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

August 22, 2002

POWERPLANTS GROUP CHAIRMAN'S FACTUAL REPORT

NTSB ID No.: DCA02MA001

A: ACCIDENT

Location: Belle Harbor, New York

Date: November 12, 2001

Time: 0917 eastern standard time (EST)

Aircraft: Airbus Industrie A300-605R, N14053, American Airlines flight 587

B: POWERPLANTS GROUP

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C: SUMMARY

On November 12, 2001, at about 0917 EST, American Airlines flight 587, an Airbus Industrie A300-605R, N14053, experienced a loss of control and crashed at Belle Harbor, New York, shortly after takeoff from John F. Kennedy International Airport (JFK), Jamaica, New York. The airplane was equipped with General Electric (GE) CF6-80C2A5 engines. The airplane had taken off from Runway 31 Left and had turned southbound as it continued to climb when control was lost and it crashed into a residential area. At the time of the accident, visual meteorological conditions prevailed. 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 passenger flight from JFK to Santo Domingo, Dominican Republic. The 2 pilots, 7 flight attendants, 246 passengers plus 5 lap children; and 5 persons on the ground were killed.

The Powerplants Group commenced its activities on November 12, 2001. The left and right engines were found at separate locations away from the main wreckage of the airplane. The left engine was recovered from in front of a gas station and the right engine was recovered from in back of a house. Neither engine had any indications of an uncontainment, case rupture, in-flight fire, preimpact malfunction, or bird strike. The thrust reversers on both engines were found in the stowed position. The auxiliary power unit (APU), which is mounted in the tailcone, was broken loose from its supports and was found laying in the tailcone slightly forward of its normal position. There were no indications on the APU of an uncontainment, case rupture, or in-flight fire. The Powerplants Group completed its on-scene activities on November 16, 2001.

Members of the Powerplants Group reconvened at American Airlines' Maintenance & Engineering Center (MEC), Tulsa, Oklahoma, on November 28, 2001, for the disassembly and examination of both of the GE CF6-80C2A5 engines. The examinations of both engines revealed the high pressure rotors' blades were bent opposite the direction of rotation and the fan and low pressure rotors' blades were bent or broken only where they were crushed by the engines' cases. There were no indications on either engine of an uncontainment, case rupture, in-flight fire, or preimpact malfunction. The Powerplants Group completed the teardown and examination of the engines on December 4, 2001.

Members of the Powerplants Group reconvened at the Honeywell engine teardown facility, Phoenix, Arizona, on December 13, 2001, for the disassembly and examination of the APU. The disassembly of the APU did not reveal any evidence of rotational damage to the compressor impellers and turbine rotors. There was no evidence of an in-flight fire, case rupture, uncontainment, or a hot air leak across a case flange.

Temperature indicating tabs were placed in several locations within the tailcone of another American Airlines A300-605R airplane. For several months, the airplane operated on American's A300 routes that included operations down into the Caribbean. The maximum temperature indicated was 160°F that was on the temp tabs directly above the bleed air duct just forward of the horizontal stabilizer center box. The maximum temperature indicated on the temp tabs that were on the vertical stabilizer access cover was 120°F.

The Port Authority of New York & New Jersey Police collected samples of fuel from the fuel truck that serviced American Airlines flight 587 before it departed JFK, and from the two tanks that supplied fuel to the truck. The fuel samples were delivered to a local laboratory for analysis and were found to conform to specifications.

D: DETAILS OF INVESTIGATION

1.0 Engine information

1.1 Engine description

The GE CF6-80C2A5 engine is a dual-spool, axial flow, high-bypass turbofan engine that features a 1-stage fan, 4-stage booster, 14-stage high pressure compressor (HPC), annular combustor with 30 fuel nozzles, 2-stage high pressure turbine (HPT), and a 5-stage low pressure turbine (LPT). The fan and booster are driven by the LPT and the HPC is driven by the HPT. The CF6-80C2A5 engine has a sea level takeoff thrust rating of 60,100 pounds flat rated to 86°F and a sea level maximum continuous thrust rating of 56,210 pounds flat rated to 77°F.[1] The engine's dry weight is 9,480 pounds.

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

1.2 Engine operating history

	Engine No. 1 (left)	Engine No. 2 (right)
Engine serial number (ESN)	695-211	690-280
Time since new	31,112 hours	25,131 hours
Cycles since new	12,282 cycles	13,216 cycles
Time since overhaul	2,887 hours	11,658 hours
Cycles since overhaul	1,072 cycles	5,421 cycles
Time since installation	694 hours	2,618 hours
Cycles since installation	264 cycles	1,229 cycles
Date of installation	August 13, 2001	July 30, 1998
Location of installation	JFK	JFK

The engines' engine condition monitoring data from October 31, 2001 to November 11, 2001, did not show any abnormal shifts in any of the recorded performance parameters: N1 rpm,[2] N2 rpm,[3] exhaust gas temperature (EGT), fuel flow, vibration, oil temperature, and oil pressure (Appendix 1).

The engines' takeoff performance data for the nine previous flights plus the accident flight showed that neither engine exceeded any of the engine operating limits.[4] The engines' takeoff performance data for the accident flight were as follows:

Parameter	Engine No. 1	Engine No. 2	Limit
N1 rpm	100.4 percent	101.0 percent	117.5 percent
N2 rpm	101.9 percent	102.4 percent	112.5 percent
EGT	814°C	831°C	960°C
Fuel flow	19,589 pounds/hour	20,358 pounds/hour	
N1 vibration	0.325	0.1	
N2 vibration	0.2	0.0	
Oil temperature	79°C	75°C	160°C
Oil pressure	69 psid[5]	68 psid	26 to 120 psid

[2] N1 is the low pressure rotor speed.
[3] N2 is the high pressure rotor speed.
[4] According to American Airlines, at 100 knots calibrated airspeed, the digital flight data acquisition unit (DFDAU) records the following engine parameters: N1, N2, EGT, fuel flow, N1 vibration, N2 vibration, oil temperature, and oil pressure. The engine takeoff performance data is then automatically downloaded through ACARS [Automatic Communications and Recording System] to an airport-based ground station that is then retransmitted via land lines to American Airlines.
[5] PSID is pounds per square inch differential that is the difference in value between two functionally related pressures occurring simultaneously at different points. In the CF6-80C2 series engine, the pressure differential is the difference between the oil pressure in the supply manifold and the bearing compartments internal air pressure, also know as vent pressure.

On November 11, 2001, the data states that the takeoff for Flight No. 988 from SJO [San Jose, Costa Rica] was a max power takeoff. The engines' takeoff performance data for the departure from SJO were as follows:

Parameter	Engine No. 1	Engine No. 2	Limit
N1 rpm	111.0 percent	110.3 percent	117.5 percent
N2 rpm	108.2 percent	107.9 percent	112.5 percent
EGT	928°C	945°C	960°C
Fuel flow	21,756 pounds/hour	21,554 pounds/hour	
N1 vibration	0.0	0.2	
N2 vibration	0.212	0.0	
Oil temperature	108°C	103°C	160°C
Oil pressure	72 psid	67 psid	26 to 120 psid

For further details on the engines' takeoff performance, refer to Appendix 2.

2.0 On-site examination

2.1 No. 1 (left) engine ESN 695-211

The No. 1 engine was recovered at the gas station at 441 Beach 129 Street, which was away from the main wreckage of the airplane. The engine was initially examined at the gas station and was then picked up and transported to American Airlines' Hangar No. 10, JFK, for further examination. The engine was then shipped to American Airlines' MEC, Tulsa, for disassembly and examination.

The engine impacted the pavement directly in front of the gas station's building between the front door and the right-hand service bay (as viewing the front of the building from the street). The engine was laying on its left side[6] partially embedded in the pavement. The engine was pointed in an easterly direction with the front of the engine facing out towards the street. Although the engine was crushed on the left side, the engine was intact and complete from the fan case to the turbine rear frame (TRF).

[6] All directions or locations on the engine, as referenced to the clock, will be as viewed from the aft looking forward (ALF), unless otherwise specified.

Photo 1: No. 1 engine with pylon still attached in front of gas station

The core of the engine between the fan frame and the LPT case was crushed inward on the left side and the core was bent. Pieces of the translating and core cowl were pushed against the engine's left side, but were mostly broken or cracked around the remainder of the engine with small areas that were burned away. There were no indications of an uncontainment, case rupture, or in-flight fire between the fan case and TRF. There were indications of a ground fire in the vicinity of the accessory gearbox, which was found broken open liberating several of the gears.

