ACCIDENT

Aircraft Type and Registration:	Dassault-Breguet Mystère-Falcon 900B, G-HMEV
No & Type of Engines:	3 Honeywell TFE731-5BR-1C Turbofan engines
Year of Manufacture:	1986
Date & Time (UTC):	20 January 2007 at 1651 hrs
Location:	Approximately 7 nm south-west of Worthing, Sussex
Type of Flight:	Commercial Air Transport
Persons on Board:	Crew - 2 Passengers - None
Injuries:	Crew - None Passengers – N/A
Nature of Damage:	Severe No 3 Engine damage, nacelle cowl holed, slight damage to the horizontal stabiliser
Commander's Licence:	Airline Transport Pilot's Licence
Commander's Age:	38 years
Commander's Flying Experience:	6,515 hours (of which 3,002 hours were on type) Last 90 days - 139 hours Last 28 days - 45 hours
Information Source:	AAIB Field Investigation

Synopsis

As the aircraft was climbing through FL130 after takeoff from Farnborough there was a loud bang and the No 3 Engine Bay fire warning activated. The crew shut down the engine and fired the extinguisher first shot; the fire warning ceased. The aircraft diverted to Gatwick and landed without further incident.

It was found that the No 3 Engine low pressure (LP) turbine assembly had suffered major disruption. Debris from the turbine assembly ruptured the engine casing, penetrated the cowling and caused slight damage to the horizontal stabiliser. Many of the fractured parts were lost overboard but the available evidence indicated that the failure had probably resulted from the fracturing of an LP turbine blade, leading to the loss of rotational restraint for the turbine stators and the spin-up and non-contained rupture of the stators.

One of the Stage 2 blades had signs of a casting defect and fracturing of this blade probably initiated the turbine assembly break-up. However, there had also been a substantial number of previous cases of Stage 3 blade fracture and it was possible that such a failure caused the turbine assembly damage. The engine manufacturer has taken measures aimed at preventing turbine blade failure. However, the possibility that casting defects could be present in Stage 2 blades produced prior to these measures and remaining in service could not be dismissed. The turbine casing had been ruptured in some of the previous cases of blade failure, but not where the newer of two available standards of casing had been fitted. The engine manufacturer issued Service Bulletins in the latter part of 2007 recommending replacement of the casing with the later standard but this modification had not been mandated.

Two Safety Recommendations have been made.

History of the flight

The aircraft, bound for Tel Aviv, departed Farnborough at 1640 hrs. Approximately 10 minutes after departure, as the aircraft climbed through FL130 in a position 7 nm southwest of Worthing the crew heard a loud noise from the rear of the aircraft. Shortly afterwards the engine fire aural warning sounded and the No 3 Engine fire warning light illuminated. The pilots noticed that the No 3 ITT[1] warning light was also illuminated and that the ITT indication was fluctuating "wildly". Indications for the No 1 and 2 Engines were normal. The pilots carried out the engine fire procedure for the No 3 engine, and declared a MAYDAY to the London Terminal Control Centre (LTCC). The crew were given immediate radar vectors for Gatwick Airport, the nearest airport; the crew accepted Gatwick since it was "fully equipped" (with rescue and fire fighting services) and had a runway of sufficient length to meet all the foreseeable performance limitations of the aircraft.

Two minutes after the MAYDAY call, the non-handling pilot announced that the fire was "under control" and that the engine fire procedure was complete. The subsequent diversion was uneventful, although on approach to Runway 26L at Gatwick there were several instances of the GPWS 'TOO LOW, FLAPS' callout. The pilots commented that the assistance provided to them by ATC had been "very professional"; approximately 12 minutes had elapsed between their MAYDAY call and the landing at Gatwick.

The operator indicated immediately after the incident that it intended to ferry the aircraft back to Farnborough using the remaining engines because there were no appropriate maintenance facilities at Gatwick. The operator also reported that the aircraft flight manual contained information about the correct procedures for conducting a two engine ferry flight and that it had received approval from the engine manufacturer for such a flight. When advised that the AAIB would inspect the aircraft at Gatwick, however, the operator decided that the aircraft would not conduct further flights until repairs had been carried out.

Ferry flights

An aircraft with one unserviceable engine would no longer meet the certification standards set for qualifying for a Type Certificate and as such the Certificate of Airworthiness would be invalid. In some cases a Permit to Fly can be issued so that the aircraft can be flown to a maintenance base. The procedure to be followed is contained in Flight Operations Department Communication (FODCOM) 28/2005, issued by the United Kingdom CAA. In addition to establishing technical and operational procedures for the safe conduct of such a flight, the FODCOM specifies that the operator must apply for a Permit in writing to the CAA. In the case of G-HMEV the operator provided evidence that such procedures were in place. In the event, no such application was made.

