AIRCRAFT ACCIDENT REPORT 3/2001

Air Accidents Investigation Branch

Department for Transport, Local Government and the Regions

Report on the accident to HS748-Series 2B, G-OJEM At London Stansted Airport On 30 March 1998

This investigation was carried out in accordance with
The Civil Aviation (Investigation of Air Accidents and Incidents) Regulations 1996

London : The Stationery Office

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ISBN 0 11 552304 9

Printed in the United Kingdom for The Stationery Office
7449/c4 12/01

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Department for Transport, Local Government and the Regions
Air Accidents Investigation Branch
Berkshire Copse Road
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December 2001

The Right Honourable Stephen Byers MP
Secretary of State for Transport, Local Government and the Regions

Sir,

I have the honour to submit the report by Dr King, an Inspector of Air Accidents, on the circumstances of the accident to HS748-Series 2B, G-OJEM at London Stansted Airport on 30 March 1998.

I have the honour to be
Sir
Your obedient servant

K P R Smart
Chief Inspector of Air Accidents

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GLOSSARY OF ABBREVIATIONS USED IN THIS REPORT

AAIB	-	Air Accidents Investigation Branch
ABP	-	Able bodied Passengers
ac	-	alternating current
AFS	-	Airport Fire Service
agl	-	above ground level
ANO	-	Air Navigation Order
ASB	-	Alert Service Bulletin
ATC	-	Air Traffic Control
CAA	-	Civil Aviation Authority
CSN	-	Cycles Since New
CSO	-	Cycles Since Overhaul
CVR	-	Cockpit Voice Recorder
Dc	-	Direct current
DFDR	-	Digital Flight Data Recorder
DRS	-	Dart Repair Scheme
DSL	-	Declared Safe Life
FAA	-	Federal Aviation Administration
FDR	-	Flight Data Recorder
FEA	-	Finite Element Analysis
FFPS	-	Flight Fine Pitch Stops
FV	-	Firth Vickers
g	-	acceleration due to gravity
HB	-	Brinell Hardness
HFC	-	High Cycle Fatigue
HP	-	High Pressure
IGPM	-	Imperial Gallons Per Minute
IP	-	Intermediate Pressure
IPC	-	Illustrated Parts Catalogue
ISA	-	International Standard Atmosphere
JAA	-	Joint Airworthiness Authorities
JAL	-	Jetstream Aircraft Limited
JAR	-	Joint Aviation Requirements
kt	-	knots
lbf	-	Pounds force
LCF	-	Low Cycle Fatigue
LP	-	Low Pressure
NGV	-	Nozzle Guide Vane
OM	-	Operations Manual
PA	-	Public Address
PSL	-	Predicted Safe Life
Rpm	-	Revolutions per minute
SB	-	Service Bulletin
TBO	-	Time Between Overhaul
TGT	-	Turbine Gas Temperature
TSN	-	Time Since New
TSO	-	Time Since Overhaul
SL	-	Sea Level
UTS	-	Ultimate Tensile Strength
1F	-	1^{st} Family
2EO	-	2^{nd} Engine Order

Air Accidents Investigation Branch

Aircraft Accident Report No: 3/2001 (EW/C98/03/7)

Registered Owner and Operator	Emerald Airways Ltd
Aircraft Type	HS748-Series 2B
Nationality	British
Registration	G-OJEM
Place of Accident	London Stansted Airport Latitude: 51° 52' N Longitude: 000° 13' E
Date and Time	30 March 1998 at 2331 hrs

Synopsis

The accident was notified to the Air Accidents Investigation Branch (AAIB) by Stansted ATC at 0020 hrs on 31 March 1998. The following inspectors participated in the investigation:

Dr D F King	Investigator in Charge
Mr A N Cable	Engineering
Mr P N Giles	Operations
Mr J R James	Flight Data Recorders

Immediately after take-off from London (Stansted) Airport, on a night flight with 40 passengers and 4 crew on board, an uncontained failure of the right engine occurred. This resulted in sudden power loss and a major engine bay fire. The Commander elected to land back on the runway. The aircraft overran the paved surface, and uneven ground in the overrun area caused the nose landing gear to collapse. After the aircraft had come to rest, with the engine bay fire continuing, the crew organised a rapid evacuation and all the occupants escaped, with little or no injury. The engine bay fire was extinguished by the Airport Fire Service (AFS), but fuel release continued for some hours.

The sustained fire around the damaged engine bay resulted from fuel system disruption caused by the engine failure. Disruption to the aircraft electrical system by the nose landing gear failure caused the Low Pressure (LP) fuel shutoff valves and engine bay fire extinguishers to be disabled after the left engine had been shut down. Additionally, electrical wiring damage had rendered the 2nd shot fire extinguisher for the damaged engine bay unserviceable prior to the accident. The crew was unable to deploy the rear door manual chutes to assist the escape from the cabin of the burning aircraft without causing significant delay to the evacuation; the escape chute system was judged to be inadequate.

The engine failure was caused by high-cycle fatigue cracking of the High Pressure (HP) turbine disc. Four similar Dart turbine failures had occurred over the previous 26 year period. These had been attributed to a combination of turbine entry flow distortion and turbine blade wear. The engine manufacturer and the CAA had concluded that the likely period before recurrence of the failure was such that additional remedial action was unnecessary, and measures aimed at fully controlling the suspected causes had not been taken. A further failure, apparently very similar to G-OJEM's, occurred in June 2001. While G-OJEM's engine failure was initially attributed by the engine manufacturer to the same causes as the previous cases, the evidence was unconvincing and major difficulty was experienced in determining the likely causes. Following protracted study and testing by the engine manufacturer, the evidence eventually indicated that an excessive gap between turbine assembly seal members, possibly influenced by the fit of the turbine clamping bolts, had probably resulted in high cyclic loads on the HP turbine disc. The disc fatigue strength had been significantly reduced by corrosion, and fatigue cracking had resulted. Mandatory modification requirements to address the causes were issued in April 2001.

The investigation identified the following causal factors:

1. Significant reduction in the fatigue strength of the HP turbine disc due to surface corrosion.

2. Inadequate control of the fit between engine turbine assembly seal members, possibly influenced by inadequate turbine clamping bolt fit, causing sufficient reduction in the natural frequency of an HP turbine disc vibratory mode to allow its excitation within the normal operating speed range and consequent excessive stressing of the disc.

3. Fuel leakage from the engine bay fuel system, resulting in a major nacelle fire.

4. Failure to identify the turbine assembly seal member fit and HP turbine disc corrosion as possible contributors to disc fatigue damage after previous similar failures.

19 safety recommendations are made.

1. Factual Information

1.1 History of the flight

The aircraft was scheduled to depart from London (Stansted) Airport at 2230 hrs on a one hour flight to Leeds Bradford Airport. A baggage problem delayed the flight and the aircraft eventually taxied at 2323 hrs, to holding point 'HP' for Runway 23. Take-off clearance was given at 2329 hrs. The First Officer was the handling pilot and the take-off was to be made with full dry power; the water methanol system was selected to standby.

The following sequence of events was established from the flight recorders. The aircraft was cleared to take-off and the First Officer called for full dry power. As engine power stabilised the First Officer called that the warning 'lights were out and the emergency panel was clear'. As the aircraft accelerated, the Commander announced 'sixty knots' and relinquished steering control to the First Officer who acknowledged and confirmed, 'full dry we have, just slightly low on the right'. No significant variation in engine rpm between the two engines could be detected from the Flight Data Recorder (FDR) recording.

The aircraft accelerated through 80 kt and, for a period of two seconds, the sound of the nose wheel running over the runway centreline lighting was recorded on the area microphone channel of the Cockpit Voice Recorder (CVR). At an airspeed of 111 kt the Commander called 'vee one, rotate', the First Officer moved the control column rearwards and the aircraft became airborne.

Less than five seconds after the 'rotate' call, at an airspeed of 115 kt and a height of between 30 feet and 100 feet agl, the sound of a sharp report followed by an engine run-down was recorded on the CVR. The aircraft yawed 11° to the right of the runway heading. As the crew asked each other what the noise had been, loud shouting could be heard from the passenger cabin. The First Officer said, as he corrected the yaw, 'something's gone' and the Commander then stated that he had taken control of the aircraft. Within eight seconds of the event the First Officer stated that an engine had stopped. Simultaneously, the senior cabin attendant, using the Public Address (PA) system, told the passengers to sit down and then advised the flight deck crew via the interphone that the right engine was on fire. Engine power was reduced and the aircraft yawed 14.5° to the left of runway heading. Four seconds later, the sound of the engine fire warning bell was recorded. Without using the PA system, the senior cabin attendant told the passengers to 'stay in your seats and make sure your seatbelts are all fastened'.