The inner liner of the inlet duct from 10 to 4 o'clock was in place on the front flange of the fan case. The remainder of the inlet duct and the entire inlet lip were separated from the fan case and were broken up.

All of the fan blades were in place in the fan disk. There was a continuous arc of fan blades between 5 and 7 o'clock that were fractured transversely across the airfoil between the root platform and the midspan shroud. The remaining fan blades were bent opposite the direction of rotation. The fan blades did not have any soft body impact damage.[7] Additionally, the fan blades did not fluoresce when examined with an ultraviolet light.[8]

[7] Soft body impact damage is characterized by the large radius of curvature of the deformation to the blade, typically a fan blade. Soft body impact damage can result from impacts with pliable objects such as birds, ice slabs, tire rubber, and plastic objects.
[8] Organic proteins such as bird remains and blood will fluoresce green when illuminated with an ultraviolet light.

Photo 2: Closeup of No. 1 engine's fan blades (FAA)

Of the six thrust reverser jackscrew drives that are installed on an A300-600 airplane's thrust reverser transcowl assembly, three intact jackscrew drives were recovered. All three of the recovered thrust reverser jackscrew drives did not have any threads showing at the forward end.[9]

The pylon remained attached to the engine. Only one of the two Halon fire bottles remained in the pylon. It was intact and still pressurized with the squib[10] in place. The fire bottle was removed from the pylon to facilitate the removal of the engine from the gas station.[11] The left engine's other Halon fire bottle, which was identified by comparing the serial number (SN) to the airplane's records, was found ruptured. It was recovered with other debris from Jamaica Bay.

2.2 No. 2 (right) engine ESN 690-280

The No. 2 engine was recovered from the backyard of the house located at 414 Beach 128 Street, which was also away from the main wreckage of the airplane. The engine was initially examined at the site and was then picked up and transported to American Airlines' Hangar No. 10 for further examination. The engine was then shipped to American Airlines' MEC for disassembly and examination.

The engine impacted the northeast corner of the house and the ground between the house and the garage. The fan and booster section was partially separated from the

[9] A thrust reverser jackscrew drive with no threads showing is indicative of the thrust reverser being in the fully stowed position.
[10] A squib is a small, electrically actuated pyrotechnic charge that forces a metal pin through a cap on the fire bottle to discharge the Halon contents.
[11] After the Halon fire bottles were removed from the pylons, the squibs were removed from the fire bottles to prevent inadvertent discharge of the Halon agent.

core with the fan and booster section pointing in a southeasterly direction and the core pointing in a northeasterly direction. Except for the separation between the fan and booster section and the core, the engine was complete from the fan case to the TRF. The fan and booster section remained attached to the core by only a short section of the fan frame and several fuel, oil, and electrical lines that were later cut to facilitate removal of the engine. Overall, the engine was burned by a ground fire, particularly around the front of the engine.

Photo 3: No. 2 engine in back of house

Photo 4: No. 2 engine's core separated from fan

There were pieces of the translating and core cowls that were against the left side of the engine, but the cowls were broken up or burned away from the right side. There were no indications of an uncontainment, case rupture, or in-flight fire.

The inlet was broken away from the inlet duct that was broken from the fan case. There was about a 270° continuous sector of the inlet lip that was missing from the rear wall of the inlet. The inlet duct was completely burned so that only carbon fiber material remained.

All of the fan blades were in place in the fan disk. There was a 180° arc of fan blades oriented towards the left side of the engine that were fractured transversely across the airfoil from adjacent to the root platform to about 1-inch outboard of the midspan shrouds. The leading edges of the fan blades had only a few small nicks and dents. None of the fan blades had any soft body impact damage. Additionally, the fan blades did not fluoresce when examined with an ultraviolet light.

Photo 5: No. 2 engine's fan blades shown after fan had been picked up (FAA)

Only one intact thrust reverser jackscrew drive was recovered and it did not have any threads showing at the forward end.

The pylon remained attached to the engine. The pylon was crushed on the top and bent towards the right in line with the HPC case and compressor rear frame (CRF) flange. Both Halon fire bottles remained in place and they were both intact and pressurized with the squibs in place. They were removed from the pylon prior to the removal of the engine from the backyard of the house.

2.3 Auxiliary power unit

The APU was found in the airplane's tailcone that was found on the southwest corner of Beach 131 Street and Newport Avenue. The APU was broken from its mounts and was found laying in the bottom of the tailcone just inside the opening of the break in the fuselage just forward of the APU's normal position. The tailpipe was separated from the APU and was found on the ground just in back of the tailcone. The APU did not have any indications of an uncontainment or in-flight fire. The area of the tailcone around the normal position of the APU did not have any shrapnel damage.

Photo 6: Auxiliary power unit

Photo 7: Tailcone over area of APU installation showing no shrapnel damage

3.0 Engine disassembly[12]

3.1 No. 1 (left) engine ESN 695-211

3.1.1 Exterior examination

The engine was received with the pylon still in place that was removed to facilitate the disassembly. The engine was crushed on the left side. There was no evidence of an in-flight fire, case rupture, or uncontainment.

Photo 8: As received condition of engine with pylon being removed (American)

3.1.2 Fan/booster

The fan case was intact, but was crushed inward against the fan disk from 6 to 11 o'clock. The Kevlar material was in place and intact. The fan blade rubstrip was rubbed from 2 to 5 o'clock. There were two adjacent imprints of fan blade tips into the fan case at about 4 o'clock.

[12] Due to the extent of damage to the engines, it was necessary at times to use an abrasive wheel and/or a plasma arc torch to facilitate the disassembly of the engine to permit the examination of the internal components.

Photo 9: No. 1 engine's fan case and rub strip showing imprint of fan blade tips

Most of the fan struts were broken and the outlet guide vanes (OGV) were deflected rearward. The fan frame rear flange was broken at 12 o'clock and from 6 to 10 o'clock.

The fan disk was intact, but was separated from the fan forward shaft and the booster spool forward flange.

The fan blade dovetails were all in place in the disk. The 10 fan blades that were oriented towards the left side of the engine were fractured transversely across the airfoil adjacent to the blade root platform. All of the remaining fan blades were full length and bent opposite the direction of rotation except for four fan blades, two groups of two adjacent blades that were located almost 180° apart that were fractured across the airfoil around the midspan shroud. The leading edges of the fan blades had a few randomly located small nicks, dents, and tears and no soft body impact damage.

Photo 10: No. 1 engine's fan disk and blades (GE)

The booster spool had a 90° circumferential fracture 2 ½-inches aft of the forward flange from 10 to 1 o'clock. It joined a 90° circumferential fracture of the forward flange from 1 to 4 o'clock that was separated from the spool and remained attached to the fan hub. The booster spool at the stage 4 had two circumferential fractures in the web to rim fillet radius from 4 to 6 and 9 to 11 o'clock. The fracture surfaces were coarse and grainy and at a 45° angle to the web surface. The booster stage 1 and 4 blades that were visible were full length and straight. The booster was not disassembled.

The fan forward shaft and fan mid shaft (FMS) were intact.

Photo 11: No. 1 engine's fan mid shaft

3.1.3 High pressure compressor

The HPC stator cases were crushed inward from the front flange back to the 8th stage bleed air manifold from 6 to 10 o'clock. The upper HPC stator case was separated from the fan frame from 9 to 12 o'clock and the lower HPC stator case remained attached to the fan frame. The upper and lower HPC stator cases were separated completely from the CRF. The HPC case horizontal split lines remained secured. The upper HPC stator case forward flange had a circumferential fracture from 10 to 11 o'clock. The HPC stator cases did not have any evidence of an in-flight fire or uncontainment. All of the variable stator vane (VSV) unison rings were broken, bent, or missing. There were six inlet guide vanes (IGV), one stage 2 VSV, and two stage 4 VSV lever arms and vane ends missing. The right hand VSV actuation lever was in place. The left hand VSV actuation lever was in place at the forward end, but separated from the case at the rear.

The HPC rotor, stages 1 through 14,[13] was intact except for the HPC stage 11 to 14 spool/shaft cone that was fractured 360° around. The fracture on the spool/shaft cone was irregularly shaped and the fracture surface was coarse and grainy and at a 45° angle to the surface. The blade rub lands on the outside of the HPC rotor had circumferential rub marks. All of the bores and webs on the inside of the HPC rotor were intact.

