages 9 through 15 showing damaged, fractured, and bent airfoils. (P&W)



Photo No. 7: Close up view of a 9th stage compressor blade showing elliptical-shaped area on the convex side of the fracture surface. (P&W)

2.2 Metallurgical examination

All of the 5th through 9th stage compressor blades were examined at P&W's Materials Laboratory, East Hartford, Connecticut.

The examination of the 5th stage compressor blades revealed fracture surfaces that were either consistent with overstress or were too damaged and smeared to determine the cause of the separation. The P&W Materials Laboratory reported that the damage to the 5th stage compressor blades as well as the other HPC blades was consistent with previous 5th stage compressor blade primary fractures. (Refer to Section 2.3)

The examination of the 6th and 7th stage compressor blades revealed that all of the blades were fractured transversely across the airfoil adjacent to the blade root platform. The examination of the blades' fracture surfaces showed mostly overstress features or were smeared due to secondary damage. The examination of the 6th and 7th stage blades did reveal a number of blades that had shallow thumbnail-shaped fatigue progressions from multiple origins on the convex side of the airfoil in the mid-chord area. The lab report stated that none of these fatigue progressions were primary to the engine event.

The examination of the 8th stage compressor blades showed that they were also all fractured transversely across the airfoil adjacent to the blade root platform. The lab report stated that the fracture surfaces were heavily smeared from secondary damage. However, there were features on the fracture surfaces that indicated that the fatigue progression was from multiple origins along the convex side of the airfoil. There were several blades that had fatigue progressions from both the convex and concave surfaces that according to the lab report was indicative of reverse bending. The lab report stated that some blades had fatigue cracks that had progressed through the thickness of the airfoil. The lab report also stated that none of these fatigue progressions were primary to the engine event.

The examination of the 9th stage compressor blades showed that they were all fractured across the airfoils at various lengths above the blade root platform. The fracture surfaces on all of the blades except one were indicative of overstress and/or were damaged due to smearing and secondary damage. There was one 9th stage compressor blade that was fractured between about 0.010- and 0.020-inches above the blade root platform that had fatigue progressions from multiple origins along the convex and concave sides of the airfoil in the mid chord area that was indicative of reverse bending. The lab report stated that the fatigue was not primary to the engine event. The lab report stated that 38 of the 9th stage compressor blades had multiple cracks in the mid chord area predominately on the convex side of the airfoil adjacent to the blade root platform. Those cracks were broken open and found to be small, shallow fatigue cracks. According to P&W, similar fatigue cracking had been noted on other PW4000 engines following repeated surge (compressor stall) events. P&W also noted that the fatigue cracking could have been the result of the unusual aerodynamic and vibratory stresses that occurred during the run down of the HPC rotor after the event occurred or due to the multiple impacts of upstream debris cascading through the engine.

2.3 5th stage compressor blade history

The 5th stage compressor blades installed in the incident engine were PN 58H305. PN 58H305 5th stage compressor blades are utilized only in the PW4000 94-inch diameter fan engines.⁸ The PN 58H305 compressor blades were introduced into the PW4000 94-inch engine via Service Bulletin (SB) PW4ENG 72-705 that was issued on February 29, 2000. The PN 58H305 compressor blade featured a wider copper nickel coating on the blade root bearing surfaces to improve blade durability at the pressure surface areas.

The PN 58H305 compressor blade was superseded by PN 50S805 5th stage compressor blade that was introduced via SB PW4ENG 72-815 on November 23, 2011. The PN 50S805 compressor blade incorporated an improved airfoil finish to improve engine efficiency and reduce the thrust specific fuel consumption.

The PN 58H305 compressor blade superseded the PN 55H705 5th stage compressor blade. The PN 55H705 compressor blade was introduced via SB PW4ENG 72-557 on March 28, 1995. The PN 55H705 compressor blade incorporated airfoil changes that were intended to improve the blade's durability.

According to P&W, there have been a number of PW4000 94-inch engine 5th stage compressor blade fractures that have been reported. The PN 55H705 compressor blade has had 22 reported airfoil fractures and 33 reported tip fractures. The PN 58H305 compressor blade has had 16 reported airfoil fractures and 2 reported tip fractures. There have been no reported fractures with the PN 50S805 compressor blade.

3.0 5th stage compressor variable stator vanes

3.1 Visual examination

All of the 5th stage variable stator vanes (VSV) were in place except for two, one in the upper case and one in the lower case, that were fractured transversely across the airfoil adjacent to the outer button. The outer buttons for the two fractured VSVs were not flush with the HPC front case flow path. The fractured VSV that was in the upper compressor case had a flat planar fracture surface along the concave side of the airfoil. (Photo No. 8) The fracture surface of the VSV in the lower compressor case half was coarse and irregular. (Photo No. 9) The 5th stage VSV airfoils had nicks and dents on the leading and trailing edges. Although the VSV rigging check showed that the VSVs were in rig, the vanes were at various angles when the front compressor case was opened.

⁸ The PW4000 94-inch diameter fan engines include the PW4052, PW4056, PW4060, PW4062, PW4062A, PW4152, PW4156A, PW4158, PW4460, and PW4462.



Photo No. 8: View of the fractured outer end of the 5th stage VSV in the upper compressor case half. (P&W)



Photo No. 9: View of the fractured outer end of the 5th stage VSV in the lower compressor case half. (P&W)

3.2 Metallurgical examination

Both of the fractured 5th stage compressor VSVs were examined at P&W's Materials Laboratory. Both of the VSVs were fractured transversely across the airfoil directly adjacent to

the outer button. The report stated the examination of the two VSVs showed that the fracture surfaces were heavily damaged. The examination of the undamaged portions of the fracture surfaces on each VSV did not show any evidence of fatigue.

4.0 *Remainder of engine*

4.1 Exterior

The engine was complete from the fan case front flange and the spinner cap to the exhaust nozzle aft edge. The tail plug was still in place on the rear of the turbine exhaust case inner duct. The engine did not have any indications of an uncontainment, case rupture, or an under cowl fire. (Photos Nos. 10 and 11) There was no notable distress to the front and rear engine mounts. (Photos Nos. 12 and 13) There were no indications of leakage of a flammable fluid.



Photo No. 10: View of left side of engine showing no indications of an uncontainment, case rupture or an undercowl fire. (P&W)



Photo No. 11: View of right side of engine showing no indications of an uncontainment, case rupture, or an undercowl fire. (P&W)

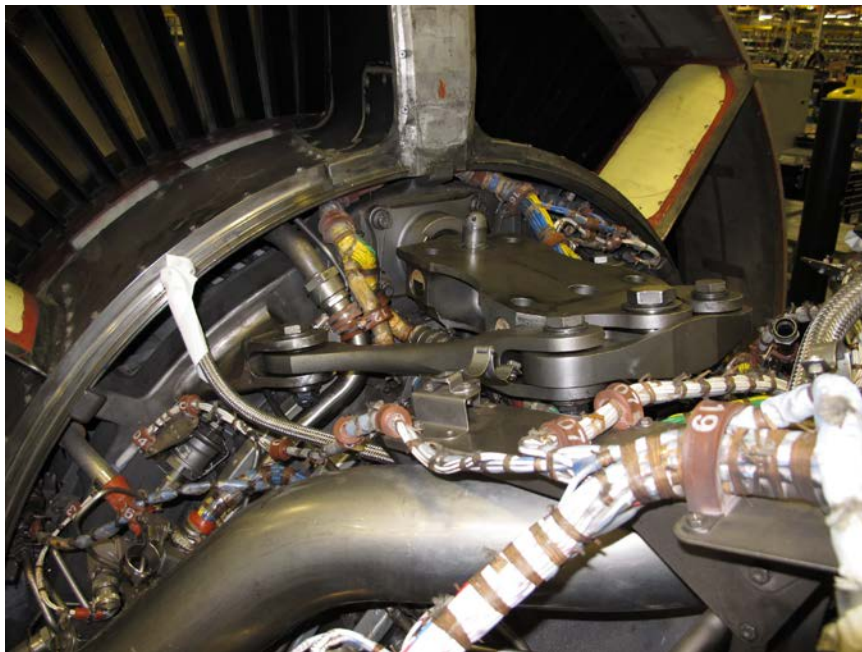


Photo No. 12: View of forward engine mount showing no apparent distress. (P&W)



Photo No. 13: View of rear engine mount showing no apparent distress. (P&W)

Neither the fan or the LPT could be rotated by hand. Later, as the turbine exhaust case and LPT were being removed, it was possible to rotate the fan rotor.

The B-nut on the rear of the hydraulic pump pressure damper was tight.