Recorded information

In addition to the FDR and CVR fitted to the aircraft, data which had been recorded from the Pease Pottage radar

Footnote

[1] Inter turbine temperature.

head was made available to the investigation. These three sources were used to reconstruct the history of flight. Both CVR and FDR retained recordings covering the period from the onset of the event until the subsequent landing at Gatwick. LP spool speed (N_1) was the only engine parameter recorded by the FDR installation; each engine was sampled and recorded at four second intervals. Spectral analysis, with particular reference to engine frequency signatures, was conducted on the CVR area microphone channel in order to corroborate the data obtained from the FDR.

The data showed that the event occurred whilst the aircraft was climbing through FL130 with all engines at 100% N_1. The heading was 167ºM and the IAS approximately 310 kt. The aircraft was located over water approximately 7 nm south-west of Worthing at the time.

The No 3 engine N_1 reduced from 100% and stabilised at 38% over a 10-second period and the No 3 engine bay fire warning activated. The engine was shut down 9 seconds later and N_1 reduced to about 22%. The other two engines were unaffected. For the remainder of the flight the No 3 Engine N_1 indicated that the fan was 'windmilling', with rotational speed proportional to airspeed. Mode C radar recordings indicated that the maximum altitude reached was FL136.

LTCC handed the aircraft over to Gatwick ATC and it was cleared to land. Seven instances of 'TOO LOW, FLAPS' were recorded during the approach. The aircraft landed 12 minutes after the event.

The model of FDR[2] fitted to the aircraft used a Group Code Recording (GCR) method of encoding data before writing the information to the magnetic tape. Overall, the quality of the recording was below average with numerous data errors. Following the engine failure, the recorded data quality deteriorated significantly with the result that there was more data in error than there was valid. The nature of GCR encoding together with the large quantity of data errors rendered large sections of the data irrecoverable.

From AAIB experience, this model of recorder is more susceptible to data errors induced through vibration of the tape transport mechanism than other tape-based recorders. Solid state recorders do not suffer from these vibration effects. In light of recording performance and continued airworthiness, the ICAO Flight Recorder Panel is reviewing the suitability of magnetic tape flight recorders with a view to amending the Standards in Annex 6 to discontinue their use. It is anticipated that such a change would also require a retrofit of existing installations and the replacement of magnetic tape recorders with those that use solid state memory as the recording medium. As the AAIB consider that this issue is being addressed satisfactorily, no Safety Recommendation is currently deemed necessary.

Aircraft description

Aircraft

The Falcon 900B is a long-range passenger transport aircraft with accommodation for two pilots and up to 19 passengers. It is a low-winged monoplane with a horizontal stabiliser mid-mounted on the fin; maximum takeoff weight is 45,500 lb (20,640 kg). The aircraft is powered by three rear-mounted turbofan engines, with the No 1 and No 3 Engines pylon-mounted on the fuselage, left and right sides respectively, and the No 2 Engine installed within the rear fuselage.

Footnote

[2] The FDR model number was 17M800-251 (commonly known as an F800) manufactured by L-3 Communications.

Powerplant

The Honeywell TFE731-5BR is a two-spool, turbofan engine with a sea-level static takeoff rated thrust of 4,750 lb. The low pressure (LP) spool consists of a three-stage axial turbine driving an axial compressor and, via a speed-reduction gearbox, the fan (Figure 1). The high pressure (HP) spool has a single-stage axial turbine driving a centrifugal compressor. Both turbines rotate clockwise (all circumferential positions noted are as viewed from the rear). At 100%, N_1 is 21,000 rpm and the HP turbine speed (N_2) is 30,300 rpm.