Photo 12: No. 1 engine's high pressure compressor rotor

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[13] The HPC rotor is an assembly that consists of a stage 1 disk, stage 2 disk, a stage 3 to 9 spool, a stage 10 disk, and a stage 11 to 14 spool/shaft.

Of the 36 HPC stage 1 blades, 20 blades were missing from the blade slots,[14] 8 were bent over opposite the direction of rotation, and 8 were fractured transversely across the airfoil adjacent to the blade root platform. There were 22 HPC stage 2 blades that were bent opposite the direction of rotation, 2 that were missing from the blade slot, and 2 that were fractured transversely across the airfoil adjacent to the blade root platform. All of the HPC stage 3, 4, and 5 blades were fractured across the airfoil adjacent to the platform. The HPC stage 6 through 14 blades had an approximately 120 to 180° arc of blades on each stage that were bent opposite the direction of rotation and the remainder of the blades were fractured across the airfoil adjacent to the platform. Several HPC airfoil pieces were recovered on the outside of the upper HPC stator case just forward of the HPC and CRF flange between 2 and 3 o'clock.

3.1.4 Compressor rear frame

The CRF did not have any indications of thermal distress, case rupture, or in-flight fire. The CRF had a 19-inch long (circumferential) by 1-inch wide (axial) rectangular-shaped hole just aft of the fuel nozzle mount pads from 2 to 4 o'clock. The edges of the hole were not curled outward. The CRF was crushed inward from 6 to 8 o'clock between the front flange and the fuel nozzle mount pads and there were circumferential cracks adjacent to the front flange at 6 o'clock and adjacent to the mount pads at 8 o'clock.

Photo 13: No. 1 engine's compressor rear frame (American)

[14] Numerous broken airfoil pieces were found in the HPC case during the disassembly.

The combustor liner was intact and did not have any indications of thermal distress. There was no metal spray material[15] on the dome or on the outside of the combustor liner.

The fuel nozzles were all in place although several nozzles had missing attaching bolts. The fuel nozzles did not have any indication of a fire. There was no build up of metal spray material on the fuel nozzles.

3.1.5 High pressure turbine

The HPT stator case was crushed inward from 5 to 10 o'clock. The HPT stator case front and rear flanges remained attached to the CRF and LPT stator case, respectively.

The HPT stage 1 and 2 disks were intact. The disks had circumferential rub marks on the rear faces of the disks' rims.

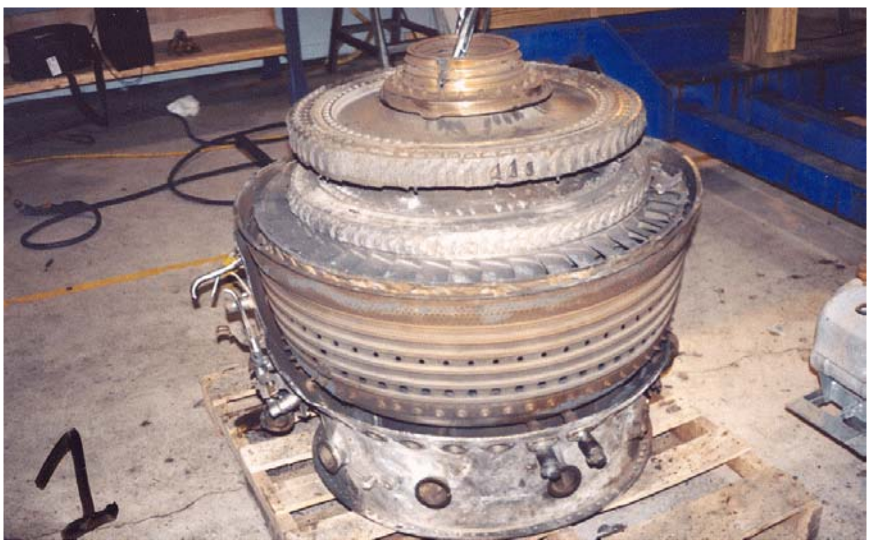

Photo 14: No. 1 engine's combustor liner and HPT stage 1 and 2 disks

All of the HPT stage 1 and 2 blades' roots were still in place in their respective disks. All of the HPT stage 1 and 2 blades were fractured transversely across the airfoil adjacent to the blade root platform. Those blades that were fractured slightly higher on the airfoil had the ends bent opposite the direction of rotation.

All of the HPT stage 1 and 2 nozzle segments were in place in the HPT stator case and intact except from 5 to 10 o'clock where the case was crushed. The HPT nozzle segments had small nicks and dents to the airfoils, but no axial cracking in the airfoils or fillet radius.

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[15] The metal spray material that is under discussion is that which would occur from internal damage and not that is normally applied to a part or component.

3.1.6 Low pressure turbine

The LPT stator case was crushed inward from 5 to 10 o'clock. The LPT case had a 360° circumferential fracture that varied from 7- to 11-inches rearward from the front flange. The LPT stator case did not have any evidence of an in-flight fire or uncontainment.

Photo 15: No. 1 engine's low pressure turbine case and the broken turbine rear frame (American)

The LPT stage 1 disk was intact except for a circumferential fracture in the rear spacer arm adjacent to the disk-to-arm fillet radius from 5 to 8 o'clock where the spacer arm was bent inward. The fracture surfaces were coarse and grainy and at a 45° angle to the spacer arm. The LPT stage 1 disk remained attached to the LPT stage 2 disk except from 5 to 8 o'clock where the spacer arms were deflected where the inner ends of LPT stage 2 nozzles were pushed inward. The disk rim was bent rearward from 5 to 8 o'clock.

The LPT stage 2 disk was intact except for a circumferential fracture in the front spacer arm adjacent to the disk-to-arm fillet radius from 5 to 8 o'clock where the spacer arm was bent inward. The fracture surfaces were coarse and grainy and at a 45° angle to the spacer arm. The disk rim was bent rearward from 5 to 8 o'clock.

The LPT stage 3 disk was intact. The disk rim was bent rearward from 7 to 9 o'clock.

The LPT stage 4 disk's bore and web were intact, but was missing an approximately 100° arc of the forward spacer flange that remained attached to the LPT stage 3 disk. The LPT stage 4 disk had a 10-inch long circumferential crack that was about 2-inches outboard from the bore, which at the clockwise end, intersected with a diagonal crack to a blade slot bottom. The fracture surfaces were coarse and grainy and were at a 45° angle to the face of the disk. Except for where the LPT stage 4 disk spacer flange was broken and remained attached to the LPT stage 3 disk, all of the flange bolts were broken and the LPT stage 4 disk was separated from the LPT stage 3 disk 360° around.

The LPT stage 5 disk was intact, but was ovalized. The LPT stage 5 disk's front spacer flange was separated from the LPT stage 4 disk rear flange along a 90° arc at 9 o'clock and the inner ends of several LPT stage 5 nozzles were sticking through the separation.

The LPT disks did not have any circumferential rub marks.

All of the LPT blades were in place in their respective disk blade slots. The LPT stage 1 and 2 blades were almost full length, but were bent slightly and missing the tip shrouds. The LPT stage 3, 4, and 5 were full length except for those blades that were oriented towards the left side of the engine and were broken at various lengths above the blade root platforms. The LPT blades did not have any rub marks or metal spray material on the airfoil surfaces.

Photo 16: No. 1 engine's low pressure turbine case cut away exposing blade and vanes (GE)

Photo 17: No. 1 engine's LPT stage 5 blades

All of LPT nozzle segments were in place and intact except for those nozzle segments that were oriented towards the left side of the engine that were crushed. The inner ends of several LPT stage 2 nozzle segments were through the LPT stage 1 rear and LPT stage 2 front spacer arms.

3.1.7 Exhaust

The TRF was crushed against the inner duct from 6 to 9 o'clock and buckled from 9 to 11 o'clock. The TRF forward flange was separated from the LPT stator case rear flange from 5 to 2 o'clock. The TRF inner duct rear face had crescent-shaped cracks at 6 and 8 o'clock. The TRF inner duct was fractured completely around the strut at 12:00 and displaced inward. TRF struts No. 2, 3, 4, and 5 were in place and straight, but the TRF had U-shaped cracks around the leading edge of struts No. 4 and 5. All of the other struts were crushed. The strut No. 12 was fractured off and the remaining TRF struts were buckled The TRF struts did not have any metal spray material or impact marks.