4.2 Fan

The spinner cap and spinner were in place and undamaged. (Refer to Photo No. 14)

The front fan case and the fan blade rub strip did not have any damage. The fan blade rub strip did not have any indications of circumferential rubbing with the fan blade tips.

The fan exit outer and inner cases did not have any damage. All of the fan exit guide vanes were in place and did not have any damage.

The front compressor hub was undamaged. All of the fan blades were in place and did not have any unrepaired damage. (Photo No. 14) The tips of the fan blades did not have any indications of rubbing with the fan blade rub strip. There were two fan blades that had areas on the leading edge that had been previously blend repaired. The mid span shrouds were not shingled and they did not have any evidence of having been shingled.⁹

⁹ Shingled is the condition of the mid span shroud of one blade overlapping the shroud of an adjacent blade in lieu of abutting at the contact surfaces.



Photo No. 14: View of front of the engine showing the spinner, spinner cap, and all of the fan blades were in place and that there was no damage to any of those components. (P&W)

The LPC stub shaft did not have any damage.

4.3 Low-pressure compressor

The fan exit inner case did not have any damage. The forward lip of the 2.5 bleed had circumferential scoring and gouging around the entire circumference. The splitter did not have any damage.

The LPC was not disassembled. The observations of the LPC module were made when it was in the assembled state.

The LPC drum was intact. There was no debris or any nicks or dents on the interior surface of the LPC drum.

The 1.6 and 2nd stage compressor blades were all in place and did not have any damage to the airfoils. The respective blade rub strips did not have any circumferential rub marks.

The 3rd stage compressor blades were all in place. There were several 3rd stage compressor blades that had nicks and dents on the trailing edges and one blade that had a small tear on the trailing edge. The 3rd stage compressor blade rub strip did not have any damage.

The 4th stage compressor blades were all in place. (Photo No. 15) There were two blades that were fractured transversely across the airfoils about 0.5- and 0.75-inches above the blade root platform. The fractured blades' fracture surfaces had shear lips at an angle to the

surface of the airfoil. (Photos Nos. 16 and 17) All of the 4th stage compressor blades had nicks and dents along the trailing edges and many of the blades had nicks and dents on the leading edges. There were two blades that had small tears on the trailing edge. There was one blade that was bent opposite the direction of rotation. (Photo No. 18) The 4th stage compressor blade rub strip had circumferential gouges around the entire circumference.

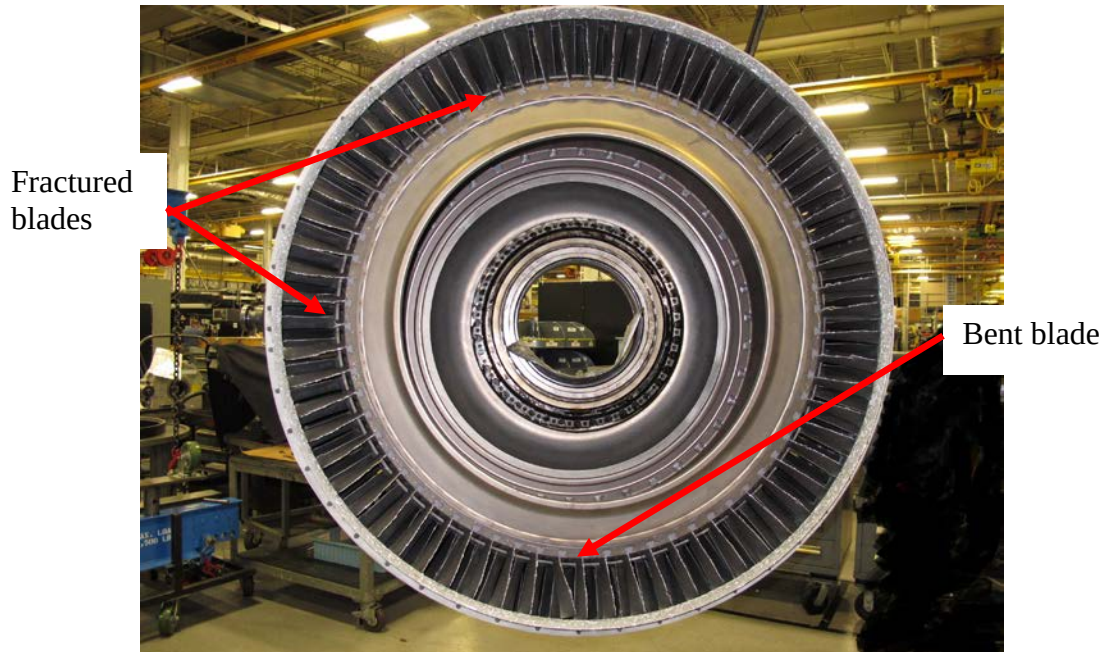


Photo No. 15: View of rear of the LPC showing all of the 4th stage compressor blades are in place, although two are fractured and one is bent opposite the direction of rotation. (P&W)



Photo No. 16: Close up view of fractured 4th stage compressor blade. (P&W)



Photo No. 17: Close up view of fractured 4th stage compressor blade. (P&W)



Photo No. 18: Close up view of bent 4th stage compressor blade also showing damage to the trailing edges of the other 4th stage compressor blades. (P&W)

The 1st, 1.6, and 2nd stage compressor stators were all in place and did not have any damage to the airfoils.

The 3rd stage compressor stator vanes were all in place. Many of the 3rd stage compressor stator vanes airfoils had nicks and dents on the trailing edges.

4.4 Intermediate case

The intermediate outer case did not have any damage. The intermediate case core struts and the flow path surfaces had nicks and dents.

The 4th stage stators were all in place. The 4th stage stators had nicks and dents to the leading and trailing edges as well as to the airfoils.

4.5 High-pressure compressor

The exterior of the compressor upper and lower front split cases and the rear compressor case did not have any damage. (Photo No. 19) The interior of the upper and lower front split cases had circumferential scoring and rub marks on the compressor blade rub strips and scoring between the VSV airfoils. (Photo No. 20) The HPC ring case did not have any thermal distress. The three HPC heatshields under the ring case were in place and did not have any thermal distress.



Photo No. 19: View of HPC showing all of the VSV rings were in place and no exterior damage to the front and rear compressor cases. (P&W)

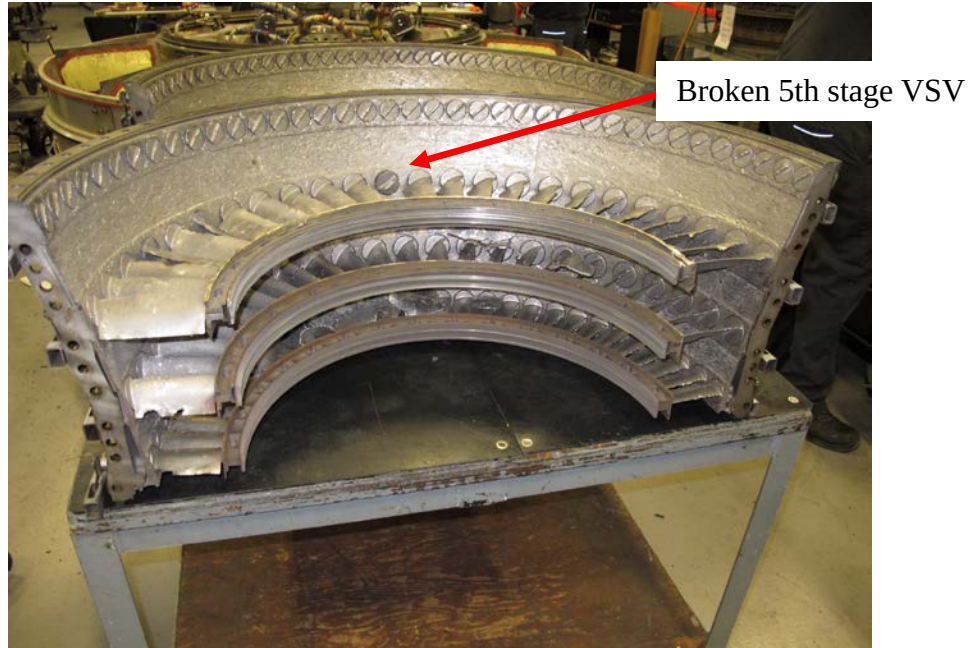


Photo No. 20: View of the interior of a front HPC case half showing circumferential scoring and gouging in the HPC blade rub strips. Also shown are the HPC IGVs and the HPC 5th stage VSV broken adjacent to the outer button. (P&W)

The VSV bell crank could be moved from the open to closed to open position using a wrench with a crowfoot wrench adapter. When the bell crank was moved, metallic sounds could be heard from within the front compressor case. When the bell crank was moved, all four stages of the unison rings moved concurrently. The rigging of the VSVs was checked and it was found to be in rig. All of the inlet guide vanes (IGV), 5th, 6th, and 7th stage unison rings were in place. The IGV and the 6th and 7th stage VSV lever arm claws were in place and engaged with the vane stems with the bolts and tab locks in place. The 5th stage VSV lever arm claws were in place and engaged with the vane stems, but there were five locations where the tab locks had twisted out of position (Photo No. 21) and the respective bolt was loose. In addition, there were two locations where the bolt and tab locks were missing. (Photo No. 22) The tack welds on the IGV, 5th, 6th, and 7th stage unison rings were intact.