The aircraft was in the air for a total period of 27 seconds before the noise of touchdown was recorded. The Commander called for brakes, to which the First

Officer replied ‘coming on’. The First Officer then suggested that he ‘fire’ the right engine fire bottle but the Commander asked him to call the fire brigade, which he then did continuously. The Flight Fine Pitch Stops (FFPS) warning horn activated 5 seconds after touchdown, 4 seconds before the aircraft ran off the end of the runway at 62 kt. The warning sounded for the remainder of the audio recording.

After the aircraft left the runway, the CVR cockpit area microphone picked up the noises of the aircraft rolling over uneven ground, the point at which the perimeter track was crossed and the final collapse of the nose landing gear. Recording on both the FDR and CVR terminated due to the removal of electrical power 7.1 seconds after the aircraft departed the paved surface. An annotated track of the aborted take-off is shown in Figures 1 and 2.

When the aircraft came to a halt, the First Officer left his seat immediately to assist with the evacuation of the aircraft. The Commander carried out a limited shutdown, which included:

HP cocks	OFF
Fuel Pumps	OFF
Water Meth	OFF
Electrics	OFF
Electrical System CBs	6 Pulled

He then satisfied himself that the cabin was clear before leaving the aircraft. All passengers and crew evacuated the aircraft without serious injury.

1.2 Injuries to persons

Injuries	Crew	Passengers	Others
Fatal	-	-	-
Serious	-	-	-
Minor/none	4	40	-

1.3 Damage to aircraft

Aircraft damage consisted of severe disruption and fire damage to the No 2 powerplant, detachment of the nose landing gear and localised disruption of the associated fuselage structure and deformation of the forward fuselage undersurface (Figure 3).

1.4 Other damage

Minor disruption of the grassed runway overrun area.

1.5 Personnel information

1.5.1

Commander:	Male, aged 61 years	
Licence:	Airline Transport Pilot's Licence	
Instrument rating:	Valid to 3 January 1999	
Base check:	Valid to 3 June 1998	
Line check:	Valid to 16 January 1999	
Medical certificate:	Class 1, valid to 30 April 1998	
	Limitations: corrective lenses for near vision	
Flying experience:	Total all types:	6,100 hours
	Total on type:	3,950 hours
	Total last 28 days:	39 hours
	Total last 24 hours:	3 hours
Previous rest period:	Off duty:	1040 hrs on 30 March 1998
	On duty:	2200 hrs on 30 March 1998

1.5.2

First Officer:	Male, aged 33 years	
Licence:	Commercial Pilot's Licence	
Instrument rating:	Valid to 14 September 1998	
Base check:	Valid to 14 September 1998	
Line check:	Valid to 10 October 1998	
Medical certificate:	Class 1, valid to 31 January 1999	
	Limitations: none	
Flying experience:	Total all types:	1,100 hours

	Total on type:		250 hours
	Total last 28 days:		35 hours
	Total last 24 hours:		2 hours
Previous rest period:	Off duty:		1040 hrs on 30 March 1998
	On duty:		2200 hrs on 30 March 1998

1.6 Aircraft information

1.6.1 General information

Manufacturer:	Hawker Siddeley Aviation Ltd (now BAE Systems plc)
Type:	HS 748 Series 2B
Aircraft Serial No:	1791
Year of manufacture:	1983
Certificate of Registration:	G-OJEM/R1, issued on 22 March 1996
Certificate of Airworthiness:	Valid until 14 April 1999
Engines:	2 Rolls Royce Dart 536-2 turboprop engines
Total airframe hours:	18,352 hours (19,122 flight cycles)

1.6.2 Aircraft weights

Maximum Take-off weight	21,096 kg
Actual Take-off weight	19,702 kg
$V_1/ V_R/ V_2$	111 kt

1.6.3 Aircraft Description

1.6.3.1 General

The aircraft is of conventional layout (Figure 4), powered by two Rolls Royce Dart Mark 536-2 gas turbine engines, each driving a 4-bladed propeller. Air Registration Board approval for the type was granted in 1962. The original manufacturer was A V Roe & Co Ltd, which became part of Hawker Siddeley Ltd, and then British Aerospace after manufacture of the type ceased in 1984. Design authority subsequently passed to Jetstream Aircraft Ltd (JAL), which then became part of British Aerospace Regional Aircraft. At the time of the accident there were approximately 260 aircraft of the type in service in various parts of the world, including 14 in the UK, carrying passengers and/or freight.

1.6.3.2 Powerplant

The engines are located forward of and generally above the wing in wing-mounted nacelles. 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 aluminium alloy cowl panels. The main landing gears and other equipment are housed in the lower part of the nacelles. An equipment bay is located in the upper part of the nacelle behind the engine, and the engine jet pipe passes beneath this bay and over the top surface of the wing.

The installation includes a water/methanol injection system to provide additional power, known as 'wet power'. The powerplant controls include a system that, in the event of loss of thrust from an engine on take-off, automatically selects full wet power on the other engine.

The range of propeller blade pitch in flight is restricted by Flight Fine Pitch Stops (FFPS). These can be withdrawn when the throttles are near the retarded position by operating a lever on the flight deck centre console to allow ground fine pitch to be selected. A warning horn sounds 3-5 seconds after the aircraft senses weight-on-wheels if ground fine pitch is not selected.

1.6.3.3 Fuel System

Fuel is carried in a single integral tank in each wing outboard of the nacelle and feeds under gravity to electric boost pumps in a collector tank located inboard of the nacelle. The pumps deliver fuel forward to the nacelle firewall via an LP fuel cock and a flexible hose. Forward of the nacelle firewall the fuel passes through a fuel heater assembly located beneath the engine hot section and through an engine firewall to the engine fuel control unit. The inlet and outlet pipes to the fuel heater (Figure 5) incorporate flexible joints to cater for small amounts of relative axial displacement and misalignment between the connectors due to engine movement. The telescoping element of each flexible joint is retained in its housing by a circlip located in a groove. The fuel heater and the associated pipes between the two firewalls is surrounded by a shroud to contain any leakage; the shroud is drained overboard via an outlet port at the bottom of the nacelle, just forward of the nacelle firewall.

Each LP fuel cock is an electrically motorised gate valve that can be closed by operation of a switch on the flight deck Emergency Panel (forward upper panel).

The dc motor of the valve is supplied from the aircraft battery busbars. The valve position is shown by an electromagnetic indicator adjacent to each switch.

1.6.3.4 Electrical System

Main aircraft electrical power is generated by two 28v dc generators, one driven by each engine with ac power provided by 115v ac inverters. With the generators off-line, electrical power is supplied by 4 Nickel-Cadmium batteries mounted on the upper forward face of the fuselage forward pressure bulkhead, which are paired to feed two battery busbars. The battery busbars are each connected to a single aircraft centre busbar via a contactor and a reverse current circuit breaker. The earth circuit cables for the batteries and for the battery contactors pass through the lower part of the forward pressure bulkhead and rearwards through the lower fuselage, outboard of the nose landing gear bay side beams. The cables are earthed by means of swaged ring tag terminals that are bolted to the side beams adjacent to the nose landing gear mounts.

Two inertial crash switches are installed one in each main landing gear bay. They are designed to operate under a longitudinal deceleration of 3g or greater, to automatically operate the engine bay fire extinguishers, switch on cabin lighting and isolate two of the batteries from the centre busbar.

1.6.3.5 Engine Bay Fire Protection Systems

A fire zone around the hot section of the engine is formed by a nacelle firewall, located 40 inch forward of the wing front spar, 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 bulkheads. Nacelle zones are defined in Figure 6.

A fire detection loop passes through Zones 1, 2 and 4 of each engine bay. Electrical resistance reduction in the detector element, due to excessive bay temperature, triggers the control unit to activate a flight deck firebell and an engine fire warning red light on the flight deck Emergency Panel.