The exhaust duct was crushed against the TRF inner duct from 5 to 10 o'clock. There were no impact marks or metal spray material on the inner diameter of the exhaust duct.

The exhaust cone remained attached to the TRF inner duct, but was crushed flat. The exhaust cone did not have any impact marks or metal spray material.

3.1.8 Bearings/oil system

The oil tank was intact.

The No. 1 bearing cage was intact, but the forward inner race was fractured into several pieces and all but four of the balls were liberated.[16] The No. 1 bearing balls and inner and outer races were dry and did not have any rotational damage. The forward and rear inner races had an approximately 120° arc of areas of spalled[17] material that corresponded to the spacing of the balls in the No. 1 bearing ball cage. Several of the No. 1 bearing balls had small areas of spalling.

The No. 6 bearing cage was fractured axially, but all of the rollers were in place although three consecutive rollers that were adjacent to the fractured cage were partially separated from the cage. The rollers were wet with oil and did not have any rotational damage.

The remaining main shaft bearings were not examined.

3.1.9 Accessories

The 14th stage air valve was in the closed position.[18]

The right-hand VSV actuator piston was extended $^{23}/_{64}$-inch from the top of the housing to the base of the nut. The left-hand VSV actuator piston could not be measured.

The variable bypass valves (VBV) that remained attached to the unison ring were in the closed position.

All of the accessory gearbox mounted components were broken or damaged and were not tested.

3.2.0 No. 2 (right) engine, ESN 690-280

3.2.1 Exterior examination

The engine was received in two pieces, the fan and the core, with the pylon still attached to the core. The pylon was removed to facilitate the disassembly. The engine was crushed and bent slightly on the left side around the area of the rear of the HPC. The engine had damage from a ground fire, particularly to the fan module.

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[16] During the on-site examination of the No. 1 engine, all of the No. 1 bearing balls were visible and were in place in the bearing cage. The bearing balls were liberated during recovery of the engine.
[17] Spalling is the cracking and flaking of particles out of a surface.
[18] The 14th stage air valve is operated electrically and actuated pneumatically. In the absence of any air pressure, the valve is spring loaded to the closed position.

Photo 18: As received condition of No. 2 engine minus the fan (American)

3.2.2 Fan/booster

The fan case was intact, but was crushed inward from 8 to 4 o'clock almost flush to the fan disk. The Kevlar material was in place and intact, but the outer wraps of material were burned. The fan blade rubstrip was missing from the fan case. There were no circumferential rub marks on the fan case under where the rubstrip should have been, but there were three adjacent imprints of fan blade tips on the right side of the case directly opposite where the case was crushed. There were two fan blade outer panels that were embedded into the fan rubstrip. All of the fan frame struts were broken and the all of the OGVs were burned.

The fan disk was intact, but was separated from the fan forward shaft.

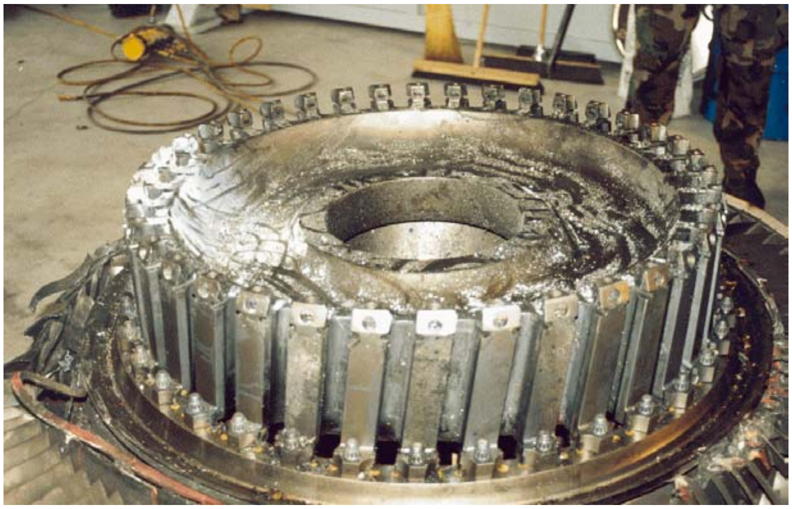

Photo 19: No. 2 engine's fan disk

The fan blade dovetails were all in place in the disk. The 16 fan blades that were oriented towards the left side of the engine were fractured adjacent to the root platform. All of the remaining fan blades were full length and bent randomly, except for one that was fractured near the midspan shroud. The leading edges of the fan blades had a few randomly located nicks and dents.

Photo 20: No. 2 engine's fan blades

The booster spool rear flange had a 180° circumferential fracture from 9 to 3 o'clock that was at the web-to-rim fillet radius except for a 6-inch long section at 12 o'clock where the fracture was 1 ½-inches inboard from the radius. There were two circumferential fractures between the seal teeth and the stage 1 blades that were 2 7/8-inches long at 10 o'clock and 4 ¾-inches long at 11 o'clock. The fracture surfaces were coarse and grainy and at a 45° angle to the web surface. The remainder of the booster spool did not have any apparent cracks. The booster stage 1 and 4 blades were all in place in the blade slots. The booster stage 1 blades were bent opposite the direction of rotation except for the blades at 9 to 1 o'clock that were fractured across the airfoil adjacent to the blade root platform. The booster stage 4 blades were full length and straight except for the blades from 7 to 2 o'clock that were fractured across the airfoil adjacent to the blade root platform. The booster was not disassembled.

The fan forward shaft and FMS were intact.

Photo 21: No. 2 engine's fan mid shaft

3.2.3 High pressure compressor

The upper and lower HPC stator cases remained attached to the fan frame and the CRF and the horizontal split lines were secured. The upper and lower HPC stator cases were intact and did not have any indications of an in-flight fire or uncontainment, but the 8th and 11th stage bleed air manifolds were crushed and buckled from 3 to 12 o'clock. The right and left hand VSV actuation levers were in place and all of the VSV unison rings were still attached. The VSV unison rings were intact and all of the vane levers were in place except for two 5th stage VSVs that were missing at 12 and 5 o'clock.

Photo 22: No. 2 engine's high pressure compressor stator case (American)

The HPC rotor, stages 1 through 14, was intact. The blade rub lands on the outside of the HPC rotor did not have any circumferential rub marks. All of the webs and bores on the inside of the HPC rotor were intact.

Photo 23: No. 2 engine's high pressure compressor rotor

The HPC stage 1 through 3 and 6 through 14 blades were bent opposite the direction of rotation. All of the HPC stage 4 blades were bent opposite the direction of rotation except for six random blades that were fractured transversely across the airfoil adjacent to the root platform. All of the HPC stage 5 blades were fractured transversely across the airfoil adjacent to the root platform. All of the HPC airfoils had nicks and dents on the leading and trailing edges.

3.2.4 Compressor rear frame

The CRF was intact and was still attached to the HPC stator case, but was separated from the HPT stator case 360° around. The CRF did not have any evidence of a case rupture or in-flight fire, but was crushed inward from 7 to 12 o'clock.

The combustor liner was intact except where it was buckled inward from 7 to 12 o'clock that corresponded to where the CRF was buckled inward. The forward louvers were split where the liner was buckled. There was no metal spray material on the dome or outside of the combustor liner.

The fuel nozzles were all in place although several nozzles had missing attaching bolts. The fuel nozzles did not have any indication of a fire. There was no build up of metal spray material on the fuel nozzles.

3.2.5 High pressure turbine

The HPT stator case was intact and did not have any evidence of an in-flight fire or uncontainment.

The HPT stage 1 and 2 disks were intact.

