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Robert A. Feeler, editorial coordinator

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On the cover: The high-pressure turbine disk from the no. 2 engine of a Hawker Siddeley HS 748 was recovered in pieces from the runway area at London (England) Stansted Airport after an uncontained engine failure. (Photo: U.K. Air Accidents Investigation Branch)

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Turbine-disk Corrosion Cited in HS 748 Engine Failure and Fire

The final report by the U.K. Air Accidents Investigation Branch said that causal factors in the nonfatal accident included the failures of maintenance personnel to identify the corrosion and to identify the inadequate fit of engine-turbine-assembly-seal members.

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FSF Editorial Staff

At 2331 local time March 30, 1998, an uncontained failure of the right (no. 2) engine occurred on a Hawker Siddeley HS 748 Series 2B during takeoff from London (England) Stansted Airport. The engine failure resulted in a sudden loss of power and a fire in the engine bay.

The flight crew landed the airplane on the departure runway (Runway 23), and the airplane overran the paved surface. Uneven ground in the overrun area caused the collapse of the nose landing gear. The airplane received substantial damage; the 40 passengers and four crewmembers were evacuated with no injuries or minor injuries.

The U.K. Air Accidents Investigation Branch (AAIB) said, in the final report on the accident, that there were four causal factors:

- "Significant reduction in the fatigue strength of the HP [high-pressure] turbine disk due to surface corrosion;

- “Inadequate control of the fit between engine-turbine-assembly-seal members, possibly influenced by inadequate turbine clamping bolt fit, causing sufficient reduction in the natural frequency of an HP-turbine-disk vibratory mode to allow its excitation within the normal operating speed range and consequent excessive stressing of the disk;
- “Fuel leakage from the engine bay fuel system, resulting in a major nacelle fire; [and,]
- “Failure to identify the turbine-assembly-seal-member fit and HP-turbine-disk corrosion as possible contributors to disk-fatigue damage after previous similar failures.”

The report said that the flight, which had been delayed about one hour because of a baggage problem, began its takeoff at 2329. The first officer was the pilot flying. The takeoff was made with full dry power; the airplane’s “wet power” water-methanol fuel-injection system was selected to “standby.” (The water-methanol system provides additional power in the event of a loss of power during takeoff; if an uncommanded power loss occurs in one engine, the system automatically selects full wet power on the other engine.)

“Less than five seconds after the ‘rotate’ call, at an airspeed of 115 knots and a height of between 30 feet [above ground level (AGL)] and 100 feet AGL, the sound of a sharp report, followed by an engine run-down, was recorded on the CVR [cockpit voice recorder],” the report said. “The aircraft yawed 11 degrees to the right of the runway heading. As the crew asked each other what the noise had been, loud shouting could be heard from the passenger cabin. The first officer said, as he corrected the yaw, ‘Something’s gone,’ and the [captain] then stated that he had taken control of the aircraft. Within eight seconds of the event, the first officer stated that an engine had stopped. Simultaneously, the senior cabin attendant … advised the flight deck crew via the interphone that the right engine was on fire.

“Engine power was reduced, and the aircraft yawed 14.5 degrees to the left of the runway heading. Four seconds later, the sound of the engine-fire-warning bell was recorded. …

“The aircraft was in the air for a total period of 27 seconds before the noise of the touchdown was recorded.”

The engine-bay fire continued while passengers and crew were evacuated from the airplane.

After the accident, debris from the Rolls-Royce Dart Mark 536-2 turboprop engine — including two pieces of the HP turbine disk, cowling parts and engine-casing parts — was found on the middle section of Runway 23 and the surrounding area.

The no. 2 engine was damaged by the uncontained failure and subsequent fire. The fuel system, nose landing gear and electrical system also were damaged. The damage to the electrical system caused the low-pressure fuel-shutoff valves and the engine-bay fire extinguishers to be disabled after the shutdown of the left engine.

Before the accident, the secondary fire extinguisher for the engine bay was unserviceable because of damaged electrical wiring. A post-accident investigation did not determine when the damage occurred.

Turbine's "Abrupt Failure" Cited

"Examination of the aircraft and the accident site made it clear that the accident had been precipitated by the abrupt failure and non-contained release of the HP turbine from the no. 2 engine," the report said. "The failure caused an abrupt loss of power from the engine and immediately initiated a substantial fire around the engine nacelle. It also caused the automatic selection of full-wet power on the no. 1 engine, which assured best climb performance but also resulted in maximum power asymmetry, and the FDR [flight data recorder] shows that the aircraft initially yawed 11 degrees right as a result."

The engine failure led to a "substantial overboard fuel leak from the fuel-heater assembly forming part of the LP [low-pressure] fuel supply line in the nacelle fire zone," the report said. The fuel leak resulted from the partial disconnection of two fuel-heater-assembly flexible joints.

"The violent breakup of the HP turbine had imposed considerable shock loads on the engine, which resulted in the gross damage to the nozzle box, the extensive fracturing of the intermediate casing and the rotational displacement of the nozzle box," the report said. "While the engine-mounting structure was apparently undamaged, it appeared that the loads had caused sufficient movement of the forward part of the nacelle relative to the wing to over-travel the fuel-heater-assembly joints and initiate the leak."

