AIRCRAFT ACCIDENT REPORT 4/2004

Air Accidents Investigation Branch

Department for Transport

Report on the accident to Fokker F27 Mk 500 Friendship, G-CEXF At Jersey Airport, Channel Islands on 5 June 2001

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At Jersey Airport, Channel Islands
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This investigation was carried out in accordance with
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Department for Transport
Air Accidents Investigation Branch
Berkshire Copse Road
Aldershot
Hampshire GU11 2HH

June 2004

The Right Honourable Alistair Darling
Secretary of State for Transport

Dear Secretary of State

I have the honour to submit the report by Mr P T Claiden, an Inspector of Air Accidents, on the circumstances of the accident to Fokker F27 Mk 500 Friendship, G-CEXF, which occurred at Jersey Airport, Channel Islands on 5 June 2001.

Yours sincerely

Ken Smart
Chief Inspector of Air Accidents

Contents

Appendices

GLOSSARY OF ABBREVIATIONS USED IN THIS REPORT

AAIB	-	Air Accidents Investigation Branch
AFS	-	Airport Fire Service
agl	-	above ground level
amsl	-	above mean sea level
ATC	-	Air Traffic Control
BCF	-	BromoChlorodiFlouromethane
CAA	-	Civil Aviation Authority
CVR	-	Cockpit Voice Recorder
DRS	-	Dart Repair Scheme
DSL	-	Declared Safe Life
efh	-	Engine Flying Hours
FDR	-	Flight Data Recorder
FEA	-	Finite Element Analysis
FO	-	First Officer
FV	-	Firth Vickers
HCF	-	High Cycle Fatigue
HP	-	High Pressure
HPT	-	High Pressure Turbine
hrs	-	hours
HV	-	Hardness Value
Hz	-	Cycles per second (Hertz)
IIC	-	Inspector in Charge
IP	-	Intermediate Pressure
IPT	-	Intermediate Pressure Turbine
ISA	-	International Standard Atmosphere
kg	-	kilogram(s)
km	-	kilometre(s)
kt	-	knot(s)
lb	-	pound(s)
lbf	-	pounds force
lbs.ft	-	pounds per feet
LCF	-	Low Cycle Fatigue
LP	-	Low Pressure
mb	-	millibar(s)
MLG	-	Main Landing Gear
MHz	-	Megahertz
MPa	-	Mega pascal
NGV	-	Nozzle Guide Vane
OM	-	Overhaul Manual
p/n	-	Part Number
psi	-	Pounds per square inch
PSL	-	Predicted Safe Life
rpm	-	Revolutions per minute
s/n	-	Serial number
SB	-	Service Bulletin
shp	-	Shaft horse power
SL	-	Sea Level
SOP	-	Standard Operating Procedures
TBO	-	Time Between Overhaul
TGT	-	Turbine Gas Temperature
TODA	-	Take-off Distance Available
TORA	-	Take-Off Run Available
tsi	-	tons per square inch
UFDR	-	Universal Flight Data Recorder
UK	-	United Kingdom
UTS	-	Ultimate Tensile Strength
VHF	-	Very High Frequency
1F	-	1st Family
2D	-	2 Orthogonal Diameters
2EO	-	2^{nd} Engine Order

Air Accidents Investigation Branch

Aircraft Accident Report No: 4/2004 (EW/C2001/6/5)

Registered Owner and Operator	Channel Express (Air Services) Limited
Aircraft Type	Fokker F27 Mk 500 Friendship
Nationality	British
Registration	G-CEXF
Place of Accident	Just after takeoff from Jersey Airport, Channel Islands Latitude: 049° 12.4' N Longitude: 002° 12.6' W
Date and Time	5 June 2001 at 1049 hrs

Synopsis

The accident was notified to the Air Accidents Investigation Branch (AAIB) by the Jersey Tower ATC watch supervisor at 1400 hrs on 5 June 2001. The following Inspectors participated in the investigation:

Mr P T Claiden	IIC and Engineering
Mr P D Gilmartin	Operations
Mr R J Vance	Flight Data Recorders

Shortly after takeoff from Runway 27 at Jersey Airport on an empty positioning flight to Bournemouth with three crew members on board, an uncontained failure occurred to the left engine at an altitude of approximately 670 feet. This resulted in a sudden and complete loss of power from the left engine and a major fire external to the nacelle, but this was extinguished during the Engine Fire Drill as the Low Pressure (LP) fuel cock was closed. The aircraft completed a left hand circuit and landed uneventfully back on Runway 27. The engine failure was caused by high cycle fatigue (HCF) cracking of the High Pressure Turbine (HPT) disc. Five similar Dart HPT failures had occurred over the previous 29 years, the most recent at London Stansted Airport on 30 March 1998 to an engine fitted in a HS 748 - Series 2 aircraft, G-OJEM. (AAIB Air Accident Report No: 3/2001.) Prior to that accident, the failures had been attributed to a combination of turbine entry flow distortion and turbine blade wear and the engine manufacturer and the CAA had concluded that the likely period before recurrence of failure was such that additional remedial action was unnecessary. While the engine failure in G-OJEM was initially attributed by the engine manufacturer to the same causes as the previous cases, major difficulty was experienced in confirming the likely causes. Following the necessarily protracted study, testing and analysis by the engine manufacturer, the evidence collected then indicated that a small gap, under running conditions, between the seal arm abutment faces of the HPT and Intermediate Pressure Turbine (IPT) discs could result in high cyclic stresses being present in the HPT seal arm radius at the disc diaphragm, and that these stresses could result in high cycle fatigue (HCF) cracking.

As a result of this, a manufacturer's Service Bulletin (SB), Modification No 1946, was issued in April 2001 and this was mandated by the CAA as an Airworthiness Directive. This modified the HPT to ensure that a positive interference or 'nip' would exist between the HPT and IPT disc seal arm abutment faces, as this was found to significantly reduce such damaging cyclic stresses. The compliance date of this SB was '*not later than 31 December 2005*'. Following the HPT disc failure to G-CEXF, which had not yet been modified, the SB was changed to a cycles based requirement, essentially with the highest cycle discs being removed from service first, and with a compliance end date of 30 June 2004. Additional study has suggested a correlation may exist between the fit of the taper bolts, which clamp the three turbine discs together, and excessive seal arm wear found during routine overhauls, although insufficient evidence was available to determine the fit of the taper bolts on the HPT from G-CEXF. This has resulted in the manufacturer amending the relevant Overhaul Manual (OM) to take account of, amongst other process changes, the individual fit of these bolts to each turbine disc.

The investigation identified the following causal factors:

1. Minimal fatigue strength margin of the engine HPT disc resulted in it being susceptible to rapid cracking if subjected to vibratory excitation, such as resonance.

2. The abutment between the HPT and IPT discs probably resulted in a small gap being present between the seal arms while the engine was operating. This allowed sufficient reduction in the natural frequency of the turbine disc vibratory mode for it to be excited while operating within the normal speed range of the engine.

3. The protracted time taken following the G-OJEM event, due to the nature of the tests required to understand the cause of the failure, precluded the timely introduction of suitable preventative action aimed at avoiding recurrence prior to the HPT disc failure on G-CEXF.

4 Fuel leakage from a severed low pressure pipe, part of the engine bay fuel system, led to a major fire, external to the nacelle.

No further Safety Recommendations were made during the course of this investigation to those already made in AAIB Air Accident Report 3/2001.

1. Factual Information

1.1 History of the flight

The aircraft was operating its fourth sector of the morning, the first two sectors having been operated by a different crew. Prior to the first sector, the commander noted that the aircraft was experiencing a notable low frequency vibration, felt through the power levers and the airframe, just after engine start. A ground engineer was summoned on board the aircraft, and confirmed the presence of a vibration. At that time he suspected one blade of the propeller on the No l engine was slightly out of track at idle RPM. The engines were shut down and a further detailed external inspection was carried out, including a satisfactory audible check of the turbines and compressor when rotating each propeller by hand. The commander restarted both engines and, although the vibration was still present, it was much reduced. It was noted that the vibration disappeared whenever the engines were operated above idle power. The commander therefore satisfied himself that the aircraft was serviceable for the intended flights from Bournemouth to Jersey and return. Both sectors were completed normally with no further sign of the vibration.

On return to Bournemouth, there was a crew change. The off-going commander briefed the on-coming commander on the vibration experienced earlier in the morning, but no entries were made in the aircraft's Technical Log. After the event, the on-coming commander reported that he had not noted any unusual vibration at all during the outbound sector or the brief accident flight prior to the failure.

The on-coming commander was a Training and Fleet Captain for the F27 and was to conduct the first two sectors of Line Training with a new First Officer (FO), accompanied by an experienced 'screen' FO occupying the supernumerary seat. The trainee FO acted as non-handling pilot on the outbound cargo sector from Bournemouth to Jersey, which was completed uneventfully. He was the handling pilot for the positioning sector from Jersey back to Bournemouth. The aircraft taxied for departure from the full length of Runway 27 and, with the surface wind from 230°M at 8 kt, the takeoff progressed normally. The aircraft became airborne approximately abeam the Control Tower and the landing gear was retracted normally. In accordance with the Standard Operating Procedures (SOPs), when passing 400 feet agl (670 feet amsl), some 17 seconds after lift off, the first officer requested 'climb power' and the commander then proceeded to reduce the fuel trimmers towards the climb power setting.

At this point the crew heard a loud 'bang' and the left engine Fire Warning activated. The FO continued to control the aircraft, keeping it straight against the potential yaw and maintaining a climb profile. The commander initiated the engine fire drill for the left engine, which involved putting the propeller lever to the Feather position, pushing the Feathering Button and pulling the Fuel Shut-Off Handle. Shot 1 of the left engine Fire Extinguishers was then activated. A few seconds later, the Fire Warning ceased, so the commander elected not to use Shot 2. He instructed the first officer to climb to 1,500 feet and make a left turn to conduct a visual left hand circuit to land back at Jersey. A MAYDAY call was made to Jersey Tower informing them of the problem and the commander's intentions. On the downwind leg, the commander took over the handling of the aircraft and conducted a single engine approach and landing on Runway 27, which was carried out uneventfully. The aircraft landed using Flap 26° at 1050 hrs.

The Airport Fire Service (AFS) had been notified and was in position when the aircraft landed. They confirmed that the fire had been extinguished and so the aircraft was taxied with an AFS escort to the parking area where it was shutdown normally. During the shutdown process, the Fuel Shut-Off Handle was inadvertently returned to ON causing some fuel to drain onto the ground. This was dealt with by the AFS personnel who also advised the crew of the spillage, following which the handle was returned to the OFF position.

1.2 Injuries to persons

Injuries	Crew	Passengers	Others
Fatal	-	-	-
Serious	-	-	-
Minor/none	3	-	-

1.3 Damage to aircraft

Aircraft damage consisted of severe disruption and fire damage to the No 1 powerplant, landing gear doors and engine nacelle, and minor damage to the left Main Landing Gear (MLG) and propeller, Figure 1.

1.4 Other damage

Minor engine debris was scattered over farmland and two houses, but with no discernible resulting damage. The larger of two segments of the HPT disc landed in a field adjacent to one of the houses, causing no damage; the smaller segment was not recovered. There were no reports of damage to property from such an object in the local area.

1.5 Personnel information

1.5.1	Commander:	Male, aged 54 years	
	Licence:	Airline Transport Pilot's Licence	
	LPC/OPC renewed:	30 November 2000	
	Line check renewed:	26 February 2001	
	Medical certificate:	Class 1, renewed 14 May 2001	
	Flying experience:	Total all types:	7,300 hours
		Total on type:	1,200 hours
		Total last 28 days:	10 hours
		Total last 24 hours:	1 hour
	Previous rest period:	In excess of 24 hours	

1.5.2	First Officer:	Male, aged 24 years	
	Licence:	Commercial Pilot's Licence	
	Type rating issued:	23 May 2001	
	Line check renewed:	N/A	
	Medical certificate:	Class 1, renewed 19 October 2000	
	Flying experience:	Total all types:	257 hours
		Total on type:	4 hours
		Total last 28 days:	4 hours
		Total last 24 hours:	1 hour
	Previous rest period:	In excess of 24 hours	

1.6 Aircraft information

1.6.1 General information

1.6.2 Aircraft weights

Manufacturer:	Fokker
Type:	F27 Mark 500 Friendship
Aircraft Serial No:	10660
Year of manufacture:	1983
Certificate of Registration:	G-CEXF, issued on 2 April 1997
Certificate of Airworthiness:	Valid until 30 June 2003
Engines:	2 Rolls Royce Dart RDa7 Mark 532-7 turboprop engines
Total airframe hours:	17,000 hours (20,000 flight cycles)
Maximum Take off weight	20,410 kg
Actual Take off weight	13,769 kg
V1/ VR/ V2	102/104/105 kt
Maximum Landing Weight	19,051 kg
Vref (Flap 26°)	95 kt

1.6.3 Aircraft Description

1.6.3.1 General

The aircraft is of conventional layout with a high mounted wing and is constructed mostly from aluminium alloy. The F27 prototype first flew in November 1955 and the Mark 500 version in November 1968. G-CEXF was manufactured in 1983 and was being operated as a freighter although, at the time of the accident, it was not carrying cargo. The F27 ceased production in 1985 and in 1996 the manufacturer, Fokker, ceased trading. The world wide fleet of Fokker products are presently supported by a new company, Fokker Services BV, based in the Netherlands, who also hold the type certificate for the F27 aircraft.

