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

May 17, 2000

POWERPLANTS GROUP CHAIRMAN'S FACTUAL REPORT

NTSB ID No.: DCA00MA006

A: ACCIDENT

Location: Nantucket, Massachusetts

Date: October 31, 1999

Time: 0150 eastern standard time (EST)

Aircraft: Boeing 767-366ER, SU-GAP, EgyptAir flight 990

B: POWERPLANTS GROUP

Group Chairman: Gordon J. Hookey
National Transportation Safety Board
Washington, D.C.

Member: Abdel Maseeh Adly Fouad
Egyptian Civil Aviation Authority
Cairo, Egypt

Member: Peter P. Maloney
Boeing
Seattle, Washington

Member: Michael S. Bartron
Pratt & Whitney
East Hartford, Connecticut

C: SUMMARY

On October 31, 1999, about 0150 EST, EgyptAir flight 990, a Boeing 767-366ER airplane, registered in Egypt as SU-GAP, crashed into the Atlantic Ocean about 60 miles south of Nantucket, Massachusetts. The airplane was operating on an instrument flight rules flight plan under the provisions of Egyptian Civil Aviation Regulation Part 121 and United States Title 14 Code of Federal Regulations (CFR) Part 129 as a scheduled international flight from John F. Kennedy International Airport (JFK), New York, New York, to Cairo International Airport, Cairo, Egypt. The airplane was destroyed by impact forces and the 4 pilots, 10 flight attendants, and 203 passengers were killed.

The left and right engines and the auxiliary power unit (APU) were recovered from the Atlantic Ocean by the Smit Pioneer and Carolyn Chouest under the direction of the U.S. Navy Supervisor of Salvage during two separate recovery efforts. In December 1999, the Smit Pioneer recovered portions of the right engine’s fan, low pressure compressor (LPC), high pressure compressor (HPC), and low pressure turbine (LPT). In March 2000, the Carolyn Chouest recovered the left engine and additional pieces of the right engine that included part of the HPC, the diffuser, and the high pressure turbine (HPT).

The left engine was intact from the fan to the turbine exhaust case. The examination of the left engine showed the engine had little, if any, rotation at the time of impact and there was no indication of any preimpact damage or fire. The engine was not disassembled.

The right engine was severely broken up. About 80 percent of the engine was recovered. The examination of the right engine showed the engine had little, if any, rotation at the time of impact and there was no indication of any preimpact damage or fire. The engine was not disassembled.

The right engine’s lubrication and scavenge oil pump was disassembled and examined at the pump’s manufacturer, Hamilton Sunstrand. The lubrication part of the pump was missing. The examination of the scavenge part of the pump did not reveal any indication of a preimpact malfunction. A metallic strand that was found on one of the magnetic chip collectors was metallurgically analyzed and determined to be iron.

Fuel samples were collected from the trucks that serviced the airplane at Los Angeles and New York before the airplane departed. The samples were tested at separate petrochemical laboratories and were found to conform to the specifications for flash point and conductivity.

D: DETAILS OF INVESTIGATION

1.0 Engine information

The engines installed on SU-GAP were Pratt & Whitney (P&W) PW4060 turbofans. The PW4060 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, and a 4-stage LPT. The engine has a takeoff thrust rating of 60,000 pounds, flat-rated to 92°F.[1]

According to EgyptAir's maintenance records, the airplane's No. 1 (left) engine was serial number (SN) 724126 and the No. 2 (right) engine was SN 724127. For further information regarding the engines' maintenance history, refer to the Maintenance Records Group Chairman's Factual Report.

2.0 Engine examination

2.1 Left engine, SN 724126

The left engine was identified by the SN that was embossed on the oil tank's data plate. The engine was intact from the fan to the turbine exhaust case. The engine did not have any indication of an uncontainment, case rupture, or fire. The left engine was recovered in March 2000, by the Carolyn Chouest, during a second recovery effort at the crash site. The left engine was examined by members of the Powerplants Group on April 10 and 11, 2000, at the former Quonset Point Naval Air Station, Davisville, Rhode Island. Before the engine had been recovered, it had been examined by the Powerplants Group on a video tape that was made by the U.S. Navy submarine, NR-1.

2.1.1 Inlet

The inlet lip was intact from about 2:30 to 10:30 o'clock[2] and the remainder of the inlet lip was not recovered. The recovered portion of the inlet lip was separated from the inlet duct and cowling. Portions of the inlet cowling up to 10-inches wide remained attached to the rear of the inlet lip from about 3:30 to 7:30 o'clock. The inlet lip was buckled at the fractured ends and at 6:30 o'clock and was crushed axially and split circumferentially at 8:00 o'clock.

2.1.2 Fan

The fan containment case was intact, but was pushed inward from about 3:00 to 10:30 o'clock. All of the fan rubstrip segments were in place. The fan containment case did not have any penetrations or bulges. The fan blade rubstrip had 10 and 5 airfoil-shaped imprints, which were similar in size and spacing to the fan blades, from 11:00 to 2:00 and 5:30 to 7:00 o'clock, respectively. The fan blade rubstrip did not have any circumferential rub marks or gouging. The electronic engine control (EEC) was missing from the fan case.

The inlet cone front and rear segments were still in place. The inlet cone rear segment was crushed inward from about 12:00 to 2:30 o'clock.

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

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

There were 37 of the 38 fan blades in the fan disk. The blade lock ring was broken where the fan blade was missing, but the pedestals on either side of the blade slot of the missing fan blade were intact. The first and third fan blades clockwise from the missing fan blade were displaced forward out of the blade slots ½- and 2 ¾-inches, respectively. There were 10 fan blades between 11 and 3 o'clock that were fractured transversely across the airfoils from 3- to 11-inches above the blade root platform. There was 1 fan blade at 3 o'clock and 12 fan blades from 9 to 12 o'clock that were bent forward and opposite the direction of rotation[3] and 1 fan blade at 10 o'clock that was bent rearward. All of the other fan blades were full length and straight.