The report said that although the fuel leak might have been stopped if the no. 2 LP fuel cock had been closed, the flight crew had no opportunity to conduct engine-fire checklists that required the LP fuel cock to be selected "OFF." After the collapse of the nose landing gear and the resulting loss of electrical power, operation of the LP fuel-cock switch and the no. 2

engine-bay fire-extinguisher switch would have been ineffective.

The engine was a Rolls-Royce Dart Mark 536-2 turboprop of the RDa7 series — a single-spool turboprop engine with a two-stage centrifugal-flow compressor, seven straight-flow combustion chambers and an axial-flow turbine. Each combustion chamber includes a flame tube containing a fuel burner.

The turbine assembly is located in the nozzle box and is connected by concentric shafts to the compressor and the reduction gearbox. The turbine assembly comprises the HP turbine stage, intermediate-pressure (IP) turbine stage and LP turbine stage, each of which consists of a turbine disk containing nimonic-steel-alloy blades that are attached by "fir-tree" sockets.

The 131 HP turbine blades (part no. RK45409) have integral platforms at their roots and shrouds at their tips, with a specified design clearance between the platforms and shrouds of adjoining HP blades.

"Wear in engine service tends to increase this clearance, which can have an adverse effect on the vibratory characteristics of the turbine assembly," the report said. "The HP turbine blades are a high-cost component of the engine, and excessively worn blades could be repaired under a *Dart Overhaul Manual* Dart repair scheme (DRS) by weld-depositing material onto the worn face(s) and machining back to the required dimension. The clearance was not measured directly but inferred from gauge checks of individual blades."

Company Revised Repair Scheme

The original DRS 297, which was introduced in 1960, said that repairs should be performed on only one face of a blade platform — on the side with the most wear. Several changes were introduced in subsequent years, including amendments in 1981 to DRS 611 to include instructions for optional shroud repair. In 1992, the inspection section of the engine overhaul manual was changed to require the inspection — during overhaul — of platforms and shrouds and the repair, in accordance with DRS 611, of wear in excess of 0.002 inch (0.051 millimeter) on either a platform or a shroud. The 1992 change followed tests that showed that gaps larger than 0.002 inch could result in increased stresses in the blade-root area. The report said that incorporation of the repair schemes was not required to be recorded in engine logbooks and that determining the standard to which a particular set of blades had been inspected and repaired sometimes was difficult. DRS 297 was canceled in

1992; by the time of this accident, DRS 611 had not been incorporated on most Dart engines in service.

The HP turbine disk (part no. RK40726) is 15.24 inches (38.71 centimeters) in diameter; is machined from a forging of a 12 percent chromium, niobium and creep-resistant steel alloy; and is coated with corrosion-resistant paint. The hub is five inches (13 centimeters) in diameter; outside the hub, there is a tapered diaphragm section with a thicker outer rim. Seal rings on the forward face and the aft face of the diaphragm form rotating parts of labyrinth seals to control internal cooling and oil-containment airflows. The IP turbine disk includes similar seal rings, and within the turbine assembly, the design gap between HP seal rings and IP seal rings at room temperature is zero, plus or minus 0.0005 inch (0.0127 millimeter), the report said.

Four Similar Failures Reported in 26 Years

During the 26 years preceding the accident, four similar Dart turbine engine failures had occurred involving high-cycle fatigue cracking of an HP turbine disk, and each of the four was attributed to "a combination of turbine entry flow distortion and turbine-blade wear," the accident report said.

The report said that Rolls-Royce and the U.K. Civil Aviation Authority (CAA) had "concluded that the likely period before recurrence of the failure was such that additional remedial action was unnecessary, and measures aimed at fully controlling the suspected causes had not been taken."

The engine manufacturer initially attributed failure of the engine on the accident airplane to the same causes that were identified in the four earlier engine failures. Nevertheless, the report said, "The evidence was unconvincing, and major difficulty was experienced in determining the likely causes."

The investigation revealed that maintenance of the accident airplane had been performed in accordance with an appropriate maintenance schedule and that there was "no evidence … of any anomaly that appeared relevant to the turbine failure."

During the last scheduled maintenance before the accident, the no. 2 combustion-chamber flame tube was replaced because of damage resulting from high temperatures. When the accident occurred, the engine had accumulated 19,420 operating hours since new and 4,659 operating hours since the last major overhaul.

Production records for the HP turbine disk showed no significant abnormalities, the report said. The disk was new when it was installed in the engine during an engine overhaul in 1981; the last overhaul before the accident was in 1987, 7,081 cycles before the accident. At the time of the accident, the HP turbine disk had accumulated 15,047 cycles since new.

Overhauled HP turbine blades, with 2,100 operating hours since new, had been installed by the manufacturer in 1976; of the 131 blades, 127 were replaced with new blades in 1984. During the 1987 engine overhaul, the blades were overhauled or replaced with other overhauled blades; that was the last time the blades were removed from the HP turbine disk.