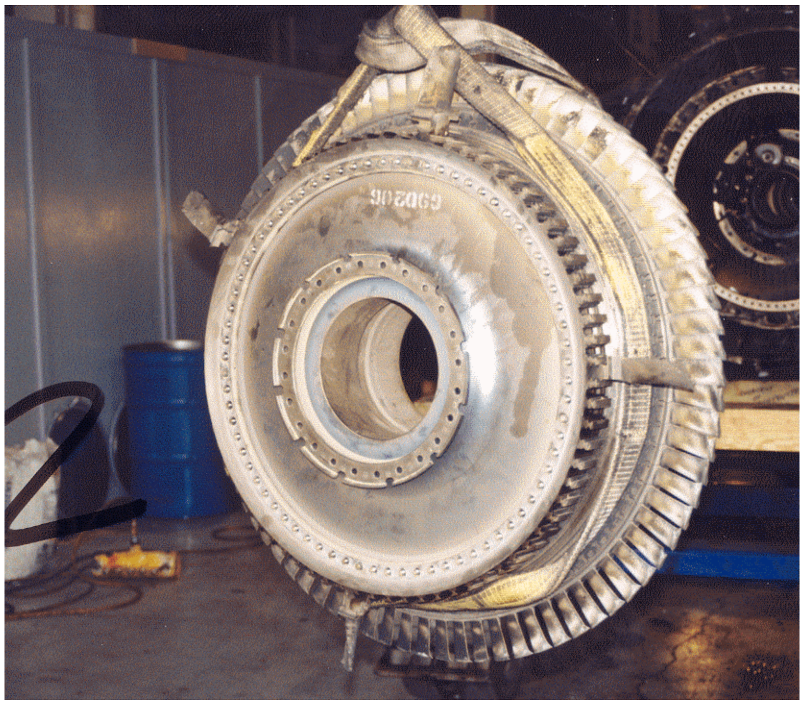

Photo 24: No. 2 engine's high pressure turbine stage 1 and 2 disks

The HPT stage 1 blades were all in place and were full length. The stage 1 blades had circumferential rub marks on the tips. The HPT stage 2 blades were all in place. All of the stage 2 blades were fractured transversely across the airfoil about ½-inch from the tip. All of the broken ends were bent opposite the direction of rotation.

The HPT stage 1 and 2 nozzle segments were all in place. The stage 1 nozzles had small nicks and dents on the trailing edge and there was one nozzle segment at 7:30 o'clock that had cracks in the airfoil adjacent to the outer fillet radius. The stage 2 nozzles had nicks and dents on the leading and trailing edges.

The HPT module was not disassembled.

3.2.6 Low pressure turbine

The LPT stator case was intact and remained attached to the HPT stator case and the TRF with all of the bolts in place. The LPT stator case did not have any evidence of an in-flight fire or uncontainment.

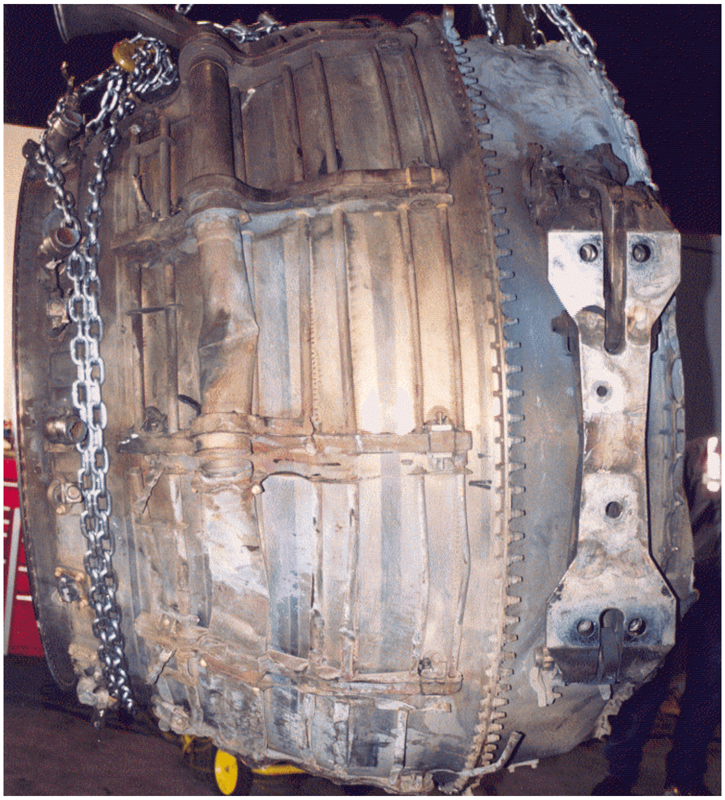

Photo 25: No. 2 engine's low pressure turbine case (American)

The bores of the LPT stage 1 and 2 disks were intact. The bores of the LPT stage 3, 4, and 5 disks were not visible.

The LPT stage 1 blades were all in place, straight, and did not have any metal spray material on the airfoils. There was an approximately 100° continuous sector of LPT stage 1 blades from 3 to 6 o'clock that had the leading edge of the airfoils up to approximately ½-inch long by $^{1}/_{8}$-inch deep and the adjacent tip shroud broken away. The fracture surfaces of the missing material were coarse and grainy and there was no indication of any circumferential rub marks. The LPT stage 5 blades were all in place, straight, and did not have any metal spray on the airfoils.

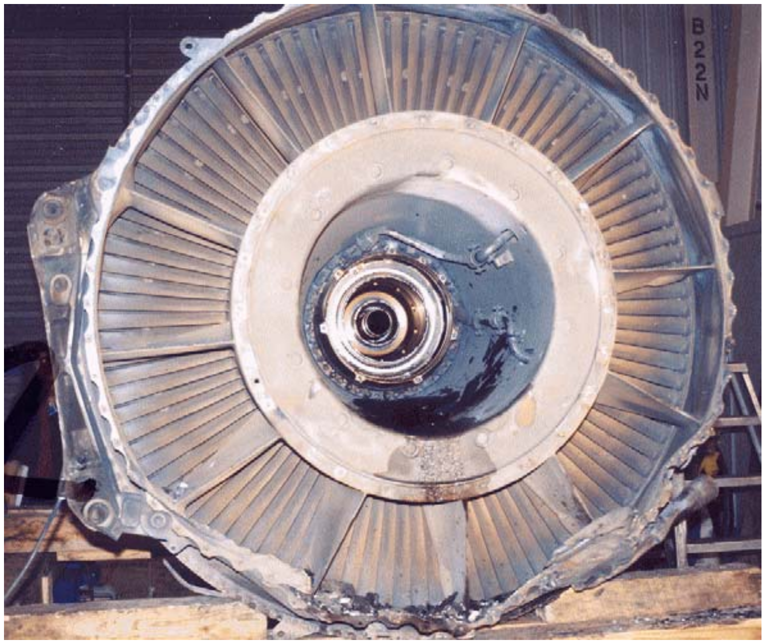

Photo 26: No. 2 engine's LPT stage 5 blade airfoils

The LPT module was not disassembled.

3.2.7 Exhaust

The TRF was intact and the struts were straight and did not have any metal spray material. The rear flange of the TRF was bent inward from 7 to 9 o'clock.

The exhaust duct was in place on the TRF rear flange, but was axially crushed from the rear forward against the TRF from 4 to 10 o'clock. The exhaust cone was in place on the TRF inner duct, and was crushed inward on the left side.

3.2.8 Bearings/oil system

The oil tank was ruptured towards the rear of the engine.

The No. 1 bearing was intact, damp with oil, and did not have any rotational damage. The bearing balls had small areas that were spalled.

The No. 6 bearing was intact, wet with oil, and did not have any rotational damage.

The remaining main shaft bearings were not examined.

3.2.9 Accessories

The accessory gearbox was broken in half. There was no indication of an in-flight fire on the accessory gearbox or attached components.

The right-hand VSV actuator piston was extended $^{55}/_{64}$ inch from the top of the housing to the base of the nut. The left-hand VSV actuator piston could not be measured.

The VBVs that were still attached to the unison ring were open.

All of the accessory gearbox mounted components were broken or damaged and were not tested.

4.0 Auxiliary power unit information

4.1 Auxiliary power unit description

The APU consists of three main components: the power section, the load compressor, and the accessory gearbox. The power section has a two-stage centrifugal flow compressor drive by a three-stage axial flow turbine that is governed by a fuel control unit (FCU) and an electronic control box. The load compressor has a single-stage centrifugal compressor that is directly driven by the power section and provides bleed air to the airplane' pneumatic system. The accessory gearbox is directly driven by the power section and carries the FCU, lubrication pumps, AC [alternating current] generator, cooling air fan, and starter motor.

4.2 Auxiliary power unit operating history

The APU installed in the airplane was a Honeywell[19] GTCP331-250H, part number 381388, serial number P-1077. According to American Airlines' records, the APU had accumulated 19,723 hours time since new and 12,104 cycles since new. The APU was installed in the airplane on September 20, 2001, and had accumulated 426 hours and 215 cycles since installation.

5.0 Auxiliary power unit teardown

5.1 External examination

As received, the APU was separated into two sections between the load compressor and the power section.