2.1.3 Low pressure compressor

The fan exit fairing and the LPC stator shrouds were crushed inward at about 1:30 and the LPC was displaced to the left. The stage 1.6, 2, 3, and 4 stator shrouds were separated axially from 2:00 to 5:30 o'clock exposing the blades and vanes, which were randomly bent in both the clockwise or counterclockwise directions. The stage 1.6 and 2 blades that were visible on the left side of the engine were straight and did not have any rotational damage. The LPC rotor was displaced from the fan disk laterally to the left and all of the visible LPC rotor front flange bolts were sheared off.

The intermediate case was intact, but the case wall was ruptured inward between the No. 1 and 2 struts.[4] The fan exit struts and case were still attached to the intermediate case from about 5 to 7 o'clock. An 80-inch long section of the fan exit case that was separated from the engine was also recovered.

2.1.4 High pressure compressor

The HPC front case upper and lower halves were still joined at the horizontal flanges, but the cases' front and rear flanges were separated from the intermediate and HPC rear cases, respectively, and all of the visible flange bolts were sheared off. The HPC front case was displaced laterally to the left and the variable stator vane (VSV) synchronizing rings between 1 and 2 o'clock were crushed down against the VSV lever arms. The inlet guide vane (IGV) VSV synchronizing ring[5] was separated from the VSV pins and bent down from 3 to 5 o'clock. The stage 8 compressor vanes and blades that were visible through the separation between the front and rear compressor cases were straight and did not have any rotational damage. The HPC front and rear cases did not have any holes, sooting, or fire damage.

2.1.5 Diffuser/combustor

The diffuser case was intact, but was pushed inward at 12:00, 12:30, and 1:00 o'clock. The diffuser case was split open adjacent to the boss where it was pushed inward at

[3] The PW4000 engine's high and low pressure rotors rotate clockwise.
[4] The struts are numbered in a clockwise pattern as viewed from the rear with the strut at 12:00 o'clock being identified as No. 1.
[5] The IGVs are at the front of the HPC directly in front of the 5th stage compressor rotor.

12:00 o'clock. The visible portions of the diffuser case did not have any indication of thermal distress.

2.1.6 High pressure turbine

The HPT case was intact. The HPT case did not have any holes, sooting, or fire damage.

The rectangular-shaped and circular-shaped turbine cooling air tubes along the HPT-LPT case flange were crushed against the rear side of the flange and against the HPT turbine case, respectively, from 1 to 3 o'clock. The rectangular-shaped tube conformed to the shape of the bolts on the rear of the flange. The active clearance control (ACC)[6] tubes on the upper part of the engine were missing. The ACC tubes on the bottom of the engine were in place and did not have any metal spray or fire damage.

2.1.7 Low pressure turbine

The LPT case was intact. The LPT case was separated from the HPT case and all of the visible flange bolts were broken. The rear of the LPT case was pushed inward from 1 to 2 o'clock. The LPT case did not have any holes, sooting, or fire damage.

All of the stage 5 and 6 turbine blades between 12 and 3 o'clock were bent or broken. The stage 6 turbine blades from 3 to 12 o'clock were all full length and did not have any metal spray material on the airfoil surfaces. The visible LPT airfoil surfaces did not have any apparent thermal distress.

2.1.8 Turbine exhaust

The turbine exhaust case was in place on the LPT case, but was crushed inward from 11:30 to 3:30 o'clock. The exhaust cone was missing. The rear bearing cover was still in place and did not have any thermal distress.

2.1.9 Accessories

The accessories were still attached to the bottom and left side of the engine. The integrated drive generator was separated from the gearbox.

2.1.10 Thrust reverser

The two complete thrust reverser actuators that were recovered, one that was attached to a thrust reverser cowl former and one that was loose, were in the stowed position.

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

2.2 Right engine, SN 724127

The right engine was identified from the part number (PN) and SN, 50B403 and R88887, respectively, that were found on the rear face of the 3rd stage compressor rotor's blade lugs that matched the PN and SN listed for the 3rd stage compressor rotor on the No. 2 engine's maintenance records provided by EgyptAir (Appendix 2). The initial recovered pieces of the right engine were examined by the Powerplants Group at Quonset Point on January 5 through 13, 2000. The engine parts that were recovered during the second recovery effort in March 2000, were examined by members of the Powerplants Group on April 10 and 11, 2000.

2.2.1 Fan

The complete fan containment case was recovered, but it was split axially at about 1 o'clock and both ends were twisted back onto itself. The fan containment case did not have any penetrations or bulges. All of the fan rub strip segments were in place except where the case was axially split. The fan rub strip did not have any circumferential gouging except for six airfoil-shaped imprints, which were similar in size and spacing to the fan blades, that were centered at 6 o'clock. The gouges in the fan rub strip were approximately ½-inch longer than the fan blade tip chord length. The EEC was missing from the fan case.

The fan disk was not recovered.

There were 12 fan blade roots recovered that had the airfoils fractured transversely at various lengths from adjacent to the blade root platform up to 7-inches above the blade root platform. The seven fan blade roots with airfoil sections that were greater than 2 inches long all had the leading edge bent towards the concave side. There were 28 pieces of fan blade airfoils recovered that ranged in size from a mid span shroud to a 22-inch long section of airfoil that included the tip and mid span shroud. The two longest pieces of airfoil, a 21-inch long piece and the 22-inch long piece were bent along the transverse axis below the mid span shroud opposite the direction of rotation and towards the direction of rotation, respectively. There were eight airfoil pieces that had the leading edge bent towards the concave side and two other pieces of airfoil had the fan blade tip bent towards the convex side. There were six fan blade pieces that had the mid span shroud. The remaining pieces of airfoil were essentially straight. The recovered fan blade tips did not have any rub marks.