At the time of the accident it was estimated that some 680 aircraft powered by variants of the Dart engine, including some 250 F27 aircraft, were still operating worldwide.

1.6.3.2 Flight controls

All control systems are operated manually by two-way cable systems, with the exception of the flaps and aileron trim tabs, which are operated electrically. Spring and balance tabs are used to provide an aerodynamic boost in order to reduce operating loads, and trim mechanisms are provided on all three control systems. The cables for these systems, and the trim systems for the rudder and elevator, all run along the upper forward fuselage in close proximity to each other where they all cross the plane of rotation of the HPT discs.

1.6.3.3 Powerplant

The engines are located forward of, and generally below the wing, in wing mounted nacelles, Figure 2. Each engine is carried on a tubular steel framework that connects mountings on the engine compressor casing to support members attached to the wing torque box. The engines are covered by hinged aluminium alloy cowl panels. The main landing gears and other equipment are housed behind the engines and an equipment bay is located in the upper part of the nacelle behind the engine. The engine jet pipe passes beneath this bay and exhausts to the outboard side of the nacelles. The installation includes a water/methanol injection system to provide additional power, known as 'wet power', with a reservoir being housed at the rear of each nacelle. The powerplant controls include a system which, when armed, in the event of loss of thrust from an engine on takeoff will automatically select full wet power on the other engine. Each engine drives a four bladed Dowty constant speed metal propeller.

1.6.3.4 Fuel System

All fuel is stored in integral outer wing tanks and collector tanks, one in each nacelle. Fuel is transferred by gravity from two outlet fittings at the inboard end of each outer wing to a collector tank located behind the engine nacelle firewall. Two canister mounted boost pumps fitted in each collector tank transfer fuel under pressure through non-return valves to a manifold, from where it passes through the manually operated 'emergency shutoff valve' (LP cock). From here the fuel passes through the fuel flow meter and on to the engine through a fuel heater located just forward of the plane of the HPT at the lower left quarter of the engine. Operation of the LP cocks is via a system of linkages and cables from two 'T' handles located at the top of the central flight panel in the cockpit.

Both sets of cables also run close together in the upper section of the fuselage where they cross the plane of rotation of the HPTs.

1.6.3.5 Engine bay fire detection system

A fire zone around the hot section of the engine is formed by a nacelle firewall, and by an engine firewall mounted on the aft face of the engine compressor casing. The cowl panels, when closed, bear against seals on each of the firewalls. An electrically operated fire warning system is provided for each engine and is divided into two systems: a primary system to detect a fire external to the engine and a secondary system for signalling an internal engine fire. The primary system comprises a capacitive Fire Wire type, and is routed in the hot section around the engine covering all critical areas, but also extends behind the nacelle firewall to cover the fuel system components installed in this area. When the system is triggered the control unit will activate a flight deck firebell and a red engine fire warning light on the flight deck glare shield panel. The secondary system comprises a temperature sensor installed in the engine breather outlet pipe. It is intended to give warning of an advanced state of mechanical failure in flight, in or around the region of the main high speed bearings, as any such failure will result in an increased breather outlet temperature.

1.6.3.6 Engine bay fire suppression system

An electrically operated two shot fire extinguisher system is provided for each engine. It comprises two cylinders each filled with some 12 lbs of BromoChlorodiFlouromethane (BCF) extinguishant pressurised by dry nitrogen, and a cartridge and pressure discharge indicator for each cylinder. When operated from a selector switch by the crew, a cylinder will discharge its contents via a pipeline through two spray rings fitted around the engine and two spray pipes fitted in the ventilation air scoop on the side of the nacelle. The spray rings around the engine are made from stainless steel, one is arranged around the intake cowling to spray aft to blanket the engine fuel system, the other is fitted just aft of the engine firewall and also sprays aft to blanket the combustion chambers and nozzle box. Effective fire suppression with this extinguishant relies upon a relatively good airtight seal being formed between the engine and nacelle firewalls and the cowling doors. The extinguishant cylinders can be fired one after the other by the selector switch on the glare shield panel, but a cylinder on the left engine, for example cannot be used with a fire in the opposite engine. The holes punched in the cowling doors by the failed sections of the HPT would have seriously reduced the effectiveness of the extinguishant in supressing any fire within the cowling.

1.6.4 Engine Description

1.6.4.1 General

The left (No.1) engine was a Rolls Royce Dart RDa7 Mark 532-7, Serial Number (s/n) 14845.

This model of Dart engine is a single shaft turboprop engine with a 2-stage centrifugal compressor, seven straight flow cannular (can) combustion chambers and a 3-stage turbine. Reduction gearing (0.0929:1) provides output power to the propeller. Dart design started in 1945 and production ceased in 1987, with a total of approximately 7,100 engines delivered. Some 1,680 engines remain in service. Early versions, with a 2-stage turbine and designed to produce around 1,000 shp, were designated as the RDa3 and RDa6 Series and initially entered civilian service on the Vickers Viscount. Developed versions were used in a number of aircraft types. The RDa7 and RDa10 Series (approximately 1,700-2,000 shp) were developments having the different, three stage, turbine; they entered service in 1958 and 1965 respectively.

The Mark 532-7 engine is a member of the RDa7 Series and was type certificated in 1961. At the time of the accident the total numbers of RDa7 and RDa10 engines produced, were approximately as follows:-

	RDa7	**RDa10**
Total Number Produced	3,500	650

Military versions of these series were designated as the RDa8 and RDa12, and were produced in significantly lower numbers.

The general layout of the Dart Mark 532-7 is shown in Figure 3. The main structure consists of, from front to rear, a reduction gear, compressor, intermediate and nozzle box casings and exhaust. Take-off power rating is listed as 15,000 rpm spool speed with a maximum allowable Turbine Gas Temperature (TGT) of 810°C (dry) and 860°C (wet), limited to 5 minutes. These limits were also specified, for an unrestricted period, for Max Continuous and Intermediate Contingency ratings respectively but were to be used only in emergency situations and/or for limited crew training. Permissible overspeed was listed as 17,000 rpm for 20 seconds and maximum overtemperature as greater than 860°C, but less than 950°C, for less than five seconds.

1.6.4.2 Combustors

Air delivered from the compressor passes through a diffuser ring and via outlet elbows into the combustion chambers. These are numbered clockwise, viewed from the front of the engine, with No 1 fitted to the top outlet elbow, and are skewed relative to the engine's longitudinal axis. Each combustion chamber consists of a fabricated welded flame tube with fixed swirl vanes fitted at the upstream end, fitted concentrically within a generally cylindrical casing. The chambers are suspended between the diffuser outlet elbows and discharge nozzles in the nozzle box. Interconnectors join the casings and flame tubes of each combustion chamber to its neighbour.

1.6.4.3 Burners

A fuel burner fitted in each combustion chamber locates in a central ring in the flame tube swirl vanes and is attached by an integral fuel inlet tube to the combustion chamber casing. The burner consists of a nozzle shroud screwed onto a threaded body, containing a spring-loaded screw-thread strainer and a swirler assembly. In the original 'air-washed' burners, the shroud passes high pressure air only, from the entry nozzle area of the combustion chamber. This arrangement could be changed to a 'fuel-washed' standard by an optional Rolls Royce modification, issued in 1966, that involved internal modification to the burner to provide a fuel bleed into the shroud annulus. The intention was to reduce the rate of carbon build-up on the exterior of the fuel discharge nozzle in service. Engine s/n 14845 was fitted with air-washed burners and the majority of Dart engines in service at the time of G-CEXF's accident had air-washed burners. Different design flow rates, known as 'burner biasing', had been used for the individual burners in an engine set since it was found, in the 1950s, that HP Nozzle Guide Vane (NGV) deterioration tended to occur at particular circumferential positions. This was due to annular variations in the compressor delivery flow and the biasing was intended to give uniform turbine entry temperatures. The correct position for a particular burner was indicated by a type number stamped on the burner inlet tube and by a varying number of flats machined onto the inlet tube thread.

1.6.4.4 Turbine Assembly

The turbine assembly is housed in the nozzle box and connected to the compressor and reduction gearbox by separate concentric shafts. It comprises, from front to rear, High, Intermediate and Low Pressure Turbine (LPT) stages, Figure 4. Each stage consists of a turbine disc with nimonic steel blades attached by means of fir-tree sockets, known as 'buckets', broached in the disc periphery.

The three turbine discs are clamped together by five taper fit bolts passing through all three discs and a further five taper bolts which only pass through the HPT and IPT discs, Figure 4. The bolts also connect the turbine assembly to the turbine-driven shafts. The assembly is located by a thrust ball bearing acting on the turbine outer shaft immediately forward of the turbine attachment flange, and a plain bearing at the forward end of the shaft.

1.6.4.5 Turbine Blades

The turbine blades have integral platforms at the blade roots and shrouds at the tips; HPT and IPT blade shrouds incorporate tip seals. The design clearance between the platforms and shrouds of adjacent HPT blades is 1.25 to 2×10^{-3} inches. Wear in engine service tends to increase this clearance and this may result in an adverse effect on the vibratory characteristics of the turbine assembly.

The HPT has 131 blades, each weighing 0.04445 lb, which are retained in the disc buckets by lock plates. The HPT blades are a high cost component of the engine and excessively worn blades are permitted to be repaired under a Dart Overhaul Manual Dart Repair Scheme (DRS) by weld depositing material onto the worn face(s), followed by a machining operation to re-establish the required dimension. The clearance is not measured directly but inferred from gauge checks of individual blades.

The original scheme (DRS 297, introduced in 1960) was concerned only with blade platforms and specified repair of one face, on either the concave or convex side of the blade, depending on which appeared to have suffered the most wear. A further scheme, DRS 611, was introduced in November 1975 to specify platform repair for aluminised blades. DRS 611 was amended in June 1981 to give instructions for optional shroud repair as well. In October 1992 the Inspection Section of the Engine Overhaul Manual was amended to require that platforms and shrouds be inspected at overhaul and that wear in either location in excess of 2×10^{-3} inch be rectified in accordance with DS 611. This followed testing in around 1990 showing that increased shroud or platform gaps could lead to increased stresses in the blade root area.

1.6.4.6 HPT Disc

The HPT disc is machined from a forging of a 12% chromium, niobium, creep-resistant steel alloy (Firth Vickers (FV) 448), produced since the mid 1960s by a single vacuum melt process (Mod 1171). It is coated with corrosion resistant paint (type PL101). The disc design had not been subjected to any significant modification since the early 1960's when the radii associated with the blade buckets were increased (Modifications 839 and 911).

The disc is 15.24 inches in diameter, with the cross section shown in Figure 4. It has a hub area around five inches in diameter with a central bore and 10 bosses near its outer edge bored and reamed to accept 'taper' bolts. Outside the hub region the disc thins to a tapered diaphragm section with a thickened outer rim that contains the blade buckets. Seal arms (rings) integral with the diaphragm, one on the forward and one on the aft face, form the rotating parts of labyrinth seals that control internal cooling and oil containment airflows. The forward and aft rings have outer diameters of approximately 10.3 inches and 9.0 inches respectively. The specified blend radius between the inner edge of the aft seal ring and the diaphragm is 0.10 to 0.15 inches.

Similar seal rings are formed on the IPT disc. With the turbine assembly clamped together, the gap (nip) between HPT and IPT seal rings at room temperature was designed to be zero, with a tolerance of +/- 1 x 10^{-3} inches. The nip between these rings closes up by approximately 1.3 x 10^{-3} inches when the engine is running and up to temperature and, therefore, if the turbine assembly is built to within drawing limits, a positive nip should always exist between these seal arms under running conditions. The IPT to LPT seal arm rings design fit is an interference of 2 to 4 x 10^{-3} inches. Cooling airflows are directed across the forward and aft faces of the HPT disc.

1.7 Meteorological information

The METAR, information 'November', timed at 1020 hrs, was:

Surface wind	230°/10 kt
Visibility	25 km
Cloud	CAVOK
Weather	CAVOK
Temperature/Dew point	+17°C / +09°C
QNH	1018 mb

1.8 Aids to navigation

Not applicable

1.9 Communications

At the time of the accident the aircraft was in contact with the Jersey Tower controller on frequency 119.45 MHz.

1.10 Aerodrome information

The runway in use was Runway 27. Its physical characteristics are:

Magnetic Heading	265°
Dimensions	1,706 x 46 metres
Surface	Asphalt
TORA/TODA	1,645/2,469 metres

1.11 Flight Recorders

1.11.1 Cockpit Voice Recorder (CVR)

The aircraft was equipped with a Fairchild A-100A CVR and the recording consisted of three channels of good quality audio information. The recorded channels were the Captain's and the First Officer's hot microphone channels and the cockpit area microphone. The recording started while the aircraft was on the ground in Jersey as the crew were in the cockpit preparing for the flight to Bournemouth, and ended when electrical power was removed from the aircraft subsequent to its landing back at Jersey. The CVR recording was subjected to a spectrographic audio analysis and this enabled a time history of the rpm of both engines/propellers to be extracted, Figure 5. These were examined for any abnormalities, but none were found.

1.11.2 Flight Data Recorder (FDR)

The aircraft was equipped with a Sundstrand Universal Flight Data Recorder (UFDR) model 980-4100-GXUS and was configured to record six channels of data; pressure altitude, indicated airspeed, magnetic heading, normal acceleration, flap position and VHF radio keying. The normal acceleration parameter was found not to be operational, but this was a known defect and procedures had been put in place by the operator to investigate and rectify the fault. No data on the aircraft's attitude or on the behaviour of its engines was recorded on the FDR and, as a result, it was not possible to identify the time of the engine failure from this data alone.