The power section did not have any indications of an in-flight fire, uncontainment, or case rupture.

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[19] The engine's data plate listed AlliedSignal as the manufacturer. AlliedSignal merged with Honeywell and the new company is known as Honeywell.

Photo 27: APU power unit (Honeywell)

The upper mount was still in place. The four bolts securing the upper mount to the power section were still in place and the safety wire was intact. Three of the four bolts were tight and the remaining bolt was loosened with just a slight amount of torque.

The insulation blankets were still in place over the compressor and turbine sections. The insulation blanket over the compressor section was pulled down about 2 inches on the forward upper left side. The insulation blanket over the turbine section was torn on the bottom where the turbine case was crushed. The clamp securing the insulation blankets to the power section was still in place.

The rods supporting the surge valve were bent and the surge valve housing was displaced rearward. The surge valve housing was creased inward 90° on the outflow side. The butterfly valve was partially open. The creased housing prevented full opening of the butterfly valve. The surge air duct was still attached to the power section, but was separated from the butterfly valve at the rear side of the butterfly valve housing.

5.2 Compressor

The compressor case was intact except for a ½-inch long circumferential split at the bottom of an inward dent that was 1-inch aft of the front flange at 6:30 o'clock.

The inlet-to-compressor case flange bolts were all in place. The break away torques were measured with a 0 - 150 inch-pounds torque wrench, SN TW 841, that was last calibrated on July 9, 2001,and was due for calibration on January 9, 2002. The installation torque for the inlet-to-compressor case flange bolts is 50 – 60 inch-pounds. The breakaway torque values were:

No.	Torque	No.	Torque	No.	Torque	No.	Torque
1	45 (np, m)	14	loose	27	36 (m)	40	30
2	25 (m)	15	loose	28	loose (m)	41	36
3	35 (m)	16	loose	29	loose	42	30
4	25 (m)	17	loose	30	loose	43	26
5	20	18	loose	31	loose	44	26
6	loose[20]	19	loose	32	loose	45	27
7	loose	20	loose	33	loose	46	loose
8	loose	21	20 (np)	34	loose	47	25
9	loose	22	loose	35	loose	48	25
10	loose	23	loose	36	loose	49	26 (m)
11	loose	24	loose	37	21	50	40 (m)
12	loose	25	45 (m)	38	25	51	50 (m)
13	loose	26	65 (m)	39	20	52	56

Note: ‘m’ is mount pad bolt; ‘np’ is nut plate bolt.

There were no heat discoloration marks on the flange surfaces.

The tie shaft was intact.

The 1st and 2nd stage impellers were intact and did not have any circumferential rub marks on the impeller blades. The 1st and 2nd stage impeller shrouds did not have any circumferential rub marks.

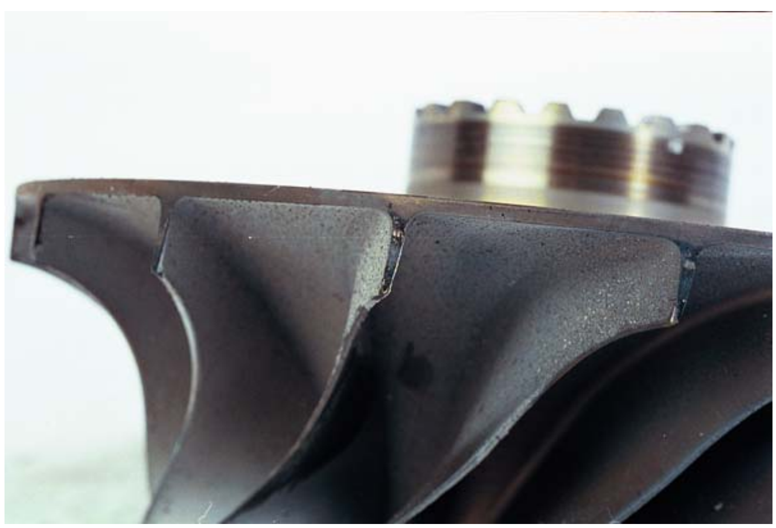

Photo 28: Close up of 1st stage impeller showing no circumferential rub marks (Honeywell)

[20] Loose indicates that the bolts were in place and snug in the bolthole, but did not require any measurable torque to break away.

Photo 29: 1^{st} stage impeller shroud coated with soot and where soot was wiped away, no impact marks from the impeller (Honeywell)

5.3 Combustor

The combustion chamber case was intact and did not have any indication of a case rupture. The case was dented inward at 6:30 o'clock between 1 and 3-inches aft of the forward flange.

The combustion chamber case bolts were all in place. The break away torques of the bolts were measured with a 0 - 150 inch pounds torque wrench, SN TW 841. The installation torque for the combustion chamber case bolts is 50 – 60 inch-pounds. The breakaway torque values were:

No.	Torque	No.	Torque	No.	Torque	No.	Torque
1	125 (np)	14	150+	27	25 (m)	40	150+
2	140	15	135	28	50 (m)	41	150+
3	135	16	150+	29	145	42	150+
4	145	17	105 (np)	30	95	43	150+
5	150+	18	150+	31	135	44	150+
6	150+	19	150+	32	110	45	150+
7	150+	20	150+	33	115	46	150+
8	150+	21	130 (np)	34	135	47	150+
9	135	22	150+	35	150+	48	140 (m)
10	150+	23	150+	36	150+	49	140 (m)
11	150+	24	150+	37	150+	50	125 (m)
12	150+	25	130 (m)	38	150+	51	missed
13	150+	26	105 (m)	39	150+	52	135 (np)

There were no heat discoloration marks on the flange surfaces.

The combustion chamber was intact and did not have any evidence of thermal distress.

5.4 Turbine

The turbine bearing support was crushed inward and forward from 3 to 10 o'clock. There was no evidence of an uncontainment in the turbine duct.

The plenum heatshield seal, which is referred to as the tadpole seal, was in place, intact, and did not have any discoloration or evidence of thermal distress. There was evidence that the silicon seal was installed, about one-third was still in place and the remainder had melted and turned to a white ash.

The turbine bearing support flange bolts were all in place and the safety wire was intact on all of the bolts. The break away torque was measured on all of the bolts using a 0 - 150 inch pounds torque wrench, SN TW 841. The installation torque for the turbine bearing support bolts is 60-63 inch-pounds. Because of the geometry of the combustion chamber case, it was necessary to use a universal adapter with the torque wrench to untorque the flange bolts. The measured breakaway torque values were:

No.	Torque	No.	Torque	No.	Torque	No.	Torque
1	less than 30[21]	9	125	17	100	25	130
2	less than 30	10	140	18	105	26	132
3	Missed	11	135	19	105	27	130
4	95	12	115	20	95	28	120
5	95	13	120	21	120	29	125
6	111	14	125	22	111	30	145
7	114	15	105	23	116	31	150+
8	110	16	130	24	120	32	150+

There were no heat discoloration marks on the flange surfaces.

The 1st and 2nd stage turbine disks and 3rd stage turbine wheel[22] were intact. All of the airfoils were still in place in the respective disks and were intact and full length. There were no circumferential rub marks on the blade tips or turbine shrouds.

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[21] Bolts No. 1 and 2 were measured with a 30 – 150 inch-pounds torque wrench, SN TW 8157, that was last calibrated on August 16, 2001, and is due for calibration on February 16, 2002. When the first two bolts were untorqued at less than 30 inch-pounds, 0 – 150 inch-pounds torque wrench, SN TW 841 was substituted

[22] The 1st and 2nd stage turbine disks are different from the 3rd stage turbine wheel in that the 1st and 2nd stage disks have individual blades inserted into each of the blade slots whereas on the 3rd stage wheel, the airfoils are integral to the wheel.