A spherical bottle in the equipment bay in each nacelle contains 12 lb of engine bay fire extinguishing medium. Each bottle has two operating cartridges and can be discharged to spray-rings in either engine bay. The wiring diagrams indicate that the first shot for an engine bay is provided by the bottle in that engine's nacelle and the second shot by the bottle in the other nacelle. Cartridges can be fired electrically by a selector/firing switch for each bottle on the flight deck Emergency Panel; each switch is locked by a hinged guard that is secured by a frangible wire tie. Electrical power is taken from the aircraft's centre busbar. Cartridge operation is indicated by two fuse indicators adjacent to each switch.

Extinguisher cartridge cable runs in the wing incorporate Breeze type plug/socket connectors (PN 508-2-11360). The cables enter the connector through an insulating sleeve and attach to pins located in a moulding. The sleeve and moulding are clamped into a shell body by a coupling nut. The sleeve and moulding are not keyed to prevent relative rotation between them.

1.6.3.6 Nose Landing Gear

The nose landing gear leg is pintle mounted on two pivot bearings near the aft end of the nose gear bay. Each bearing is fastened to a longitudinal trussed side beam forming the sides of the bay. A landing gear drag strut attaches to a fitting mounted on the aft wall of the bay. With the landing gear extended, the nosewheel tyres are located 20.7 feet forward of the mainwheel tyres.

1.6.3.7 Cabin Safety Equipment

Two doors are installed at the rear of the cabin, a passenger door on the left and a baggage door on the right, both of which serve as emergency exits (Figure 4). Access to the baggage door from the cabin is via the toilet door. With a normal aircraft ground attitude the rear door sills are approximately 6 feet above ground level; however, with the aircraft resting on its mainwheels and nose fuselage this height increases to approximately 11 feet.

Two emergency exit hatches are located over the centre wing section, one on either side of the cabin. Each provides an exit aperture 19 inch wide x 26 inch high.

A crew/freight door is fitted at the front left side of the cabin. It is not an approved emergency exit but can be used if circumstances permit. The flight deck has a sliding window on each side, with adjacent escape ropes, that can be used as emergency exits.

Each rear door is provided with an escape chute, consisting of an approximately 12 foot long canvas chute, located in an underfloor stowage compartment in the rear vestibule area and deployed manually. Each chute consists of a centre panel with left and right wing panels. The inboard end of the centre panel is pre-attached to anchor rings in the stowage compartment by two webbing straps, each terminating in a spring clip. Each wing panel has a spring clip at its inboard upper corner, which can be manually attached to a catchplate on each side of the doorframe, and has three handling loops near its outboard end. The anchor points and clips are colour coded; the Emerald Airways Ltd Operations Manual

(Page 9.10.2, issued 31 March 1995) notes that compartment anchorages are coloured one yellow and one blue and the doorframe anchorages are coloured one green and one red. A 15 foot long nylon rope, knotted at 12 inch intervals to assist grip, is fastened to the aft hook. The stowage compartment hatch, hinged on its forward edge and forming part of the cabin floor, has a lifting ring at one corner and is placarded, 'LIFT TAG FOR ESCAPE CHUTES', on the outer surface and with operation instructions on the inner.

Chute deployment involves opening the stowage hatch, lifting out the rolled chute and throwing it and the rope out of the doorway and attaching the wing panel spring hooks to the catchplates. The Operations Manual specifies that two Able Bodied Passengers (ABP) should then shin down the rope to the ground and use the handling loops to hold out the end of the chute for cabin occupants to slide down. It is noted that in the event of a nose undercarriage collapse the ABPs "should, if possible, make an attempt to keep the slide as high as possible to reduce the angle."

It was reported that some HS748 aircraft, including G-OJEM, had previously been fitted with automatically deployed inflatable escape slides at the rear doors. These had been removed because of serviceability problems.

The basic philosophy of an emergency evacuation is to use all available exits. On land, the doors are considered the primary escape routes and the overwing emergency exits as secondary routes because of their smaller size and the greater fire potential in the wing area.

Cabin crew member seats are installed adjacent to the cargo door and the rear left door. A removable phone handset clipped into a socket above the left rear cabin door connects to the aircraft interphone and cabin PA systems. The controls for the phone are installed in the socket, which is located 4.3 feet above the pan of the rear cabin crew seat and 1.3 feet left of the seat centre. A portable battery powered loud hailer is also provided, stowed in an overhead locker.

1.6.4 Engine Description

1.6.4.1 General

The No 2 engine was a Rolls Royce Dart Mark 536-2, Serial Number 334 ('Engine 334'), a single spool turboprop engine with a 2-stage centrifugal compressor, 7 straight flow can combustion chambers and an axial flow turbine (Figure 6). 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. 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 civil service on the Vickers Viscount. Developed versions were used in a number of aircraft types. The RDa7 and RDa10 Series (approximately 1,700-2,750 shp) were developments having a different, three stage, turbine; they entered service in 1958 and 1965 respectively.

The Mark 536-2 engine, the engine type involved in G-OJEM's accident, 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 and their accumulated service time were approximately as follows:

	RDa7	RDa10	RDa7 & RDa10
Total Number Produced	1,500	500	2,000
Accumulated Service Time – Millions of Hours	68	14	82

Military versions of these series were designated as the RDa8 and RDa12; their combined total accumulated service time was less than 1 million hours. The engine manufacturer estimated that at the time of G-OJEM's accident there were approximately 2,800 turbines in service of the type installed in G-OJEM's No 2 Engine.

The general layout of the Dart Mark 536-2 is shown in Figure 6. The main structure consisted, from front to rear, of reduction gear, compressor, intermediate and nozzle box casings. A flight deck placard listed the Take-off power rating as 15,000 rpm spool speed with a maximum allowable Turbine Gas Temperature (TGT) of 810°C dry and 920°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 1,000°C TGT, with no more than 5 seconds above 920°C.

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 (Figure 7). The chambers are suspended between the diffuser outlet elbows and discharge nozzles in the nozzle box. An interconnector joins the casings and flame tubes of each combustion chamber to its neighbour. The combustion chambers were not routinely serial numbered.

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 334 was fitted with fuel-washed burners. The majority of Dart engines in service at the time of G-OJEM's accident had air-washed burners.

Different standards of fuel-washed burner were approved for the Dart; for the type fitted to Engine 334 the swirler assembly consisted of a distributor plate and a swirl chamber, clamped between internal registers in the body and shroud (Figure 8). With the burner assembled, the distributor plate and swirl chamber seated against the respective register and against each other. High pressure fuel from the fuel control unit passed through the strainer and via 8 holes in the distributor plate to an annular channel formed in the aft side of the swirl chamber, and thence via two tangential slots to a central cavity in the chamber. The swirl chamber slots imparted a spiral motion to the flow and formed the controlling restriction in the fuel flow path through the burner. The main fuel flow exited the cavity by a discharge orifice and entered the flame tube as a conical spray. A 0.010 inch diameter hole in the base of each slot bled fuel into the shroud annulus to provide the fuel-wash flow. The body and shroud were prevented from unscrewing by a sprung lock ring.

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, possibly due to asymmetry in the bleed chamber. 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.

The burners had been the subject of a considerable number of modifications over the years. Identification of the burner Part No that specified the correct burner standard for a particular engine application and combustor position was given in the Rolls Royce Dart Parts Catalogue, Section 73-1. The burner Part No listing against various pre- and post-modification combinations was extensive and complex. The burners were not routinely serial numbered. The correct identifiers for G-OJEM's No 2 Engine were:

BURNER NO	TYPE NO	NO OF TUBE THREAD FLATS
1, 4 & 6	146	2
2 & 3	23 A.F.	0
5 & 7	57 B.E.G	1

A common practice among overhaul agencies had been to identify engines for which fuel-washed burners were fitted by applying a 'FUEL WASHED' caption to the outside surface of each combustor casing, either by printing or decal. No identification was provided where air-washed burners were fitted.

Regular cleaning of burners was recommended (Rolls Royce Dart Maintenance Manual, Section 73-1), to remove carbon accumulations from the discharge nozzle exterior in order to 'ensure maximum life of the hot end components and trouble free starting'.

Procedures for fuel burner testing (Rolls Royce Dart Overhaul Manual, Section 73-1) required test rig checks of flow rate, spray pattern and external leakage. Burners that failed to meet the requirements were to be rejected. Procedures were given for lapping parts of the burner, generally with a polishing medium, to rectify faults in the flow rate or spray pattern. These included measures to increase the flow rate of an air-washed burner, but this was not permissible for a fuel-washed burner and it was noted:

"CAUTION: DO NOT ATTEMPT TO INCREASE THE FUEL FLOW OF MOD.1536 OR MOD.1894 FUEL BURNERS [fuel-washed burners] BY DEEPENING THE SLOTS. IF AN INCREASE IN FUEL FLOW IS REQUIRED USE SELECTIVE ASSEMBLY OF THE SWIRL CHAMBER."