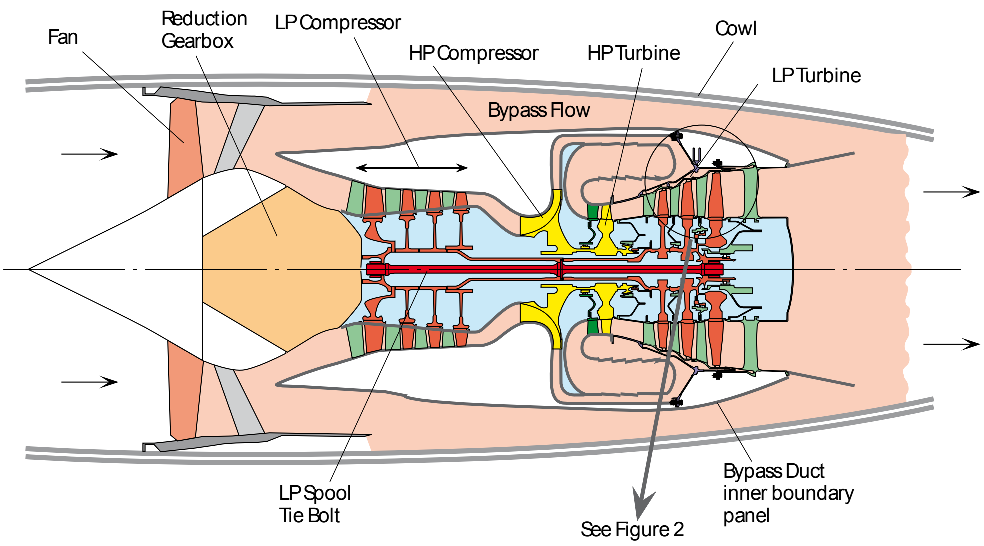

Figure 1

G-HMEV Engine Schematic

The turbine assemblies are of conventional configuration, with each turbine stage consisting of a series of radial aerofoil-section blades installed in fir-tree slots formed in the rim of a turbine disc. A ring of static nozzle guide vanes (NGVs) at the inlet to the HP and LP turbines controls the flow of gases from the combustion chamber. The flow onto the LP turbine 2nd and 3rd stages is directed by a ring of stator vanes upstream of each stage. Integral blade tip shrouds connect the LP turbine NGVs and stator vanes together (Figure 2). The Stage 2 stators are rotationally keyed to the Stage 3 stators, which are bolted to the aft flange of the interstage turbine transition duct (ITTD), a casing with a Y-shaped cross section that surrounds the LP turbine assembly.

The LP turbine blades each have an integral tip platform with a knife-edge profile that fits against the respective NGV or stator vane shrouds to control gas leakage at the tip. An integral platform at the root of

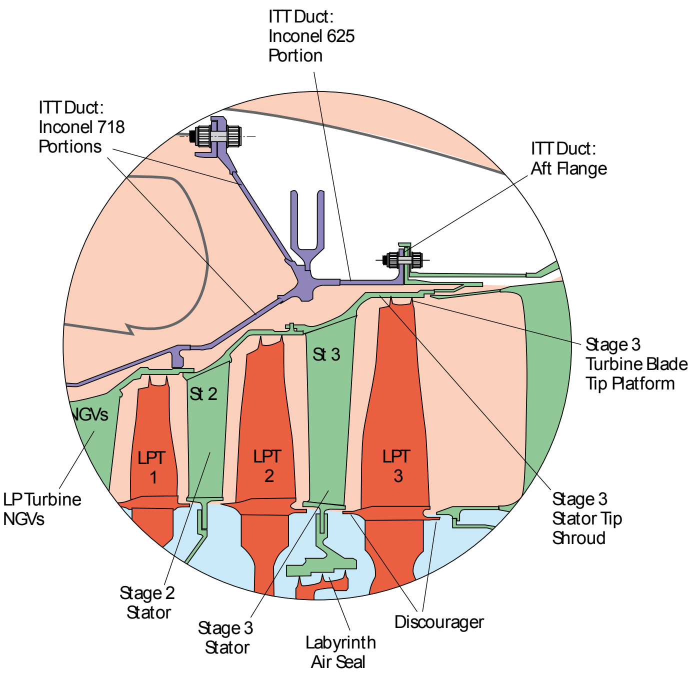

Figure 2

LP Turbine Schematic

each blade incorporates 'discouragers' to control gas leakage between the root of the blade and the adjacent stators.

The Stage 2 turbine has 52 blades, each around 3.4 inches long with a chord of 1.2 inches. They are produced by machining a casting of IN100 alloy, a nickel-chromium steel alloy, and protected with an aluminide coating.

A firewire sensing element for the engine bay fire detection system runs around the outside of the ITTD near to its aft flange. The hot section of the engine is covered by two relatively lightweight steel panels forming the inner wall of the bypass duct in this region. The cowls for the pylon-mounted engines of the Falcon 900 are of double-walled composite construction.

Examination

Aircraft

Examination of the aircraft revealed an approximately 6x6 inch triangular hole in the upper portion of the No 3 Engine cowl. The hole was at the longitudinal station of the LP turbine and located at around 2 o'clock. A 4-inch long scratch in the undersurface of the right horizontal stabiliser, near to the tip, appeared likely to have been caused by debris ejected from the engine.

No 3 Engine

The rear part of the No 3 Engine ITTD had been cut through circumferentially just forward of its aft flange, round 310°. The edges of the cut had been bent outwards, producing a gap in the duct of up to 2 inches (Figure 3). An 8-inch length of the circumference on the left side remained intact. The panels covering the engine hot section had sustained multiple impact damage in a band centred on the cut in the ITTD, together with overheat discoloration of the paint in this area. The damage included extensive holing of the panels; one area of holing concided with the hole in the cowl.

The engine manufacturer and the maintenance organisations responsible for the aircraft and engines provided an excellent level of co-operation and assistance with the investigation. The No 3 Engine (Part No 3075330-3. Serial No P95127C) was strip examined under AAIB control at the manufacturer's facility in Phoenix, Arizona, USA, with representatives from the USA National Transportation Safety Board (NTSB) and the USA Federal Aviation Administration (FAA) present.