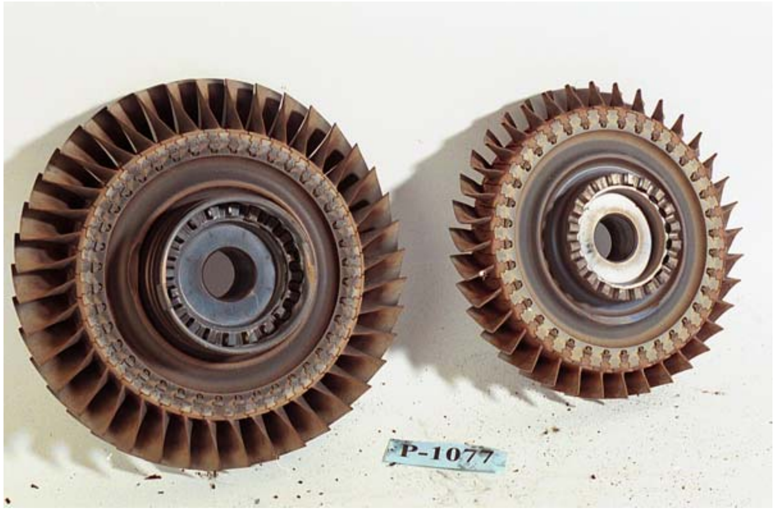

Photo 30: 1st and 2nd stage turbine disks showing no bent blades (Honeywell)

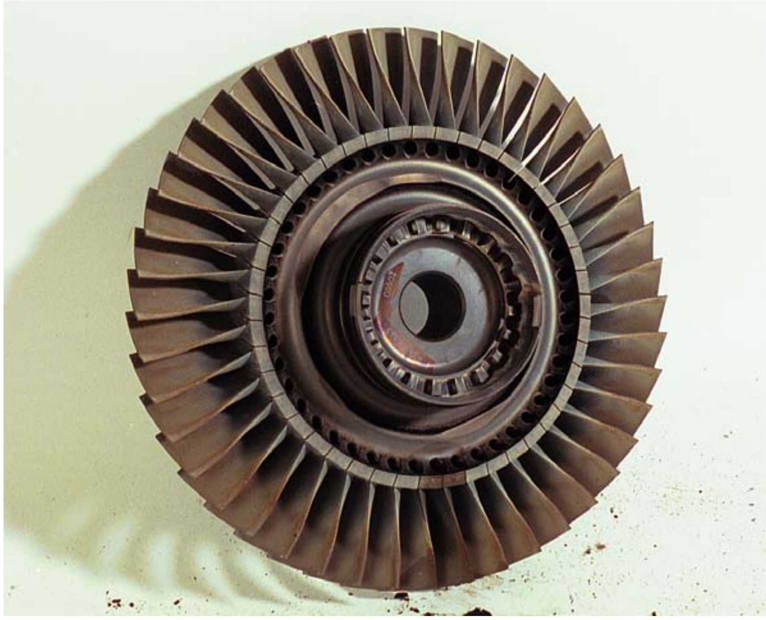

Photo 31: 3rd stage turbine wheel (Honeywell)

The turbine stators did not have any damage except for two 2nd stage stators, which were located about 120° apart and adjacent to the mount lugs, that had diagonal fractures across the airfoil. The turbine shrouds were all intact.

The tie shaft stretch force using standard overhaul tooling to loosen the retaining nut was 4,500 psi.

5.5 Bearings

The compressor bearing was not removed.

The turbine bearing housing was intact and was coated with coke[23] and soot. The quill shaft was jammed in the rear turbine bearing housing.

The turbine bearing was intact, wet with oil, free to rotate, and did not have any rotational damage. The turbine bearing cavity had a slight amount of residual oil.

5.6 Accessories

The fuel nozzles were all in place. The fuel manifolds were found on the engine and did not have any indication of an in-flight fire, but were broken in several places. The rubber sleeves over the manifolds were burned and melted from a ground fire.

The accessories were broken and no functional testing was accomplished.

5.7 Gearbox/load compressor

As received, the gearbox and load compressor was separated from the power section. The gearbox and load compressor housing did not have any holes or penetrations, but was sooted. The inlet guide vane actuator was detached. The generator housing was missing, exposing the generator windings. The gearbox was not disassembled. The oil cooler was missing. The filter housing was cracked and the filter bowl was separated from the housing. The gearbox sump was not drained or checked for residual oil.

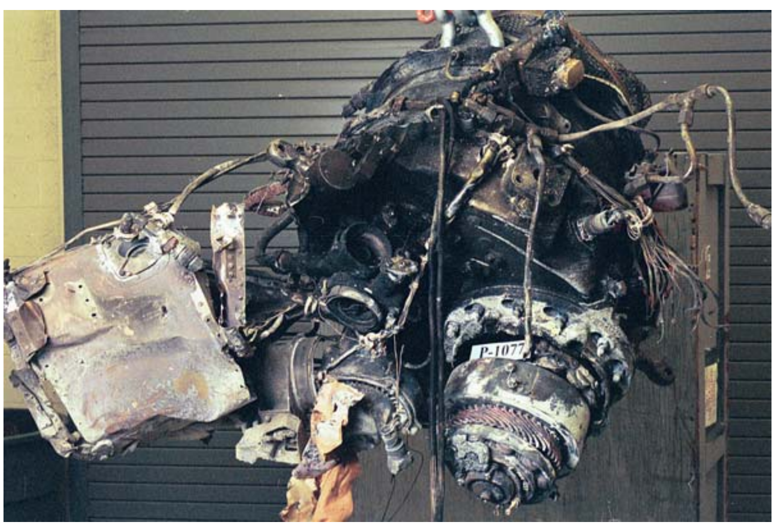

Photo 32: Gearbox and load compressor broken away from APU power section (Honeywell)

[23] Coke is the hard, crystalline residue of turbine engine oil.

6.0 Auxiliary power unit installation

According to Airbus, the APU is installed within a titanium-walled fireproof enclosure in the airplane's tailcone. The APU's enclosure is located between fuselage stations C95 and C101. The front of the APU enclosure is approximately 12.6 feet aft of the vertical stabilizer's rearmost mount lugs, which are located between fuselage stations C86 and C87. The APU has three mounting points with vibration isolators: left side of the compressor case, right side of the compressor case, and top of the turbine case. The APU is supported within its enclosure by seven struts: three to the left forward mount, two to the right forward mount, and two to the rear mount.

Intake air to the APU is provided through a fireproof duct from the bottom of the airplane to the APU's plenum chamber. The APU's exhaust gases are directed by an insulated tailpipe from the APU exhaust out through the aft end of the tailcone.

Bleed air from the APU that is used for air conditioning and engine starting exits the enclosure at about 2 o'clock, about 3-inches from the right side of the engine and 10-inches from the top. The bleed air duct runs forward from the APU enclosure along the right side of the fuselage about 10-inches from the skin, crosses over to the left side of the airplane over the horizontal stabilizer center box, runs along the left side of the fuselage about 10-inches from the skin up to the aft pressure bulkhead where it turns downward to run along the underside of the airplane on the left side. The bleed air duct within the tailcone is covered with a thermal insulation shield.

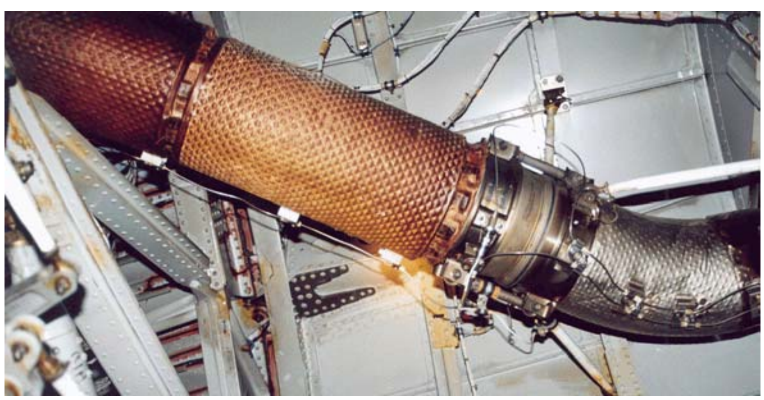

Photo 33: Example of APU bleed air duct exiting firewall in tailcone showing overheat detection wires in American Airlines A300 N70054

The APU's enclosure has two independent overheat or fire detection wires routed around the entire circumference of the enclosure. The APU bleed air duct has a single overheat detection wire along the full length of the bleed air duct. The overheat or fire detection wires in the APU enclosure will activate the warning system at 154°C. The overheat detection wire on the bleed air duct will activate the warning system at 124°C. An APU fire or overheat warning would illuminate the APU fire T-handle in the cockpit in the center of the overhead panel and show an APU fire warning message on the Electronic Centralized Aircraft Monitor (ECAM) panel. A bleed air duct overheat warning would be displayed with a warning light on the lower right side of the overhead panel and a duct overheat warning message on the ECAM panel.