"No record was found to suggest that DRS 297 or DRS 611 (inspection and, if necessary, repair of blade platforms) had been carried out on the blades at this time or previously," the report said. "The records were inconsistent with regard to blade life at the time of release from the 1987 overhaul, but investigation indicated that the total time since new at the time of the accident was 7,242 hours for 127 of the blades, 14,759 hours for three of the blades and 15,842 hours for one blade."

All of the combustion chambers, with burners, had been replaced with overhauled units two years, or 1,246 operating hours, before the accident. Five of the combustion chambers had been inspected 293 operating hours before the accident, and the no. 2 combustion-chamber flame tube was replaced 23 operating hours before the accident because of over-temperature damage, which the report said was "not unusual." The original burner was refitted after replacement of the combustion-chamber flame tube but was not recalibrated; recalibration was not required or recommended.

The approved lives of the relevant components, determined by Rolls-Royce and approved by the CAA were:

- 6,200 operating hours between engine overhauls;
- 20,000 flight cycles for HP turbine disks;
- 25,000 operating hours for HP turbine blades; and,
- 3,000 operating hours between overhaul for combustion chambers, although the operator performed overhauls every 2,200 hours.

The last on-wing condition assessment of the engine before the accident had not included an internal inspection of the no. 2 combustion chamber or the no. 5 combustion chamber because of difficulties in accessing the two combustion chambers. The inspectors said that the combustion chambers had not been checked and that there was an oil leak from the accessory gearbox; otherwise, they said that they had found the engine's major components in good condition.

The investigation revealed that part of the no. 2 engine HP turbine disk separated while the engine was being operated at takeoff power and that the resulting damage was "fully consistent with the large forces generated having caused the turbine clamping bolts to fracture and the main part of the disk to break free and both disk portions to penetrate the nozzle box casing and the [cowling]."

Report Says Damage Could Have Been Worse

The report said that damage resulting from the HP turbine disk failure could have been more extensive.

"The two major portions of the disk exited the nacelle with considerable energy," the report said. "It was fortunate that … their trajectory happened to be such that they both missed the cabin. The debris also missed electrical wiring looms and the LP fuel heater in the engine bay. Wiring damage in a previous [accident] had apparently contributed to catastrophic consequences, and a direct hit on the fuel heater by a portion of the disk could well cause fuel leakage at a much higher rate than occurred in [this accident].

"There was also an appreciable chance of a portion of non-contained disk striking the fuselage. This would have the potential to directly [endanger] a number of passengers close to the rotation plane, to cause appreciable fuselage structural damage and to disable various aircraft systems. It appeared that flight-control-system vulnerability could be a particular concern, as all the primary [controls] and secondary controls mechanical runs could lie in the path of a single piece of fractured disk released toward the fuselage just below the horizontal."

The investigation revealed that the disk failure followed the development of a fatigue crack in the inner-blend radius between the diaphragm of the HP turbine disk and the rear-seal ring. The fatigue crack progressed through about 40 percent of the circumference of the disk and about 80 percent of its thickness.

"The disk was sufficiently weakened that normal operating loads were sufficient to extend the crack radially from either end and to fracture the remaining 20 percent of the cracked section, causing a substantial portion of the disk to detach, with consequent severe power plant disruption," the report said. "There was no doubt that the circumferential cracking of the HP turbine disk had been responsible for the sudden disruptive failure of the disk."

The fatigue crack originated at an area of surface corrosion with corrosion pits up to 0.004 inch (0.102 millimeter) deep. The report said that the corrosion was "reportedly quite usual" and probably had reduced the fatigue strength of the HP disk but was not the sole cause of the disk failure.

High-cycle Fatigue Blamed for Initiation of Crack

Investigators found no gross overstress, over-temperature or low-cycle fatigue (LCF) effects on the disk. The report said that "the evidence strongly indicated that HCF [high-cycle fatigue] had been responsible for the initiation and progression of the circumferential crack" and that the fatigue may have occurred because of stress levels near the ultimate tensile strength for the material.

"Thus, it was concluded that the fracture had resulted from the failure region having experienced fluctuating loads that, superimposed on the steady loads, had exceeded the capability of the material, and that this had resulted from a vibratory condition of the disk," the report said.

Investigators were unable to determine how many load cycles had been involved in the fatigue-crack progression.

The engine was originally designed in the 1950s, when specialists knew relatively little about the fatigue damage that could be caused by cyclic loading. Subsequent tests were conducted to assess HCF loading effects for HP turbine blades but not for the disk diaphragm.

"Following a number of disk failures in service, further stress analysis had shown that the blend radius areas between the diaphragm and the seal rings were highly stressed by the steady-state loading on the disk with the engine at maximum power," the report said. "Only a small margin existed for superimposed alternating stresses without fatigue development in these areas being likely."

The tests also had shown that larger gaps between the blade platforms "were likely to accentuate the vibration by reducing the natural frequency of the disk-vibration mode, and the manufacturer had believed that the same would be true for increased shroud gaps," the report said. "However, no quantitative test data on the fluctuating stress levels in the seal-ring-radius areas resulting from these features had been available."