The examination revealed that many of the components exposed to the hot gas path had been coated with a silvery metallic deposit, consistent with the deposition of fine aluminium debris ground from the LP and HP compressor shrouds by, respectively, the LP compressor blades and the HP impeller. Such an effect reportedly would commonly result in the event of operation of this engine type with major imbalance present. With this

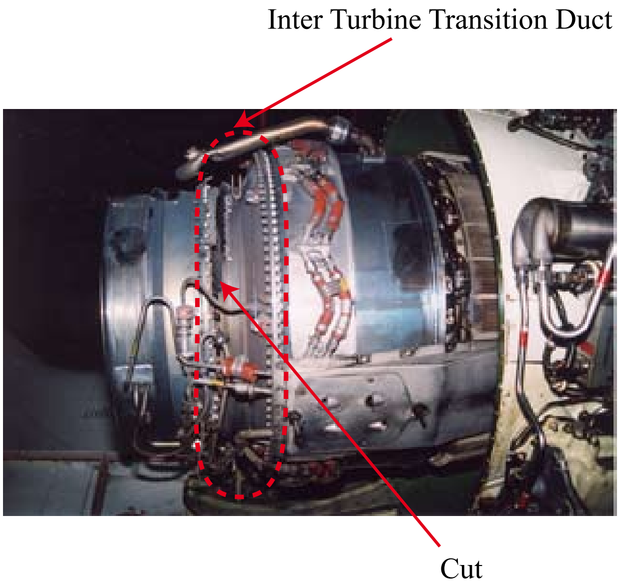

Figure 3

No 3 Engine Turbine Assembly

exception, no damage or anomaly was found upstream of the LP turbine. In particular, there were no signs of hard object impact on the HP turbine blades or on any other upstream gas path components.

No 3 Engine LP Turbine assembly

All of the LP turbine Stage 1 blades had suffered severe impact damage to their trailing edges and, for four blades, localised impact damage to the leading edge.

The outer portion of all Stage 2 blades had broken off, generally at around 1.5-2.0 inches from the root platform. However, three of the blades had fractured at around 0.5 inches from the root platform and one (No 5, numbered from a clocking index mark on the disk) had fractured at the platform (Figure 4). The outer portion of all Stage 3 blades had broken off, generally at around 70% span. Only a relatively small amount of the debris fractured from the turbine assembly remained with the powerplant, most of it having been ejected overboard. In particular, most parts of the Stage 2 and Stage 3 stators were absent.

The nature of the damage indicated that the disruption had resulted from a failure in the Stage 2 or Stage 3 of the LP turbine assembly. The leading edge damage to the Stage 1 turbine blades was consistent with the effects of limited forward penetration of debris into this region. The fracture surfaces of those stator and turbine blade parts that were available had features consistent with failure due to overload and no signs were found of pre-existing fractures, with the exception of the Stage 2 No 5 blade, as described below.

Stage 2 LP Turbine No 5 Blade

The fracture surface of the No 5 blade exhibited a discoloured region, extending over the rear one-third of the section, where the surface had a darker

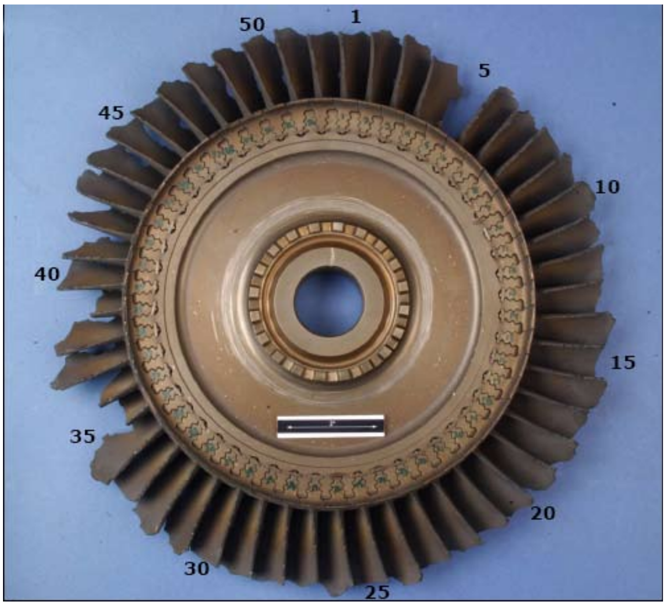

Figure 4
Stage 2 LP Turbine

appearance than for the remainder of the fracture (Figure 5). Detailed scanning electron microscope (SEM) examination of the discoloured region showed features of heavy oxidisation and areas with a smooth appearance, lacking typical fracture features. The characteristics indicated that these areas had been unbonded, ie the material had been separated before the blade had failed. Features evident with the SEM on the