The APU has a dedicated Halon fire bottle with a single discharge nozzle located at about 7 o'clock on the forward bulkhead to spray the agent into the APU enclosure for fire suppression. The fire extinguishing agent can be discharged into the APU enclosure with the APU fire handle in the cockpit. When the cockpit is not occupied, the fire extinguishing agent can be discharged from a switch on the nose wheel strut. To permit unsupervised ground operations, the A300-600 airplane's APU is equipped with an automatic fire extinguishing system. If a fire is detected by both wires, the system will automatically discharge the Halon fire extinguishing agent into the APU enclosure.

7.0 Airplane tailcone heat study

Sets of temperature tabs, which read from 120 to 180°F and 180 to 250°F,[24] were applied to several locations inside the tailcone of American Airlines A300-605R, N70054. The airplane operated on American's A300 routes for several months that included operations down into the Caribbean. The temperature tabs were examined on the evening of August 1, 2002, at JFK while N70054 was undergoing an overnight inspection.

Location of tabs	Maximum temperature
Top of frame C83 adjacent to fuselage skin	120°F
Fuselage skin on left side between frames C85 and C86, directly above the bleed air duct	160°F
Vertical stabilizer access cover between frames C85 and C86	120°F

[24] Temperature tabs have windows that are marked for specific temperatures. If the material in a particular window turns black, that indicates the tab was exposed to temperatures greater than the marked temperature. The range of two tabs that were installed on N70054 were 120 to 180°F, which had windows marked 120, 140, 160, and 180°F; and 180 to 250°F, which had windows marked 180, 200, 230, and 250°F.

Photo 34: Temp tabs on frame C83 showing exposure to120°F

Photo 35: Temp tab over bleed air duct on left side of fuselage showing exposure to 160°F

Photo 36: Temp tabs on vertical stabilizer access cover showing exposure to 120°F

8.0 Fuel testing

According to a Port Authority of New York & New Jersey Police report, American Airlines flight 587 was serviced with 8,513 gallons of jet fuel from Ogden Aviation's truck No. 612 before the airplane departed JFK. Shortly after the airplane crashed, two Port Authority detectives obtained a one gallon sample of jet fuel from truck No. 612 as well one gallon samples from Ogden tanks No. 314 and 315.[25] The police report further states that samples were flown by police helicopter to Newark International Airport and given to Dr. Nabil Mohtadi, Nobil Petro Testing, 86 Doremus Avenue, Newark, New Jersey, for an eight point fuel test and a full fuel test (Appendix 3).

The results of the eight-point tests on the fuel samples from truck No. 612 and tanks No. 314 and 315 were reported in Nobil Petroleum Testing Laboratory Analysis Reports No. 1101-173, 1101-174, and 1101-175, respectively, all dated November 12, 2001 (Appendix 4). The results of the eight-point tests were as follows:

Test	Specification	Truck 612	Tank 314	Tank 315
Gravity, API at 60°F	min 37/max 51	43.0	43.1	43.1
Flash point, Tag closed cup, °F	min 100	119	119	119
Copper corrosion, 2 hours at 212	max No. 1	1a	1a	1a
Freezing point, °C	min –40.0	-45.5	-46.5	-46.5
Distillation, IBP, °F		324	320	320
Rcvd, 10 percent, °F	max 400	356	356	356
Rcvd, 50 percent, °F		406	406	406
Rcvd, 90 percent, °F		168[26]	470	470
End point, °F	max 572	526	524	524
Recovery, vol pct		99.0	99.0	99.0
Residue	max 1.5	0.7	0.9	0.9
Color, Saybolt	min +16	+21	+21	+21
Water reaction, separation rating		1	1	1
Interface rating	max 1b	1	1	1
Change in volume, ml		0	0	0
Microseparometer index (MSEP)	min 85	92	91	91

The results of the full tests on the fuel samples from truck No. 612 and tanks No. 314 and 315 were reported in Nobil Petroleum Testing Laboratory Analysis Reports No. 1101-173, 1101-174, and 1101-175, respectively, all dated November 12, 2001 (Appendix 6). The results of the full tests were as follows:

[25] According to the Port Authority Police, tanks No. 314 and 315 were sampled because they were the tanks that supplied the fuel to truck No. 612 that serviced flight 587 before it departed JFK.

[26] The recovered 90 percent value of 168°F was found to be a typographical error. Nobil Petroleum Testing provided the Safety Board with the worksheet from when the test was accomplished and a corrected laboratory analysis report that indicated the correct value for 90 percent recovered should have been 468 (Appendix 5).

Test	Specification	Truck 612	Tank 314	Tank 315
Gravity, API at 60°F	min 37/max 51	43.0	42.9	43.1
Doctor test	sweet	Negative	Negative	Negative
Mercaptan sulfur, wt pct	max 0.003	Cancelled	Cancelled	Cancelled
Sulfur, X-ray, wt pct	max 0.30	0.090	0.093	0.085
Aromatics (FIA), vol pct	max 25	17.1	16.9	17.4
Distillation, IBP, °F		320	320	324
Rcvd, 10 percent, °F	max 400	358	356	356
Rcvd, 20 percent, °F		374	370	372
Rcvd, 50 percent, °F		410	406	406
Rcvd, 90 percent, °F		476	470	468
End point, °F	max 572	530	524	526
Recovery, vol pct		99.0	99.0	99.0
Residue	max 1.5	0.8	0.9	0.7
Loss, vol pct	max 1.5	0.2	0.1	0.3
Acidity, total	max 0.10	0.010	0.017	0.014
Flash point, Tag closed cup, °F	min 100	119	119	119
Freezing point, °C	min –40	-46.5	-45.5	-46.5
Viscosity, KIN CST at -20°C/-4°F	max 8.0	5.166	5.224	4.907
Heat of combustion	min 42.8	43.185	43.168	43.173
Smoke point, mm	min 18	21	21	20
Naphthalenes, vol pct	max 3.0	1.32	1.30	1.38
Copper corrosion, 2 hrs at 212	max No. 1	1a	1a	1a
Gum, existent (unwashed)	max 7	1	1	2
Water reaction: separation rating		1	1	1
Interface rating	max 1b	1	1	1
Volume change		0	0	0
Electrical conductivity, pS.m		11@72°F	5@72°F	8@72°F
Thermal stability		Pass	Pass	Pass
Pressure drop, mm	max 25	0	0	0
Tube deposit rating	max 3	No. 1	No.1	No. 1

9.0 Metallurgical testing

During the teardown of the two engines, the left and right hydraulic pumps from each engine were found. The drive shafts from each of the four hydraulic pumps were removed and shipped to the Safety Board's Materials Laboratory for examination. The left engine's hydraulic pumps drive shafts were identified as 1A and 1B and the right engine's were identified as 2A and 2B.

Shaft piece 1A was fractured on the pump side of the "O" ring seal. According to the Materials Laboratory's report, the metallurgical examination revealed that various portions of the fracture were on flat planes and that all portions of the fracture contained smear marks on linear fracture features that indicated the fracture occurred under direct shear loading.

Additionally, the splines on the gearbox end contained an imprint pattern indicating excessive bending loading.

Shaft piece 1B was not fractured and there was no evidence of any torsional damage. One of the protruding ends of the roll pin though the pump end of the shaft was sheared off in the longitudinal direction.

Shaft 2A was fractured through the reduced section shear point adjacent to the splines on the pump end of the shaft. According to the report, most of the fracture was on a slant 45° plane that is typical of an overstress bending fracture. One side of the fracture was deformed slightly outward indicating that some amount of torsional loading may have been present when the fracture occurred.

Shaft 2B was fractured adjacent to the splines on the gearbox end. According to the report, the fractures were typical of an overstress bending fracture. One of the protruding ends of the roll pin through the splines on the pump end of the shaft was bent forward.

For further details, refer to Materials Laboratory Factual Report No. 02-046, dated May 17, 2002 (Appendix 7).

Gordon J. Hookey
Powerplants Group Chairman

APPENDIX

1. Left and right engines' engine condition monitoring data

2. Left and right engines' takeoff performance data

3. Port Authority of New York and New Jersey Police report on obtaining the fuel samples and transferring them to Nobil Petro Testing

4. Nobil Petro Testing reports on the 8-point tests accomplished on the fuel samples

5. Nobil Petro Testing laboratory worksheet for sample from tank No. 314 and corrected laboratory analysis report showing correct 90 percent recovered temperature

6. Nobil Petro Testing reports on the full tests accomplished on the fuel samples

7. NTSB Materials Laboratory Factual Report No. 02-046, dated May 17, 2002