The investigation revealed that the failed engine's combustion chambers and their flame tubes were "essentially intact," that no foreign objects had entered the engine or any components upstream of the HP turbine, and that a partial blockage of the flow of fuel to the turbine was unlikely.

Until the results of engine-rig tests became available in 2000, the engine manufacturer had believed that the HP turbine disk failure had been caused by "excessive fuel-burner asymmetry, with a probable contribution from excessive HP-turbine-blade platform and [HP-turbine-blade] shroud gaps," the report said. Results of those tests, however, revealed that "neither fuel-burner-flow asymmetry … nor excessive HP-turbine-blade gaps had a significant effect on alternating stress levels in the HP turbine-disk-seal-radius area."

Instead, the report said, "A gap between the HP [turbine-seal rings] and IP turbine-seal rings that had probably been enlarged by in-service wear had reduced the disk resonant frequency to within the normal operating speed range of the engine. The combined effects of resonance and disk corrosion had caused high-cycle fatigue cracking of the disk."

The report recommended that the engine manufacturer continue research to determine the causes of deviations from fuel-burner-flow installation settings and to develop methods of preventing such deviations.

After the accident, the manufacturer issued service bulletins (SBs) intended to prevent Dart HP-turbine-disk failures; CAA required compliance with those SBs.

Nevertheless, the report said, "A significant number of disk failures had previously occurred over many years, all with potentially serious or catastrophic consequences, without effective measures to prevent further failures having been taken. Attempts were made during the investigation to assess the reasons for this, in order that measures could be recommended aimed at ensuring more prompt and effective action in similar situations in the future."

The recommendations included the following:

- "The engine manufacturer [should] include a requirement in the engine overhaul manual for an as-received burner-flow check for the Dart engine at all maintenance-shop visits. The manual should also include requirements for any turbine-disk action necessitated by the flow-check results that fully reflect the latest knowledge on the possible effects of burner-flow deviation on the disks”;

- “The CAA [should] review and [should] revise, as necessary, the requirements for the recording of maintenance actions with the aim of ensuring that information essential to the further operation and maintenance of the aircraft is readily available”; and,

- “The CAA [should] require the engine manufacturer to designate repair schemes in such a way that the standard that has been used on any particular occasion is readily apparent.”

The report said that, although the CAA knew about previous HP disk failures and “had been involved with the manufacturer in assessing the continued airworthiness of the engine and judging it to be satisfactory,” there had been little action regarding possible HCF damage to the HP turbine disk as a result of fuel-burner flow deviation. In addition, the primary factor in this accident — the gap between the HP turbine-seal ring and the IP turbine-seal ring — was not previously identified.

“The manufacturer and the CAA had apparently been satisfied with the continuing airworthiness of the engine type on the basis of failure-rate statistics,” the report said. “HP turbine disk failures, including diaphragm failures, had continued. …

“The reasons for the lack of effective action over a number of years, while failures had continued, could not be fully established. A significant cost would have been associated with a requirement for replacement of the disk with a redesigned version, or for early incorporation of DRS 611, and this was undoubtedly a factor that had militated against such action. However, simple and low-cost measures, such as determining fuel-burner flow deviation by means of an as-received check had not been taken. This was quite inconsistent with the manufacturer’s sustained belief that excessive flow deviation had a major responsibility for the failures.

“The likely effectiveness of the improvements to the engine manufacturer’s investigation review and risk-assessment process made since [this] accident could not be fully assessed.”

Based on the findings, the report included the following recommendations:

- “The engine manufacturer and the CAA [should] reassess the susceptibility of the three-stage Dart turbine [disk] to HCF failure and ensure that effective action aimed at preventing recurrence has been taken”; and,

- “The CAA and the engine manufacturer [should] consider the need for further improvement to their systems intended to ensure effective action to prevent recurrence following potentially catastrophic in-service failures of U.K.-type-certificated equipment used on public transport aircraft.”

The report said that, in 1991, the engine manufacturer had said that there had been 27 failures of the part no. RK49121 HP turbine disk in which part of the disk had detached and that all 27 events were attributed to HCF. No additional events were reported from the time of the manufacturer’s statement in 1991 until one additional event, which occurred in June 2001. (The report said that the event was “apparently very similar” to this accident and that the disk failure modes probably were similar.)

Of the 27 events, four were disk-diaphragm failures in which relatively large pieces of debris were released, as was the 2001 event. Four of these five events involved engines of the RDa7 series; one involved an engine of the RDa10 series.♦

[FSF editorial note: This article, except where specifically noted, is based on the U.K. Air Accidents Investigation Branch Accident Report No. 3/2001 (EW/C98/03/7). The 70-page report contains diagrams, photographs and tables.]