The CVR and FDR recordings were time correlated using the VHF transmissions from the aircraft. From this it was possible to identify the point at which the engine failed on the FDR time history and a plot of the recorded FDR data is at Figure 6. From this it was determined that the engine failure occurred 17 seconds after takeoff, by which time the aircraft was approximately 400 feet above the runway (670 feet amsl). The absence of recorded aircraft attitude prevented any analysis of the aircraft's behaviour when the engine failed or during the subsequent circuit and landing.

1.12 Aircraft and site examination

1.12.1 General

Examination showed that the left engine had suffered severe non-containment damage in the region of its turbine assembly, and this was associated with serious mechanical and fire damage to the left nacelle.

1.12.2 Aircraft Examination

Examination of the aircraft at Jersey Airport revealed all damage to be confined to the left engine and its nacelle, left propeller and left main landing gear (MLG). The left engine hinged cowls had remained in place but exhibited two major exit holes, one each in the outboard and lower cowls, in line generally with the plane of rotation the HPT. Severe distortion was present to one of the steel struts of the engine support frame adjacent to the lower left attachment at the firewall, where the attachment fitting itself was distorted and no longer connected to the engine support frame. This distortion was also in line with the plane of the HPT. The mechanical linkages to the engine HP fuel cock, power and propeller controls, which also pass through this region, had been severed. This mechanical damage was consistent with a high speed (and hence high energy) failure of the HPT and the subsequent non-containment of one or more sections. The low pressure fuel pipe from the nacelle firewall to the engine had also been severed in this region and it was readily apparent that fuel had discharged from the open end into the slipstream, caught fire and caused severe heat damage to the structure of the nacelle. Lesser damage to the left MLG had resulted as this remained retracted for the duration of the fire. The cowls were generally free from any effects of a sustained fire and the wing showed no evidence of fire damage at all.

1.12.3 Accident Site

Following the accident, internal parts of the left engine and small sections of its cowling were discovered on the ground and across the roof of a house some 200 metres beyond the end of the runway, slightly to the left of the extended runway centreline. These parts comprised mainly sections of HPT blades and Nozzle Guide Vanes (NGVs). It was apparent from the relative positions of the damage to the engine and cowlings that the HPT disc itself had departed the engine in two sections and that one section had been ejected horizontally to the left, the other also to the left but downwards along a path some 15° from the vertical. An assessment of this damage also suggested that it was the smaller of the two sections which had been ejected horizontally. The nature of the landscape, which included hedgerows, ditches, ponds, farmyards, and soft sandy soil, was not conducive to a thorough search. Initially, a visual search was conducted over several days for the HPT sections in areas where it was estimated they might have impacted the ground, but initially to no avail. Several days later, metal detectors were employed and the main section of the HPT disc was discovered some 12 inches below the surface of a field in the area containing HPT blades. The smaller of the HPT disc sections was not recovered.

1.12.4 Nacelle

The left engine cowl doors had remained in place but ragged holes had been punched in the outboard and lower doors in the region of the HPT and it was evident that this had been caused by impact from engine debris. There was evidence of only slight fire damage to the inside of these doors but major fire damage was present to the lower section of the nacelle from the firewall to its aftmost point. In many areas, holes had been burnt through the nacelle skins.

1.12.5 Powerplant

The hot section of the engine had been severely damaged, with the nozzle box/turbine module burst open in the plane of the HPT over a 150° arc around the left and lower side. The HPT was not present within the engine but the severely damaged IPT, and a surprisingly lightly damaged LPT, remained within the module. The intermediate casing had fractured over 360°, detaching the nozzle box from the rest of the engine, and this had rotated approximately 25° in the same direction of rotation as the engine spool. The nozzle box rotation had caused the combustors to disengage from the nozzle box discharge nozzles and/or the compressor diffuser outlet elbows, and from the combustion chamber interconnect tubes.

Detailed examination of the engine was undertaken, and it was disassembled by the engine manufacturer under AAIB supervision. Damage had occurred to all turbine stages, progressively decreasing rearward through the engine, and all 10 (taper) bolts, which clamp the three turbine discs to the engine shaft, had failed. The bolt failures, the fracture of the intermediate casing and the rotation of the nozzle box were all consistent with the effects of a sudden HPT disc failure. This induced massive out of balance forces, resulting in contact of the turbine stages against the nozzle box casing and NGVs, whilst the engine spool was rotating at high speed. With the exception of the HPT disc, no evidence was found to indicate turbine or combustor overtemperature, oil starvation, bearing failure, or any other pre-accident defect with the engine.

1.12.5.1 HPT disc

The major portion of the HPT disc was recovered from the area beyond the end of the runway together with a few fragments of cowling, HPT blades and HP NGVs. Most HPT blades had been fractured near to their fir tree attachments and the damage was too severe to allow any estimation to be made of blade platform or shroud gaps.

The disc portion recovered measured some 15 x 11.5 inches overall and weighed 7.973 kg (17.57 lb), Figure 7. The main part of the disc fracture was circumferential, following the inner radius between the disc aft seal arm and the diaphragm around approximately 40% of the circumference. There was an approximately radial fracture at one end, and a secondary circumferential fracture at the other, which also led to an approximately radial fracture. The face of the main circumferential fracture was mostly flat, and generally banded with a series of light-coloured conchoidal markings. The fracture face had not suffered any appreciable impact damage and only minimal degradation by oxidation. The secondary fracture was also generally flat and similarly marked, Figure 8.

Detailed laboratory examination was carried out by the engine manufacturer in conjunction with the AAIB, using optical microscopy, scanning electron microscopy and transmission electron microscopy of replicas of the fracture. Hardness checking was also carried out. The evidence indicated that the circumferential cracks had been caused by high-cycle fatigue (HCF) and, although corrosion pitting was present on the face of the disc in the region of the two crack faces, their likely origins were not associated with any such corrosion. Pits generally had depths of between 2.2 and 5.5 x 10^{-3} inches and were similar to numerous others associated with fairly extensive light corrosion present on the disc surface, particularly in the aft seal ring inner radius area.

The manufacturer and an overhaul agency reported that in-service corrosion of this type and extent was quite usual.

There were no signs of any other anomaly at the origins of the two related fatigue cracks, such as a material defect, inclusion or forging lap, and it was judged that any such defect would have been apparent. The fatigue had propagated circumferentially over a distance of approximately seven inches, mainly against the direction of disc rotation, and forward over approximately 80% of the section thickness. Adjacent to the primary origin, in the direction of rotation, a secondary area of fatigue cracking was present with an origin on the front face of the disc diaphragm. Overload fracture of the remaining part of the section and (generally) radial tear off in tensile rupture at both ends had then completed the fracture to release the smaller section of disc. Detailed inspection of the larger disc segment failed to find any additional cracking. The macro and microstructure of the disc material was normal and hardness was satisfactory, with measured values in the range 309-314 Vickers Hardness Value (HV) against a specification requirement of 299-334 HV. There were no signs of gross overstress, which would have been likely to result in multiple cracks. Evidence that the failure was transgranular and without creep related features, as shown by a lack of plastic deformation near the origins, indicated that a low cycle fatigue (LCF) mechanism had not been involved and that the disc had not failed due to over-temperature.

The main fatigue fracture surface exhibited between 35 to 50 (or possibly more) lighter coloured narrow bands, with darker regions between. It was concluded that each light region was an arrest band that represented a divider between separate periods of vibration that had been sufficient to cause crack growth in HCF during the propagation stage. It was not possible to correlate the arrest bands to flight cycles. On the strikingly similar fracture faces on the HPT from another HPT disc failure on G-OJEM (see paragraph 1.16), some indistinct striation features were evident in the HCF areas between the bands, but the available detail was insufficient to allow an accurate striation count over the fracture surface. The best estimate of striation densities, in that event, was in the order of 20,000 per millimetre, and HCF load reversal cycles involved in the propagation of the crack was estimated to be in the order of one to five million.

In the case of G-CEXF, evidence was found of fretting/galling on the HPT to IPT disc seal arm abutment faces, and the location of this damage relative to the fatigue crack is illustrated in Appendix C-3. The seal arm face on the IPT disc showed generally light galling, judged not to be of significance in a loss of dimension context, but this was visibly more severe over the arc between bosses 1 to 5. Light galling was also present between bosses 7 and 8 on both the HPT and IPT discs, although the loss of the smaller disc segment precluded examination of the seal arm face between, approximately, boss locations 3 and 6. Expert examination by tribologists from the manufacturer indicated that such wear could be consistent with the results of a 'rolling contact' type of mechanism, (see paragraph 1.18.3.6). The taper bolt hole location bosses were examined to determine if any obvious wear was evident on the contact surfaces. All these contact surfaces on the forward side of the disc were free from any evidence of wear. On the rear side of the disc, five of the bosses surfaces were damaged, and this precluded comment on their pre-failure condition. Optical binocular microscopic examination of the remaining four found them to be free of evidence of significant wear, although some patchy black, thin deposits were seen on several. The thickness of the material was assessed as being less than 0.1×10^{-3} inches; analysis of this material found it to be consist of oxides of the disc material. However, residual machining marks were still clearly discernable on the boss faces, indicating that there had been little or no relative motion between the discs.

1.12.5.2 Combustors

All seven combustor cans were recovered from the engine and none appeared to have sustained any damage as a result of the HPT failure. On examination at the manufacturer's facility at East Kilbride, all were judged to be in good condition, except the No 7 can, which exhibited a small heat distressed area to a section of the first cooling ring. This was associated with several small cracks, which linked through air holes in the air casing, but the type and extent of this damage was judged as minor and typical of a proportion of the cans seen at overhaul. Such minor damage is known not to give rise to any asymmetry in the gas flow between cans at the entry to the turbine module. The aircraft operator operates a policy of replacing the cans at 2,000 hour intervals, at which time they are removed as complete assemblies. Seven overhauled cans were fitted in October 2000 but the No 1 can was replaced one month later. This was due to the fact that it was a can owned by different operator and shipped in error to Channel Express by the overhaul agency, and subsequently returned.

1.12.5.3 Fuel Burners

The seven fuel burners were removed and examined and then checked on a test rig for flow rate and flow pattern. All the burners were of the air washed type, all passed the required leakage check at 1,500 psi and were found to be free from streaking. Each burner was tested on a production rig at the manufacturer and spray patterns, fuel and airflow rates were measured. Although several of these flow rates were slightly outside overhaul/release limits, it was reported that this was not atypical for in-service units. Data was also obtained for the burner fuel flows of the set removed, in October 2000, with the previous combustor cans. No significant deviations were noted.

1.13 Medical and pathological information

Not applicable

1.14 Fire

These was some evidence that a flash type of fire had occurred within the hot section of the engine compartment and this probably resulted from the disengagement of the combustor cans as the nozzle box was forcibly rotated. The main fire occurred as a result of engine debris severing the low pressure fuel pipe between the firewall and the fuel heater at the lower left side of the engine. It was evident that fuel had streamed aft, outside the nacelle, and had ignited to form a torching flame in the slipstream. The effects of this flame were evident along the length of the lower left side of the nacelle, where many skin panels had been burnt through and where heat damage had been caused to structural elements within the nacelle. The main landing gear had remained retracted for the duration of the fire and suffered relatively minor heat damage at the lower end of the drag strut.

1.15 Survival aspects

An engine failure after takeoff is one of the eventualities that flight crews are trained for and regularly examined upon. As such they would be expected to carry out the appropriate procedures correctly. This was the case with G-CEXF. Despite a fire external to the nacelle, which was extinguished during the fire drill, it was possible to continue to fly the aircraft around a left hand circuit and carry out a successful single engine landing. With only the three flight crew members on board, and without the necessity for an emergency evacuation, there were no survival aspects relevant to this accident.

1.16 Tests and research

The failure of the HPT disc on G-CEXF was remarkably similar to several previous failures, the most recent of which occurred to a HS 748 Series 2 aircraft, G-OJEM, at London Stansted Airport, on 30 March 1988 (AAIB Air Accident Report 3/2001). Following that event, a lengthy period of re-evaluation of the turbine assembly was conducted by the manufacturer in conjunction with the AAIB. This included testing and analysis of the HPT and its interaction with the IPT, detailed reassessment of previous similar cases and previous test data, re-analysis of the HPT fatigue strength and vibrational characteristics and the development of a Finite Element Analysis (FEA) mathematical model of the disc and its vibrational behaviour. In addition, a substantial programme of testing on a similar engine was set up by the manufacturer, aimed at measuring the stresses experienced by an operating HPT disc, with various configurations of engine components and operating conditions considered to be relevant. This work was directly relevant to these two most recent HPT disc failures, enabling the one from G-CEXF to be quickly understood. Since the accident to G-CEXF, additional confirmatory work, and a re-assessment of the turbine attachment taper bolts installation, has been carried out. Although reported upon in detail in AAIB report 3/2001, much of the outcome of this previous work is relevant to the failure associated with G-CEXF and, therefore, is included in this report.