2.2.2 Intermediate case

The intermediate case bearing support structure was broken into several pieces, of which only the No. 1, 1.5, and 2 bearing support flanges were recovered. A 21-inch long by 2-inch wide section of the No. 1 bearing support remained attached to the intermediate case bearing support structure. The No. 1.5 bearing support was crushed axially between 4 and 8 o'clock. The front half of the No. 1.5 bearing support was missing from 9 to 7 o'clock. The No. 1.5 bearing support had an axial fracture at 11:30 o'clock. The No. 2 bearing support mount ring was broken into two pieces with fractures at 12 and 2 o'clock. All of the core struts and bypass struts were broken off of the intermediate case. Two bypass struts were recovered: one that was bent at the inner end and had the leading edge crushed axially and the other was bent at the outer end.

The top of the bevel drive gear had two 180° circumferential fractures in the inner and outer diameter of the gear web. The bevel drive gear teeth did not have any apparent wear distress. The bevel drive gear had four locations where there were three adjacent gear teeth chipped.

2.2.3 Low pressure compressor

The LPC/LPT coupling was broken circumferentially aft of the No. 1 bearing and axially from the front through the anti-icing air holes. About 50 percent of the circumference of the front of the LPC/LPT coupling was recovered. A 14-inch long by 8-inch wide axial section of the LPC/LPT coupling from just aft of the forward section bell was recovered.

The four-stage LPC rotor drum was broken up into the individual stages that were broken up into several pieces. A 16-inch long section of the LPC rotor drum front flange had the imprint of four fan blade roots on the front face of the flange.

There were two pieces of the LPC rotor stage 1.6 rim recovered that were 33- and 28-inches long and that had 26 and 22 blade slots, respectively, out of 94 blade slots. There were no blades found in the stage 1.6 blade slots. There were 23 1.6 stage compressor blades recovered that ranged in length from being fractured transversely across the airfoil adjacent to the blade root platform to full length (about 5 ¾-inches long).

There were four pieces of the LPC rotor stage 2 rim recovered that were 14 ½-, 12 ½-, 10-, and 10- inches long that had 10, 9, 7, and 7 blade slots, respectively, out of 88 blade slots. There were no blades found in the stage 2 blade slots. There were 15 2nd stage compressor blade pieces recovered that ranged in length from being fractured transversely across the airfoil adjacent to the blade root platform to full length (about 5-inches long).

There were four pieces of the LPC rotor stage 3 rim recovered that were 40-, 38-, 12-, and 13-inches long that had 24, 22, 7, and 8 blade slots, respectively, out of 72 blade slots. The four pieces of the stage 3 rim had six, two, two, and one blades, respectively, that were fractured transversely across the airfoil adjacent to the blade root platform. In addition, the 40-inch long piece of the stage 3 rim had one blade that was 3 ½-inches long that was bent opposite the direction of rotation. There were 17 3rd stage compressor blade pieces recovered that ranged in length from ½ inch above the blade root platform to full length (about 4 ¼-inches long).

There were four pieces of the LPC rotor stage 4 rim recovered that were 21-, 30-, 13 ½-, and 10-inches long that had 12, 18, 8, and 6 blade slots, respectively, out of 66 blade slots, along with two pieces of the stage 4 rotor web that were 4 ½- and 15-inches long. The respective pieces of the stage 4 rotor rim had 8, 6, 7, and 2 blades that were fractured transversely across the airfoil adjacent to the blade root platform. The 21-inch long piece of stage 4 rotor rim had two blades that were fractured across the airfoil 1-inch above the blade root platform and the 10-inch long piece had one full length blade (about 4 ¾-inches long), all of which were bent opposite the direction of rotation. There were three other 4th stage compressor blade pieces recovered: two blades that were full length and bent opposite the direction of rotation and one blade that was 1 ½-inches long and bent towards the direction of rotation.

The 1.6, 2nd, 3rd, and 4th stage compressor blades had nicks and dents on the leading and trailing edges of the airfoils and were either straight or were bent towards or opposite the direction of rotation.

There were 6 of 88 1st stage, 8 of 96 1.6 stage, 15 of 84 2nd stage, 5 of 78 3rd stage, and 9 of 64 4th stage compressor vanes recovered. All of the vanes, which ranged in length from 1-inch long to full length, had nicks and dents on the leading and trailing edges and were either straight or were bent towards the direction of rotation or opposite the direction of rotation. There were two 11-inch long pieces of the 2nd and 3rd stage compressor stator shroud with the adjacent blade rub strip that had the imprints of blade tips into the rubber.

2.2.4 High pressure compressor

There were 19 pieces of the forward HPC case upper and lower halves, which included the variable stator vane unison rings, recovered. The upper half of the forward HPC case was missing all of the inlet guide vane (IGV) stage stator ring and all of the 5th stage stator ring except for a 12-inch long section at 12 o'clock. The lower half of the forward HPC case was missing from 3 to 7 o'clock and all of the IGV and 5th stage stator rings. The forward edges of the forward HPC case upper and lower halves were bent radially outward. The 6th and 7th stage variable stator vanes that remained in the forward HPC case upper and lower halves were bent rearward and did not have any impact damage on the leading and trailing edges. There were seven pieces of the HPC IGV stator ring recovered that accounted for 34 of the 76 IGV positions and nine pieces of the 5th stage compressor stator ring recovered that accounted for 35 of the 48 vanes. The HPC rear case was intact except for a 2-inch by 2-inch hole at 12 o'clock about 3 -inches aft of the front flange and a 3 ¾-inch wide section that was missing from the front flange from 3 to 6 o'clock. The recovered HPC front and rear cases did not have any holes, sooting, or fire damage.