There was no requirement, when an engine was undergoing a repair or overhaul shop visit, for the burners to be flow checked in their as-received condition and reportedly this was seldom done.

1.6.4.4 Turbine Assembly

The turbine assembly is housed in the nozzle box and connected to the compressor and the reduction gearbox by separate concentric shafts. It comprises, from front to rear, High, Intermediate and Low Pressure (HP, IP and LP) turbine stages (Figure 9). 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 5 taper bolts passing through taper-reamed holes in the discs and a further 5 taper bolts clamp the HP and IP discs together. The bolts are designed with an interference taper fit in each turbine disc, with a bolt draw (axial protrusion of bolt when unloaded) of 5.25×10^{-3} inch 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; HP and IP blade shrouds incorporate tip seals. The design clearance between the platforms and shrouds of adjacent HP blades is $1.25\text{-}2.0 \times 10^{-3}$ inch. Wear in engine service tends to increase this clearance, which can have an adverse effect on the vibratory characteristics of the turbine assembly (Para 1.18.3.7).

The HP turbine has 131 blades (PN RK45409), each weighing 0.04445 lb, which are retained in the disc buckets by lock plates. The HP turbine blades are a high cost component of the engine and excessively worn blades could be repaired under a Dart Overhaul Manual, Dart Repair Scheme (DRS) by weld depositing material onto the worn face(s) and machining back to the required dimension. The clearance was not measured directly but inferred from gauge checks of individual blades.

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 frettage. 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 0.002 inch be rectified in accordance with DRS 611. This followed testing in around 1990 showing that increased shroud or platform gaps could lead to increased stresses in the blade root area (Para 1.18.3.7).

The incorporation of the schemes was generally recorded in engine overhaul workpacks compiled by overhaul agencies but was not required to be recorded in the Engine Log Book. The repair scheme designation remained the same through the above changes and it was reportedly sometimes difficult or impossible to subsequently ascertain the standard to which a particular set of blades had been inspected and repaired. DRS 297 was cancelled in August 1992. It was reported that DRS 611 had not been incorporated on most Dart engines in service at the time of G-OJEM's accident.

1.6.4.6 HP Turbine Disc

The HP turbine disc (PN RK40726) 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 1960s when the radii associated with the blade buckets were increased (Modifications 839 and 911).

The disc is 15.24 inch in diameter, with the cross section shown in Figure 10. It has a hub area around 5 inches in diameter with a central bore and 10 bosses near its outer edge bored and reamed to accept the taper attaching bolts. Outside the hub region the disc thins to a tapered diaphragm section with a thickened outer rim that contains the blade buckets. Seal rings formed 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 inch and 9.0 inch respectively. The specified blend radius between the inner edge of the aft seal ring and the diaphragm is 0.10-0.15 inch.

Similar seal rings are formed on the IP turbine disc. With the turbine assembly clamped together, the design gap (or 'nip') between HP-IP seal rings at room temperature is zero, with a tolerance of $\pm 0.5 \times 10^{-3}$ inch; the IP-LP seal ring design fit is an interference of $2\text{-}4 \times 10^{-3}$ inch. Cooling airflows are directed across the forward and aft faces of the HP turbine disc.

1.7 Meteorological information

The METAR at 2320 hrs was:

Surface wind	270°/5 kt
Visibility	>10 km
Cloud	Overcast, base 2,200 feet
Weather	Nil
Temperature/Dew Point	+8°C/6°C
QNH	1016 mb

1.8 Aids to navigation

Not applicable

1.9 Communications

At the time of the accident the aircraft was in contact with the London Stansted Tower controller on frequency 123.8 MHz. This frequency was recorded and a speech transcript was made.

1.10 Aerodrome information

The physical characteristics of Runway 23 were:

Magnetic Heading	227°
Dimensions	3,048 x 46 metres
Surface	Asphalt
TORA/ASDA	3,048 metres

Runway lighting	
Centreline	High Intensity Bi-directional, 15 metre spacing
Edge	High Intensity flush, 46 metre spacing
End	Red wing bars

The overrun area is grassed and is crossed by the Airside Road about 80 metres from the end of the paved area.

1.11 Flight Recorders

In accordance with requirements of the Air Navigation Order (ANO) the aircraft was fitted with a 25 hour Digital Flight Data Recorder (DFDR) and a four channel, 30 minute CVR. Prior to being entered in the UK Register, the aircraft had been fitted with extra transducers for pitch, roll, lateral acceleration and control surface positions consistent with the requirements of a USA Federal Aviation Administration (FAA) compliant DFDR installation. These extra aircraft parameters were in excess of those required to meet the minimum requirements applicable to an aircraft of this type.

1.11.1 Digital Flight Data Recorder Description

The DFDR to be fitted to the aircraft was identified in the Illustrated Parts Catalogue (IPC) as a Plessey PV1584. This part number is generic only and does not specify in enough detail which particular variant to fit. There are a number of different recorders available with this main part number, each of which is specified with a suffix letter; e.g. PV1584A, PV1584B, etc. All the models look very similar and will all fit into the same avionics rack in an aircraft. However, the operation and aircraft interface requirements of each of the variants is subtly different.

The operator of the accident aircraft had access to two of the variants; PV1584A and PV1584D. The main difference between these two is the programming of the multi-purpose inputs to suit the aircraft installation. Each of the main analogue inputs can be configured to allow connection to various types of transducer, i.e. a synchro, potentiometer, high or low level dc, ac phase ratio etc. To configure an input, the 'input type' control line associated with the multi-purpose input should be connected to the 'transducer type' output line. This connection is made as part of the aircraft wiring, typically on the back of the DFDR mounting rack. The programming of the inputs on a PV1584D is achieved by the selection of one of eight 'programmes' using three DFDR input control lines. These 'programmes' are pre-determined by the DFDR manufacturer and allow for up to eight different types of installation. They are generally used to switch between operation under FAA or Civil Aviation Authority (CAA) requirements where different sampling rates of some parameters are mandated. These differences in the installation and operation of the various DFDRs can be confused further when the modification state of the DFDR is taken into account. For example, a PV1584D with modifications 5 and 6 embodied will cause the equipment to function as if it were a PV1584L, with different sampling rates and parameter word-slot assignments in the data frame. At the time of the accident, the DFDR installation on G-OJEM was configured for a PV1584D with the programming inputs selected for FAA operation. It should be noted that the minimum recording rate of normal acceleration is 4 times per second under FAA requirements whereas the CAA stipulate 8 times per second.

To allow the aircraft to be entered on the UK Register the DFDR installation should have been altered to ensure that the sampling rate of the recorded parameters satisfied the requirements of the CAA. This requirement was overlooked by both the CAA surveyor and the operator of the aircraft.

1.11.2 DFDR Calibration and Mandatory Readout

The installation on G-OJEM was required to be calibrated and the DFDR read out on an annual basis, as set down in the aircraft maintenance schedule. At the time of the last assessment, carried out on 6 February 1998 by a JAR 145 approved organisation, the aircraft was fitted with a PV1584D (without modifications 5 and 6 embodied). Calibration of the installation required the aircraft transducers to be exercised over their full range and, using an appropriate test set connected to the DFDR, a note taken of the raw DFDR data being recorded at various points of transducer travel. Test results sheets were provided for the maintenance engineer to note down the values obtained. However, the results sheet did not state a range of expected values at each test point to enable the engineer to assess correct transducer operation. Following the accident, the altitude transducer was removed from the aircraft, tested by the manufacturer and found to be extremely worn. From the date of manufacture of the transducer it is likely that it was originally fitted to the aircraft at the time of build and had not been overhauled since. The degree of wear was such that no useful altitude information pertinent to the accident was recovered.

As part of the annual check, CAA Specification 10 requires a read-out from a completed flight and an assessment of the standard of the recorded data. A copy tape of the DFDR recording was submitted to an approved replay organisation to fulfil this requirement. It is industry practice to replay the copy tape through appropriate equipment, select the first sample of each parameter that appears in every one second subframe of data and plot the parameters in the form of line traces. This method is generally employed even if the parameters are sampled more than once in the subframe. In the case of the normal acceleration, for a CAA compliant recording the parameter should have been sampled and recorded eight times in each second, whereas an FAA compliant recording (like that from G-OJEM) would only have four samples in the same period of time. This error in sampling rate was not detected by the replay organisation as there currently is no stipulation that all of the samples of a particular parameter in a data frame should be inspected.