1.17 Organisational and management information

At the time of the failure on G-OJEM, the method by which the manufacturer conducted their internal investigations into engine failures was essentially project based. The group within the company responsible for an engine type, or group of engines, in this case the Small Engine Division (Dart, Spey and Tay engines), would conduct their own investigations, liase with external agencies and authorities, and report to the Chief Airworthiness Engineer for that group. During the investigation into the G-OJEM failure, a necessarily large and time-consuming technical effort aimed at understanding the problem was made by the manufacturer. There was also an apparent diffuse approach adopted by the manufacturer to accident or incident investigation at that time and associated problems in co-ordination were experienced by the AAIB. Responsibilities for aspects of the investigation were divided between an appreciable number of sections and individuals within the company. Both the personnel and the structural arrangements within the company changed extensively over time and no single point contact with the company was designated. However, early in 1999, the decision to establish a company wide 'Single Best Approach' to investigation was taken by the manufacturer and, as a result, various objectives were identified, including the need to:-

Provide timely investigation support to authorities worldwide.

Assist authorities in expeditiously making accurate safety findings.

Identify any engine involvement and need for continued airworthiness action.

Gather information for safety risk management.

It was decided to base the company wide system on two centres, with respect to the investigation of accidents/incidents, with one based at Bristol in the UK and the other in North America at Indianapolis. It remains the declared policy of the manufacturer to offer participation in all investigations, under the direction of national investigative authorities, of accidents/incidents involving aircraft where a fatality or serious injury has occurred, or where there is a potential that the manufacturer's products may have contributed to the cause. This is now achieved by the establishment of a team already experienced in accident investigation at each centre who provide a lead investigator following any relevant event. Such a designated person is the focal point for all investigation team communication, and is empowered to determine investigation support requirements on a priority basis and commit company resources, and be responsible for the timely production of a final report to be submitted to the government investigation authority. The same person, through the heads of such centres, reports within the company to the Chief Airworthiness Engineer. The accident to G-CEXF was the first to be investigated using the new company investigation system and this was shown to be effective.

1.18 Additional information

1.18.1 Engine History

G-CEXF was built in 1983 and delivered to Egypt in 1984, where it was operated until 1996 as SU-GAE. Engine s/n 14845 was installed as a new engine at initial build and remained with the aircraft until the time of the accident. Significant points in its history are as follows:

- Engine last overhauled in November 1995 at the manufacturer's facility in East Kilbride, and released as a RDa7 535-7
- Installed on F27 SU-GAE for 10 hrs
- Re-rated as a 532-7 engine and installed on Channel Express F27 G-CEXF in May 1996 (same aircraft, new operator)
- Installation ground run and first flight in June 1997
- Operated for 3,700 hrs between June 1997 and June 2001
- Nine power checks carried out over this period, the last one on 4 May 2001
- HPT disc failure on 5 June 2001

Note: It was considered that the presence of the fatigue crack, prior to the HPT disc failure, would not have affected the normal operation or performance of the engine.

1.18.2 Component Lives

The approved lives for the relevant components at the time of the accident, as determined by the engine manufacturer and approved by the CAA, were:

Component	Approved Life
Engine	6,000 hours time between overhaul (TBO), plus a 10% extension agreed by the CAA
HPT Disc	20,000 cycles Declared Safe Life (Group A component)
HPT Blades	25,000 hours (recommended)
Combustors (with Burners)	3,000 hours TBO (2000 hours as a policy decision by the operator)

1.18.3 HPT disc

1.18.3.1 HPT disc history

The subject HPT disc was machined, by IHI in Japan, from a forging supplied by Rolls Royce between 1981/1982. As a fully machined part, it was received back to the engine manufacturer in January 1984 as p/n RK 33466, s/n LW 430 and, after inspection, was transferred to their East Kilbride facility in February 1984. The HPT disc part number was subsequently changed to RK45565 following minor modification and allocated to the subject engine, s/n 14845.

1.18.3.2 Disc Crack Inspection Criteria

Turbine disc inspection/check procedures (Rolls Royce Dart Overhaul Manual, Section 72-6-1) defined permissible crack lengths in the blade attachment grooves and serrations, but required rejection of the disc if any other cracking were to be found.

1.18.3.3 HPT Disc Environment

Operating temperatures, pressures and loading experienced by the HP turbine disc had been predicted by the engine manufacturer from rig test results and analysis. The most severe conditions were generally at take-off power and values for takeoff at Sea Level International Standard Atmosphere (SL ISA) conditions are given below. Temperatures during transient conditions, during the period when take-off power had been set and engine conditions were in the process of stabilising, and for the 'soaked' (ie, temperature stabilised) condition were also predicted. Relevant predicted values were:

LOCATION	**TEMPERATURE - °C**	
	Transient	Soaked
Rim	430	430
Rear Seal Ring Inner Radius	242	319
Bore	168	275

Internal air pressures on the disc are in the order of 60 psi for the inner part of the forward face, 47 psi for the outer part, and 42 psi for the aft face, with the engine at maximum power.

The predominant steady-state disc loading consists of centrifugal forces applied by the blades, the taper bolts and the disc's own inertia. With the loading values predicted, the resultant radial force on a disc half is estimated to be in excess of 173,000 lbf. Further forces, in the axial direction, result from pressure loading on forward and aft faces of the disc and from taper bolt clamping loads. In addition, axial and radial loads can be applied to the tip of the rear seal ring by its contact with the forward seal ring of the IPT disc; these would be dependent upon the combination of dimensional tolerances and disc deformation under the effects of pressure, temperature and centrifugal loading. Aerodynamic forces applied to the disc by the blades are predominately circumferential, due to aerodynamic loads, but also have an axial component. The disc can also experience forces from blade bending moments, both in and out of the plane of the disc, and blade torsion. Fluctuation in the blade loading, such as could result from inlet flow distortion, can also apply out-of-plane alternating forces to the disc rim.

1.18.3.4 Disc Life

Information from the engine manufacturer indicated that the HPT disc design had included considerations of static and fatigue loading strengths. The static strength was concerned with the magnitude of the steady loading stress that could be withstood without failure. HCF considerations related to fluctuating loads, and the characteristic that the repeated application and removal of a load that induced maximum stresses well below the maximum static strength value, could cause fatigue cracking to initiate and propagate. LCF considerations concerned relatively infrequent load fluctuation, often those that normally occur once per flight cycle, such as turbine disc loads that would be expected to reach their maximum with take-off power set. In this context, LCF was considered to cover the <100,000 load reversal regime. HCF related to much more frequent load fluctuation, such as the load reversal that can occur with each vibration cycle under vibratory conditions, and concerned the >100,000 load reversal regime.

Based on the above, the engine manufacturer had determined a 'Declared Safe Life' (DSL) for the HPT disc in the RDa7 engine of 20,000 flight cycles (18,000 flight cycles for the RDa10), with the fatigue strength of the central bore originally considered to be the critical area. The DSL was based on a 'Predicted Safe Life' (PSL) that was determined by endurance cyclic rig testing of a production standard disc at peak stresses and temperatures, that equalled or exceeded those experienced in engine service. The nominal cycle parameters were intended to represent the loading experienced during a generalised flight cycle, and thus addressed LCF considerations. The manufacturer operated its normal practice for the design of critical components, whose failure could hazard an aircraft, of applying a number of factors to the test results to allow for the inherent scatter experienced in fatigue mechanisms and to provide a safety margin. This was intended to meet the applicable airworthiness requirements, that the probability of a hazardous failure should be Extremely Remote ($<10^{\sim7}$/engine flying hour (efh)) for a 'Class A' component. The effect of corrosion on the fatigue strength had been estimated by specimen testing. It was originally concluded by the manufacturer that corrosion would reduce the fatigue life but, with typical corrosion pit depths, not to the extent of needing to reduce the DSL. Subsequent testing studied the fatigue strength of the radiused area between the rear seal ring and the diaphragm with simulated corrosion pits present of 5×10^{-3} inches deep. A PSL of 20,000 cycles was determined, providing zero margin from the DSL. However, following further specimen testing, the engine manufacturer reported their revised conclusion in May 2001, that corrosion pitting would, in fact, appreciably reduce the allowable alternating stress in the aft seal ring inner radius area that could be tolerated for an extended fatigue life. HCF had been generally addressed by consideration of the vibratory stress characteristics determined by strain gauge testing. The design of the HPT included consideration of HCF of the blades, but not of the disc itself. Following G-OJEM's accident, the manufacturer assessed the HCF life of the disc using stress levels estimated from the available information. The 1ife, in the event of disc resonance, could not be accurately determined because of uncertainty over exact stress levels and the likely period of resonance, but was estimated to be in the order of tens of flights, should the disc experience a limited period of resonance each flight. It was generally accepted that the propagation phase of any disc fatigue cracking that occurred under a HCF vibratory mechanism, would be very rapid compared to the initiation phase, with typically 90-95% of the total cycles to failure being required for crack initiation.

1.18.3.5 Steady Stresses

In 1992, as part of the investigation into previous Dart RDa7 and RDa10 HPT disc failures, the engine manufacturer conducted a finite element analysis of the stress distribution imposed on the HPT disc by the steady-state loading at take-off power. Minimum drawing dimensions for the HPT disc were used. Particular attention was paid to the stress levels in the radii between the diaphragm and the forward and aft seal rings. These regions were found have high maximum stresses and steep local stress gradients. The stresses were higher in the transient than in the heat-soaked condition:

CONDIT-ION	**SEAL**	**RADIUS**	**TEMPER-ATURE °C**	**PEAK STRESS MPa**	**MINIMUM 0.1% PS MPa**	**MINIMUM UTS MPa**
Heat-Soaked	Front	Inner	340	641	618	755
		Outer	349	556	615	751
	Rear	Inner	319	780	627	766
Transient	Front	Inner	278	734	644	786
		Outer	294	605	638	778
	Rear	Inner	242	865	659	805

MPa = Mega Pascal [1 MPa = 0.06475 tons per square inch (tsi)]

The theoretical stress exceeded the ultimate tensile strength of the material in some areas and it was predicted, using a recognised technique, that in practice these stresses would be relieved to a significant extent by local yielding of the material. With allowance for this effect, the predicted stresses were at a level that left a relatively low margin for superimposed alternating stresses, without fatigue development in these areas being likely.

1.18.3.6 Vibratory Characteristics

Engine manufacturer's analyses, following previous HPT disc failures, identified the various resonant vibratory modes of the HPT assembly. These comprised various families of types of vibratory displacement, each with a number of modes with individual resonant frequencies. In the case of the '1st Family' (1F) modes, disc sectors experienced simple bending displacement perpendicular to the plane of the disc, repeated at discrete frequencies that were harmonics of the disc rotational frequency. If excited, these were the most likely of the modes to impose appreciable alternating stresses in the seal arm radius areas. The displacement pattern rotated opposite to the disc rotation direction at disc rotation speed and thus formed a standing pattern, with a given part of the disc experiencing repeated sinusoidal displacement parallel to the disc rotational axis. This results, effectively, in a 'rolling contact' mechanism between the HPT/IPT seal arm faces. Resonance in any of the modes could only occur if the excitation force were above a certain threshold level; similarly, the amplitude of any disc vibration that occurred would increase with an increase in the magnitude of the force. Thus, excitation of any of the modes was more likely when the engine was operating at high power and the available excitation energy was greatest.

Most of the 1F mode frequencies were below the harmonics of the spool speed at high engine power, but the predicted frequency of a 2nd Engine Order (2EO) mode, at approximately 540 Hz, was only marginally above the 15,000 rpm, take-off rating 2nd harmonic of 500 Hz. The displacement pattern in this mode had nodes (positions of zero displacement) located on two orthogonal diameters (2D), and the mode was thus referred to as the 1F, 2D/2EO mode. It was intended that excitation forces would remain below the threshold, and/or the excitation frequencies would be outside the range of resonant frequencies (typically 115% minimum). The likelihood of the mode being excited would be increased by factors that tended to increase the excitation energy and/or to reduce the resonant frequency sufficiently for it to coincide with a low harmonic of the engine spool speed at a high power setting.