The HPC rotor was axially compressed such that the face of the stage 5 compressor disk was 4 ½-inches forward of the HPC rear case front flange. The stage 5 compressor disk was intact, but the integral spacer arm was axially compressed. There were five stage 5 compressor blades missing from the blade slots and all of the other stage 5 compressor blades were fractured transversely across the airfoil adjacent to the blade root platform. The stage 6 compressor rotor rim was intact, but was bent up. There were at least two non-adjacent blade pedestals that were fractured adjacent to the slot bottom. The fracture surfaces of the missing pedestals had a grainy appearance. The stage 7 and 8 compressor rotor rims were fractured axially at 11 o'clock and telescoped over the stage 6 compressor rotor. The fracture surfaces on the ends of the stage 7 and 8 compressor rotors were at 45° angles and were grainy.

There were several stage 6 compressor blades that were missing from the blade slots; two that were bent opposite the direction of rotation; although there were no spanwise marks on the concave surfaces, and all of the other stage 6 compressor blades were fractured transversely across the airfoil adjacent to the blade root platform. There were 10 stage 7 and 1 stage 8 compressor blades that were full length, but were bent slightly opposite the direction of rotation and did not have any spanwise marks on the concave airfoil surfaces. The stage 7 and 8 rotors had several blades missing from the blade slots and all of the other blades were fractured transversely across the airfoils adjacent to the blade root platforms.

Out of the 40 5th stage compressor blades, 12 blades that were full length and one blade root with the airfoil fractured transversely adjacent to the blade root platform were recovered. All of the 5th stage compressor blade airfoils were straight except for one that was bent opposite the direction of rotation. Out of the 51 6th stage compressor blades, 1 full length blade, 1 blade root, and 3 pieces of blade were recovered. The full length 6th stage compressor blade was bent opposite the direction of rotation. Out of the 56 7th stage compressor blades, 1 piece of airfoil that was straight and 1 piece of airfoil that was bent opposite the direction of rotation were recovered. Out of the 53 8th stage compressor blades, 9 pieces of airfoil were recovered. All of the recovered HPC airfoils had nicks and dents on the leading and trailing edges.

There were 13 HPC IGV pieces recovered that ranged in length from 1 ½- to 4 ½-inches long. All of the recovered HPC IGV airfoils were straight except for one fractured vane that was 2 ½-inches long and had the fractured end bent towards the concave surface. There were 13 5th stage compressor stator vane pieces recovered. Five of the 5th stage compressor vanes were full length (about 4 ½-inches long) and straight. Out of the other eight recovered 5th stage compressor stators: one was full length and bent to the convex side, one was full length and bent to the concave side, three were pieces of airfoil, and three were blade roots. There were 25 6th stage compressor vane pieces recovered. Eight of the 6th stage compressor vanes were bent across the outer end of the airfoil so the outer puck was towards the concave side similar to those 6th stage vanes that remained in the forward HPC case. Of the remaining recovered 6th stage vane pieces: three were straight, seven were pieces of airfoil, and eight were pieces of the inner and outer pucks. There were six 7th stage compressor vane pieces recovered. Four of the recovered 7th stage compressor vanes were full length and two were parts of the airfoil. The recovered full length 7th stage compressor vane airfoils were bent so the outer puck was towards the concave side of the airfoil similar to the vanes that remained in the forward HPC case. All of the recovered HPC compressor stator vanes had nicks and dents on the leading and trailing edges.

2.2.5 Diffuser/combustor

The diffuser case had a 21-inch long circumferential by 6-inch wide axial hole between 11 and 1 o'clock and two 11-inch by 11-inch dents at 6 and 7 o'clock, all about 13-inches aft of the front flange. The diffuser case did not have any apparent thermal distress. The visible portions of the combustor liner and fuel nozzles did not have any apparent thermal distress.

2.2.6 High pressure turbine

The HPT case was intact. The stage 1 turbine blades were all in place and intact. The stage 1 turbine blades did not have any impact marks on the convex surfaces and there were no rub marks on the blade tips. The stage 2 turbine blades were all in place, but all were fractured across the airfoil at varying lengths from adjacent to the blade root platform up to 3 ½-inchs above the blade root platform. The stage 2 vanes had dents on the trailing edges, but there were no circumferential marks adjacent to the dents. The visible HPT airfoil surfaces did not have any apparent thermal distress.

2.2.7 Low pressure turbine

There were eight LPT outer transition duct panel pieces recovered. The panel surfaces did not have any metal spray.

The LPT case was fractured 360° around between 2- and 5-inches aft of the front flange, just in front of the 5th stage turbine vanes, and axially into several pieces of which six were recovered. The front flange of the LPT case remained attached to the diffuser case. The recovered pieces of the case did not have any holes and the exterior of the case and a small piece of the ACC tubing did not have any metal spray or fire damage. The honeycomb rubstrips did not have any circumferential scoring.

The LPT shaft was recovered with and remained in the HPC to HPT assembly. The LPT shaft was full length, but appeared to be bent at the forward end.

The 3rd, 4th, 5th, and 6th stage turbine disks were intact and still joined together. The 5th stage turbine disk-LPT shaft mating flange was not deformed, but the disk flange snap had an approximately 1-inch long section that was bent inward slightly. One of the LPT shaft flange bolts remained in the disk and was sheared off flush to the mating surface. The rotating airseals did not have any circumferential marks and the knife edge seals were full height except for several random locations where there were short sections bent forward. All of the blades that remained in the disks were fractured transversely across the airfoil adjacent to the blade root platform. The blade root platforms did not have any circumferential marks. The blades that remained in the blade slots in the LPT rotor were 115 of 128 3rd stage, 127 of 130 4th stage, 117 of 126 5th stage, and 56 of 108 6th stage. There were 96 airfoils of the 476 3rd, 4th, and 5th stage turbine blades and 71 6th stage turbine blade airfoils recovered.[7] Almost all of the LPT blade airfoils were straight and none of the airfoils had any metal spray material on the airfoil surface.