Photo No. 21: Close up view of a 5th stage VSV lever arm with the tab lock twisted out of position. (P&W)

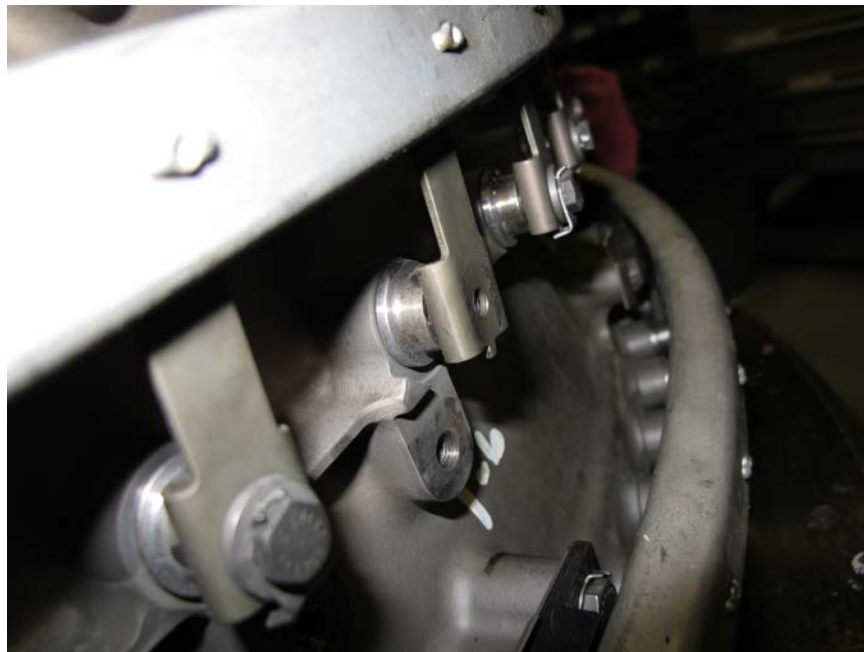


Photo No. 22: Close up view of a 5th stage VSV lever arm showing the bolt and tab lock missing. (P&W)

The IGV inner shroud ring halves were loose in the front of the HPC with no vane stems present. The IGV inner shroud ring lugs were all in place and did not have any rub marks. All of the IGVs were fractured transversely across the airfoils adjacent to the outer button. (Refer to Photo No. 20) The 6th and 7th stage VSVs were all in place. All of the VSV airfoils had nicks

and dents on the leading and trailing edges. Although the VSV rigging check showed that the VSVs were in rig, the vanes were in various angles when the front compressor case was opened.

The 5th stage compressor disk, rear compressor front hub, 6th-to-12th stage compressor drum, and 13th-to-15th stage compressor drum were intact. The 5th stage compressor disk and forward portion of the 6th-to-12th stage compressor drum had circumferential rub marks on the outer surfaces. There were indications of the orange RTV rubber in the 5th stage compressor disk's blades slots at the blade lock locations.

The 8th stage compressor stator segments were all in place and all of the 8th stage compressor stator vanes were in place. There was one vane that was pulled out of the outer shroud, although it remained in place and the inner end of the vane remained in the inner shroud. There were three other vanes that were almost completely pulled out of the outer shroud, but the inner ends remained in the inner shroud. There were numerous other 8th stage compressor vanes that were partially pulled out of the outer shroud. All of the airfoils had nicks and dents on the leading and trailing edges and on the convex and concave airfoil surfaces. The trailing edges were bent in the direction of rotation.

The 9th stage compressor stator segments were all in place and all of the 9th stage compressor stator vanes were in place in the segments. The stator vanes had nicks and dents on the leading and trailing edges and on the convex and concave surfaces of the airfoils. The trailing edges were bent in the direction of rotation.

All of the anti-vortex tubes were noted to be in place as viewed from the outside of the high pressure compressor drum.

The 10th stage compressor blades were all in place. (Refer to Photo No. 6) Of the 62 10th stage compressor blades, there were 17 blades that were fractured transversely across the airfoil adjacent to the blade root platform. The blades that were fractured adjacent to the blade root platform were distributed randomly around the entire rotor. The fracture surfaces on those blades were coarse with an irregular shape and texture. The remaining 10th stage compressor blades were fractured across the airfoil between about 1- to 1.5-inches above the blade root platform. The blades had nicks and dents on the leading and trailing edges and on the concave sides of the airfoils. The leading edges were worn down at an angle from the blade root platform to the fractured end. There were several blades with pieces missing from the trailing edges as well as tears in the trailing edges. The ends of the blades were bent opposite the direction of rotation. (Photo No. 23)



Photo No. 23: Close up view of damaged 10th stage compressor blades. (P&W)

The 10th stage compressor stator segments were all in place and all of the 10th stage compressor stator vanes were present, although there were two vanes that had the center part of the airfoil, about 1.25- and 1.75-inches, missing. The vanes had nicks and dents on the leading and trailing edges and on the convex and concave surfaces of the airfoil. There were several vanes that had tears on the trailing edges. The trailing edges were bent in the direction of rotation. (Photo No. 24)



Photo No. 24: Close up view of the damaged 10th stage compressor stator. (P&W)

The 11th stage compressor blades were all in place. (Refer to Photo No. 6) Of the 68 11th stage compressor blades, there were 4 that were fractured transversely across the airfoil adjacent to the blade root platform. The four blades that were fractured adjacent to the blade root platform were distributed around the entire rotor. All of the remaining 11th stage compressor blades were fractured across the airfoil about 1.25-inches above the blade root platform. The blades' leading and trailing edges and concave airfoil surfaces had nicks and dents. The leading edges were worn back at an angle from the platform to the fractured end. There were several blades that had tears in the trailing edges. The ends of the blades were bent opposite the direction of rotation. (Photo No. 25)



Photo No. 25: Close up view of damaged 11th stage compressor blades. (P&W)

The 11th stage compressor stator segments were all in place and all of the 11th stage compressor stator vanes were in place. There were six compressor stator vanes that were fractured between the leading and trailing edges adjacent to the outer shroud. The leading and trailing edges and the airfoils' convex and concave surfaces had nicks and dents. There were pieces missing from the trailing edges. The trailing edges were bent in the direction of rotation. (Photo No. 26)



Photo No. 26: Close up view of damaged 11th stage compressor stator. (P&W)

The 12th stage compressor blades were all in place. (Refer to Photo No. 6) Of the 72 12th stage compressor blades, there were 26 blades that were fractured transversely across the airfoil adjacent to the blade root platform. The blades fractured adjacent to the platform were distributed around the entire rotor. The remaining 12th stage compressor blades were all fractured across the airfoil between about 0.9- and 1.25-inches above the blade root platform. The fractured ends of the longer blades were bent opposite the direction of rotation. All of the 12th stage compressor blade airfoils had nicks and dents on the leading and trailing edges as well as the concave side of the airfoil. There were several 12th stage compressor blades that had tears in the trailing edge adjacent to the blade root platform. There were several of the 12th stage blades that were fractured adjacent to the blade root platform that had areas with planar elliptical-shaped fracture surfaces. The examination of those fracture surfaces showed that the elliptical-shaped areas were on the concave sides of the airfoil radiating towards the convex side surfaces. Additionally, the fracture surfaces were grainy with no indications of arrest marks and were surrounded by shear lips. (Photo No. 27)



Photo No. 27: Close up view of damaged 12th stage compressor blades. (P&W)

The 12th stage compressor stator segments were all in place and the 12th stage compressor stator vanes were all present. There were 13 airfoils that were fractured between the leading and trailing edges adjacent to the outer shroud. The leading and trailing edges and the airfoils' convex and concave surfaces had nicks and dents. All of the vanes had pieces missing from the trailing edge between the midspan area and the outer end. There was one vane that was missing the middle approximately 1.25-inches of the airfoil. There was a second vane that was fractured transversely across the airfoil at the inner end and the outer portion of the airfoil was pulled out of the outer shroud and was missing. The missing vane was from the clockwise end of the segment. The outer diameter of the shroud at the location of the missing vane had the remains of the two tack welds and there appeared to be evidence of braze material in the airfoil's slot. (Photo No. 28)