On 26 February 1998 the aircraft tech log showed that the DFDR fail light came on intermittently. On the day after this entry, the PV1584D was replaced with a PV1584A by the operator of the aircraft and the defect report was cleared. The DFDR calibration procedure states that in the event of having to replace the DFDR, no recalibration is necessary and none was carried out. As the aircraft did not have the 'transducer type' wiring links fitted to the mounting rack, the main effect of changing the DFDR variant was to render the subsequent recordings of most of the parameters unintelligible. The affected parameters were Fine Airspeed, Pitch, Roll, Normal and Longitudinal Acceleration, Rudder, Aileron, Pitch Trim and both Engine Torques.

During the investigation, it was found that the operator had one other aircraft of this type which had been fitted with a PV1584A (G-BVOV) and two others fitted with the PV1584D. The situation was brought to the attention of the operator and the DFDR on G-BVOV was replaced with a PV1584D on 28 April 1998.

1.11.3 Accident flight

The data for the accident flight has been derived from the unaffected DFDR recorded parameters (Airspeed Coarse, Heading, Engine RPM, Flap and Push-to-talk), the Cockpit Voice Recording and recorded data from the Stansted radar head.

During the first 15 minutes of the CVR recording the loading of the passenger luggage was completed. The crew decided to run through the pre-flight checks as far as they were able. Bug speed was calculated as 111 kt and the QNH, unchanged from that previously noted, was 1016 mb. The First Officer, who was to fly the sector from the right-hand seat, asked for the captain's briefing, which the Commander stated to be standard right-hand seat. The First Officer acknowledged this, added that standard emergency procedures would be used if required and that they would follow the 'Buzad 4 Romeo' departure with an initial altitude of 3,000 feet. He further said that they would fly visually back for a landing on Runway 23 should they have any problems and this was agreed by the Commander.

Once clearance was obtained, the right engine was started, followed by the left engine. Both engine starts were uneventful. During the engine starts the senior cabin attendant used the PA system to give a safety briefing to the passengers and the crew received confirmation of the 'Buzad 4 Romeo' departure.

The crew carried out the post-start checks and then taxi checks whilst they manoeuvred to hold short of Runway 23. Flap 7.5 was selected and the Water / Methanol system set to 'Standby'.

Having lined up on the runway the crew performed a full and free check of the flying controls, the First Officer checking control column and control wheel movement and the Commander operating the rudder pedals. The events during the take-off run are given in Section 1.1.

1.12 Aircraft and site examination

1.12.1 Accident Site

Following the accident, a significant quantity of engine debris was found scattered on the mid section of Runway 23 and its surrounds, over an area extending between 1250-1800 m from the start of the runway (Figure 1). The debris included nacelle cowl and engine casing parts, together with fragments of the NGVs, blades and disc associated with the HP turbine. The HP turbine disc was found in two major pieces, each 35-45 m from the runway centreline, with the larger piece on a runway turn-off south-east of the runway and the smaller on the grass to the north-west.

G-OJEM came to rest in the overrun area beyond the end of Runway 23, within the airport boundary. Tyre marks on the runway and overrun area could be traced to the aircraft and thereby positively attributed to G-OJEM's landing ground roll. They showed that the aircraft had touched down on the left side of the runway with its right mainwheels, 2,600 m from the start of the runway and 18 m left of the centreline, while tracking 3° right of the runway heading. Nosewheel and left mainwheel touchdown followed approximately 40 m later. The aircraft then turned left until parallelling the runway centreline and ran along the left side of the runway to its end, 440 m from initial touchdown. The tyre markings in places showed a regular variation in density, characteristic of the normal operation of the anti-skid system whereby the brakes are momentarily released when heavy braking causes imminent wheel locking.

The overrun area at the end of the runway was level, generally firm and relatively smooth, with a covering of short grass. It contained a series of low-mounted approach lights and was crossed at an angle by a disused metalled road. The surface of the 6 m wide road was very slightly elevated above the level of the surrounding ground; a survey showed an upslope onto the road from the east of 8 inches over a distance of 4.5 m and a downslope off the road to the west of 15 inches over a distance of 11 m.

Tyre marks showed that after leaving the runway G-OJEM continued to roll straight on all three landing gears until encountering the road, 78 m from the end of the runway. At this point the tyre marks associated with all three landing gears ceased. Main gear tyre contact was re-established on the second half of the road, but marks and wreckage distribution showed that the nosewheel tyres had not re-contacted the ground until 18 m beyond the road leading edge. The ground on the west side of the road was somewhat softer than that between the runway and the road and the nose landing gear wheels broke the surface at the point where they re-landed. The marks and wreckage distribution showed that the nose landing gear had collapsed rearwards at this point and come to rest shortly thereafter. The aircraft slid for a further 22 m on the forward fuselage undersurface and the mainwheels before coming to rest 118 m from the end of the runway, 558 m from the initial touchdown point.

1.12.2 Aircraft Examination

1.12.2.1 General

Examination showed that the No 2 engine had suffered severe non-containment damage, fire had damaged the No 2 nacelle and the adjacent part of the wing, and the nose landing gear leg had been torn off.

1.12.2.2 Nacelle

The No 2 nacelle cowl panels remained in place but had been holed on the top centreline in the region of the HP turbine and it was clear that this damage had been caused by impact from engine debris. Additionally, most of the cowl panels had been fire damaged, including the lower and aft portions of the engine bay cowls and most parts of the jet pipe and landing gear bay cowls. The fire damage also extended to the wing adjacent to the inboard side of the nacelle, to the inboard wing flap and to the right main landing gear tyres. The damage generally consisted of soot blackening, together with destruction of the paint coating in many areas. It was more extensive and generally more severe on the inboard than the outboard side and on exterior rather than interior surfaces. In some areas the cowl panel material had suffered visible local overtemperature damage, in one case causing loss of material, but the fire damage had not resulted in significant breaching of the cowls. The wing damage consisted of sooting and paint blistering of the external surface of the leading edge fairing, and moderate sooting of parts of the top and bottom exterior surfaces of the wing torque box.

Some of the fire damage had directional features indicative of having occurred while the aircraft was in flight and/or during the ground roll. The characteristics were consistent with the effects of relatively brief internal overheating in the nozzle box region, plus a more prolonged overheating emanating both from the hole in the top of the cowl area and from the aft lower part of the engine bay while a rearward airflow was present. They also indicated that overheating originating at the aft lower part of the engine bay, without any appreciable airflow present, had occurred.

1.12.2.3 Nose Landing Gear

The nose landing gear detached as the result of failure of the fuselage supporting structure for the leg and failure of the drag link attachment. The right hand pintle bearing block for the leg had separated from the support structure with only localised damage to the side beam, but on the left side the aft part of the beam had been torn out with the leg. The failures were consistent with the effects of rearward overload applied to the wheels.

1.12.2.4 Electrical System

The studs attaching both left battery earth cables to the structure were fitted to the part of the left side beam in the nose landing gear bay that was torn out by the gear collapse. The structure had remained with the detached landing gear and had broken out the terminal ring tags for the cables, causing their disconnection from aircraft earth. In addition, the earthing wires for the right battery contactors severed as a result of tensile overload. Tests showed that the wiring damage had isolated all four batteries from the aircraft electrical system, as would be expected. The inertial crash switches had not tripped in the accident. All four batteries were at normal voltage (26-27 v) and temporary repairs to the damaged earth circuits restored normal battery power to the aircraft.

1.12.2.5 Fuel System

Both LP fuel cock switches were found On and both cocks were open. Checks during the investigation with battery supplies restored to the aircraft showed that both valves and the associated indicators operated normally when selected by the flight deck switches. Operating time was approximately 2 seconds.