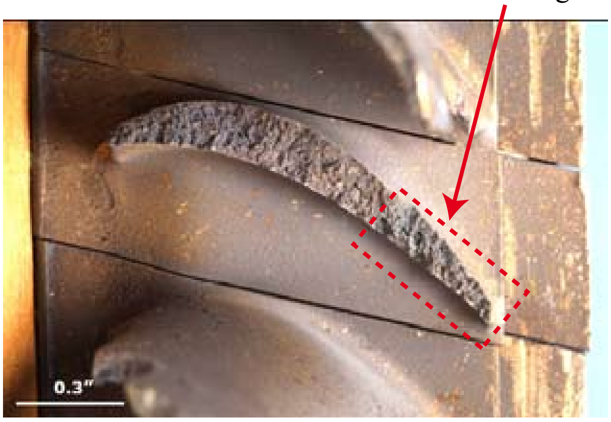

Figure 5
Stage 2 LP Turbine No 5 Blade

un-discoloured part of the fracture indicated that it had resulted from overload. The chordwise extent of the pre-existing crack was around 0.3 inches.

A section cut through the discoloured region revealed a relatively thick coating on the fracture surface, with the appearance of an oxidation product (Figure 6). The sectioning also revealed a secondary crack, beneath the separation fracture surface and generally parallel to it, also coated with the oxidation-type material. The secondary crack extended to the surface of the blade at the trailing edge but, on the plane of the section, was not open at the surface. Some alloy depletion was evident in the parent material beneath the coating layer on both the separation fracture and the secondary crack, also indicative of oxidation effects. Energy dispersive x-ray analysis of the coating layer material revealed the presence of the IN100 base metal elements and of oxygen, again indicative of an oxidation product. Aluminium was not present in a high concentration.

The presence of the secondary crack and the oxidisation both of its surface and of the discloured region of the separation fracture were evidence of pre-existing cracks in this area that had been open at the blade surface while exposed to the hot oxidising environment. The features were indicative of a casting defect; the metallographic appearance suggested that this had been a 'hot tear' (see below). The absence of appreciable aluminium in the oxidisation layers suggested that the cracks had not been open at the blade surface when the aluminide blade coating had been applied.

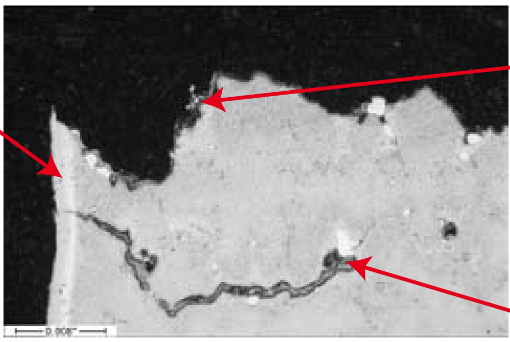

Figure 6

Section through Discoloured Region of Stage 2 LP Turbine No 5 Blade

Maintenance history

Maintenance documents indicated that the No 3 Engine had been converted from a TFE731-5A model to a TFE731-5BR-1C in 1998. A new LP turbine Stage 3 disc with all new Stage 3 blades (Part Number (PN) 3060690-1) had been installed at this time.

The engine had undergone repairs in early 2006, apparently to rectify a problem with excessively low margins from the allowable limits for turbine temperature and spool speed at takeoff power. All new LP turbine Stage 2 blades had been installed at this time. These blades had accumulated 624 hr/188 cycles from new at the time of the accident. The next scheduled inspection of the blades, including fluorescent penetrant inspection (FPI), would have been at the subsequent Major Periodic Inspection of the engine, required every 2,100 operating hours.

The records indicated that routine FPI and eddy current inspection of the LP turbine Stage 3 blades for cracking had been carried out during the 2006 repair work, the last inspection of the blades. These inspections are reportedly normally accomplished without removing the blades from the disc. At the time of the accident the Stage 3 blades had accumulated 3,109 hr/1,304 cycles from new and 624 hr/188 cycles since the 2006 inspection.

The last aircraft and engine check had been a 300/400 hr Check, followed by a ground engine run, completed immediately before the accident flight. At the time of the accident the engine had accumulated 8,409 hr/3,366 cycles from new.

Background

TFE731 engine

The Garrett (subsequently Honeywell) TFE731 engine was first certificated in 1972, as the TFE731-2. Development produced -3 and -4 versions, followed by the -5 model, a higher power version that was certificated in 1983. These four versions are referred to as the 'Classic' models. At the time of the accident the -4 and -5 engines, which constituted the majority of the Classic fleet, numbered approximately 2,795, with a total operating time of around 12.7 million hours.

Further developments generated -20, -40, -50 and -60 models, referred to as 'NG' (Next Generation) models. At the time of the accident approximately 1,894 NG engines had been produced, with a total fleet operating time of around 4.1 million hours.

LP Turbine Stage 2 Blades

The LP turbine Stage 2 blades are manufactured by filling a casting mould with the molten IN100 alloy in a vacuum furnace. The cooling rate of the casting during solidification is controlled by the mould insulation and the surrounding temperature.