Photo No. 28: Close up view of damaged 12th stage compressor stator. (P&W)

The 13th stage compressor blades were all in place. (Refer to Photo No. 6) All of the blades were fractured across the airfoils between about 0.25- and 1.5-inches above the blade root platform. The blades' airfoils were battered with nicks and dents on the leading and trailing edges as well as the concave side of the airfoil. The longer blades were bent opposite the direction of rotation. There was one blade that had a tear on the leading edge of the airfoil. (Photo No. 29)



Photo No. 29: Close up view of damaged 13th stage compressor blades. (P&W)

The 13th stage compressor stator segments were all in place and all of the 13th stage compressor stator vanes were present, although there was one vane that was missing approximately 0.6-inches from the center of the airfoil. There were five vanes that were fractured axially between the leading and trailing edges adjacent to the outer shroud. The vanes' airfoils had nicks and dents on the leading and trailing edges and the convex and concave sides of the airfoil. All of the vanes were missing parts of the trailing edges and the trailing edges were bent in the direction of rotation. (Photo No. 30)



Photo No. 30: View of 13th stage compressor stator. (P&W)

The 14th stage compressor blades were all in place. (Refer to Photo No. 6) The 14th stage compressor blades varied in length from about 0.5- to 1.25-inches long. The blades had nicks and dents on the leading and trailing edges and on the concave surfaces. The trailing edges also had tears at the inner ends of the blades. Almost all of the airfoil ends were bent opposite the direction of rotation. (Photo No. 31)



Photo No. 31: Close up view of damaged 14th stage compressor blades. (P&W)

The 14th stage compressor stator vane segments were all in place and the 14th stage compressor stator vanes were all in place. The vanes' airfoils had nicks and dents on the leading and trailing edges and on the convex and concave surfaces. The vanes' trailing edges also had tears primarily towards the outer diameter. Many of the 14th stage compressor stator vane trailing edges were bent in the direction of rotation. (Photo No. 32)



Photo No. 32: Close up view of damaged 14th stage compressor stator. (P&W)

The 15th stage compressor blades were all in place. (Refer to Photo No. 6) All of the blades had nicks, dents and tears in the leading and trailing edges. Most of the blades had sections of the leading and trailing edges from about the midspan area outboard to the tip missing. Most of the blades were full length. There were two 15th stage compressor blades that were fractured across the airfoil about 0.75- and 0.87-inches above the platform. There were also several blades that had the airfoils bent forward. (Photo No. 33)



Photo No. 33: Close up view of damaged 15th stage compressor blades. (P&W)

The 15th stage compressor stators were in place in the diffuser case. All of the airfoils were battered with nicks and dents on the leading edges as well as the convex and concave surfaces. (Photo No. 34)



Photo No. 34: Close up view of damaged 15th stage compressor stator vanes. (P&W)

The HPC rear hub and HPC rear shaft were in place on the HPC. The diffuser outer and inner knife edge seals were full height with only some minor chipping of the hard face on the edge of the knife edge.

4.6 Diffuser/combustor

The diffuser case did not have any thermal distress. There was a large quantity of broken compressor blade airfoil fragments as well as BB-sized metal particles in the bottom of the diffuser case. (Photo No. 35)



Photo No. 35: Close up view of the interior of the diffuser case showing small metal particles in the bottom of the case. (P&W)

The combustion chamber was in place in the diffuser case. The outer and inner combustion chamber liners were intact and did not have any thermal distress. The thermal barrier coating on the inner and outer liners was intact. The outer combustion chamber liner was a light brown color while the inner combustion chamber liner was a dark gray color. (Photos Nos. 36 and 37)



Photo No. 36: View of interior of outer combustion chamber liner showing no thermal distress.



Photo No. 37: View of interior of inner combustion chamber liner showing no thermal distress.

The fuel nozzles and manifolds were all in place. (Photo No. 38) The fuel nozzles and splash plates did not have any indications of thermal distress or streaking. The splash plates were sooted. (Photo No. 39)



Photo No. 38: View of combustor liner showing all of the fuel nozzles were in place. (P&W)



Photo No. 39: Close up of fuel nozzles showing no thermal distress and the sooted splash plates. (P&W)

The 1st stage turbine vanes were all in place and intact. There was a light coating of metal spray on the 1st stage turbine vanes' airfoils. (Photo No. 40)



Photo No. 40: Close up view of 1st stage turbine vanes showing light coating of metal spray.

4.7 High-pressure turbine

The HPT rotor could be turned smoothly with very light resistance. When the HPT rotor was turned, the gear box lay shaft turned concurrently.¹⁰ When the rotor was initially turned, it could only be turned a short distance and metallic clanging sounds could be heard. Subsequently, the HPT rotor could be turned and no sounds were heard.

The HPT case was intact and did not have any thermal distress.

The HPT module was not disassembled. The observations of the HPT were in the assembled state.

The 1st stage turbine hub was intact and the 1st stage turbine front side plate was in place on the hub. There was no evidence of oil streaking on the front face of the 1st stage turbine hub. (Photo No. 41) All of the 1st stage turbine blades were in place in the hub, but all of the blades were fractured about 2-inches above the blade root platform. All of the fracture surfaces were coarse and jagged. (Photo No. 42)



Photo No. 41: View of front of HPT stage 1 disk and front side plate showing disk intact and front side plate in place with no oil staining. (P&W)

¹⁰ When the HPT rotor was turned, the gear box had been removed from the engine.



Photo No. 42: Close up view of 1st stage turbine blades showing coarse fracture surface on ends of the blades. (P&W)

The 2nd stage turbine hub was intact. The 2nd stage turbine hub web outboard of the bore-to-web fillet radius had a dark bronze-colored discoloration. The 2nd stage turbine rear side plate was in place on the hub and intact. The 2nd stage turbine rear side plate had a dark brown-colored discoloration. (Photo No. 43)

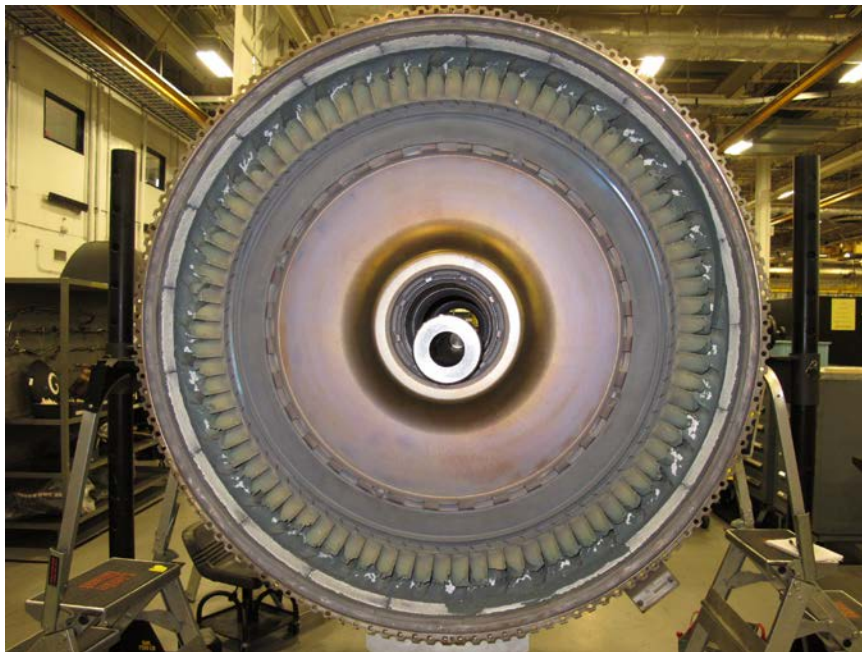


Photo No. 43: View of 2nd stage disk and rear side plate showing disk was intact and rear side plate was in place. (P&W)

All of the 2nd stage turbine blades were in place in the disk. (Refer to Photo No. 43) There were two 2nd stage turbine blades that were fractured about 1.75- and 2-inches above the blade root platform. The remainder of the 2nd stage turbine blades were fractured on the leading edges about an inch above the blade root platform that continued to about 2.25 inches above the platform at the mid chord area to the trailing edge. The fracture surfaces were coarse and grainy. The coating on the 2nd stage turbine blades had a green tint and was flaking off the airfoils adjacent to the fractures. (Photo No. 44)



Photo No. 44: View of 2nd stage turbine blades showing they were in place and had coarse fractured ends. (P&W)

The 2nd stage turbine vanes were all in place. All of the 2nd stage turbine vanes had a green tinted metal spray material on the convex airfoil surfaces that in places was delaminating. Many of the vanes had the leading edges and/or the trailing edges partially burned away. There were several vanes, primarily in the bottom portion of the engine that had the vanes' convex airfoil burned away exposing the internal air cavity. Those 2nd stage turbine vanes that had burn throughs that exposed one or two of the internal cavities on the leading edge also had metal slag trailing behind the burned area on the convex and concave sides of the airfoil.