At the time of the AAIB Inspector's arrival at the aircraft 4 hours after the accident, fuel was found to be draining from the engine LP fuel pipe drain, near the aft end of the No 2 engine bay, in a steady stream, at an estimated rate of 1-2 IGPM (Imperial Gallons per minute). The flow had reportedly been continuous since the extinguishing of the fire around the No 2 nacelle. It was stopped by operation of a manual shut-off valve in the No 2 engine LP fuel supply line in the wing. Examination of the fuel heater assembly revealed that the circlips retaining two of the four flexible pipe joints had been displaced from their grooves, thus allowing both joints to partially disengage to the point where substantial leakage paths were created. Electrical bonding cables bridging the joints tended to resist complete joint disengagement. With the nose down attitude in which the aircraft came to rest, the fuel heater assembly was below the level of the bottom of the fuel tanks. Fuel from the leakage paths would flow into the shroud and overboard via the drain port. The damage was consistent with the effects of overload caused by excessive forcible longitudinal extension of the assembly.

Minor fuel leaks from three fuel tank access panels in the left wing undersurface were found, apparently from the joints between the panels and the wing lower skin. The highest leak rate was 1 drop/second. Such leakage was reportedly not uncommon.

1.12.2.6 Engine Bay Fire Extinguishing System

Examination showed that the fire extinguisher switches were both in their central position and were gated, but that the wire locking on both switch guards was broken. None of the four fuse indicators indicated that extinguishers had fired, the fire extinguisher bottles remained charged and none of the cartridges had been fired. Continuity checks showed that the circuits through the cartridges from the flight deck switches were functional, except that for the 2nd shot for No 2 engine bay, which was open circuit. This was found to be due to the detachment of one of the cables from the cartridge circuit Breeze connector in the wing. The damage was consistent with the effects of cable overstressing as the result of rotation of the sleeve relative to the moulding during fastening and/or unfastening of the connector, reportedly a known problem with this type of connector. There was no evidence to indicate when this damage or the disconnection had occurred.

1.12.2.7 Cabin Safety Equipment

The cabin passenger and baggage doors at the rear of the cabin, the overwing emergency exit hatches, the cargo door and the flight deck DV windows were found to operate without difficulty.

Both rear slides were found stowed. They were deployed without difficulty but climbing down the rope was found to be an awkward process as the upper part of the rope lay against the fuselage side and tended to trap the fingers. It was noted that the colours of the slide anchorage points and clips did not correspond to the description in the Operations Manual (OM). The stowage compartment clips were red and blue respectively, rather than yellow and blue, and the doorframe anchorages (clips and catchplates) were respectively white and yellow, rather than green and red. The stowage compartment anchor rings were found to be uncoloured, apart from by the turquoise paint scheme of the remainder of the compartment.

1.12.2.8 Powerplant

Fire damage was evident within the No 2 engine bay, particularly in Zone 2, but had not caused appreciable component damage.

The hot section of the engine had been severely damaged, with the nozzle box burst open in the plane of the HP turbine over a 150° upper arc and the HP turbine having exited the engine. The intermediate casing had fractured over 360°, detaching the nozzle box from the rest of the engine, and the nozzle box had rotated approximately 45° in the same direction as the engine spool (the engine exhaust duct having rotated in the jet pipe). The nozzle box rotation disrupted the combustors and caused their disengagement from the nozzle box discharge nozzles and/or the compressor diffuser outlet elbows, and from the combustion chamber interconnectors.

Detailed examination of the engine was undertaken and the engine was stripped by the manufacturer under AAIB control. Severe damage had occurred to all NGV and turbine stages, progressively decreasing rearward, and all the turbine clamping bolts had sheared. The clamp bolt failure, the fracture of the intermediate casing and the rotation of the nozzle box were consistent with the effects of a sudden HP turbine seizure against the nozzle box casing and NGVs, while the engine spool was rotating at high speed. No signs were found to indicate turbine or combustor overtemperature, oil starvation, bearing failure, or any other pre-accident anomaly with the engine, with the exception of the HP turbine disc and the fuel burners (see below).

1.12.2.9 HP Turbine

The HP turbine disc (PN RK40726, SN LV450) was recovered from the runway area in two main pieces (Figure 11), together with a few small fragments. The HP turbine blades had all been fractured near to their fir tree attachments. The damage was too severe to allow any estimation of blade platform or shroud gaps, or of seal ring wear and prevented accurate dimensional checking of the disc.

The main disc portions, designated as ‘A’ and ‘B’, measured 11.5x3.5 inch and 15.0x11.5 inch overall and weighed 3.5 lb and 17.6 lb respectively. The main part of the disc fracture was circumferential, following the inner radius between the disc aft seal ring and the diaphragm for approximately 40% of the circumference, with an approximately radial fracture at either end.

The face of the circumferential fracture was generally flat, and banded with a series of light-coloured conchoidal markings (Figure 12). The fracture face on Piece A had suffered appreciable impact damage; that on Piece B was little damaged by impact but the features had been degraded by oxidation to a significant extent.

Detailed laboratory examination was carried out by the engine manufacturer under AAIB direction, using optical microscopy, scanning electron microscopy, transmission electron microscopy of replicas of the fracture surface, sectional microstructure assessment and hardness checking. The fracture was also examined by the Materials Centre of DERA, Farnborough. The evidence indicated that the circumferential crack had been caused by high-cycle fatigue (HCF), originating at a corrosion pit in the inner radius between the diaphragm and the aft seal ring. The pit had a maximum depth of 0.004 inch and was 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 reported that in-service corrosion of this type and extent was quite usual.

There were no signs of any other anomaly at the origins, 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 7 inches, mainly against the direction of disc rotation, and forward over approximately 80% of the section thickness. Overload fracture of the remaining part of the section and generally radial tear-off in tensile rupture at both of its ends had then completed the fracture and released Piece A. Detailed inspection of the disc pieces failed to find any additional cracking.

The microstructure of the disc material was normal and hardness was satisfactory, with measured values in the range 316-319 HB against a specification requirement of 286-321 HB. There were no signs of gross overstress, which would have been likely to result in multiple cracks. Characteristics of the failure indicated that a low-cycle fatigue mechanism had not been involved and that the disc had not failed due to overtemperature.

The fatigue fracture surface exhibited approximately 150 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. Some indistinct striation features were evident in the HCF areas between the bands, with striation densities in the order of 20,000/mm. The available detail was insufficient to allow an accurate striation count over the fracture surface. The best estimate of the number of HCF load reversal cycles involved in the propagation of the crack was in the order of 1-5 million.

1.12.2.10 Combustors

No signs were found of significant combustor anomalies having preceded the engine failure. The first deflector strip in the No 2 combustor flame tube was found slightly distorted; it was concluded that this had probably occurred before the engine failure but would have had no significant effect on the engine. The No 2 combustor had been replaced 23 operating hours before the accident (Para 1.18.2) due to local overtemperature damage. The replaced combustor was located awaiting repair and the damage was found to consist of holing of the flame tube and the destruction of a small part of the first deflector strip; only a small amount of material was missing, insufficient to have lodged on the HP NGVs. Comparison of the replaced and the replacement combustors showed that the damage to each was in a similar location.

No material had detached from any of the flame tubes fitted at the time of the accident. Analysis of foreign debris found lodged in the combustors indicated that it had not originated within the engine and was consistent with cowl material that had entered after the turbine failure.

1.12.2.11 Fuel Burners

The fuel burners remained in place and undamaged, with the exception of separation of the end fitting of the No 1 burner supply pipe at an engine firewall connector. The fracture had occurred at a brazed joint. Detailed examination showed that the braze penetration was below optimum, but concluded that the failure had resulted from overload caused by combustion chamber displacement when the nozzle box had rotated after the turbine had failed.

The 7 fuel burners were removed and examined and then checked on a test rig for flow rate and flow pattern. Major difficulties were encountered in determining whether the burners were of the air-washed or fuel-washed type, even after all identification markings had been compared by the engine manufacturer with the Rolls Royce Dart Parts Catalogue listing. They were initially identified as air-washed burners and tested accordingly on a manufacturer's calibrated test rig. The testing was repeated in accordance with the fuel-washed burner requirements when they were subsequently finally identified as fuel-washed burners. All the burners passed the required leakage check. The test results included:

BURNER NO	FLOW AT 1,000 PSI				SPRAY	
	RATE IGPH	REQUIREMENT IGPH		PERCENTAGE EXCEEDANCE * %	ANGLE ° (Requirement 100-110°)	ASYMMETRY° (Requirement 2° Max)
		Min	Max			
1	25.87	25.00	25.60	1.0	100	6
2	26.49	23.40	24.00	10.4	97	7
3	27.33	23.40	24.00	13.9	91	1
4	26.40	25.00	25.60	3.1	98	2
5	21.70	21.80	22.40	-0.5	99	7
6	25.30	25.00	25.60	0.0	91	5
7	23.41	21.80	22.40	4.5	96	0

*Measured flow rate exceedance below minimum or above maximum requirement, as a percentage of the minimum or maximum requirement respectively.