1.18.3.7 Pre G-OJEM situation

Prior to the accident to G-OJEM, the engine manufacturer had concluded from theoretical modelling and analysis of available test data that the predominant relevant loads tending to cause disc resonance would be from turbine blade flapping loads imposed on the disc rim. These could excite a resonant mode if sufficient repeated variation in the gas loads on individual blades at the relevant frequency were experienced as the disc rotated. The possible causes of such loading were blockage of the output from a combustion chamber, blockage of HP NGV apertures, or excessive asymmetry in fuel burner flows. The limited test data available, prior to G-OJEM's accident investigation, suggested that the frequency of the 1F, 2D/2EO mode would be reduced by excessive gaps between the HPT blade platforms, due to wear, thereby increasing the likelihood of excitation. The manufacturer also believed that the same would be true for excessive blade shroud gaps. Engine rig testing had identified a number of situations where combinations of the above defects, in combination with particular operating conditions, caused elevated alternating stresses in the disc rim. The most severe case was due to 2D/2EO vibration occurring at various engine shaft speeds in the range 14,000 to 15,500 rpm, depending on the engine configuration, and believed to be due to 1F blade flapping. Results included the following:

TURBINE BLADE STANDARD AND STRESS MEASUREMENT POSITION	CONFIGURATION	PEAK–PEAK STRESS MPa	FREQ-UENCY Hz	SHAFT SPEED rpm	ENGINE ORDER
Pre DRS 611	Standard Build	9.266	1190	10,200	7*
Disc Rim	Burner blanked	40.15	517	15,500	2
	Discharge Nozzle blocked	72.59	467	14,000	2
	Wide Platform Gaps	37.07	1657	14,200	7*
	Wide Platform Gaps & Discharge Nozzle blocked	84.94	473	14,200	2
DRS 611	Standard Build	46.33	444	13,320	2
Diaphragm (front face)	Discharge Nozzle blocked	58.69	479	14,370	2
DRS 611	Standard Build	35.52	444	13,320	2
Diaphragm (rear face)	Discharge Nozzle blocked	44.79	492	14,760	2

* See paragraph 1.18.3.9

As shown, the alternating rim stresses associated with the 2D/2EO mode were somewhat increased by burner blockage or wide platform gaps; they were particularly increased by discharge nozzle blockage, especially in combination with wide blade platform gaps. Only limited test data on disc diaphragm stresses had been obtained and none existed to indicate the effects on the diaphragm of fuel burner blockage and/or excessive blade shroud and platform gaps. However, the data did show some increase in diaphragm stresses due to discharge nozzle blockage. This had suggested to the manufacturer that fuel burner asymmetry could have a similar effect, as differences in flow between burners would also result in circumferential distortion of the turbine entry flow parameters. The results showing a substantial increase in rim stresses due to fuel burner blockage had supported this view. The manufacturer had concluded that the effects on the disc diaphragm dynamic loading of flow distortion due to burner asymmetry could be quantified by a Fourier analysis statistical transformation method, applied to an engine set of burner flow rates. The transform decomposed the distribution of burner flows into a series of harmonic components and the magnitude of a component represented the strength of the corresponding engine order distortion. As a baseline, the standard burner biasing produced a 2D Fourier Index of 1.2. However, test data on the effect on diaphragm stresses of either burner asymmetry or wide blade gaps did not exist.

1.18.3.8 Post G-OJEM situation

Until further testing had been completed, following G-OJEM's accident (see below), the engine manufacturer concluded that this disc HCF failure had also resulted from excessive burner flow asymmetry, possibly in combination with excessive blade gaps, particularly as several burner flows were found to be significantly outside limits on that engine. In an attempt to verify this by relating diaphragm stresses to the rim stress data, the manufacturer spent a considerable time developing a 3-Dimensional Finite Element Analysis (FEA) computer model. The results of this analysis were also relevant to the G-CEXF investigation. In addition to these influences (excessive burner flow asymmetry and excessive blade gaps), the investigation identified the possibility that the degree of nip (or interference) between the HPT and IPT seal arms might affect the vibratory characteristics of the disc, and hence the alternating stresses experienced in the diaphragm to seal arm radii. It seemed that a gap between the rings could significantly reduce the frequency of the HPT disc lF, 2D/2EO mode and render it susceptible to excitation. Analysis with the FEA model indicated that the frequency of the mode would reduce from 110% of maximum engine speed when the HPT seal arm was constrained by contact with the IPT seal arm, to 90% without this constraint.

Analysis by the manufacturer has shown that build clearance at the seal arm will close up as the engine is run up to full power by 1.3×10^{-3} inch (on an ISA day). In the 1F, 2D mode, the 'unconstrained' axial displacement at the HPT rear seal arm is also 1.3×10^{-3} inch, so to allow the 2D mode to occur without the HPT and IPT seal arms making contact requires a 2.6×10^{-3} inch cold build gap. Strain gauge results from the rig tests showed that, with a 3×10^{-3} inch cold build gap the, 2D mode can occur without HPT to IPT seal arm contact and that measured stress values did not change significantly when the cold build gap was increased from 3×10^{-3} inches to 10×10^{-3} inches. A cold build gap of between 1.3×10^{-3} inches and 2.6×10^{-3} inches would result in a gap, under running conditions, which could allow a 2D resonance to be formed at the HPT seal arm, the amplitude of this resonance being limited by the actual size of this gap, and hence axial movement between to HPT and IPT seal arms. Under these conditions, a rolling contact mechanism could be produced around the HPT seal arm and, over a period of time, exacerbate the problem by increasing the running gap through wear between the seal arm abutment faces.

The investigation into G-OJEM's failure identified no further factors that were likely to have affected the HPT vibratory characteristics. Assessment by the manufacturer indicated that abnormalities elsewhere in the engine, such as the compressor, reduction gear or propeller, could have a vibratory effect on the turbine only in the torsional sense and definitely would not be expected to influence the out-of-plane bending vibratory characteristics of the disc. No means by which fuel control unit anomalies could significantly influence disc vibration were found. Turbine blade tip seal rubs had apparently been common over the life of the engine type, but evidence of resultant fatigue damage reportedly had not been found. Thus the factors identified as possibly relevant to G-CEXF's HPT disc failure (given the apparent absence of NGV blockage and burner flow asymmetry) were:

The degree of HPT to IPT seal arm nip

Excessive turbine blade platform and/or shroud gaps

1.18.3.9 Test Data - Pre G-CEXF Accident

Following G-OJEM's accident, the manufacturer set up a test programme of strain gauge measurement on the diaphragm and rim of the disc of a similar operating test engine. The testing determined the stresses with the engine initially set up to specification and then with various configurations of burner flow asymmetry, blade platform and shroud gaps and seal arm abutment. Operating conditions included fast and slow engine acceleration and deceleration profiles, combined with variations in power loading on the engine and use of dry power and of water-methanol injection. The fluctuating stress levels in the seal ring radius areas were estimated from the values determined at the nearby strain gauge locations, using predictions from the FEA model. The testing did not start until July 1999, due to difficulties in commissioning the test rig, and the programme was protracted because of the deep engine disassembly required for each configuration change. The first results were available to the investigation in mid 2000 and subsequently, further information provided for the first consistent explanation of G-OJEM's failure, but only shortly before the failure on G-CEXF.

The testing reportedly showed generally significantly higher alternating stresses in the seal ring areas under transient temperature conditions (ie engine acceleration) than during steady state running. The test results were summarised as follows:

BURNER FLOW ASYMMETRY	HPT BLADE PLATFORM & SHROUD GAPS	HPT-IPT SEAL RING GAP - INCH	SEAL RING ALTERNATING STRESS INCREASE FACTOR	
			FRONT	REAR
Standard	Standard	Standard	1	1
As for G-OJEM	Standard	Standard	Minimal change	Minimal change
Standard	Standard	3×10^{-3}	2.2	5.8
As for G-OJEM	Standard	3×10^{-3}	1.7	4.3
Standard	Worn	3×10^{-3}	2.1	6.3
As for G-OJEM	Worn	10×10^{-3}	1.6	5.8

Thus, there was a major increase in alternating stress level in the seal ring radius areas associated with excessive seal ring abutment clearance. This was particularly the case for the aft seal ring inner radius area where a 3 x 10^{-3} inch HPT to IPT seal arm gap caused estimated alternating stress levels to increase by a factor of between 4 and 6.3. This was consistent with the effects of a reduction in the frequency of the 1F 2D/2EO vibration mode to below the maximum operating speed of the engine. The associated fatigue endurance level, without disc corrosion present, was predicted to increase from 30%, with an interference of 2 to 3 x 10^{-3} inches, to 70% with a gap of 2 x 10^{-3} inches. This is summarised in the following table:

HP-IP TURBINE DISC SEAL GAP/NIP **inch**	**FATIGUE ENDURANCE LEVEL** **%**	**HPT TURBINE DISC 1F/2D FREQUENCY AS PERCENTAGE OF MAXIMUM SPOOL SPEED (15,000 RPM)** **%**
2 - 3 x 10^{-3} nip	30	108.5
2 x 10^{-3} gap	70	98

With a 5 x 10^{-3} inch corrosion pit present, and a seal ring gap, the fatigue endurance level exceeded 100%. The stresses with corrosion present, but with an interference at the seal abutment faces, were acceptable. The testing also demonstrated high alternating rim stresses associated with two other vibration modes. One was a 2F, 7D/7EO resonance, excitable at an engine speed of 14,500 rpm and sensitive to excessive HPT blade platform gaps. The second was a 1F, 7D/7EO resonance, excitable at an engine speed of 10,000 rpm and sensitive to excessive HPT blade shroud gaps.

Following the testing, the manufacturer commenced a programme to assess the amount of wear found on HPT to IPT seal arm abutment faces on engines returned from service for overhaul and to develop recommendations and requirements aimed at controlling the wear.

1.18.3.10 HPT to IPT disc seal arm abutment

Prior to the two most recent disc failures, there were no specific manufacturer requirements or recommendations to inspect for wear of the seal abutment faces during turbine overhaul and little information from service experience on this matter was available. However, one overhaul agency for some two years prior to the G-CEXF event, performed measurements, known as 'drop checks', between the seal arm faces and bosses of HPT/IPT/LPT discs, to establish the cold build gap/interference of such discs that came from engines undergoing maintenance in their workshops. In addition, draw checks (see paragraphs

1.18.5/6) on the turbine taper bolts were also carried out and assessed in relation to patterns of wear on the seal arm abutment faces.

From a sample of 40 engines, some 30% were found with a gap outside drawing limits in the range + 1 to 3.5 x 10^{-3} inch, 38% with a gap within drawing limits of 0 to 1 x 10^{-3} inches, 10% were flush and 22% exhibited an interference (nip). Of the assemblies examined, a gap existed on 68%, only four occurrences of significant wear were reported and unworn discs were sometimes seen with a gap of over 2 x 10^{-3} inches. No disc assembly was seen with a gap in excess of 2.6 x 10^{-3} inches around 360° at the HPT/IPT seal arm interface. The four cases of significant wear ranged from 1.5 to 5 x 10^{-3} inch, but the figures quoted were for maximum loss of material on the IPT seal arm face. Corresponding wear was present on the mating face on the HPT arm, but less severe, which suggested that the actual gaps were slightly bigger that the measurements quoted above. In studying all the above data, it became apparent to the manufacturer and overhaul agency that there could be a possible link between poor, or loss of, taper bolt draw and excessive wear on the seal arms. Of the 40 HPT discs examined, four were identified as having a measure of inadequate bolt draw together with abnormal wear on the seal arm abutment face.

1.18.4 Other Dart HPT Disc Failures

1.18.4.1 General

A 1991 report by the engine manufacturer indicated at that time that HPT discs had suffered 27 cases of failure where part of the disc had detached. All had been attributed to HCF cracking. The failures had all occurred on RDa7 and RDa10 engines; none had occurred on RDa8 or RDa12 military engines, for which the total operating time accumulated was comparatively low. Two further cases of HPT disc rim failure were reported during the 1990s, up to the time of G-OJEM's accident, but no cases were known of any Dart IPT or LPT disc failures.

1.18.4.2 Rim Failures

Of the 27 reported cases, 21 had been 'rim failures', where a crack had originated at a turbine blade bucket and propagated through the outer part of the disc to a neighbouring bucket, thus releasing a relatively small portion of the disc together with a number of blades. The size of the released portion differed greatly but on average constituted around eight 'fir trees', or 6% of the circumference. In six cases, debris had penetrated the engine cowl. Some of the failures had occurred in the climb or cruise, but most were during takeoff. Little information was available on the effects of the failure, beyond the sudden loss of engine power, often at a critical point. In one case, the electrical wires to the feathering pump had been severed by the debris and prevented the propeller from being completely feathered.

Eleven of the rim failures had occurred on the RDa7 and ten on the RDa10. However, there was a considerable difference in accumulated operating times between the types and the failure rate per operating hour was considerably lower for the RDa7 than the RDa10, by a factor of around 1:4. Almost half of the cases were considered by the manufacturer to have been resolved, and the majority of these had been attributed to turbine entry flow distortion due to significant thermal degradation of combustion chamber flame tubes. Thirteen further cases of HCF cracking of the disc rim were known to have been found on inspection at overhaul or repair, nine on the RDa7 and four on the RDa10. LCF cracking in the bucket grooves was common in HPT discs that had accumulated appreciable operating time. Allowable limits were specified in the Overhaul Manual and this had not led to any failures which resulted in diaphragm rupture.

1.18.4.3 Hot Rupture Failures

Two disc failure cases had been hot rupture diaphragm failures, both on RDa7 type engines, where combustion chamber deterioration had led to fuel burner damage and the release of excess fuel into the combustion system. This caused overheat and failure of the HPT disc. Modification action aimed at preventing recurrence had been taken.

1.18.4.4 Diaphragm Failures

The remaining five previous cases, including G-OJEM, had been disc diaphragm failures. In these cases considerably larger pieces of debris were released than in the rim failure cases and the potential for aircraft damage was greater. Of the five cases, four were on RDa7 engines and one on the RDa10. The available information from the manufacturer's investigations, in summary, was as follows:

1. HS748, Dart 531, Philippines, 29 April 1972:

At 60 kt during the take-off run a loud bang was heard, followed by a No 2 engine bay fire warning. The takeoff was aborted. Engine bay fire extinguishers were ineffectual against the engine bay fire, which was subsequently extinguished by the AFS. The failure was caused by the separation of a 10 x 4 inch portion of the HPT disc. The turbine bolts had sheared and both the separated and main portions of the disc had been released through the left side of the nozzle box at an angle of 30° below the horizontal, thus barely missing the fuselage. The fuselage sustained 15 impacts and had been punctured in several places but no debris entered the cabin. The disc failure was found to have resulted from a resonant HCF mechanism that had probably originated from a frettage fatigue crack in the rear seal abutment face. The resonant condition was thought to have been excited by a partial blockage of the turbine entry gas flow by a front cooling strip from the No 4 combustion chamber flame tube that had detached and lodged on the NGVs. This was calculated to have produced a flow distortion 2D Fourier Index of 2.5.