The 1st stage turbine BOAS were all in place. The 1st stage turbine BOAS all had their thermal barrier coating present, but they also had some chipping from the aft edge and the corners. The 1st stage turbine BOAS did not have any circumferential scoring or rub marks. The 1st stage turbine BOAS had a dark gray metallic coating on the surface, but there were areas where the white color of the thermal barrier coating was visible. There were no indications of impact marks or deformation on the 1st stage turbine BOAS.

The 2nd stage turbine BOAS were all in place. The 2nd stage turbine BOAS all had their thermal barrier coating present. The 2nd stage turbine BOAS did not have any impact marks or deformation. There was a light green-colored metallic coating on the surface of the 2nd stage

turbine BOAS Nos. 4 and 5 and 9 and 10¹¹ at about 2 to 3 o'clock and 5 to 6 o'clock, respectively.¹² The coating over BOAS Nos. 4 and 5 was attached at the counter-clockwise end of the No. 4 segment and peeled off for the remainder. The coating on the Nos. 9 and 10 BOAS was attached to the full length of the segment. The 2nd stage BOAS that did not have the metallic coating were a white color.

The HPT active clearance control¹³ manifolds did not have any damage.

4.8 Low-pressure turbine

The LPC drive shaft was intact and did not have any stains or circumferential rub marks. (Photo No. 45)



Photo No. 45: View of LPC drive shaft after it was removed from the engine and separated from the LPT showing it was intact and did not have any rub marks. (P&W)

The LPT case had a 360° circumferential separation about 0.62-inches forward of the rear flange. (Photo Nos. 46, 47, and 48) The forward part of the LPT from the rear face of the front flange to the separation was about 22.25 inches. The measurement of an exemplar LPT case between the rear face of the front flange and the front face of the rear flange showed it to be

¹¹ Gas turbine engine industry convention is to number parts in a clockwise pattern, as viewed from the rear, with the No. 1 part located at the 12:00 o'clock position.

¹² All references to position or directions, as referenced to the clock, will be as viewed from the rear, looking forward, unless otherwise specified.

¹³ The active clearance control system provides air to externally cool the turbine cases to minimize the thermal growth of the cases. This reduces the gaspath leakage between the turbine blade tips and turbine case air seals to improve an engine's fuel efficiency.

about 23.06 inches. The exterior of the LPT case wall on both sides of the separation had a dark band of discoloration in comparison to other parts of the case. On the forward side of the separation, the dark band varied from about 0.5- to 0.87-inches wide and on the aft side, the band varied from about 0.25- to 0.38-inches wide. There were three small outward dents in the LPT case in the approximate plane of the 5th stage turbine rotor. The remainder of the LPT case's exterior surfaces did not have any damage. The exterior surface of the LPT case had impingement marks consistent with air flow from all of the turbine case cooling pipes. The interior of the LPT case had circumferential scuff marks in the plane of the 5th stage turbine rotor and BOAS from about 4:30 to 10 o'clock. There was circumferential scoring and gouging on the 6th stage turbine vanes' aft rail and in the plane of the 6th stage turbine rotor and BOAS around the entire circumference of the case. The 6th stage BOAS anti-rotation plates were still in place on the LPT case with their fingers intact.

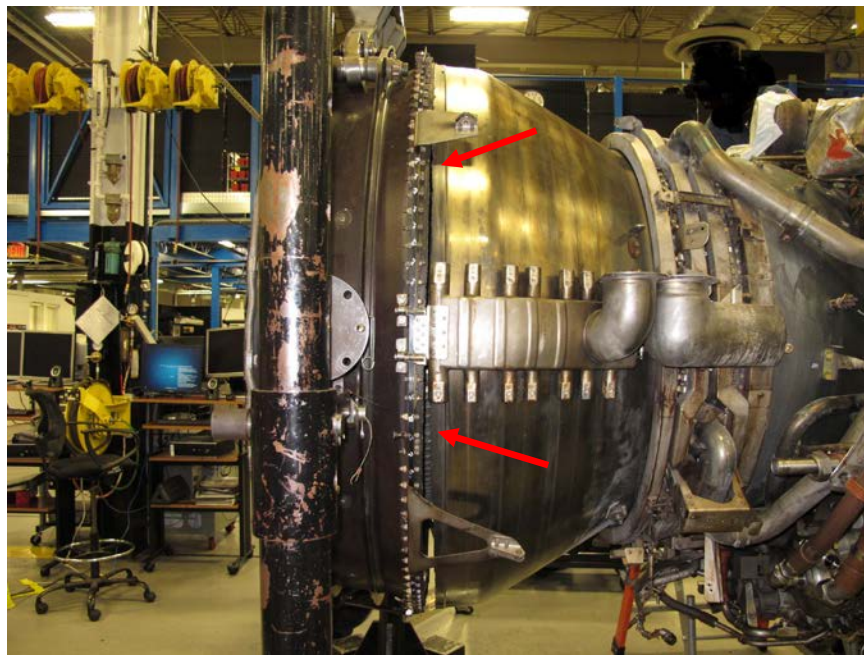


Photo No. 46: Right side of engine showing circumferential split in LPT case. (P&W)



Photo No. 47: Left side of engine showing circumferential split in LPT case. (P&W)

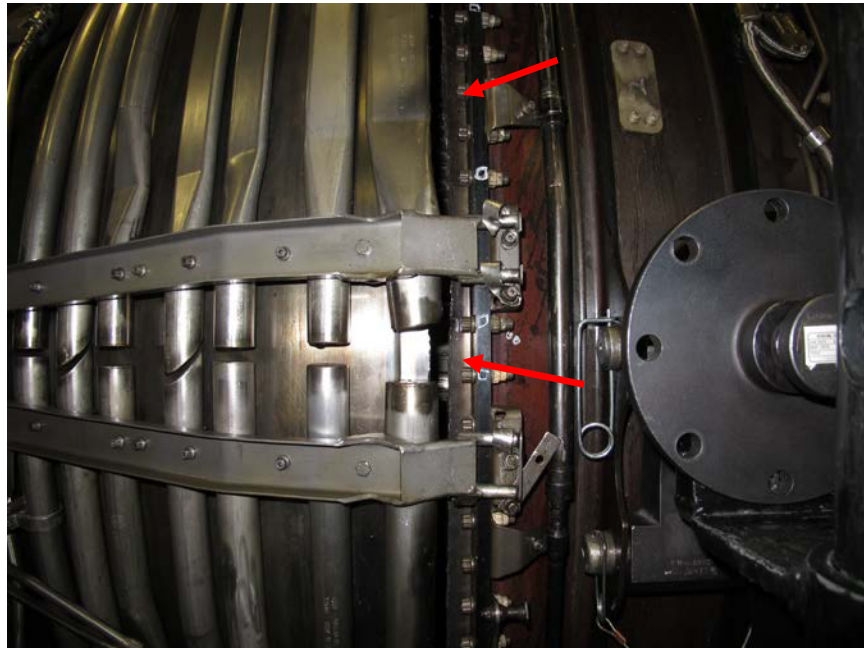


Photo No. 48: Close up of left side of engine showing proximity of split in LPT case to the rear flange. (P&W)

When placing a fragment of the forward part of a 6th stage BOAS in the normal installed position in the LPT case, it was noted that the cut in the segment was aligned with the cut in the LPT case. Similarly, when placing a fragment of the aft part of a 6th stage BOAS in the normal installed position in the LPT case, the cut in the aft segment was aligned with the cut in the aft portion of the LPT case. The angle of the edges of the separation of the LPT case corresponded to the angle of the cut edges on the 6th stage turbine BOAS. (Photo No. 49) The LPT case inner

diameter that was not protected by the front and rear fragments of the 6th stage BOAS was scuffed and circumferentially scored 360° around. With these parts in position, the scuffing and circumferential scoring could be seen to extend from the aft rail of the 6th stage turbine vane continuous to the aft C-channel portion of the 6th stage BOAS.