There were 6 complete and 15 partial (outer foot, but no airfoils) out of 42 3rd stage turbine vane clusters, 3 complete and 27 partial out of 44 4th stage turbine vane clusters, 2 complete and 14 partial out 38 5th stage turbine vane clusters, and 1 complete and 14 partial out of 36 6th stage turbine vane clusters recovered. The vane airfoils were battered on the

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[7] The PW4000 6th stage turbine blade is hollow and is readily identifiable from the other LPT blades that have a solid airfoil.

trailing edges and were either straight or bent opposite or towards the direction of rotation, but none of the airfoil surfaces had any metal spray material.

2.2.8 Exhaust

The turbine exhaust case was missing the outer case wall and two struts from 11 to 2 o'clock. The outer case wall was bent outward and rearward from 3 to 6 o'clock where the five struts that were, going in a counterclockwise direction from 2 o'clock, had one separated at the outer end, two separated at the inner end, and two missing completely. The remainder of the struts were buckled towards the convex side and the case's outer wall was rotated about 10° counterclockwise. The turbine exhaust case mount rail had the mount bolt at 11 o'clock still in place and the mount bolt hole at 1 o'clock was torn open to the mount rail edge. The turbine exhaust case's and turbine exhaust duct's gaspath surfaces did not have any apparent scoring from debris. The No. 4 bearing compartment cover was ruptured and the edges of the hole were petaled rearward.

2.2.9 Bearings

The No. 1 bearing ball cage was battered and bent and was split axially through one of the ball pockets. There was one ball pocket that was cracked through on one side. The interior surfaces of the ball pockets did not have any rotational distress.

The No. 4 bearing outer race was intact, and was cocked and jammed into the No. 4 bearing housing in the turbine exhaust case. The No. 4 bearing outer race bearing raceway did not have any rotational distress.

2.2.10 Gearbox/accessories

None of the gearbox housing was recovered except for two fragments that were still attached to the Integrated Drive Generator Quick Attach Detach (IDG QAD) ring and the lubrication and scavenge pump. There were no internal gearbox components recovered.

The angle gear box bevel drive gear with a 7 ½-inch long section of shaft was recovered. The roller and ball bearing cages were intact, but all of the rolling elements were missing from both bearings.

The lubrication and scavenge oil pump was recovered and had one end of the housing broken open. The fuel/oil cooler and bypass valve end cover at the oil-in end with the mount bracket was recovered. The air starter, engine-driven hydraulic pump, and IDG QAD ring were also recovered.

The oil pressure transmitter, 15th stage manifold, pneumatic air precooler, starter air valve, oil filter cover, thrust reverser isolation valve, and two 2.9 bleed valves were recovered.

None of the engine accessories had any sooting or fire damage.

2.3 Auxiliary power unit

The exterior of the APU did not have any holes or fire damage. The compressor blades in the inlet of the APU were not damaged. The APU exhaust duct was crushed inward. The APU load compressor inlet was crushed axially. The APU was not disassembled.

2.4 Cowling

Several pieces of engine cowling were recovered. All of the cowling pieces were tagged indicating that they had been recovered as floating debris, except for a piece of the core cowl that was recovered by the U.S. Navy SupSalv operation. It was not possible to identify on which engine the recovered floating pieces of cowling were installed. None of the recovered pieces of cowl had any sooting or fire damage.

There were two pieces of the inlet cowl recovered that were an “L”-shaped piece that was 37-inches long (circumferential) by 20-inches wide (axial)[8] and one that was 24-inches long by 32-inches wide.

There was one 27-inch long by 19-inch wide piece of the fan cowl recovered that had a part of the EgyptAir logo on the exterior surface.

There were eight pieces of the thrust reverser cowl recovered. The recovered pieces of the thrust reverser cowl were a 45-inch long by 23-inch wide piece that had a pressure relief door still in place, a 44-inch long by 17-inch wide piece, a 21-inch long by 21-inch wide piece, a 28-inch long by 32-inch wide piece that was marked “Thrust reverser actuator access,” a 26-inch long by 25-inch wide piece, a 59-inch long by 44-inch piece that had an access hole for a door that was missing, a 78-inch long by 49-inch wide piece, and a 32-inch long by 14-inch wide piece.

There was one 14-inch long by 9-inch wide piece of the core cowl that was recovered from the bottom during the SupSalv operation.

The left and right halves of the thrust reverser cowling latches, which are at the bottom of the cowling, were recovered from the bottom during the SupSalv operation.

There was one 68-inch long by 20-inch wide piece of the thrust reverser outer duct support ring that included a blocker door that was recovered. It was not possible to determine the position of the thrust reverser at impact.

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[8] Length will be in the circumferential direction and width in the axial direction.

3.0 Lubrication and scavenge pump

3.1 Lubrication and scavenge pump examination

The lubrication and scavenge pump was shipped to Hamilton Sunstrand, the pumps manufacturer, Rockford, Illinois, and was examined in the presence of the Powerplants Group that consisted of the Group Chairman and P&W on January 28, 2000.[9]

The gearbox mounting flange was attached to a piece of the gearbox. The lubrication stage of the pump and the No. 3 bearing scavenge elements were missing. The lube inlet core was missing. The lubrication and scavenge pump gearbox drive gear was intact. The pump housing did not have the Service Bulletin (SB) 5008437-79-4 jackscrew that were added to facilitate removal of the pump from the gearbox. The SB was issued on September 15, 1994. According to Mr. Ryszaw Wlaznik, Hamilton Sunstrand Engineering, all PW4000 lubrication and scavenge pumps that were overhauled at Hamilton Sunstrand after the SB was issued would have had the jackscrew holes incorporated.