MAINTENANCE ALERTS

Missing Tail-rotor Bumper Plug Goes Unnoticed During Rebuilding And Inspection

A Schweizer 269B helicopter was being air taxied from one location to another at Victoria (British Columbia, Canada) International Airport when anti-torque control failed and the aircraft yawed to the right. At an altitude of about 10 feet (three meters), the pilot quickly lowered the collective and the helicopter was landed hard.

“A main-rotor blade cut through the upper surface of the tail boom, and the skid gear broke at all attachment points,” said the accident report by the Transportation Safety Board of Canada. The pilot was not injured.

When the aircraft's anti-torque system was inspected, investigators found that the bumper plug was missing from the recently installed tail-rotor drive shaft. The bumper plug restricts the aft movement of the tail-rotor drive shaft and prevents the splined drive of the drive shaft from becoming disengaged from the tail-rotor transmission input gear.

The report said, "The helicopter had been partially assembled from an assortment of parts originating from three different helicopters of similar type in various conditions. This work was performed at the Island Flight Support maintenance facility under the authority of Starwest Aviation … . [Nevertheless], Starwest Aviation did not hold a helicopter rating on its maintenance approval. An aircraft maintenance engineer (AME) was hired to perform the assembly work.

"The AME was instructed to install the tail-boom assembly onto the helicopter, with its [tail-rotor] drive components already installed. He was unfamiliar with the helicopter type and was unaware of the bumper plug. After installing the tail boom, he provided worksheets indicating that he had inspected the [tail-rotor] drive shaft."

Following the initial assembly at Starwest Aviation, the assembly work was completed by A&L Helicopter Maintenance, which signed for the helicopter's airworthiness certification.

"A&L Helicopter Maintenance personnel performed a 200-hour inspection and replaced the tail boom and installed the [tail-rotor] drive shaft," said the report. "A 200-hour inspection requires that all preceding inspections be carried out, namely the 100-[hour], 50-[hour] and 25-hour, daily and special inspections. The 25-hour and daily inspections make specific reference to an end-play and backlash check for the [tail-rotor] drive shaft. Performed correctly, these checks are meant to detect an incorrect installation and a worn or missing bumper plug … . It is probable that the missing bumper plug went unnoticed after the installation of the drive shaft."

The report said that the Schweizer 269B maintenance manual describes in detail the procedure for replacing and inspecting the bumper plug in the model.

Nevertheless, "the Schweizer illustrated parts catalog for the 269B and earlier models does not have a reference for the bumper plug," said the report.

Uncontained Engine Failure Occurs After Inspection on Same Day

During takeoff at Newark (New Jersey, U.S.) International Airport on Sept. 5, 2000, a Continental Airlines

McDonnell Douglas DC-10-30 experienced a decrease in no. 2 engine N_1 [low-pressure fan speed] to 78 percent five seconds after N_1 stabilized at the target 104 percent. The engine-fail light and the master warning light illuminated. The captain, who was the pilot not flying, took control of the aircraft and rejected the takeoff at an indicated airspeed of about 90 knots, shutting down the no. 2 engine. After clearing the runway, the aircraft was stopped on a taxiway and was towed to the gate. None of the 14 crewmembers or 230 passengers on board was injured.

"Examination of the engine revealed that the low-pressure-turbine case was fractured around its circumference, at the back side of the second-stage vanes," said the accident report by the U.S. National Transportation Safety Board (NTSB). "In addition, from the nine o'clock position to the two o'clock position, a [2.25]-inch [5.72-centimeter]-wide strip of the metal case was missing, from over the top of the second-stage vanes. A visual examination through the opening in the case revealed that all of the second-stage vanes were missing."

The second-stage low-pressure vanes of the engine, a General Electric Aircraft Engines (GEAE) CF6-50C2, consisted of 16 segments held in place by eight nozzle locks. Thirteen segments were recovered from the debris field, as were pieces of engine cowling and other parts.

The engine was examined by a GEAE facility in the United Kingdom, under the supervision of NTSB.

"The investigation revealed [that] all second-stage-turbine nozzle locks had failed," said the report. "The nozzles had exited through a hole in the engine case."

The accident was the second known uncontained failure of a CF6-50C2 engine, NTSB said. After the previous failure — which also occurred on an aircraft operated by Continental Airlines — the airline issued a directive to inspect for broken nozzle locks using visual techniques and tactile techniques.

The GEAE Powerplant Group report on its examination said, "[Continental Airlines] paperwork indicated that the inspection was performed on engine SN 455-276 [the serial number of the engine involved in the second failure] on May 29, 2000, with no nozzle-lock discrepancies detected."

On Aug. 4, 2000, Continental Airlines issued Temporary Revision (TR) No. 00-72-01 to its DC-10 maintenance manual. With this change to the manual, an engine could remain in service until its next shop visit with one third-stage nozzle lock broken or one fourth-stage nozzle lock broken, but an engine was not allowed to remain in service if any second-stage nozzle lock was broken, NTSB said.