2. Viscount, Dart 527, Israel, 2 October 1979:

After takeoff a loud noise was heard, followed by a No 3 engine bay fire warning. After securing the engine, the flight was completed. The failure had resulted from the detachment of a 9 x 2.5 inch portion of the HPT disc. The nozzle box was holed but the turbine bolts remained intact and the disc parts were retained within the engine cowl. The disc failure was attributed to HCF cracking from a large number of origins situated in the outer blend radius between the diaphragm and the front seal. The disc exhibited many fine surface fissures that were abnormal but were subsequently concluded not to have influenced the failure. The No 2 burner was found to have a flow rate approximately 25% in excess of the specified maximum, due to incorrect torquing of the nozzle shroud onto the body, giving a calculated flow distortion 2D Fourier Index of 6.9. The distortion was thought to have excited a resonant vibration in the disc.

3. NAMCYSll, Dart 542-10J, Caribbean, 25 June 1989:

As power was being set for takeoff and the engine was accelerating through approximately 10,000 rpm, a loud bang was heard, followed by an engine bay fire warning. A 17 x 3.75 inch portion of the HPT disc had detached which holed the nozzle box and exited the nacelle. The remainder of the disc remained in the engine. The fuselage was damaged by debris. The failure was attributed to HCF cracking at the inner blend radius between the diaphragm and the rear seal arm. The crack originated at a 1×10^{-3} inch deep corrosion pit. Rig testing showed five of the burners to have satisfactory flow rates; no information was available on the other two, which were not returned to the engine manufacturer. The cause of the failure could not be assessed. The disc failure features were very similar to those exhibited by both the G-OJEM and G-CEXF failures.

4. Fokker F27, Dart 532-7, Colombia, 20 August l99l:

At 80 kt during the take-off run a loud bang was heard, followed by an engine bay fire warning. The failure was caused by the detachment of two portions of the HPT disc, 8 x 3 inches and 3.5 x 3 inches in size. The turbine bolts had sheared and the two detached portions of the disc had been released from the engine. The main part of the disc lodged in the engine cowl. One portion of the disc embedded itself in the fuselage, which also sustained numerous other debris impacts. The disc failure was attributed to a resonant HCF mechanism that had originated from a 3×10^{-3} inch deep corrosion pit in the inner blend radius between the diaphragm and the front seal arm. One HPT blade was found to have failed in fatigue cracking just above the platform, at an unknown point; this was not considered to have had any effect on the disc failure. The resonant condition that caused the disc failure was attributed to flow distortion caused by excessive asymmetry in burner fuel flows following non-standard overhaul. The flow for 5 of the burners was found to be outside the Overhaul Manual limits, by up to 24% below to 22% above, giving a flow distortion 2D Fourier Index of 4.2.

1.18.4.5 Diaphragm Failure Case Summary, Figure 9

In all of the above cases of HPT disc diaphragm failure the available information from the engine manufacturer indicated that disc fracturing had been the result of a vibratory HCF type mechanism, with no evidence of LCF effects. The failures had generally originated at corrosion pits but these were considered to have been sites that had been exploited by the mechanism, rather than a cause of the cracking, and this view was supported by the nature of the failure on G-CEXF.

Evidence that there had been abnormal turbine inlet flow distortion was found in three of the cases. In two of these cases the cause was assessed as being gross burner fuel flow asymmetry. In the remaining case the suspected failure cause was not established; the engine manufacturer was unable to flow check two of the burners. The engine casing and the cowl were holed in three of the cases and part or all of the HPT disc exited the engine. The fuselage sustained damage from debris impact in three of the cases and in one of these the main disc parts only narrowly missed the fuselage.

In these four previous diaphragm failure cases the operating service accumulated by the discs at the time of failure varied between approximately 7,500 to 13,000 hours/6,200 to l5,000 cycles from new and 154 to 5,000 hours since overhaul. No other cases were known of a HPT disc found with diaphragm cracking, for example during overhaul or repair, and no cases were known where evidence of LCF cracking of the diaphragm had been found.

1.18.5 Principal of Dart taper bolt design

As a result of the failure on G-CEXF, the engine manufacturer conducted a review of the design of the taper bolt system used to install the three turbine discs in the Dart engine. Critical to the structural integrity of the turbine rotor in the assembled condition is the correct fitment of the taper bolts used to clamp the three turbine discs together and attach these to the turbine shafts. The bolts provide a positive joint that is capable of maintaining the balance of the rotor under the variable load and temperature conditions experienced by the turbine when operating. By design, they eliminate radial and circumferential slip. The design is complex as it relies on the end load in the bolt to provide the clamping forces, and this is a function of both bolt 'draw' and bolt tension. At the various interfaces between the disc bosses, tapered sections of these bolts are drawn into appropriately tapered holes by the action of tightening the nut. The assembly is engineered such that within specified limits (5×10^{-3} inches to 25×10^{-3} inches) an approximately equal draw should be achieved at each tapered interface. The maximum end load in the bolt is directly under the nut, but this tension is reduced at each fitted taper section between the nut and the bolt head. The total reduction in end load in the bolt towards the head depends on the amount of bolt draw at each taper interface, and the remaining available tension compresses the flanges of the turbine shafts.

The total torque to be transmitted by the turbine is 2,200 lbf.ft and the torque split between the HP, IP and LP turbines is estimated in the ratio of 43:36:21. An analysis of this model of Dart engine indicated that these torque levels could not be transmitted by friction alone between the faces of the mating bosses, so the tapered sections of the bolts would experience shear forces at the disc interfaces. Diagrams illustrating the location and principle of the tapered sections, and a theoretical loads analysis for a turbine stack, are shown in Figures 10 and 11.

1.18.6 Measurements on turbine discs removed from engine s/n 14845 (G-CEXF)

The three turbine discs from the failed engine were taken to a RR Dart overhaul agency in the UK where dimensional checks had been habitually conducted on 'in-service' discs for several years. Attempts were made to establish by detailed inspection and measurement if any dimensions, as manufactured, were outside drawing limits and the taper bolt 'draws' which existed at the time of the failure. However, the severe damage to the HPT and lesser damage to the other two discs precluded a comprehensive set of measurements and it was not possible to determine directly that either a nip or a gap had been present at the abutment face of the HPT and IPT seal arms.

Visual examination of the condition of the HPT/IPT seal arm abutment faces not directly affected by accident damage revealed areas of light/heavy frettage damage at specific locations around the circumference. The locations of this damage on both discs are shown relative to the boss locations, the fatigue cracks and their origins in Figure 12. An assessment of any damage in the sector of the fatigue cracks could not be made on the HPT disc as the missing segment was not recovered.

1.18.7 Possible causes of seal arm wear

During the investigation, consideration was given to the possible causes of seal arm wear. The tests and analyses, conducted prior to the G-CEXF failure, indicated that excessive (cold) clearance between the HPT and IPT disc seal arms can allow axial movement between the seal arms, in addition to a 2D resonance at the HPT disc seal arm. The amplitude of this resonance is limited by the size of the gap and the frequency of excitation. In other words, any gap, under engine operating conditions, between the seal arm abutment faces of more than 1.3×10^{-3} inches could allow the HPT disc seal arm to vibrate freely without making contact with the IPT arm. Lesser amplitudes of vibration could still occur, however, with lesser gaps, the combination of the axial movement and 2D resonance resulting in the rolling contact wear mechanism around the inner HPT seal arm.

Non parallel boss face

If the two abutting faces of the bosses were not parallel, the seal arm faces may be in interference on one side with a corresponding clearance on the other. This could result in progressive wear around the circumference, raising the likelihood of an arm resonant condition becoming established and further wear occurring.

Zero IPT bolt draw with seal arm interference

Under this condition, the bolt would pick up no load from the IPT, and the torque load would be transmitted to the seal arm abutment face. This is likely to result in wear and could influence the failure mechanism directly.

Zero IPT bolt draw with seal arm clearance

Excessive movement between the HPT and IPT discs, due to the bolts not being effective at the taper sections, could result in significant seal arm wear.

1.18.8 Possible reasons for the loss of bolt draw

Although the turbine discs have a finite life, 20,000 cycles in the case of the HPT, they are often re-worked in overhaul agencies within that life on an opportunity basis as engines are overhauled; a HPT disc may be re-worked two, three or possibly four times during its life. Operations carried out may include lapping of the boss and seal arm faces to remove minor fretting, removal of any corrosion, crack checking and the re-application of the corrosion protection paint. At the time the HPT disc from G-CEXF was last re-worked, prior to the failure on G-OJEM, there were no requirements in the engine Overhaul Manual to re-ream the tapered holes following any lapping operation. In addition, taper bolt draw checks on re-assembly were conducted with either the three turbine discs as a stack, which could mask the loss of bolt draw at any one interface, or by using spacers with tapered holes to represent one or two of the discs. The drawback with this method is that the spacers were made to a nominal dimension and did not accurately represent any particular disc, and thus could give a false indication of actual bolt draw. Changes implemented by the overhaul agency mentioned in paragraph 1.18.3.10, and also as detailed in a revised version of the relevant manufacturer's Overhaul Manual, now allow for the dimensional differences between the spacers and any particular disc to be taken into account when establishing the bolt draws on individual discs. This should preclude loss of draw at any particular interface and thus minimise the possibility of inducing, or exacerbating, wear at the seal arm faces.

1.18.9 Possible HPT disc modifications

In the course of various failure investigation studies the engine manufacturer had assessed possible modifications aimed at reducing the susceptibility of the HPT disc to HCF. Measures applicable to existing discs included shot peening, to produce compressive surface stresses, paint changes to improve corrosion resistance, geometry changes to reduce peak stress levels (such as reducing the thickness of the seal rings) and applying a hard coating to blade platforms to reduce wear. Those applicable only to a disc redesign included an increase in the diaphragm thickness, an increase in the blend radius between the seal rings and the diaphragm and a change in the disc material. A stress re-analysis by the manufacturer in 1992 indicated that seal ring thickness reduction would have little effect on peak stresses. Increasing the minimum inner blend radii of the seals would result in appreciable reduction in peak stresses in these areas, particularly for the rear seal. A change in the disc material to Inco 718 was predicted to at least double the alternating stress capacity at each of the blend radii. Most of these changes, however, were not incorporated as a design change of the disc would then require re-certification. In consideration of the time scales required to achieve this, and of the age and diminishing size of the fleet, such changes were not thought to be a practical proposition.

1.18.10 Remedial action

The engine that failed had been developed from one of the earliest gas turbine engine designs and the type had been in service for nearly fifty years. A substantial number (some 1,680) of similar engines remain in service in commercial aircraft. Several major fatigue failures, similar to the type that happened with G-OJEM and G-CEXF, had occurred previously and had been attributed by the engine manufacturer to a combination of turbine entry flow distortion and turbine blade wear. Some measures aimed at controlling these aspects had been taken. The engine manufacturer and the CAA had judged, on the basis of a statistical analysis, that the likely period before recurrence of the failure was such that additional remedial action was unnecessary. The factors since determined as the probable causes of these failures had not been identified prior to the investigation of the accident to G-OJEM, and thus the HPT design had not been subject to remedial action. Following the engine rig testing after G-OJEM's accident, the manufacturer issued Dart Service Bulletin Da72-533 (Modification 1946) in February 2001, which was later revised (Revision 1) and re-issued on 12 April 2001. The modification was aimed at providing a positive nip of 2 to 3 x 10^{-3} inches between HPT and IPT disc seal arms, and improved corrosion protection for the HPT disc. The positive nip is achieved by a metal spraying process onto the seal arm abutment face, to build the surface up and grinding this back to produce a raised surface when compared to the original. The SB was classified as Mandatory by the CAA, with accomplishment required at the next disassembly that afforded access to the relevant area, or by the end of 2005, whichever was earlier. However, following the near identical failure on G-CEXF this SB was revised to take account of HPT disc cycles and re-issued in July 2001.

The revised requirements were as follows:

1 *All engines*

After 31 July 2001, no engine may be dispatched from an overhaul facility without SB Da72-533 (Mod 1946) embodied

2 *Engines fitted with HPT discs with more than 15,000 cycles since new at 31 July 2001*

Remove from service for rework in accordance with SB Da72-533 within 300 cycles

3 *Engines fitted with HPT discs with more than 12,000 cycles since new at 31 July 2001*

Remove from service for rework in accordance with SB Da72-533 within 600 cycles

4 *Engines fitted with HPT discs with more than 9.000 cycles since new at 31 July 2001*

Remove from service for rework in accordance with SB Da72-533 within 900 cycles

5 *All engines*

After 30 June 2002, no engine may continue in service fitted with a HPT disc with more than 12,000 cycles since new without SB Da72-533 being embodied

6 *All engines*

After 30 June 2003 no engine may continue in service if fitted with a HPT disc with more than 6,000 cycles without SB Da72-533 being embodied

7 *All engines*

After 30 June 2004, no engine may continue in service without SB Da72-533 embodied.