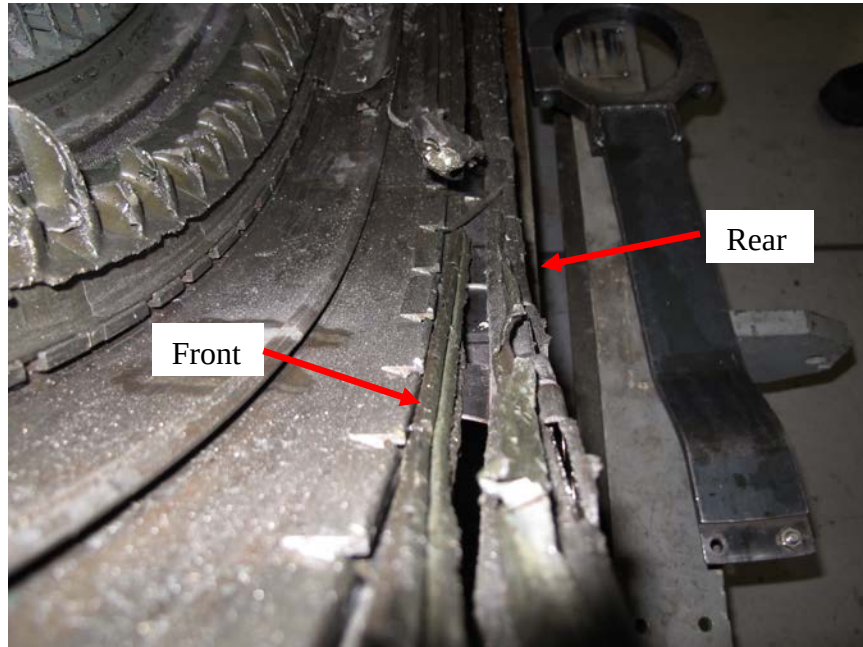


Photo No. 49: View of some fragments of the front and rear 6th stage BOAS that were placed inside of the LPT case showing the cuts in both match each other. (P&W)

To facilitate the disassembly of the engine, 14 5-inch by 2-inch 0.040-inch thick plates made from a material similar to that of the LPT case were welded on the sides of the case to hold the rear flange to the rest of the LPT case when it was removed from the engine. (Photo No. 50)



Photo No. 50: View of one of the metal plates that was welded onto the LPT case to facilitate removal of the LPT and turbine exhaust case from the HPT case.

There were 34 shards of metal that ranged in length from about 0.25 to 2-inches long and about 0.13 inches thick recovered from the turbine debris. The pieces were a dark color similar to that observed on the edges of the separation of the LPT case and had machining marks on the outer diameter that matched the machining marks on the LPT case. (Photo No. 51) The radius of the shards matched the radius of the LPT case in the area of the cut.



Photo No. 51: Close up view of metal shard found in turbine debris that had machining marks similar to machining marks on outside of LPT case. (P&W)

The upper and lower turbine case cooling manifolds did not have any damage on tubes Nos. 1 through 6.¹⁴ The No. 7 tube had denting that was primarily oriented radially inward into the tube and forward in the approximate plane of the rear air holes from about 4 to 1 o'clock. Some of the dents along the top of the tube were elongated circumferentially and there was scoring damage in the bottom of the dents. There were dried black oil stains at about 2:30, 5, 6, 7, 9, 10:30, and 12 o'clock. (Photo No. 52) The P-clamps on the aft end of the bridge brackets were twisted. The steel wool cushions on the P-clamps on the aft end of the bridge brackets were crushed and deformed. At about 9 o'clock, three of the four P-clamps were fractured. The end of the upper No. 7 tube, at about 9 o'clock, was dented in the direction of rotation. The links connecting the posts at the end of the turbine case cooling air manifold bridge brackets were buckled except for the one at about 9:15 o'clock that was missing the midspan of the link.



Photo No. 52: View of the upper and lower turbine case cooling air manifolds showing no damage to manifolds Nos. 1 through 6, but damage to manifold No. 7. (P&W)

The outer and inner transition duct segments were all in place. The flow path surface of the outer transition ducts was coated with a green tinted metallic coating. The inner transition ducts' flow path surface had a light metallic coating. There were 11 pieces of metal spray material that matched the contour of the 2nd stage turbine BOAS laying in the bottom of the engine on the outer transition ducts. (Photo No. 53)

¹⁴ The turbine case cooling manifolds are numbered 1 through 7 from front to rear.



Photo No. 53: View of outer transition duct segments showing green-tinted metal spray material on segments as well as metal spray material on the bottom of the engine. (P&W)

The 3rd, 4th, 5th, and 6th stage turbine disks were all intact. (Photos Nos. 54, 55, 56, and 57, respectively) All of the 3rd, 4th, and 5th stage turbine blades were in place in their respective disks. There were seven 6th stage turbine blades missing from the disk. Most of the 3rd stage turbine blades were fractured about 1.75-inches above the blade root platform, approximately in line with the height of the inboard ends of the broken 3rd stage turbine vane cluster airfoils. (Photo No. 58) The remaining 3rd stage turbine blades, and all of the 4th and 5th turbine blades were fractured transversely across the airfoils adjacent to the blade root platform. (Photos Nos. 59 and 60) The visual examination of the fractured blades' fracture surfaces showed that all were coarse and grainy.



Photo No. 54: View of 3rd stage turbine disk showing that it was intact. (P&W)



Photo No. 55: View of the 4th stage turbine disk showing that it was intact. (P&W)



Photo No. 56: View of the 5th stage turbine disk showing that it was intact. (P&W)



Photo No. 57: View of 6th stage turbine disk showing that it was intact. (P&W)



Photo No. 58: Close up view of 3rd stage turbine blades showing coarse fracture surface.
(P&W)



Photo No. 59: Close up view of 4th stage turbine blades showing coarse fracture surface.
(P&W)



Photo No. 60: Close up view of 5th stage turbine blades showing coarse fracture surface.
(P&W)

The 6th stage turbine blade roots that remained in the disk were all fractured transversely across the airfoil adjacent to the blade root platform. (Photo No. 61) Of the 128 6th stage turbine blades, there were 6 randomly located blade roots that were missing from the disk. The examination of the debris that was recovered from the engine and cowl at NRT and what was removed from the turbine exhaust case during the disassembly of the engine identified 116 airfoils. The recovered airfoils were straight with most having transverse fractures at the inner and outer ends. Most of the recovered airfoils were about 9-inches long. An exemplar 6th stage turbine blade was measured and the length of the airfoil was about 9.5-inches long. The recovered blades had nicks and dents on the leading and trailing edges as well as on the convex surface of the airfoil. The recovered 6th stage turbine blades did not have any necking or thermal distress. There were 10 6th stage turbine blade airfoils that still had a portion of, or all of, the tip shroud still in place. In addition, there were 93 6th stage turbine blade shrouds, either all of the shroud or part of a shroud, that were recovered. Most of the shrouds were fractured across the airfoil adjacent to the underside of the shroud although there were some that had part of the airfoil up to about 2.25 inches. Most of the shrouds still had the knife edge in place and most of the knife edges were full height. The shroud pieces were battered with nicks and dents on all of the surfaces. The shrouds did not have any indication of the circumferential rubbing, heat discoloration, or thermal distress. The cross shroud and cross notch surfaces were square to the axis of the shroud.



Photo No. 61: Close up view of 6th stage turbine blades showing coarse fracture surface. (P&W)

The 3rd stage and 4th stage turbine BOAS were in place in the LPT case. The 3rd and 4th stage BOAS honeycomb remained in place, but the honeycomb on both had deep circumferential scoring consistent with the respective turbine blade shroud knife edges.

The 5th stage BOAS were in place from about 10 to 4 o'clock. The honeycomb was rubbed down to the metal backing plate.

The engine has 12 6th stage BOAS. There were nine full length pieces and three partial pieces of the 6th stage turbine BOAS forward C-channel recovered. It appeared as if the pieces had been cut circumferentially at an angle from the inside out. The outer diameter surfaces of the forward end of the 6th stage turbine BOAS did not have any circumferential scoring. There were one full length and two partial length pieces of the aft end C-channel on the 6th stage BOAS recovered. The aft end pieces had circumferential scoring on the inner diameter surface along with material transfer and heat check cracking. There were no pieces of the center portions of the 6th stage BOAS honeycomb identified. All of the identified pieces of the 6th stage BOAS were recovered from the loose debris.