Information from the engine manufacture indicated that occasionally the casting could suffer intergranular cracking during or shortly after solidification, a defect known as a 'hot tear'. This could apparently result from an inappropriate cooling regime or possibly because of physical disturbance of the mould and tended to occur towards the aerofoil root, where the thermal gradients were relatively high. It was intended for such a defect to be detected by inspection of the casting. The standard technique, after grit blasting of the casting, was FPI, intended to reveal the presence of a crack that extended to the surface. The engine manufacture stated that the critical crack length in the root fillet region of the blade (ie the crack length at which failure during the next engine cycle would be expected) was 0.250-0.375 inches.

For an approximately 10-year period the castings had been manufactured by a contractor in the USA. Two feeds to the mould had been employed, one at the tip and the other at the root. In early 2006 the production process had been changed and manufacture moved to Mexico. One of the changes was for a single feed to the mould, at the root. The Casting Number altered but the blade PN remained the same. At the time of G-HMEV's accident, around 55,000 TFE731 blades had been produced using the revised process.

In October 2006 a TFE731 engine suffered an LP turbine failure during a production test run, as the engine was nearing maximum power in a test cell. Investigation by the engine manufacturer found evidence of a hot tear defect in an LP turbine Stage 2 blade, which was found to be a revised production standard blade.

Further assessment by the engine manufacturer as a result of the failure found that a number of revised production standard blades exhibited hot tear defects that had remained undetected by the standard inspection process. It was concluded that hot tears in blades produced by the previous casting process had tended to be more open at the surface and therefore more readily detected than with the revised process. As a result, a thermal cycle was added to the revised production process whereby the cooled blade castings were re-heated to 2,000ºF and then re-cooled, prior to the FPI.

Following the failure, revised production standard blades installed in turbine assemblies that had not returned to service were re-inspected. As the aluminide coating on a completed blade could cover and hide a hot tear, affected LP turbines were removed and subjected to an overspeed spin before FPI in order to apply a controlled overload to the blades, with the aim of opening up any hot tear defects present. An eddy current inspection method was also developed, but was considered impractical for general usage.

The engine manufacturer estimated that the operating time before complete fracture of a blade with a defect such as that found on the No 5 blade from G-HMEV's failed engine would be less than 1,000 hours.

LP Turbine Stage 3 Blades

Information from the engine manufacture indicated that several versions of LP turbine Stage 3 blade had been employed on the TFE731 engine and others were in development at the time of G-HMEV's accident. The original type of blades (PN 3074755) had suffered a substantial number of in-service cases of high-cycle fatigue cracking and fracturing near the tip, apparently associated with a torsional resonant vibration mode and also possibly related to excessive bowing of the stator shrouds. At the time of G-HMEV's accident this type of blade was no longer in service.

Redesigned stator shrouds and redesigned blades (PN 3060690-1) with an elevated resonant frequency were introduced. This type of blade was fitted to G-HMEV's No 3 Engine. A minor variation of this version (PN 3060690-2) was also produced. At the time of G-HMEV's accident the PN 3060690 blade was fitted to approximately 40% of the TFE731 fleet.

The blade was made from a nickel-chromium steel alloy casting, treated with a hot isostatic press process to reduce porosity, and machined to the required dimensions. Variabilities in the shape of the cast blade could be corrected by 'straightening' (bending and twisting), while cold, in order to produce a casting that was within the final machined dimensions. Unlimited straightening was permitted for this version of blade.

One of the features of the blade that raised the resonant frequency above the normal operating range was a highly waisted profile (ie pronounced reduction in chord) at around 75% span. A region of the blade leading edge at the waist was found to experience relatively high operating stresses, with a normal maximum stress of around 90 ksi (thousands of pounds per square inch). However, this version of the blade remained in service for a number of years without major problems.

Several cases of blade fracture were then experienced, apparently affecting a particular batch of blades installed during 1999 and 2000. A Service Bulletin (TFE731-72-3691, initial issue date 12 August 2004) recommended replacing blades from this batch at the time of certain aircraft or engine maintenance checks or engine disassembly operations. However, failures of blades that were not from the suspect batch subsequently occurred. The failures were attributed to excessive local stress, probably related to residual stresses introduced by straightening operations during manufacture.

A further version of the blade (PN 3060788), with restrictions on the amount of straightening allowed, was developed as a replacement for the suspect batch of PN 3060690 blades. The operating stress in the highly stressed leading edge region was significantly reduced, but the blade remained susceptible to stress concentrations produced by any nicks in the leading edge and a number of failures occurred.

Two further versions of blade were in development at the time of G-HMEV's accident. One of these (PN 3061823) is made from a different material and has a different camber and less pronounced waisting. Its resonant frequency remains outside the LP turbine operating speed range and normal maximum peak stresses are significantly reduced (in the order of 50 ksi). No straightening during manufacture is permitted. Start of production was planned for December 2008.