1.18.11 Failure reassessments

The engine manufacturer had attempted to identify any common causal factors for the HPT disc failures and, in l991, had conducted a study of previous cases to attempt to reassess their causes. This was updated after G-OJEM's accident. It was concluded that, in all cases, the disc fracturing had been the result of a vibratory HCF type mechanism, with no evidence of LCF effects, and this view appears to be confirmed by the nature of the failure on G-CEXF. It was also concluded that no changes had been made to the design of the HPT assembly over the time that it had been in service, that the manufacturer considered were likely to have adversely influenced its susceptibility to fatigue damage. No failures had occurred in 'lst Life' engines (ie, service between engine construction and its first overhaul or repair), but five of the cases had occurred on 1st Life HPT discs. In the cases that were considered resolved, an anomaly that would have produced turbine entry flow distortion was identified. This was commonly flame tube deterioration in the case of the rim failures and in the case of the diaphragm failures either flame tube deterioration, leading to NGV blockage, or fuel burner asymmetry. No common background factors such as component service life or overhaul provenance were evident, except that all of the RDa7 failures had occurred on engine models that were used in twin-engined aircraft and had a higher rated power than the types used in 4-engined aircraft. This remains the case.

1.18.12 Risk analysis

In a 1993 review, the engine manufacturer made a statistical assessment of the rate and effects of the previous failures. This was in the predicted context of a diminishing level of Dart operations, a long time scale for the incorporation of design changes to the disc and potential operator resistance to any significant changes. It was noted that the rim failure rate had been stable for many years and that changes had been made available in recent years to improve the flame tube material and control the blade platform and shroud gaps. Failure rate calculations were based on those cases where the mechanism had not been identified; those cases where the cause was considered to have been determined were not taken into account. The manufacturer had concluded that the resultant catastrophic risk rate was sufficiently low that corrective action was not necessary.

Following the failure to G-OJEM in 1998, the basic cause of the diaphragm failure was considered to be understood, although it was thought possible that asymmetric burner flows could still have influenced that particular failure. This was subsequently shown not to be the case. In order to assess the risk of further failures occurring for the same reason and to establish a time frame in which to incorporate Mod 1946, a risk analysis was conducted using a constant hazard rate model. With this model it would be expected that a low life disc will be equally susceptible to a diaphragm failure as an older item. The results for the three failures at that time taken to count against this problem gave a risk of 1.8×10^{-7} failures/engine flight hour (efh), with a reaction time for hazardous events of 5,333 hours. This analysis also predicted that if the modification was not incorporated then there would be 1.98 failures of the diaphragm up to the projected time of fleet withdrawal in 2015. The initial issue of SB Da72-533 (Mod 1946) was to achieve modification of all HPT discs by the end of 2005, which was well within the required period, and the analysis predicted with full Mod incorporation by 2005 there would be 0.46 additional failures by 2015.

Following the failure to G-CEXF, this risk assessment was re-evaluated as, from the current understanding of the mechanism of failure, a time/age related model was considered by the manufacturer to be more appropriate than the use of a constant hazard rate model. (Although it is reasonable to assume that all discs should have met the design intent at the time of manufacture (ie, within drawing limits, resulting in no gap at the abutment faces under running conditions when assembled), a review by the manufacturer of all the Service Bulletins pertaining to the turbine area of the engine, prior to the publication of SB Da72-533, failed to identify any inspection action that would check the relative positions of the seal arm faces). The model therefore considers the time since new of a disc, rather than time since overhaul. The outcome of this re-evaluation of the risk analysis showed that the current (at the time of the failure) risk was above the required threshold requirement of 1×10^{-8}/efh. The model demonstrated that a rapid decrease in the risk could be achieved by early removal of high life disc/high utilisation engines and that the risk would be reduced to below the threshold level by June 2003. The analysis also indicated that the expected number of failures will be a maximum of 0.2 by the end of the new compliance period of June 2004, and this was considered to be sufficiently below unity that no more failures would be likely to occur. The model conservatively takes no account of on-going routine overhaul activity in the fleet embodying Mod 1946, or for operators planning fleet modification ahead of the mandatory compliance dates to support their operational needs. It also indicated that for the first 12 month period of action, the risk of 3.8×10^{-7}/efh would be reduced to 4.6×10^{-8}/efh.

2 Analysis

2.1 The Flight

2.1.1 The engine failure

Examination of the aircraft and the debris recovered from the fields immediately beyond the end of the runway left no doubt that the accident had been precipitated by the sudden, non-contained, failure of the HPT disc in the left engine. There had been no evidence of any abnormal indications to the crew on the flight deck that could have alerted them to the impending failure and as such there was every reason for the crew to operate the aircraft in a normal manner. Evidence from the crew, the FDR, CVR and location of the released engine parts, indicated that the failure had occurred some 17 seconds after lift off whilst the engine was still at maximum power and when the aircraft was at an altitude of some 670 feet. The failure caused a complete loss of power from the left engine and, due to the complete severance of the low-pressure fuel pipe immediately forward of the nacelle firewall, a substantial fire external to the nacelle.

2.1.2 The crew handling of the emergency

At the start of the duty, the commander had been briefed by the previous crew on the vibration experienced earlier in the morning, but no entries had been made in the aircraft's Technical Log. However, the commander did not note any unusual vibrations during the outbound sector from Bournemouth to Jersey, or at the commencement of the accident sector, and the vibration experienced earlier in the day was not considered to be relevant to the HPT disc failure.

In the post accident debrief, the three flight deck crew members indicated that the aircraft handling by the first officer during the failure sequence, the actioning of the Engine Fire Drill by the commander and the subsequent handling of the single engine return for an approach and landing at Jersey, were all completed as per Standard Operating Procedures, and were without problem. The only adverse comment related to the subsequent fuel spillage once the aircraft had been shut down in the parking area. Had the Fuel Shut-Off Handle not been returned to the ON position, then the fuel leak would probably not have occurred. It must be noted that the crew composition, the very good daylight weather conditions and the light weight of the aircraft at the time were all significant factors in assuring that the effects of the failure were minimised and the flight brought to a safe and successful conclusion.

2.1.3 The fuel leakage

The sustained fuel leak from the left engine nacelle resulted from the complete severance of the low pressure fuel pipe between the nacelle and engine firewalls by a section of the HPT disc as it departed the engine. When this happened there would have been a number of likely ignition sources in this region of the engine including combustor flame release, hot surfaces associated with the running engine, frictional heating and sparking associated with the HPT release and engine oil that had already possibly ignited. All the evidence indicated that the leaking fuel had ignited very soon after the HPT failure and that it was this fuel fed fire which had caused all the damage to the nacelle. It was extinguished after approximately 27 seconds by the crew action of closing the LP fuel cock as part of their FIRE actions. The fact that the left main landing gear was retracted for the duration of the fire, and that the fire was extinguished before more serious structural damage could occur to the nacelle, resulted in minimal damage to the landing gear and its operating system. This in turn enabled the aircraft to be landed normally.

2.2 HPT Failure

2.2.1 The HPT disc failure

The evidence showed that part of the left engine HPT disc separated while the engine was operating at take-off power and that the engine disruption was fully consistent with the effects of this separation. The separation would have subjected the remaining section of the disc to severe rotational imbalance loads, and signs were found of jamming interference of this disc section with the nozzle box casing. The damage found was fully consistent with the large forces generated causing the turbine taper bolts to fail at the disc interfaces, thus allowing the main part of the disc to break free. Both disc sections were released with sufficient energy to penetrate the nozzle box casing and the cowl, with the larger section severely damaging an engine support strut, causing it to break free from its firewall fitting. The engine investigation therefore concentrated on establishing, in conjunction with the manufacturer, the reasons for the HPT disc failure.

2.2.2 HPT disc failure consequences

The two major portions of the HPT disc exited the nacelle with considerable energy, as evidenced by the appreciable damage that each caused, and it was fortunate that the exit point from the nozzle box and both their trajectories happened to be such that they both cleared the fuselage. There appeared to be no features that would cause a non-contained portion of a fractured disc to preferentially exit the nozzle box and/or nacelle at a particular circumferential location, apart from possible local shielding in some areas by the engine mounting struts. The larger section of G-CEXF's HPT disc, in fact, made contact with the lower left strut, causing severe deformation, and wrenched it from the firewall attachment fitting. Previous experience of gas turbine engine non-containment events suggested that substantial debris tends to be contained within approximately +/-5° of the original plane of rotation of the failed component. In the case of the F27, with these assumptions, substantial non-contained HPT disc debris would not strike the wings or propeller but could strike and penetrate the fuselage. The fuselage subtends an arc of approximately 10° from the engine positions and the probability of a fuselage strike by a 12 inch square portion of disc, for example, was estimated to be in the order of 1 in 10. All primary and secondary flight controls (with the exception of the aileron trim, which is electrically signalled) are operated by mechanical (cable) systems that cross the turbine discs' planes of rotation where they run in the crown of the forward fuselage.

Thus, there is a risk that all of the flight control runs are potentially exposed to damage or being cut through following such a turbine disc failure, although the probability of such a fragment of HPT striking these control runs within the fuselage is significantly less than 1 in 10. On G-CEXF, all controls to the engine and the LP fuel pipe were severed. The control cables, which operate the LP fuel cocks, also run through this same area. This raises the possibility that, in any similar event, it might not be possible to isolate a leaking fuel supply and hence extinguish a fire, and/or that the undamaged engine might be inadvertently shut down by a false signal applied to the LP cock cables. However, with the risk of HPT disc failure being estimated at 4.6×10^{-8}/efh after the first 12 months of remedial action (Mod 1946, post July 2002), together with a substantially less than 1 in 10 chance of a released segment striking the controls runs in the fuselage, the probability of such an event occurring is estimated at less than 4.6×10^{-9}/efh, ie, extremely remote.

2.2.3 HPT disc fracture mode

It was clear from the detailed examination that two fatigue cracks had developed in the inner blend radius between the diaphragm of the HPT disc and the arm of the rear seal. The cracks had collectively progressed circumferentially for approximately seven inches, or around 40%, of the circumference, and axially through about 80% of the disc thickness. The evidence indicated that, at this point in the propagation of the cracks, the disc was sufficiently weakened for normal operating loads to extend the crack approximately radially from either end. This fractured the remaining 20% of the section, thus releasing a substantial portion of the disc, with consequent severe disruption to the powerplant and nacelle. There was no doubt that the circumferential fatigue cracking of the HPT disc had been responsible for the sudden disruptive failure.

Detailed examination found no signs to suggest that the properties of the disc material were outside the specification requirements, or that there had been localised anomalies that could have triggered the fatigue cracking. The corrosion pits, up to 5.5×10^{-3} inches deep, were characteristic of the fairly extensive light corrosion present on the disc surface. The fact that such corrosion was reportedly quite usual on in-service discs indicated that, while the pitting was likely to have reduced the fatigue strength of the disc, it had not acted as a focus, unlike the case with G-OJEM, for the crack which ultimately developed to failure.

No signs of gross overstress, overtemperature, or LCF effects on the disc were present and the evidence all indicated that HCF had been responsible for the initiation and progression of the circumferential crack. Some evidence indicated that the fatigue had occurred due to stress levels near to the UTS for the material. Thus it was concluded that the fracture resulted from the area of the seal arm to diaphragm blend radius experiencing fluctuating loads that, superimposed on the steady loads, had exceeded the capability of the material. These fluctuating loads had resulted from a vibratory condition of the disc.

A characteristic of the disc material was that prominent striation features associated with fatigue progression on the crack surface were unlikely to form. Also, the striations that might have been present were rendered more indistinct by light corrosion of the surface formed whilst the disc section lay in the damp ground for several days after the event. Attempts to establish the number of cycles involved in the main fatigue crack progression were not successful, but previous attempts, following the similar G-OJEM event, provided a broad indication of the likely range, ie, in the order of 1 to 5 million load reversals. It was likely that many times this number of cycles had been required to initiate the crack but variability in the mechanism prevented firm conclusions from being drawn as to the likely initiation cycle count. Thus no real estimate could be made of the number of load cycles that had been involved in producing the crack.

In summary, the HPT disc failure was very similar to that which occurred on G-OJEM and had resulted from extensive HCF cracking. This resulted from relatively high stress load fluctuation associated with a vibratory condition of the disc; no firm evidence on the cumulative vibration period or number of flight cycles involved was available.

2.2.4 HPT disc life and stress background

The Dart was an early design of gas turbine and the turbine that failed in Engine s/n 14845 was of a type that had been basically designed in the 1950s, when there was a lower level of knowledge and awareness of the fatigue damage effects of cyclic loading. LCF damage considerations for the HPT disc, covering the possible effects of the loading variation experienced by the disc in the course of a typical flight cycle, had led to the determination of a PSL of 20,000 flight cycles for the disc as used in the RDa7 engine type. HCF loading effects, resulting from possible vibratory conditions, had been assessed for the HPT blades by strain gauge testing, but this was not the case for the disc diaphragm. Following a number of disc 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 disc with the engine at maximum power. Little margin existed for superimposed alternating stresses, even without corrosion pitting being present, without the likely development of fatigue in these areas.

Although the predicted margin was lower for the aft inner radius than for the other three locations, previous diaphragm failures had also originated in two of the other locations. No evidence was available to ascertain the reasons for this inconsistency, but the difference in the margins was relatively small and it was possibly due to error in the predictions, to local variability in the individual discs involved or to chance scatter in the fatigue mechanism. The apparently similar vibratory conditions having caused HCF cracking of the rim on some occasions and the diaphragm on others had not been established at that time.