The 3rd stage turbine vane clusters' inner and outer shrouds were all in place. There was a continuous sector of 16 3rd stage turbine vane airfoils that had the midspan burned out between 5 and 7 o'clock. All of the vanes between 7 and 5 o'clock had thermal distress that ranged from the leading edge partially burned away to almost all of the airfoil burned away. The vanes had a green tinted metal spray on the airfoils. The vanes between 4 and 5 o'clock and 7 and 8 o'clock had the midspan leading edge burned away and the material was sagging. (Photos Nos. 62 and 63) All of the remaining airfoils had their coating either blistered/wrinkled or flaking. At the bottom of the engine, there was metal slag between the airfoils.



Photo No. 62: View of 3rd stage turbine vanes leading edges showing section burned out completely and other vanes partially burned away. (P&W)



Photo No. 63: View of 3rd stage turbine vanes trailing edges showing section burned out completely and other vanes partially burned away. (P&W)

The 4th stage turbine vane clusters had all of the outer shrouds in place. The complete clusters were in place except at about 2:30 o'clock where the airfoils were fractured at the midspan area and the inner shroud was missing, at about 3 o'clock where the airfoils were cracked and the inner shroud remained, and between 3:30 and 8 o'clock where the airfoils

inboard of the midspan area and the midspans were missing. (Photo No. 64) The remaining 4th stage turbine vane airfoils had nicks and dents.



Photo No. 64: View of the 4th stage turbine vanes showing a section was missing between the arrows. (P&W)

There was a continuous arc of 5th stage turbine vane clusters that were in place between 10 and 4 o'clock. The airfoils on all of the vane clusters were battered with nicks and dents. There was a single 5th stage turbine vane cluster with the airfoils fractured adjacent to the outer shroud at around 9 o'clock. (Photo No. 65)

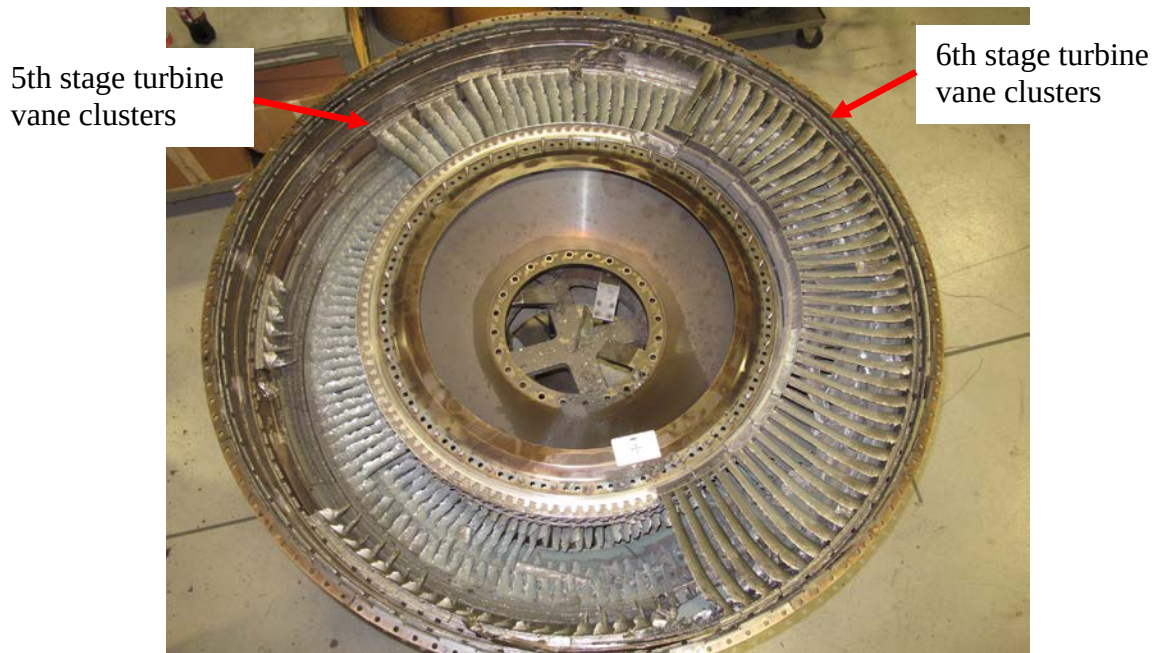


Photo No. 65: View of the 5th and 6th stage turbine vanes showing missing vanes. (P&W)

There was a continuous arc of 6th stage turbine vane clusters that were in place from about 1 to 5:30 o'clock. There were 2 6th stage vane outer shrouds between 12 and 1 o'clock with the airfoils fractured directly adjacent to the shroud. All of the airfoils were battered with nicks and dents. (Refer to Photo No. 65)

The LPT case-to-turbine exhaust case (TEC) flange bolts were all in place. The breakaway torque on about 50 bolts¹⁵ that were located around the entire LPT case-TEC flange was measured in the loosening direction using a Sturtevant Richmond Model MD 6001 0 to 600 inch-pound torque wrench SN S0659873 that was tagged indicating the calibration was due on April 13, 2018. The torque on the bolts that were checked varied from about 240 to over 600 inch-pounds. The remaining bolts were reported to be tight when they were removed to separate the turbine exhaust case from the LPT case.

4.9 Turbine exhaust case

The exterior of the turbine exhaust case did not have any damage. The interior (gas path) surface had numerous very minute nicks and dents. (Photo No. 66) The exterior and flow path surfaces of the turbine exhaust case had a light coating of oil.

¹⁵ There are 132 LPT case-to-turbine exhaust case flange bolts.



Photo No. 66: View of interior surface of turbine exhaust case and turbine exhaust case strut showing impact nicks, dents, and gouging on both. (P&W)

The turbine exhaust case struts were in place and the inner and outer standup welds were intact. The strut's convex surfaces had numerous nicks and dents. (Refer to Photo No. 66) Turbine exhaust case strut No. 1 had a dent on the leading edge in the direction of rotation. Strut No. 3 had a small hole on the convex side leading edge adjacent to the stand up weld. Strut No. 4 had gouges on the leading edge at the mid span area.

The exhaust cone was in place on the rear of the turbine exhaust case. The exhaust cone's interior surface had some very minor impact marks and dents. (Photo No. 67) The interior surface of the exhaust cone was wet with a coating of oil. The tail plug was in place on the inner duct. The tail plug had a few minor impact marks and dents. The tail plug was wet with a coating of oil. There was an airfoil piece laying in the bolt cut out at about 11 o'clock.

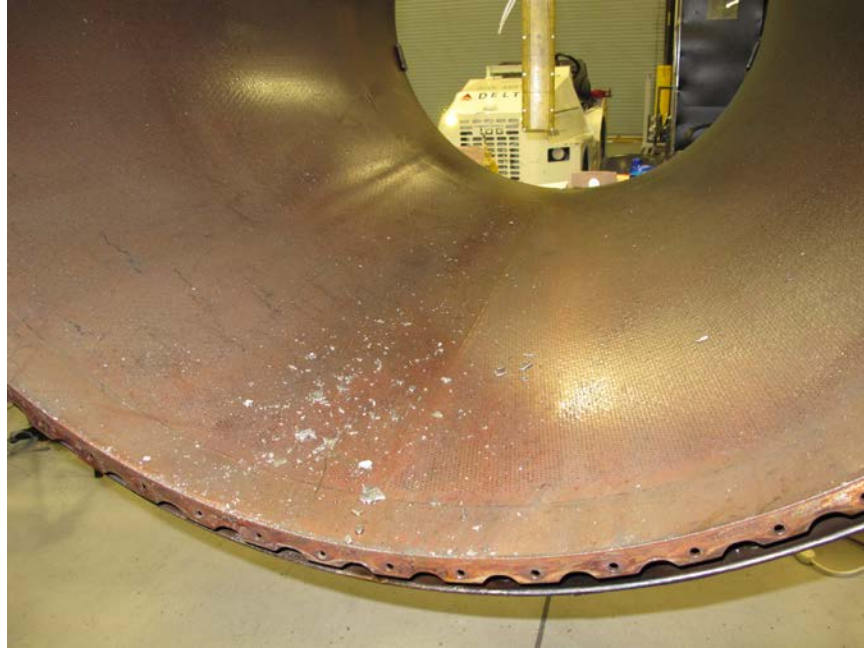


Photo No. 67: The interior of the exhaust cone showing small nicks and dents as well as metallic debris. (P&W)

The No. 4 bearing compartment cover was in place and did not have any thermal distress. The No. 4 bearing oil pressure and scavenge lines within the compartment were in place. There was no coke¹⁶ or sooting around the rear cover or on the pressure and scavenge tubes. The oil in the No. 4 bearing compartment had an amber color and did not have an acrid odor.