The No. 1, 1 ½, and 2 bearing and angle gearbox magnetic chip collectors were missing. The No. 3 and 4 bearing magnetic chip collectors remained. The No. 3 magnetic chip collector had a strand of metallic material that was collected for further examination (Refer to Section 3.2 for the metallurgical examination of this particle). The No. 4 magnetic chip collector did not have any apparent metallic material.

The gear stack cover was broken off and the stationary bearings and the gear stack, as viewed through the inlet ports, were shifted forward about ¼-inch. The pump housing had to be cut axially to facilitate removal of the pump gear stack. The scavenge pump gear elements for the gearbox, angle gearbox, No. 1, 1 ½, and 2 bearings, and No. 4 bearings did not have any apparent damage or circumferential marks on the gear teeth ends. All of the pump bearings did not have any damage. The pump inner diameter wall for the gearbox pump elements had fine circumferential scoring that was typical for service run units according to Mr. Wlaznik. The pump inner diameter wall for the No. 3 bearing had circumferential scoring that appeared to be greater than normal for service run units according to Mr. Wlaznik. The pump inner diameter walls for the other scavenge element gears did not have any circumferential scoring.

3.2 Metallurgical examination of metallic particle

The strand of metallic material that was collected from the lubrication and scavenge pump No. 3 magnetic chip collector was submitted to the Safety Board's materials laboratory for metallurgical analysis. The strand was measured using the scanning electron microscope and determined to be 0.03 inches long and have a diameter that varied from 0.0003 to 0.0008 inches. The X-ray energy dispersive spectroscopic analysis of the strand produced a spectrum containing iron and oxygen as the major peaks. For further details, refer to Materials Laboratory Factual Report No. 00-030, dated February 2, 2000 (Appendix 3).

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[9] All of the Group members were advised of the examination of the lubrication and scavenge pump, but the other Group members declined to participate in the examination.

4.0 Oil pressure transmitter

The oil pressure transmitter was shipped to Meggitt Avionics, the transmitter's manufacturer, Manchester, New Hampshire, and was tested in the presence of the Powerplants Group Chairman on January 31, 2000.[10]

The oil pressure transmitter, PN 4118-01124, SN 891402, was intact. When the inlet tube fitting was removed from the housing, oil drained from the port. When the vent tube fitting was removed from the housing, water drained from the port. The vent screen had dried mud on the screen. The electrical connector pins had corrosion.

The oil pressure transmitter was connected to a transmitter test station, model number T520603, SN 4, calibration tracking number 06245, which had a sticker indicating it was last calibrated on 9/15/99, and that the next calibration was due on 9/30/00. As soon as 28 volts direct current was applied to the transmitter, the gage on the test station indicated that the oil pressure transmitter was shorted out.

The oil pressure transmitter was connected to an Associated Research AC [alternating current] Hypot [high electrical potential] tester, model number 412, calibration tracking number L3549, which had a sticker indicating that it was last calibrated on 9/8/99, and that the next calibration was due on 9/30/00. When approximately 200 volts AC was applied to the transmitter, a light on the tester illuminated indicating that there was internal electrical leakage.

5.0 Fuel testing

5.1 JFK fuel

On October 31, 1999, following the crash of the airplane, an NTSB Northeast regional office staff member traveled to JFK and obtained a sample of jet fuel from the truck that serviced flight 990 before the airplane departed to Cairo. On January 25, 2000, the fuel was tested by Saybolt, a petrochemical laboratory in Kenilworth, New Jersey, for flash point, specific gravity, and electrical conductivity in the presence of the Powerplants Group Chairman.[11]

5.1.1 Specific gravity

The specific gravity of the fuel sample was measured using a Paar density meter, model No. DMA-48, in accordance with American Society of Testing and Materials (ASTM) Standard D4052, by Mr. Steve Dirago, a Saybolt laboratory technician. The density meter had a sticker that indicated it had been calibrated on December 3, 1999, and was due for calibration in March 2000. Before the start of the test, the test chamber of the unit was cleaned with acetone and blown dry with low pressure shop air. To test the fuel for specific

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[10] All of the Group members were advised of the oil pressure transmitter testing, but they declined to participate.
[11] All of the Group members were advised of the testing of the fuel, but they declined to participate.

gravity, a small amount of fuel was decanted from the sample using a sterile syringe. The tip of the syringe was inserted into a port on the right side of the test unit and some fuel was pumped into a glass chamber that is viewable through a small window in the front of the unit. A light in the unit was turned on and the fuel was pumped in and drawn out slightly to ensure that there were no air bubbles in the sample to be tested. When it was certain that there were no air bubbles in the sample, the light was turned off and the test was started. The unit automatically cooled the sample to 15.5°C (the light must be off or it will warm the sample and result in false test readings), measured the specific gravity, and then printed out the specific gravity of the fuel sample. After the test was completed, the fuel sample was withdrawn from the test chamber in the unit and returned to the container. The test chamber was then cleaned with acetone and blown dry with low pressure shop air. The calibration of the unit is checked daily using toluene with a process know as control charting. The control charting check that was accomplished just prior to testing the fuel sample resulted in a density reading of 31.98 for the toluene test fluid.[12]

The specific gravity of the fuel sample was 0.7905 (Appendix 4).