Engineering rig testing following the G-OJEM failure had established the various frequencies and modes of natural vibration of the disc and suggested that a 2D vibration causing out-of-plane bending of parts of the disc diaphragm was the most significant. The mode involved diametrical maximum out-of-plane displacement in a pattern that rotated around the disc and thus imposed repeated load reversals on elements of the disc. Cyclic variation of the bending stress in the diaphragm/seal arm radii generated by the centrifugal stress field would inevitably be associated with such diaphragm bending vibration. The frequency corresponded to engine shaft speed in the range 14,000 to 15,500 rpm. The engine manufacturer at that time had considered that the mode could be excited by blade flapping loads imposed on the rim disc if sufficient variation in the gas loads on individual blades were experienced as the disc rotated past the exhausts from the seven combustor cans. The possible causes of circumferential gas load variation were blockage or restriction of the output from a combustion can or NGV aperture, or excessive burner fuel flow differences. The limited test results available at that time had supported this, as had the conclusions of some of the previous HPT disc failure investigations, but no evidence of either was discovered on the engine from G-CEXF. The test data had also indicated that increased gaps between the blade platforms were likely to accentuate the vibration by reducing the natural frequency of the disc vibration mode, and the manufacturer had believed that the same would be true for increased shroud gaps. However, no quantitative test data on the fluctuating stress levels in the seal arms radius areas, resulting from these features, were available at that time.

2.2.5 Causes of the HPT disc failure

All of engine s/n 14845's combustion cans, and their flame tubes, were essentially intact and no evidence was found of foreign objects having entered the engine, or of any parts of the engine upstream of the HPT having separated before the disc failure. Therefore, it was concluded that it was very unlikely that partial blockage of the gas flow path to the HPT had influenced the disc failure.

The investigation into the causes of the HPT disc failure necessarily relied heavily on the information and expertise generated and supplied by the engine manufacturer, in particular as a result of the failure on G-OJEM, the investigation of which was directly relevant to this investigation. For an extended period, prior to the rig testing results obtained in 2000, the manufacturer attributed such failures to the effects of excessive fuel burner asymmetry with a probable contribution from excessive HPT blade platform and shroud gaps. No flow assymmetry of any significance was found with engine s/n 14845's burner set, the flow deviation from the normal and the value of the 2D Fourier Index was not of significance. The 2D Fourier Index, calculated from the flow rates of a burner set, was intended to provide a measure of the flow distortion and the potential of the set to excite the HPT disc to resonance. However, the results of the measurements, in response to G-OJEM's accident, of burner flows for engines returned from service showed a significant number had an Index appreciably greater than that for the G-OJEM engine without HPT HCF damage being evident. This strongly indicated that the susceptibility of any particular HPT to HCF damage is influenced by factors other than, or additional to, those causing flow disturbance. The engine manufacturer had, at the time of the G-OJEM accident, considered that the main additional factor was the size of the platform and/or shroud gaps.

While this appeared to be a possibility, no evidence to positively confirm this conclusion was found, as the quantitative effect of excessive gaps was unknown and, where disc failure had occurred, it had not been possible to measure the gaps due to damage. The size of the HPT blade platform and shroud gaps for both engines s/n 334 (G-OJEM) and s/n 14845 (G-CEXF), also, could not be assessed due to the level of damage. Following the failure on G-OJEM, measurements on other engines provided some data on platform and shroud wear rates and indicated that both were highest for blades that had not been repaired to the DRS and lowest for new blades. However, the wear rates experienced by individual engines varied so widely that no sensible prediction could be made for either engine s/n 334 or s/n 14845.

Engine rig testing after G-OJEM's accident indicated that neither fuel burner flow asymmetry to the degree found on that engine, nor excessive HPT blade gaps, had a significant effect on alternating stress levels in the HPT disc seal arm radius areas. The evidence suggested that these factors were more relevant to fatigue in the HPT disc rim area. The predominant adverse effect on the diaphragm stress levels, which could affect the HPT disc resonance frequency, was related to the amount of nip (or gap) between the HPT and IPT seal arms. It appeared that the degree of contact between the seal arms was likely to affect the characteristics of the disc out-of-plane bending vibration, but no evidence to quantify the effect had been available prior to the rig tests. The significance of this effect was shown by the large variation in the fatigue endurance level with changes in the room-temperature gap or nip between the seal arms, from 30% with an interference of 2 to 3 x 10^{-3} inches, to 70% with a 2 x 10^{-3} inch gap. A four to six times increase in the stress levels at the rear seal ring inner radius was associated with a 3 x 10^{-3} (or more) inch room-temperature seal arm gap. Such a gap allows full amplitude HPT disc resonance in the 1F, 2D/2EO mode, without constraint by contact between the HPT and IPT disc seal arms. This has the effect of reducing the natural frequency of the mode sufficiently toput it within the normal operating range of the engine spool speed. The +/-1 x 10^{-3} inch maximum build gap allowed by design tolerances is less than the 1.3 x 10^{-3} inch cold build gap necessary to allow the onset of vibration and, therefore, a gap should not be present under running conditions. As very few HPT diaphragm failures have occurred, compared with the number of engines in service over the years, it is likely that the great majority run with little or no gap present between the HPT and LPT seal arm faces. The consequent zero gap or nip between the seal arms serves both to constrain the axial displacement of the HPT seal arm, and hence the onset of the vibratory condition, and to minimise or preclude wear of the seal arm abutment faces. Cold build gaps of more than 1.3 x 10^{-3} inches, but less than 2.6 x 10^{-3} inches, would not be sufficient to allow full-amplitude resonance, but more a partial-amplitude vibration constrained by contact between the seal faces. This is likely to cause wear of the HPT and IPT seal arm faces by the rolling contact mechanism, resulting in enlargement of the gap over time, and increased vibratory stresses in the seal arm/diaphragm radii, with the possible consequence of HCF cracking.

Changes to the relative positions of the disc boss and seal arm abutment faces could result from maintenance practices, such as lapping, to remove minor surface defects, and it is possible that this could lead to loss of interference between the seal faces and non-parallel boss faces. If so, this may induce an interference/clearance around the seal arm faces which, under operating conditions, could induce wear. Wear at the seal arm and boss faces might also occur as a consequence of loss of draw of the turbine taper bolts. This view was based on a 'small sample' analysis of the data collected by an overhaul agency over several years of inspecting Dart turbine discs, where loss of material/abnormal wear of seal arm abutment faces was seen on four of the 40 discs examined. These four, however, were also associated with varying degrees of loss of bolt draw.

Visual examination of the condition of the faces at the HPT/IPT seal arm abutment faces from G-CEXF revealed areas of non-accident related light/heavy frettage at specific locations around the circumference. The condition of the faces of the HPT and IPT bosses, however, indicated that there had been little or no relative movement between the HPT and IPT discs, which mitigated against a loss of bolt draw having been present in the turbine assembly on the subject engine.

The damage sustained by the HPT in the accident precluded any meaningful measurement of the aft seal arm height, and hence any direct determination of the interference or clearance with the forward arm on the IPT. The presence of fretting damage, particularly as this was determined by the manufacturer to have resulted from a 'rolling contact' type of wear mechanism, suggests that a gap was likely to have been present between the HPT and IPT seal arm faces under running conditions, which in turn allowed a measure of vibration to occur and consequent HCF cracking to develop. Thus it would seem likely that a cold build gap in excess of 1.3×10^{-3} inches was present at the seal arm abutment faces after the turbine stack was last assembled.

Until the further investigation carried out after the accident to G-CEXF and the consequent changes to maintenance practices, little or no guidance was contained in the Overhaul Manuals used by overhaul agencies with regard to the importance of maintaining the correct nip between the turbines in the stack, and how this could be inadvertently degraded by the then current maintenance procedures. Changes now implemented by the manufacturer to the Overhaul Manuals, and the re-issue of SB Da72-533, should assure that the risk of HCF cracking of the HPT is reduced to a level where another failure of the same type would not be expected to occur during the remainder of the fleet life. However, due to the practicality of implementing Mod 1946 into the affected engines, the risk analysis carried out by the manufacturer indicated that the risk of another such failure would not have been reduced to an acceptable level until after June 2003, albeit that a rapid decrease in the risk after the first 12 months of implementation of Mod 1946 would be achieved by targeting high life HPT discs in high utilisation engines. Up to the date of this report, no further HPT disc diaphragm failures are known to have occurred.

3. Conclusions

(a) Findings

(i) The crew were properly licensed, medically fit and adequately rested to conduct the flight.

(ii) The accident was precipitated by the catastrophic failure of the No 1 engine immediately after takeoff resulting in a sudden loss of power and an immediate substantial fire external to the nacelle. No prior warning was available to the crew.

(iii) The No 1 engine failure resulted from the fracture and non-containment of the HPT disc.

(iv) The turbine break-up resulted in significant fuel leakage from the low pressure fuel pipe and this was the major fuel source for the fire, which posed a considerable hazard to the aircraft.

(v) The commander's decision to fly a tight left hand circuit and land the aircraft as soon as possible was wise in the circumstances.

(vi) The HPT disk failure resulted from extensive HCF cracking caused by resonant vibration of the disc. It was likely, after a lengthy initiation period, that the cracking had developed relatively rapidly.

(vii) The HPT disc resonance probably resulted from the presence of a gap between the HPT and IPT seal arms that had reduced the disc resonant frequency sufficiently for excitation to occur within the normal engine speed range.

(viii) The HPT disc was of an old design and had a relatively low fatigue margin.

(ix) A substantial number of previous catastrophic HPT disc failures had occurred.

(x) HPT disc failures, prior to that on G-OJEM, had been attributed by the engine manufacturer to turbine entry flow distortion, possibly in conjunction with the effects of turbine blade wear. Testing following G-OJEM's accident and analysis following the G-CEXF failure did not support this.

(xi) Effective action to address the suspected causes of the previous HPT disc failures and prevent recurrence had not been taken prior to the failure on G-OJEM.

(xii) The failure of the HPT disc on G-CEXF occurred shortly after the issue of SB Da 72-533 in April 2001, which was intended to prevent such failures. However, there was insufficient opportunity for the HPT disc on G-CEXF to have been modified. SB Da 72-533 was re-issued in July 2001 with revised compliance times, which took into account the age, in cycles, of the HPT disc.

(b) Causal factors

The investigation identified the following causal factors:

1 Minimal fatigue strength margin of the engine HPT disc resulted in it being susceptible to rapid cracking if subjected to vibratory excitation, such as resonance.

2 The abutment between the HPT and IPT discs probably resulted in a small gap being present between the seal arms while the engine was operating. This allowed sufficient reduction in the natural frequency of the turbine disc vibratory mode for it to be excited while operating within the normal speed range of the engine.

3 The protracted time taken following the G-OJEM event, due to the nature of the tests required, to understand the cause of the failure, precluded the timely introduction of suitable preventative action aimed at avoiding recurrence prior to the HPT disc failure on G-CEXF.

4 Fuel leakage from a severed low pressure pipe, part of the engine bay fuel system, led to a major fire, external to the nacelle.

4 Safety Recommendations

No new Safety Recommendations are made in this report. However, following the accident to G-OJEM, 19 recommendations were made in AAIB Report 3/2001 to the aircraft operator, the engine manufacturer and the CAA. Many of these were related to the specific details of that particular accident, aircraft type and operator. Two of the recommendations, however, were made directly in respect of the HPT failure and are pertinent to this investigation. These are re-produced below, together with the responses (dated 7 December 2001) from the CAA.

Safety Recommendation 2001-20

It is recommended that the engine manufacturer and the CAA reassess the susceptibility of the three-stage Dart turbine to HCF failure and ensure that effective action aimed at preventing recurrence has been taken.

CAA Response

The CAA accepts this recommendation

The CAA assessed the problem with the engine manufacturer and issued Airworthiness Directive 007-02-2001 on 22 February 2001. The directive is aimed at preventing recurrence of failure of the three-stage turbine disc owing to HCF by ensuring effective damping is present on the disc diaphragm.

The CAA revised this airworthiness Directive on 1 July 2001 in response to additional data, ie, the failure on G-CEXF. This revision instructed considerable reductions to the compliance period and targeted the higher cycle discs.

CAA Status – Closed

Safety Recommendation 2001-21

It is recommended that the CAA and the engine manufacturer consider the need for further improvement to their systems to ensure effective action to prevent recurrence following potentially catastrophic in-service failures of UK type-certificated equipment used on public transport aircraft.

CAA Response

The CAA accepts this recommendation

The CAA has reviewed its system for ensuring the effectiveness of actions taken to prevent recurrence of potentially catastrophic failures of UK type-certificated equipment on public transport aircraft. The CAA has concluded that the performance of its current systems has been sufficient to contain catastrophic failures on UK type-certificated equipment to an extremely low rate. Given its commitment to refining these systems where possible and necessary to ensure that this remains the case, the CAA concludes that it has met the intent of this recommendation.

CAA Status – Closed

Since the failure of the HPT disc on G-CEXF, the engine manufacturer has transferred the responsibility for its 'small engines', which includes the Dart engine series, to their facility based in Germany. The type certificate for these engines is now held by the German LBA, the national Airworthiness Authority.

P T Claiden

Inspector of Air Accidents
Air Accidents Investigation Branch
Department for Transport
June 2004

Unless otherwise indicated, recommendations in this report are addressed to the regulatory authorities of the State having responsibility for the matters with which the recommendation is concerned. It is for those authorities to decide what action is taken. In the United Kingdom the responsible authority is the Civil Aviation Authority, CAA House, 45-49 Kingsway, London WC2B 6TE or the European Aviation Safety Agency, Office G-12 02/74, Rue de Genève 12, B-1049 Brussels, Belgium.