Summary of previous failure cases

LP Turbine Stage 2 Blades

The engine manufacturer provided information on the failure of the Stage 2 blade in the production test bed in 2006. The casting defect was located just above the fillet between the root platform and the aerofoil, a relatively highly stressed area, and extended across approximately 75% of the section. The manufacturer concluded that stress concentrations created by the defect had caused the blade to fracture under normal loading conditions. With the possible exception of G-HMEV's accident, no similar failures to engines in service had been reported.

LP Turbine Stage 3 Blades

The engine manufacturer was aware of 65 previous cases of fracture of PN 3060690 Stage 3 blades in service, 44 on Classic engines and 21 on NG engines. Around 66% of the failures occurred on the TFE731-5B version of the engine. The failures had occurred at a blade operating time of between 811-6,000 hr from new. Six in-service failures of the PN 3060788 blade had occurred.

In some of the previous cases of blade failure the fracture surfaces were missing or damaged. However, the engine manufacturer considered that reliable fracture and materials analysis results had been obtained in around one third of the cases. This had led to the conclusion that blade fracturing had typically occurred when excessive stresses led to a small chordwise intergrannular crack in the leading edge at around 75% span that had then extended in low-cycle fatigue. Above a critical crack length of approximately 0.25 inches the remaining part of the blade cross-section had become overloaded and suffered rapid fracture.

The blade material is relatively notch-sensitive and thus a nick in the blade surface, as could be caused by hard object impact occurring during engine running or maintenance operations, tended to act as a significant stress concentrator. The blades were therefore considered to be quite sensitive to leading edge damage, particularly in the region of relatively high operating stresses at around 75% span. Additionally, testing and calculation reportedly showed that significant residual stresses could be introduced during blade manufacture by straightening operations on the casting. Because of the particular profile of the blade, the cold-working associated with straightening tended to be concentrated at the waist region, producing residual stresses in this area that could add to the relatively high leading edge operating stress in the same region.

LP Turbine blade failure effects

In many of the Stage 3 blade failure cases little further damage resulted, but in some cases the broken portion of blade caused other Stage 3 blades to break. Blade debris could then pile up and be dragged round in contact with the stator shrouds and could sever the shroud ring, thus separating the Stage 2 and Stage 3 stator rings from the aft flange of the ITTD. The consequent removal of the rotational restraint for the stator rings would lead to their spinning under the influence of aerodynamic forces on the vanes, while retained generally centralised by the inner labyrinth air seals. The forces would also tend to drive the stator rings aft, causing them to suffer damage from contact by the turbine blade discouragers. It was predicted that the rings would burst at around 2,000 rpm; debris from the rotating stators would then impact and damage the ITTD.

In four of the previous Stage 3 blade failure cases part of the ITTD circumference was cut through and engine debris non-containment occurred. In one of these cases debris struck the aircraft fuselage, causing denting and scratching but no penetration.

All four of the cases where the ITTD was ruptured occurred on Classic engines. Some cases of Stage 2 and Stage 3 stator spinning on NG engines occurred, but in none of them was the ITTD penetrated. This was attributed to the significantly stronger material used for the aft portion of the duct on the NG engines (see below).

Interstage turbine transition duct

The main part of the ITTD is fabricated from welded Inconel 718 for both Classic and NG engine types. The aft portion of the duct is also of Inconel 718 for NG engines, but is of Inconel 625 for Classic engines. The Ultimate Tensile Strength of the two materials at 1,200ºF is in the order of 158 ksi for Inconel 718 and 50 ksi for Inconel 625.

The engine manufacturer had plans in place at the time of G-HMEV's accident for a programme to modify Classic engine ITTDs by replacing the aft portion with an Inconel 718 component, as for the NG engines. Service Bulletins to incorporate this modification were issued on 12 September 2007 (Nos TFE-731-72-3727 and TFE-731-72-3728, applicable to different engine models). The Service Bulletins noted that compliance addressed a safety issue and that the manufacturer recommended accomplishment:

> *'at the next major periodic inspection (MPI), next access (next access is defined as removal of the ITT duct), or within three years of release of this service bulletin, whichever occurs first.'*

The FAA stated their intention to issue an Airworthiness Directive (AD) to mandate incorporation of the Service Bulletins. At the time the Service Bulletins were issued in late 2007, approximately 2,800 engines in service (all of the Classic engines) were fitted with the original standard of ITTD. On 4 April 2008 the FAA issued a Notice of Proposed Rulemaking (NPRM) to this effect (USA Federal Register Docket No FAA-2008-0264). The NPRM required comments by 3 June 2008. It was anticipated that the AD would be issued in July 2008.