The GEAE Powerplant Group report said, "According to Continental DC-10 airplane zonal inspection requirements at the time of the incident, the fan-thrust reverser and core cowls are to be opened every 1,650 hours or 400 cycles to perform visual inspections of the engine and pylon. As part of the engine/pylon zonal inspection, the nozzle locks are visually inspected in accordance with the DC-10 maintenance manual. According to the Continental work card [for] engine SN 455-276, the zonal inspection was performed on Sept. 5, 2000, the same day as the incident. The zonal inspection work card did not indicate that there were any anomalies with the nozzle locks.

"Examination of the zonal inspection work card revealed that the inspection was listed as a general visual and servicing inspection. No specific reference was found for examination of the nozzle locks."

The work card, however, said that "zonal inspection is a general visual inspection of all components, systems, installations and structure including, but not limited to, electrical, hydraulic, pneumatic, fuel and mechanical systems, including, but not limited to, wiring, tubing, plumbing, ducting, clamps, fittings and brackets, [and] primary and secondary structure … inspecting for conditions such as cracking, corrosion, chafing, leaks, loose/missing fasteners, damage, delamination, dust and lint accumulation, inadequate drainage or insufficient corrosion-inhibiting coatings and for other circumstances which could lead to the above conditions."

NTSB said that the probable cause of the incident was "the sequential failure of the second-stage low-pressure-turbine nozzle locks, which allowed the second-stage turbine-nozzle segments to rotate within the low-pressure turbine case, machining a hole whereby the nozzle segments exited the engine. A factor was the failure of the mechanic to detect any failed second-stage nozzle locks during the zone inspection."

Skin Crack in ATR-42 Fairing Leads To Emergency Landing

An Avions de Transport Regional (ATR)-42-300 was en route from Dublin (Ireland) Airport to Cork (Ireland) Airport when, as it was descended through Flight Level 100 (approximately 10,000 feet), airframe vibrations began.

"General cockpit vibration was described as severe, with the level of vibration increasing toward the rear of the aircraft," said the incident report by the Ireland Air Accident Investigation Unit. "The level of vibration was sufficiently high to cause the crew to briefly consider a forced landing, rather than continuing to Cork Airport, as the flight path crossed a large urban area."

The pilot declared an emergency to air traffic control and was cleared for an approach to Cork Airport.

"During the approach … it was noted by the flight crew that the level of vibration reduced after the selection of flap 15 degrees and the lowering of the undercarriage," the report said. "Further selection of flap to 30 degrees brought about the cessation of all airframe vibration, as sensed by the crew."

The ATR-42 was landed at Cork Airport without further incident. There were no injuries among the three crewmembers and 45 passengers.

When the aircraft was inspected for damage, inspectors discovered that there was a crack in the trailing-edge section of panel 291BL, a nearly flat, rectangular wing-fuselage fairing located above the fuselage centerline directly forward of the wing forward mainspar. Panel 291BL consists of composite material: an outer skin of two layers of Kevlar and an inner skin of three layers of Kevlar. Between the inner skin and outer skin is a lightweight honeycomb of hollow hexagonal forms whose vertical walls brace the Kevlar skins.

"The [panel 291BL] was found to be cracked on its upper (outer) skin, along a line parallel to its trailing edge located approximately 70 millimeters [2.73 inches] forward of the trailing edge," said the report. "The crack could be seen with the naked eye. There also appeared to be damage to the inner skin corresponding to the external crack. … It was also noted that there was a marked reduction in the bending stiffness of the panel in the area of the visible crack, when the panel was manually subjected to modest bending loads."

ATR previously had issued three service bulletins (SBs) about wing-to-fuselage panels, including panel 291BL: SB ATR-42-53-006B (1991, revised 1994); SB ATR 42-53-0091 (1994); and SB ATR 42-53-0082 (1994, revised 1998). The French aviation regulatory agency, Direction Générale de L'Aviation Civile (DGAC), issued two Airworthiness Directives (ADs) about panel 291BL: 94-162-056 (B) (1994) and 97-159-071 (B) (1997).

The report said that after the SBs and ADs were issued, seven reported incidents of in-flight vibration occurred that were associated with panel 291BL. The requirements of the pre-1998 SBs and ADs had been met on the incident aircraft, the report said. A more recent SB, ATR42-57-0059 (2000), classified by the manufacturer as "recommended," had not been accomplished.

The report said, "In normal use, the panel [291BL] was subjected to a negative aerodynamic pressure (lift) on the external upper face. This force tended to suck the panel up into the shape of an inverted 'U.' This force was resisted by the bending stiffness of the panel. This, in turn, produced compressive loads in the inner skin and tensile loads in the outer skin. Due to the effects of telegraphing, the inner skin's ability to resist repeated applications of compressive loads was reduced. ["Telegraphing" refers to sagging of the skin between the supporting vertical walls of honeycomb, in a pattern similar to the sagging of telephone wires or telegraph wires between poles.] This ultimately led to failure of the inner skin along the line of the trailing edge.

"Due to the failure of the inner skin, the outer skin now had to carry in increased load which led, in turn, to failure and cracking of the outer skin. The result of this damage was to significantly reduce the ability of the panel to withstand the bending stress. The net result was loss of bending stiffness, which allowed the panel to deform, under aerodynamic forces, into a slightly inverted 'U' shape."