Thus, two adjacent burners, Nos 2 & 3, were found to have flow rates somewhat in excess of the maximum specified. Three others had a lower exceedance and one had a flow rate slightly below the minimum requirement. The flows produced a 2D Fourier Index (Para 1.18.3.7) of 4.3 for the burner set.

Burner strip examination showed that 2 of the 8 distributor plate holes in No 4 burner were blocked with a carbon-like deposit but this was not considered significant. Corrosion pitting of the distributor plate and/or the swirl chamber was found for all of the burners, in some cases to an appreciable degree. In many cases the pitting was present in the annular contact areas between the distributor plate and both the burner body and the swirl chamber and it appeared possible that this could have compromised the metal-metal seal between these components in some areas. No evidence was available to confirm whether the corrosion had occurred before or after the accident. No other anomalies in any of the burner components or their assembly were found. In particular, no signs were found on Nos 2 or 3 Burners of a loose shroud, broken components or eroded flow passages.

The measured spray angle for all the burners was at or somewhat below the minimum specified and for four of them the asymmetry in the cone angle was somewhat greater than the maximum specified. These values reportedly were well within the usual range for as-received burners returned from service.

1.13 Medical and pathological information

Not applicable

1.14 Fire

Passengers seated on the right side of the cabin behind the wing reported seeing light-coloured sparks emanating from the jet pipe of the No 2 engine during the take-off run. This developed around the point of lift-off into a shower of sparks and an orange glow in the jet pipe, followed by a dull bang and then flame from the top of the nacelle. The fire around the nacelle developed until it was described as a sheet of flame over the inboard side of the nacelle. The fire continued after the aircraft had come to rest.

The Airport Fire Station is situated on the south-west end of the main terminal complex, about 2,700 metres from the accident site. The initial call to the Station was at 23:31 hrs. All appliances were at the fire station at the time of the call; six AFS vehicles attended G-OJEM and it was estimated that they were at the site within two minutes of the initial call.

The fire around the No 2 nacelle was initially tackled by a high output monitor from the rear. Another monitor was used to protect the left side of the aircraft fuselage. Meanwhile the Station Officer and Sub Officer went to the forward door to check that no occupants remained on board. The latter then arranged for one of the AFS vehicles to redeploy to the area where the crew and passengers had assembled to provide lighting and first aid, and confirmed that all 44 occupants had been accounted for. Subsequently, a thorough search of the inside of the fuselage was made by a fire-fighting team wearing breathing apparatus. Two sidelines were run out to cool the No 2 engine and surrounding area and a ladder was used to gain access with cooling foam directly into the engine. The AFS attempted, with the assistance of the flight crew, to stem a substantial fuel leak from the bottom of the engine compartment but was unable to do so and a foam blanket was re-applied to the area as necessary. A total of about 90,000 litres of finished foam was used. Four ambulances had been requested; however, it later transpired that only two occupants had suffered injuries, in both cases minor.

During the AFS operations a portable generator used to supply lighting ceased to operate and could not be restarted. It was also reported that communications difficulties were experienced, with the officer-in-charge reporting that he was unable to communicate effectively with the fire fighting teams.

1.15 Survival aspects

The passengers had been given a normal emergency demonstration and the No 2 cabin attendant (No 2) had also given a personal briefing on the operation of the overwing emergency exits to the passengers seated adjacent to them. The cabin lighting had been dimmed for take-off and all window blinds were raised. The

No 1 cabin attendant (No 1) occupied the forward facing seat by the left rear door and the No 2 occupied the rear facing seat on the flight deck bulkhead by the front right door.

When the nacelle fire became evident shortly after rotation several passengers started to shout. The No 1 reported that some unfastened their seat belts and stood up. She had to unfasten her seat belt momentarily to get the PA handset from its stowage in order to regain control of the cabin and to tell the passengers to sit down. She then told the Commander via the interphone that there was a fire in the right engine, again having to unfasten her seat belt in order to reach the interphone controls.

There was no time to give a full emergency brief or to prepare the escape chutes, but the No 1 did tell the passengers to stay in their seats and to ensure that their seatbelts were "all fastened". Unfortunately in the confusion some passengers heard this as "unfastened" and unfastened their seatbelts as they adopted the brace position.

Following collapse of the nose landing gear the aircraft came to a halt in a nose down attitude, with fire burning around the No 2 nacelle. The cabin area had suffered no damage and none of the occupants had been injured in the impact. All emergency lighting appears to have functioned normally. The passenger seated by the left overwing exit had no problems opening it and evacuation started almost immediately, with the encouragement of the No 2, who had left her seat.

Meanwhile the No 1 checked for fire outside the left rear door and then opened it. By this time passengers had moved towards the door and were standing on the emergency chute stowage hatch. The fire was still burning fiercely and smoke was starting to enter the cabin. Because of this, she decided that the most expeditious course of action was to evacuate the passengers directly from the doorway, which was some 11 feet above the ground.

The First Officer left the flight deck and opened the freight door. It opened easily and, with the nose landing gear collapsed, the doorway sill was only a short distance above the ground. The No 2 then began to direct passengers towards this doorway while the First Officer stood just outside and encouraged the passengers in their escape.

It is estimated that the evacuation was accomplished in less than a minute, with 16 passengers escaping through the two rear exits, a further 16 through the overwing exits and the remaining 8 through the freight doorway. The First Officer and the No 2 also left the aircraft through the freight doorway, as did the Commander after he had assured himself that there was no-one left on the aircraft.

Once the last passenger had left through the rear exit the No 1 moved forward through the cabin and carried out a thorough check of all the seats before returning to the rear door and leaving the aircraft.

With assistance from some of the passengers, the crew assembled the passengers at what they considered to be a safe distance from the aircraft and did a head count by checking the passengers off in groups of ten. The passengers and crew were later transported to a reception area in the terminal building.

1.16 Tests and research

An extensive programme of research and testing was undertaken by the engine manufacturer, in co-operation with the AAIB, in an attempt to assess the causes of the HP turbine failure. This included detailed reassessment of previous similar cases (para 1.18.4) and previous test data, re-analysis of the HP turbine fatigue strength and vibrational characteristics (para 1.18.3.8) and the development of a computerised Finite Element Analysis 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 HP turbine disc, with various configurations of engine components and operating conditions thought to be relevant.

1.17 Organisational and management information

The engine that failed had been developed from one of the earliest gas turbine engine designs and the type had been in service for over forty years. A substantial number of similar engines remain in service in commercial aircraft. A number of turbine failures similar to the one that precipitated G-OJEM's accident had occurred previously. These had been attributed by the engine manufacturer to a combination of turbine entry flow distortion and turbine blade wear and 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 determined as the probable causes of G-OJEM's failure had not been identified prior to the investigation of G-OJEM's accident and had not been subject to remedial action.

The engine manufacturer reported that improvements to their process for investigating and resolving potentially catastrophic service failures had been made since G-OJEM's accident. These included the formal review of investigation findings by an independent Technical Directorate and the refinement of risk assessment methodologies and investigation closure requirements.

1.18 Additional information

1.18.1 Operational

1.18.1.1 Aircraft landing performance

The manufacturer was asked to provide an estimate of the landing roll distance under the following conditions:

Flap setting	-	$7^1/_2°$
IAS at touchdown	-	106 kt
Left propeller	-	10,000 rpm
Right propeller	-	0 rpm
Headwind	-	5 kt
Temperature	-	+8°C
QNH	-	1016 mb.

The estimated distance was 1,000 feet, with an estimated increase of 25 to 30% if the FFPS was not withdrawn. The estimates were made on the assumption of a touch down and landing roll on a normal paved surface. It was not possible to estimate with any accuracy the effect on the landing distance in the circumstances of this accident.

1.18.1.2 Aircraft library

During the course of the investigation a number of inconsistencies and omissions were noted in the information contained in the various manuals constituting the aircraft library. They were brought to the attention of the operator, who undertook to rectify them.