5.1.2 Flash point

The flash point of the fuel sample was determined using a Fisher Tag Closed Cup Tester ASTM D-56 in accordance with ASTM Standard D-56, by Mr. Salem Bedad, a Saybolt laboratory technician. The calibration of the Fisher Tag Closed Cup Tester is checked by checking the flash point of nDecane, an ASTM calibrant fluid, that has a known flash point of 123°F. The records show that the flash point for the nDecane during the last calibration check, which was accomplished on January 24, 2000, was 123°F. About 50 milliliters of the fuel sample was placed into a clean metal cup. The cup was then set in a holder so that the bottom of the cup was setting in a bath of water that was at room temperature. A cover, which has a small window that is normally closed, was placed over the cup. The tester has a small gas jet adjacent to the cup that can be rotated down to the window in the cover. The window is rigged to open when the gas jet is rotated down to the cover. The water bath is heated electrically to warm the fuel to produce vapors. A rheostat on the base unit controls the amount of heat input to the water to control the temperature increase of the fuel to 2°F per minute. The temperature of the fuel in the cup and the water bath were measured using ASTM 9F thermometers, SN 79799 and F97-517, respectively. The thermometer calibration log indicated the thermometers were last calibrated on May 13, 1999. As the fuel was warmed to produce vapors, the gas jet, which was lit, was rotated down to the window at every 1°F increase of temperature of the fuel. The flash point of a fuel sample is the temperature of the fuel when the lit gas jet ignites the vapors in the cup.

Two flash point tests were conducted and the vapors in the cup ignited at 114°F for both tests. The barometric pressure during the flash point tests was 746

[12] Control charting checks the density of a known test fluid to ensure the readings stay within a required range. If the readings fall out of the required range, the unit must be adjusted. The upper and lower limits for the density of toluene is 31.999 and 31.969, respectively.

millimeters of Hg. The flash point of the fuel sample when corrected for the barometric pressure was 115°F[13] (Appendix 4).

5.1.3 Electrical conductivity

The electrical conductivity of the fuel sample was measured using an Emcee Electronics, Inc. Model 1152 Digital Conductivity Meter SN 12343 by Mr. Bedad, in accordance with ASTM Standard D2624-95, "Standard Test Methods for Electrical Conductivity of Aviation and Distillate Fuels." The meter had a sticker that indicated it was last calibrated on September 3, 1999, and that the next calibration was due on September 3, 2000. The meter consists of two parts: the meter box, which has a liquid crystal display (LCD) indicator and two buttons that are marked 'M' and 'C', and a probe that is inserted into the bottom of the meter box. The calibration of the meter is checked by pressing the button marked 'C' and the LCD display must show 10 times the number marked on the probe. The probe was annotated with the number 40 and the LCD indicated 400. The probe was rinsed off with acetone to rinse away any potential residue. The probe was dipped into the fuel sample and after approximately 3 seconds, the button "M" was pressed. The electrical conductivity value of the fuel sample was displayed on the LCD.

The electrical conductivity of the fuel sample was 0 pS/m (Appendix 4). The temperature of the fuel sample at the time of the test was 62.5°F, as measured on an ASTM 12F-8F thermometer, SN K96-749, which according to the thermometer logbooks was last calibrated on May 14, 1999.

5.2 Los Angeles fuel

On October 31, 1999, following the crash of the airplane, a Federal Bureau of Investigation special agent traveled to the Los Angeles International Airport (LAX) and obtained a sample of jet fuel from the truck that serviced flight 990 before the airplane departed to JFK.

On November 24, 1999, the LAX fuel was tested by Core Laboratories, a petrochemical laboratory in Carson, California, for flash point and electrical conductivity in accordance with ASTM Standard D-56 and D-2624, respectively. The flash point for the fuel was 117°F and the electrical conductivity was 11 pS/m at 68°F (Appendix 5).

6.0 Fuel flow transmitter tests

During the readout of the digital flight data recorder (DFDR) data, it was noted that after the right and left fuel control switches were moved to cutoff, which was followed immediately by a drop of the respective N2[14] and exhaust gas temperature (EGT) indications, that the indication of fuel flow continued for about four seconds. At the request of the Safety Board, Boeing performed an inflight shutdown test of the PW4060 engines on a new production Boeing

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[13] The flash point temperature is corrected for pressure as follows: $FP_{corrected} = FP_{observed} + 0.06(760\text{-}P)$.
[14] N2 is the high pressure rotor speed.

767 with fuel flow transmitters identical to those on Flight 990 to study the effects of fuel flow transmitter run down on the DFDR data. In addition, Boeing advised the Safety Board that Ametek, the fuel flow transmitter's manufacturer, conducted a laboratory test to document the run down time of the fuel flow transmitter. In a February 11, 2000, letter to the Safety Board, Boeing provided the results of Ametek's laboratory tests, the Engine Instrument and Crew Alerting System (EICAS) fuel flow indication logic, and the results of the Boeing 767 engine inflight shutdown tests (Appendix 6).

6.1 Laboratory test

The fuel flow transmitter operates by fuel entering the housing and being directed into a stationary swirler cap that swirls the fuel. The swirling fuel is directed over a rotor and magnet assembly causing the assembly to turn. The rotor and magnet assembly has two magnets that generate a start and stop pickoff signal as they pass by an electrical coil. The time differential between the start and stop signals is used to determine the fuel flow rate.[15]

Boeing reported that Ametek performed a bench test of a fuel flow transmitter. The test procedure was to establish a steady state fuel flow of 1,300 PPH and then close the fuel valve upstream of the transmitter. When the fuel valve was closed, the fuel flow transmitter produced a start-stop signal pair, which is indicative of the rotating element turning, for about 4-5 seconds.