The panel's deformation allowed its trailing edge to be tugged loose by aerodynamic forces and deflected into the airflow, the report said.

"This produced a substantial step in the airflow over the top of the wing and caused substantial turbulent eddies to form," said the report. "When these turbulent eddies passed over the tailplane and elevators, the result was the severe vibrations felt by the crew."

The report concluded that the incident had the following causes:

- "The cause of the in-flight vibration was the displacement of the trailing edge of the panel 191BL, due to the loss of bending stiffness of the panel; [and,]
- "The bending stiffness of the panel 129BL deteriorated due to a combination of cyclic loading of the panel and telegraphing of the inner skin of the panel, which in turn caused structural degradation of the inner skin and ultimately a crack on the outer skin."

ATR has begun a program to address issues raised in the investigation, the report said. Two SBs have been issued: SB ATR-42-53-0123, which requires all panels 291BL to be inspected for damage and cracks within three months of Nov. 6, 2001, and thereafter every 1,000 hours, until the panel 291BL is replaced in accordance with SB ATR-42-53-0125, which requires the replacement of the panel 291BL with a panel of improved design by Nov. 1, 2004. The DGAC has issued an AD making the ATR SBs mandatory.

Models later than the ATR-42-300, as well is the larger variant ATR-72, have a differently designed panel in the panel-291BL position. Those models have not experienced similar problems, the report said. ♦

NEWS & TIPS

Stationary Chuck Has Compact Design

With a compact design and square body, the SMW Autoblok Model MACC-3 stationary chuck is designed for work-holding applications in tight spaces. Applications include reworking brake pads or regrinding rotors. The MACC-3 is available with base sides of four inches (10 centimeters [cm]), five inches (13 cm), six inches (15 cm) and eight inches (20 cm). The square size and low-profile body (1.8 inches; 4.7 cm) makes the unit easier than conventional round chucks to mount on a horizontal table or to stack units, the manufacturer said.

According to the manufacturer, the MACC-3 applies clamping force up to 5,600 pounds (2,540 kilograms) using soft jaws. The unit has a through hole for shafts or “swallowed” parts, and has a sealed body designed to promote long service life and low maintenance. The chuck can be mounted either horizontally or vertically using two precision screws.

For more information: SMW Autoblock, 285 Egidi Drive, Wheeling, IL 60090 U.S. Telephone: (888) 224-8254 (U.S.) or +1 (847) 215-0591.

Battery Charger Dissipates Sulfuric Acid Deposits

VDC Electronics has developed three “maintenance chargers” for lead-acid batteries that emit high-frequency pulses, causing sulfuric acid crystals that have formed on the battery’s storage plates to break down. The sulfuric acid that is released returns to the cell’s electrolyte. Desulfated “clean” plates enable batteries to be charged more quickly, to be capable of delivering their rated full power and to have extended service life, the manufacturer said.

The product line includes maintenance chargers for 12-volt batteries, 24-volt batteries and 36-volt batteries. Besides charging the batteries, the units can maintain batteries at full charge while avoiding overcharging, the manufacturer said. Test circuitry in the chargers determines the battery's condition and state of charge. Light-emitting diodes (LEDs) indicate usage modes: power, charge, maintain and desulfate-condition.

For more information: VDC Electronics, 83 Cedar Lane, P.O. Box 5537, Englewood, NJ 07631 U.S. Telephone: (800) 379-5579 (U.S.) or +1 (631) 423-8220.

Shear Mounts on Vise

A shear and rod cutter can be mounted on a vise, as well as on a standard workbench. The Heinrich Co. No. 1 Portable Metal Shear includes a latch to permit vise mounting, which facilitates moving the unit to other workstations or job sites.

The product's shearing capacity is 1/8-inch (3.18-millimeters [mm]) mild steel and the product's rod-cutting capacity is 5/16-inch (7.94-mm) mild steel. The 24-inch (61-centimeter) handle mechanism is designed to provide easy cutting action, and the unit permits shearing of wide sheets, straight cuts or outside curves.

For more information: Heinrich Co., 2707 South Memorial Drive, Racine, WI 53403 U.S. Telephone: +1 (262) 634-4229.

Inspection Systems "Learn" from Operator

Multiaxis automated ultrasonic inspection systems, manufactured by Nutronik and distributed in North America by Krautkramer, include a Teach and Learn mode that allows operator input of the dimensions and contours of the component to be tested. The systems also will accept specifications from commercial computer-aided design (CAD) software.

Three different inspection methods are available using the inspections

Ultrasonic Inspection System

systems: immersion, water-jet (squirter) coupling and water-flow coupling. Systems are designed for the aircraft maintenance industry in custom-built formats.

For more information: Krautkramer Ultrasonic Systems, 50 Industrial Park Road, Lewistown, PA 17044 U.S. Telephone: +1 (717) 447-1210.