4.10 Bearings

The engine was not disassembled sufficiently to be able to examine the Nos. 1 and 2 ball bearings.

The No. 1.5 roller bearing was intact, wet with oil, free to rotate, and did not have any rotational damage. The No. 1.5 bearing aft carbon seal assembly was in place and intact. The No. 1.5 bearing aft carbon seal carbon element was intact and did not appear to have any damage. The No. 1.5 aft carbon seal carrier moved freely on the springs.

The No. 2 bearing support outer diameter mating face did not have any galling or surface damage. The No. 2 bearing support is located in the intermediate case.

The No. 3 bearing was intact and all of the bearing rollers were in place. The bearing was wet with oil, the rollers turned smoothly but not freely, and there was no rotational distress on the rollers.

The No. 3 bearing carbon seal plate was intact and did not have any damage.

¹⁶ Coke is the hard, crystalline residue of turbine engine oil.

The No. 3 bearing oil slinger was intact and did not have any damage. The oil slinger had a black oil sludge in the inner diameter gallery. According to the Delta aviation maintenance technicians, this was a normal condition.

The No. 4 roller bearing was intact, wet with oil, free to rotate, and did not have any rotational damage.

4.11 Oil system

The oil tank did not have an acrid odor.

The No. 4 bearing compartment oil vent line at about 8 o'clock where it crossed the LPT – turbine exhaust case flanges was dented and had a hole on the underside.

The magnetic chip detectors were pulled for examination. (Photo No. 68)

Gearbox	1 small particle
Oil tank	1 metallic flake and 1 very small particle
Nos. 1, 1.5, and 2 bearings	Several very small particles
Accessory gear box	Several very small particles
No. 4 bearing compartment	1 small particle
No. 3 bearing compartment	Several very small particles and 1 metallic sliver



Photo No. 68: Magnetic chip detectors with minute bits of debris. (P&W)

5.0 Engine core cowl

The inboard and outboard core cowls were intact. (Photos Nos. 69, 70, 71 and 72) The cowl hinges, latches and hooks were in good condition and the blow-out doors were intact and closed.



Photo No. 69: View of the outboard core cowl exterior showing no penetrations. (P&W)



Photo No. 70: View of the outboard core cowl interior showing no penetrations. (P&W)



Photo No. 71: View of the inboard core cowl exterior showing no penetrations. (P&W)



Photo No. 72: View of the inboard core cowl interior showing no penetrations. (P&W)

The inboard core cowl had 2 small splits, about 0.5- and 1.13-inches long, located about 17.5 inches from the aft edge of the cowl and about 16 inches from the bottom. (Photo No. 73) The second split had a piece of material trapped in between the separated aluminum. Internal engine debris was found on the internal lower structure of the cowling. This debris appeared to include blade tips, case and airfoil fragments. (Photo No. 74) There was evidence of debris impact affecting the condition, in some cases missing, of the aft end of the inner cowl

primer/paint at or near the plane of the 6th stage rotor. On the outer surface of the core cowl at the aft end, at or near the plane of the 6th stage rotor, there were a few locations evident on the cowl of dimpling in the plane where the cracks were noted. The forward inboard core cowl skin lower area near the bottom had areas of the aluminum bonded skin missing. The area of the missing aluminum skin was about 2.5 inches by 3.25 inches. (Photo No. 75) This area was covered with shavings-like machined material and was covered with oil consistent with the material missing prior to this event.



Photo No. 73: Close up view of the two splits in the skin of the inboard core cowl. (P&W)



Photo No. 74: Close up view of airfoil debris on the inside of the inboard core cowl. (P&W)



Photo No. 75: View of damage to inside of inboard core cowl. (P&W)

The outboard core cowl had evidence of internal engine debris on the internal lower structure of the cowl. (Photo No. 76) This debris appeared to include blade tips, case, and airfoil fragments. There was evidence of debris impact affecting the condition, in some cases missing, of the aft end of the inner cowl primer/paint at or near the plane of the 6th stage rotor. On the outer surface of the cowl at the aft end, at or near the plane of the 6th stage rotor, there were a few locations evident on the cowl of dimpling in the plane where the splits were noted.



Photo No. 76: Close up view of airfoil debris on the inside of the outboard core cowl. (P&W)

6.0 *Pilot reports*

Because of the scheduled length of the flight, the flight crew was an augmented crew consisting of the primary captain and first officer (FO) and a reserve captain and FO. The flight crew at the time of the event was the primary crew with the captain being the pilot flying and the FO being the pilot monitoring. The reserve captain and FO both reported that they were in the crew rest area for their scheduled rest period, although all of the pilots reported that the reserve pilots returned to the cockpit to assist. The captain and FO reported that about 3 hours after they departed NRT when they were about 200 miles southwest of the CARTO waypoint on the G344 airway at FL 320, they heard what was described as a thump or a bang that was followed by the airplane yawing to the left. The pilots reported that the airplane was stabilized with the rudder. The pilots also reported that following the thump or bang, the No. 1 engine's EPR and N1 rpm decreased and the EGT increased towards the redline limit, and the oil quantity was 9 quarts. The pilots reported that when the No. 1 engine's EGT reached the redline limit, they retarded the thrust lever to idle. The Engine Failure Checklist was accomplished and the VNAV drift down procedure was initiated while turning 45° off track to the right. The pilots declared an emergency and the airplane descended to FL 280. After consulting with Delta Air Lines' dispatch and maintenance, it was determined that the airplane would divert back to NRT. Upon approaching NRT, fuel was jettisoned to reduce the airplane's weight to below the maximum landing weight. The airplane made a three-engine landing at NRT without further incident. After the airport fire department personnel had checked the No. 1 engine to determine it was safe, the airplane taxied under its own power to the gate where the passengers and crew deplaned normally.

7.0 *Flight data recorder data summary*

From the time the airplane departed NRT up until just prior to the incident, the four engines' N2 rpm, EPR, EGT, and Wf¹⁷ were evenly matched. The vibration levels on the No. 1 engine were slightly lower than the other three engines.

At subframe¹⁸ reference number (SRN) 180253, the No. 1 engine's EPR, N2, EGT, and Wf were 1.29, 91 percent, 410°C, and 6,944 pounds per hour (pph), respectively. The No. 1 engine's throttle resolver angle (TRA) was 58.89°. Concurrently, the EPR, N2, EGT, and Wf for the Nos. 2, 3, and 4 engines were as follows: No. 2: EPR – 1.29, N2 – 92 percent, EGT - 431°C, Wf – 7,136 pph; No. 3: EPR – 1.29, N2 – 92 percent, EGT - 415°, Wf – 6,944 pph; and No. 4: EPR – 1.29, N2 – 91 percent, EGT - 447°C, and Wf – 7,008 pph. The EPR commanded was 1.29.

At SRN 180254, the No. 1 engine's EPR had decreased to 0.76, N2 had decreased to 86 percent, the EGT had increased to 418°C, Wf had decreased to 6,400 pph, and the TRA remained at 58.89°. Between SRN 180254 and SRN 180263, the EPR decreased from 0.76 to 0.66, N2

¹⁷ Wf stands for weight of fuel, or fuel flow, in pounds per hour (pph).

¹⁸ A subframe is equal to 1 second of time.

decreased from 86 to 66 percent, the EGT increased from 418 to 591°C, Wf had decreased from 6,400 pph to zero, and the TRA increased from 58.89 to 60.29°.

Between SRN 180269 and 180281, the Wf indication alternated between zero and 576 pph.¹⁹ During this time period, the EGT increased from 591 to 609°C.

Between SRN 180282 and 180371, the Wf varied from 576 pph down to a low of 512 pph to a high of 608 pph and back down to 544 pph. During the same time period, the EGT initially decreased from 605 to 566°C and then increased to 751°C, the EPR was steady at 0.65, and the N2 varied between 43 and 47 percent. Also within this same time period, from SRN 180282 to 180312, the TRA increased from 67.15 to 71.63° and the commanded EPR increased from 1.37 to 1.54.

Between SRN 180325 and 180337, the TRA changed from 71.63 to 33.4° where it essentially remained for the duration of the recording.

At SRN 180370, the No. 1 engine's fuel cutoff parameter changed from Run to Cutoff. The Wf decreased to zero and remained at zero for the duration of the recording. The EGT had reached at maximum of 751°C²⁰ and then decreased for the duration of the recording.

¹⁹ When the airplane was taxiing at NRT prior to takeoff, the Wf was about 1,500 pph.

²⁰ At SRN 183158 and lasting for approximately 100 minutes, the recorder EGT data was consistent with unreliable data.