6.2 Flight test

Boeing reported that on the Boeing 767 airplane, the engines' N1 rpm,[16] N2 rpm, EGT, and engine pressure ratio (EPR)[17] data is provided directly to the DFDR. The engines' fuel flow data is sent to the DFDR through the EICAS. Boeing stated that the EICAS incorporates the following filter logic for the fuel flow signal: "When no fuel flow activity exists or its input sample is less that 300 pounds per hour, and persists for 1.5 seconds, then set the resulting filtered value to zero. While this time delay is running but not expired, then the previous sampled value prior to the activation of the time delay shall continue to be input to the filter."

On January 11, 2000, Boeing performed a fuel flow DFDR test on the first flight of a new production 767-300 airplane that was equipped with PW4060 engines and Ametek fuel flow transmitters identical to the configuration that existed on Flight 990. The test procedure required the engine throttles be set to establish a fuel flow of 1,300 pounds per hour (PPH), which was the fuel flow recorded on the DFDR for Flight 990 when the fuel control switches were moved to the cutoff position. The test airplane's engines were then to be shut down individually by moving the fuel control switches to the cutoff position and observing the cockpit fuel flow indication.

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[15] PW4000 Maintenance Manual Section 73-31-01 Description
[16] N1 is the low pressure rotor speed.
[17] EPR is a measurement of engine power output as a ratio of the total pressure of the gases in the PW4060 exhaust pipe ($P_{t4.95}$) divided by the total pressure of the air entering the engine inlet (P_{t2}). EPR is equal to $P_{t4.95}/P_{t2}$.

The flightcrew reported that for both engines after the fuel control switch was moved to cutoff, the respective fuel flows remained steady at about 1,300 PPH for about 4-5 seconds before briefly dropping to zero, then momentarily increasing to 2,200 PPH before remaining steady at zero. The flightcrew also reported that the engines' N2 indication decayed almost immediately after the fuel control switch was moved to cutoff.

Boeing downloaded the DFDR data after the engine shutdown flight test. The data shows that when the left engine was shut down, the airplane was descending through 33,000 feet, speed was 280 knots calibrated airspeed (CAS), fuel flow 1,300 PPH, N2 rpm about 72 percent, and EGT was about 226°C. Because the DFDR records the fuel control switch position once every four seconds, the decay of N2 and EGT was used to determine when the fuel control switch was moved to the cutoff position. The left engine's data show that at time frame 527991, both N2 and EGT begin to decrease. The indication of fuel flow remained constant at about 1,300 PPH until time frame 527993 when it dropped to zero. At time frame reference 527810, the fuel flow indication increased to about 2,200 PPH, and then decreased back to zero. The data show that when the right engine was shut down, the airplane was descending through 28,200 feet, speed was 280 knots CAS, fuel flow 1,300 PPH, N2 rpm about 72 percent, and EGT was about 218°C. The left engine's data show that at time frame 527991, both N2 and EGT begin to decrease. The indication of fuel flow remained constant at about 1,300 PPH until time frame 527993 when it dropped to zero. At time frame reference 527995, the fuel flow indication increased to about 2,200 PPH, and then decreased back to zero.

7.0 Low oil pressure warning

During the readout of the DFDR data, it was noted that the DFDR recorded low oil pressure warnings[18] for both left and right engines for a period of about 16 seconds during the airplane's descent. The warnings occurred during the airplane's descent about 3-4 seconds after negative-G condition initiated and continued until shortly after the airplane returned to a positive-G condition.

The Safety Board requested P&W to provide an assessment of the effect of the negative-G condition on Flight 990's PW4060 engines. In a November 30, 1999, letter to the Safety Board, Mr. Michael Bartron of P&W's Flight Safety Office and a member of the Powerplants Group, stated that the PW4060 engine's type certificate data sheet, E24NE, states that temporary interruption of oil pressure associated with negative-G operation for a period of up to 30 seconds is permissible. Mr. Bartron further stated that the operation of the PW4060 engine with low oil pressure for a period of 16 seconds should not have a significant effect on engine operation (Appendix 7).

Boeing advised the Safety Board that during the Boeing 767 airplane/PW4000 engine flight test program, an airplane did undergo negative-G testing. In accordance with 14 CFR 25.943 and Advisory Circular (AC) 25-7A, the airplane was tested under negative-G conditions to demonstrate acceptable operation of the airplane's fuel system and engine's oil

[18] The low oil pressure warning occurs when the engine's oil pressure drops below 70 pounds per square inch.

system. The AC requires that the duration of the negative-G test condition should be a minimum of 7 seconds with a total accumulation of at least 20 seconds of negative-G operation. Boeing advised that the negative-G certification test was conducted on August 20, 1987. There were four test conditions performed that were performed at 10,00 to 15,000 feet, 275 to 350 knots, at high engine thrust settings. Boeing stated that in each of the test conditions, the engine oil pressure began to decrease almost immediately with the onset of negative-G operation. The test conditions, which ranged in length from 0.8 to 8 seconds, obtained a maximum negative-G condition of -0.80 g and the lowest engine oil pressure recorded was 65 pounds per square inch. The low oil pressure warning occurred about 1.5 seconds after the onset of the negative-G condition and stopped about 1.5 seconds after positive Gs were obtained (Appendix 8).

[redacted]

Gordon J. Hookey
Powerplants Group Chairman

[redacted] 5/10/00

APPENDIX

1. Photographs

2. EgyptAir Engineering Section's life limited component list for PW4060 engine SN 724127

3. Safety Board's Materials Laboratory Report No. 00-030, dated February 2, 2000

4. Saybolt Laboratory results of testing JFK fuel

5. Core Laboratory results of testing LAX fuel

6. Boeing letter, dated February 11, 2000, regarding Engine Fuel Flow FDR Data

7. P&W letter, dated November 30, 1999, regarding effects of negative-G operation on PW4060 engine and copy of type certificate data sheet

8. Boeing data sheet on results of 767/PW4000 negative-G flight test