Eddy-current Instrument Can Be Hand-held

Phasec 2s, a compact eddy-current instrument from Hocking that weighs 2.06 pounds (0.94 kilogram), features a 5.5-inch (14-centimeter)-diagonal display with 480 pixels by 320 pixels. Light-emitting diode (LED) backlighting makes viewing clear under any lighting conditions, the manufacturer said.

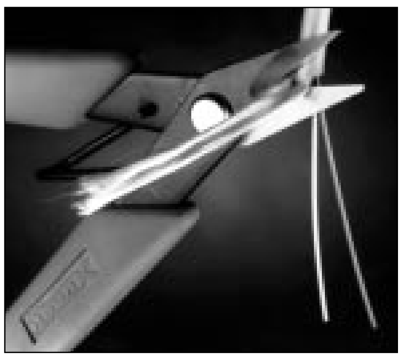
The unit has an operating frequency of 100 hertz (Hz) to six megahertz (MHz), with a gain of 0 decibels (dB) to 85 dB in 0.1 dB steps. Phase rotation of 0 degrees to 360 degrees in 0.1-degree steps allows for precise adjustments and positioning, the manufacturer said.

The Phasec 2s can be connected to a personal computer for specification of settings, data transfer and printing. Power is supplied by a lithium-ion battery that enables the instrument to be used continuously for as long as 14 hours without recharging.

For more information: Agfa NDT, Krautkramer Ultrasonic Systems, 50 Industrial Park Road, Lewistown, PA 17044 U.S. Telephone: +1 (717) 242-0327.

Shear Cleanly Cuts Kevlar

An ergonomic scissor designed for cutting Kevlar fiber insulation has been introduced by Xuron. The Model 9180 Kevlar Shear includes one serrated blade edge that prevents the fibers from sliding, and the other blade edge is highly sharpened to provide a clean, uniform cut, the manufacturer said.



Kevlar Shear

Featuring cushioned rubber grips and a return spring that opens the blade after each cut, the product is designed to be comfortable to hold in any size hand because there are no finger loops. Applications include preparing fiber-optic cable.

For more information: Xuron Corp., 60 Industrial Park Road, Saco, ME 04072 U.S. Telephone: +1 (207) 283-1401.

Drive-through Berms Contain Spills

Ultra-Containment Berms from Ultra-Tech capture spills from tanker trucks and other vehicles, helping to meet environmental spill-containment requirements, the manufacturer said. The berms consist of a flat base with side walls that can be driven through, which then revert to their upright position. The manufacturer describes them as "drive in–drive out sidewalls." For storage, the berms can be rolled up.

The berms are designed to be used while vehicles are being loaded, unloaded or washed, and are said to keep spills from entering storm-water drains. The material is rugged, chemical-resistant Copolymer-2000,

Containment Berm

the manufacturer said. Standard and custom sizes are available.

For more information: UltraTech International, 11542 Davis Creek Court, Jacksonville, FL 32256 U.S. Telephone: (800) 353-1611 (U.S.) or +1 (904) 292-1611.

Tape Withstands Heat

A pressure-sensitive, glass-fabric tape with a silicon adhesive — 17-FibGx — has high abrasion resistance, temperature stability and strength, the manufacturer said.

Applications are primarily electrical, including splicing and positioning in coils, and reinforcing slot insulation and armature insulations. High-temperature applications include transformers, solenoids and appliances.

The tape is rated for continuous use in the range from –100 degrees Fahrenheit (F; –73 degrees Celsius [C]) to 500 degrees F (260 degrees C). It is available in widths from 0.25 inch (0.635 centimeter) to 22 inches (56 centimeters).

For more information: CS Hyde Co., 461 Park Ave., Suite 300, Lake Villa, IL 60046 U.S. Telephone: (800) 461-4161 (U.S.) or +1 (847) 265-6903.♦

Call for Nominations

Flight Safety Foundation–Boeing Aviation Safety Lifetime Achievement Award

The Flight Safety Foundation (FSF)–Boeing Aviation Safety Lifetime Achievement Award recognizes an individual for his or her lifetime commitment and contribution to enhancing aviation safety. Nominees should have devoted efforts spanning two decades or more to enhance civil aviation safety and/or military aviation safety beyond the normal expectations of their particular job assignments. Nominations can be posthumous. The recipient of the award — established by the Foundation and The Boeing Co. — will receive a handsome, wood-framed, hand-lettered citation and complimentary registration for the recipient and spouse or guest for the joint meeting of the FSF 56th annual International Air Safety Seminar (IASS), the International Federation of Airworthiness 33rd International Conference and the International Air Transport Association. The recipient's name will be inscribed on a specially designed trophy displayed at the Museum of Flight in Seattle, Washington, U.S.

The nominating deadline is Feb. 28, 2003. The award will be presented in Bangkok, Thailand, at the FSF 56th annual IASS, Nov. 17–20, 2003.

Flight Safety Foundation

Review selection criteria and submit your nomination(s) via our Internet site.
Go to http://www.flightsafety.org/life_achievement_award.html

For more information, contact Ann Hill, director, membership and development, by e-mail: hill@flightsafety.org or by telephone: +1 (703) 739-6700, ext. 105.