Discussion

The evidence showed that the major disruption and non-containment of the No 3 engine during the climb had resulted from a break-up in the LP turbine assembly that had caused extensive rupturing of the ITTD surrounding the turbine. The engine bay fire warning that occurred very shortly after the break-up probably resulted from the impingement of hot engine gases, escaping through a substantial gap created in the duct, onto the firewire element fitted around the engine in this area. The pilots encountered no difficulties in carrying out the fire drill and the warning ceased shortly thereafter. An effective and helpful ATC service expedited the crew in diverting and landing without further difficulties.

Debris ejected through the gap cut in the ITTD penetrated the bypass duct wall and the engine cowling. It appeared likely that debris had contacted the horizontal stabiliser, albeit without causing substantial damage. However, the effects on the aircraft could have been more severe had the debris been ejected through the cowl at a different rotational position. A similar non-contained failure of an engine mounted within the fuselage, such as the No 2 engine of the Falcon 900, would appear to entail the risk of significant damage to aircraft systems and possibly to the structure.

Most of the parts broken from the LP turbine assembly had been ejected from the engine and lost into the sea and some fracture surfaces on the parts that remained had been damaged. Positive evidence as to the cause of the turbine assembly break-up was therefore not available.

However, service experience suggested the type of failure mechanism that had occurred. It had been found that in some cases a Stage 3 blade failure could initiate a cascade failure of the other blades in the stage, and that the resultant damage could cause the Stage 2 and Stage 3 stators to spin up and burst. Impact of the stator debris would damage the ITTD, in some cases to the extent of penetrating the aft limb of the duct. It appeared possible that similar effects would result from a Stage 2 blade failure. The damage to the available parts from G-HMEV's No 3 Engine was similar to that which had previously resulted from the above failure sequence and thus indicated that the disruption had originated with the failure of a Stage 2 or Stage 3 LP turbine blade.

Positive evidence was found of a defect in the Stage 2 No 5 turbine blade, consistent with a hot tear formed during casting. It appeared likely that a defect of the type found could cause a separation fracture of the blade under normal operating loads within the blade's operating time since new and that this had led to the turbine disruption. However, there was no evidence to determine whether the fracture of this blade had in fact initiated the turbine assembly break-up sequence, or had resulted from it, if the initiation event was the failure of a Stage 3 blade. Stage 3 blades of the standard fitted to G-HMEV's No 3 Engine had previously suffered a number of failures, apparently due to surface nicks and/or because of residual stresses that could be introduced at manufacture.

It was therefore concluded that the turbine disruption had probably resulted from the failure of the Stage 2 blade due to the casting defect present, but could have been caused by a Stage 3 blade failure.

The alterations introduced to the process for inspecting Stage 2 blades were intended to improve the detection of significant hot tear defects in the castings. However, it appeared that a substantial number of revised production process blades that had entered service before the improved defect detection process had been applied could be subject to an elevated risk of hot tear defects, in common with the No 5 blade from G-HMEV's engine. The first opportunity to detect such defects would normally be the FPI carried out at the next Major Periodic Inspection, required every 2,100 operating hours.

The engine manufacturer's planned introduction of an improved Stage 3 blade, with lower peak operating stress and a prohibition on operations during manufacture that might excessively increase residual stresses, was intended to address the failure problem affecting these blades.

While the above failure sequence had led to a number of cases of ITTD penetration on Classic engines, experience suggested that NG engines were unlikely to suffer duct penetration in similar circumstances because of the significantly stronger alloy used for the aft limb of the duct. Thus incorporation of the modification that upgraded the ITTD on Classic engines to the NG standard appeared likely to eliminate the problem of non-containment in the event of a turbine blade failure.

Safety Recommendations

The above measures, to improve the Stage 2 blade inspection process, to introduce an improved Stage 3 blade and to modify the ITTD on Classic engines, indicated a concerted aim by the engine manufacturer to resolve the problem. However, it appeared that it might take an extended time period for the measures to be incorporated across the engine fleet and, as none of them had been mandated, there was no certainty as to the level of take-up. In view of the appreciable number of previous cases of blade failure and resultant non-containment and the potential hazard to the aircraft of non-containment, the following Safety Recommendation is made:

Safety Recommendation 2008-013

It is recommended that the FAA comprehensively review the measures already proposed by the manufacturer aimed at preventing non-contained LP Turbine assembly failures of Honeywell TFE-731 engines, including the proposed timescales for incorporation of the measures across the fleet, with the aim of ensuring an adequate standard of airworthiness.

In view of the experience indicating that the upgraded version of the ITTD is likely to prevent possibly hazardous debris non-containment in the event of an LP turbine assembly break-up, the following Safety Recommendation is made:

Safety Recommendation 2008-014

It is recommended that the FAA require the timely incorporation of Honeywell Service Bulletins (Nos TFE-731-72-3727 and TFE-731-72-3728) for the fitment of an upgraded standard of Inter-Turbine Transition Duct to Honeywell TFE-731 engines, in order to ensure that the modification is embodied across the engine fleet within a reasonable timescale with the aim of eliminating the non-containment hazard posed by an LP turbine blade failure.