1.18.1.3 Engine failure during take-off

The advice given in the Operations Manual was ambiguous. Part 9.7, Handling Abnormal, stated that, "If an emergency arises after V_1, the take-off should normally be continued." It was noted that Part 9.11, Contaminated Runways, was more specific and stated that, "If failure occurs almost immediately after lift-off, the pilot should use his discretion as to whether to continue or abandon take-off." In the same paragraph it did, however, state that, "If failure occurs after the aircraft is airborne and the speed is close to V_2, the take-off should be continued." The company was unable to explain this ambiguity but have confirmed that the Joint Aviation Authorities (JAA) based manuals, which are being put in place, will ensure that the Commander will be given the discretion to land ahead following a major failure after lift-off, if he/she considers that it is the safer option.

The Handling Abnormal section also noted that a swing away from the dead engine should be anticipated when the FFPS on the live engine was withdrawn during the ground roll. It stated that with a crosswind from the live engine side it was desirable to delay the selection of ground fine pitch after touchdown until the airspeed had fallen.

1.18.1.4 Emergency drills

Although drills for Engine Fire in the Air/on the Ground, and Imminent Overrun were available to the crew, the timescale and nature of the event was such that it is unreasonable to expect them to have carried out any formal drill until the aircraft had come to a halt.

An Evacuation Drill is set out in the Emergency Drill Check List and would have been the most appropriate. It required the following actions:

"AIRCRAFT	STOP IMMEDIATELY
PARK BRAKE	SET
HP COCKS	BOTH FEATHER
LP COCKS	BOTH SHUT
FIRE EXTINGUISHERS	BOTH SHOTS
FUEL BOOSTER PUMPS	OFF
FLAPS	FULLY DOWN
DUMP VALVE	OPEN
CABIN EMERGENCY LIGHTS	ON
ATC	NOTIFY
ANNOUNCE (ON PA)	"EVACUATE! EVACUATE USE ONLY --------EXITS/DOORS: RELEASE YOUR SEAT BELTS AND GET OUT!""

Although some items were actioned, the formal drill as such was not carried out.

All emergency training was carried out on the aircraft and consequently touch drills were the normal practice; however, the Commander could not recall practising the above drill, certainly not in the recent past, and the First officer said that he had never practiced the drill. The company training system has been modified to allow more emphasis to be placed on this type of drill and to stress the importance of items such as the LP cock. This is a control which, unlike the associated HP cock, is seldom if ever operated by flight crews except in an emergency situation.

At the time of the accident this drill did not appear in the Emergency Drill Check List, however, the company subsequently took the necessary action to include it and to emphasise the fact that it is a memory action drill.

1.18.2 Aircraft History

Records indicated that the maintenance of the aircraft was specified to be in accordance with Maintenance Schedule JEA/HS748/5 (Issue 1 applicable at the time of the accident). The aircraft had valid Certificates of Registration, Airworthiness and Release to Service. A Maintenance Statement and Certificate of Maintenance Review had been issued at the last scheduled maintenance check (P13 Period Check), conducted on 9 February 1998, and was valid at the time of the accident.

The records indicated the following summary history of Engine 334:

	ENGINE SN 334				
DATE	TSN * hr	EVENT	TSO hr	CSO cycle	OPERATION
27-10-61	0	Manufactured as Mk 101.	0	0	61-70: Operated by RAF.
6-4-65	1,362	Overhaul and conversion to Mk 102.	1,329		
27-4-71	3,058	Overhaul and conversion to civil Mk 532-2L.	1,696		70-73: Storage.
15-1-76	5,811	Overhaul and conversion to Mk 534-2.	2,753		73-90: Operated in civil aircraft in the Azores.
25-4-79	7,892	Repair.	2,081		
13-10-81	9,750	Overhaul and conversion to Mk 535-2. HP Turbine Disc SN LV450 installed new.	0	0	
3-5-84	12,178	Hot section inspection. Nozzle box stripped, turbine discs overhauled. 127 new HP turbine blades fitted, 4 reused.	2,427	3,884	
29-8-86	14,661	21 HP NGVs replaced.	4,911	7,966	
28-3-87		Overhaul due to loss of performance DRS 209 carried out on HP Turbine Disc.			
27-1-90	17,343	Removal due to Hot Section deterioration. Engine commenced long term storage.	2,682	4,476	90-95: Storage.
16-5-94		Hot section repair. 60 overhauled and 10 new HP NGVs fitted. HP turbine blades not removed from disc. Engine released as Mk 536-2 for earlier of 4,000hr/5,000 cycle.			
18-8-94		Installed as No 2 on G-OJEM (then G-BKAL)			
31-3-95		First flight since 27-1-90.			
21-5-95	17,404	Aircraft and engine exported to New Zealand.	2,743	4,494	95-96: Operated in New Zealand.
5-3-96	18,174	Aircraft and engine returned to UK.	3,513	5,219	
12-4-96		Overhauled combustors fitted. Operation with the accident operator commenced.			1996-Accident: Operated in the accident aircraft by the accident operator.
Sept-97	19,027	In-Field Condition Assessment, excluding internal inspection of Combustors 2 & 5.	4,366	6,560	
9-2-98	19,297	Last Scheduled Maintenance Check (P13 Period Check). No 2 Combustor Flame Tube replaced due to overtemperature damage.	4,636		
31-3-98	19,420	Accident. Aircraft TSN 18,352 hr, CSN 19,122 cycle.	4,659	7,081	
30-3-02		Engine Overhaul due (Aircraft TSN 19,897 hr)			

* Engine Operating Hours:

TSN	- Time since New	CSN	- Cycles since New
TSO	- Time since Overhaul	CSO	- Cycles since Overhaul

Production records for the HP turbine disc (SN LV450) showed no significant abnormalities. This included sample tests on the original cast ingot on 14 February 1980, which gave room temperature values of:

0.2% PROOF STRESS tsi (MPa) (the tensile stress required to cause 0.2% plastic strain)	ULTIMATE TENSILE STRENGTH - tsi (MPa) (the tensile stress required to cause fracture)	BRINELL HARDNESS
57.0 (880)	65.9 (1018)	321

A disc manufacturing concession for a marginally oversized taper bolt hole listed in the records was not considered relevant. The disc was fitted new to Engine SN 334 in 1981 during overhaul of the engine by the manufacturer. Its last overhaul was in 1987, 7,081 cycles before the accident; DRS 209, a standard overhaul procedure for the repair of worn disc sealing lands, was carried out. At the time of the accident the disc had accumulated 15,047 cycles since new.

Overhauled HP turbine blades, with 2,100 operating hours since new, had been fitted to the engine by the manufacturer in 1976; 127 of these (out of 131) had been replaced with new blades during a hot section inspection in 1984. They were last overhauled, or substituted with overhauled replacements, at the time of the 1987 engine overhaul and this was the last occasion on which the blades had been removed from the disc. No record was found to suggest that DRS 297 or DRS 611 (inspection and, if necessary, repair of blade platforms) had been carried out on the blades at this time, or previously. The records were inconsistent with regard to blade life at the time of release from the 1987 overhaul, but investigation indicated that the total time since new at the time of the accident was 7,242 hours for 127 of the blades, 14,759 hours for 3 of the blades and 15,842 hours for 1 blade.

All of the combustors, with burners, were replaced with newly overhauled units 2 years / 1,246 hours prior to the accident. Five of the assemblies had originated from a UK overhauler; the other two, Nos 2 and 3, were ex-stock of a defunct operator. Records indicated that burner flow rates measured on a test rig during the overhaul were all within limits. Five of the combustors had been inspected 293 hours before the accident. The No 2 combustor flame tube was subsequently replaced 23 hours before the accident, after it was found to have sustained local overtemperature damage; such damage was reportedly not unusual. The original burner was refitted at this time; it was not recalibrated and there was no requirement or recommendation to do so.

No evidence was found from the records of any anomaly that appeared relevant to the turbine failure. The last on-wing condition assessment of the No 2 Engine prior to the accident did not include internal inspection of the Nos 2 & 5 combustors, due to access difficulties. With this proviso and the exception of an oil leak from the accessory gearbox, the inspection agency reported that the engine major components were in good condition.

1.18.3 HP Turbine

1.18.3.1 Component Lives

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

COMPONENT	APPROVED LIFE
Engine	6,200 hours TBO (time between overhaul)
HP Turbine Disc	20,000 cycles Declared Life (Group A)
HP Turbine Blades	25,000 hours (Group B)
Combustors (with Burners)	3,000 hours TBO (Operator used 2,200 hours)

1.18.3.2 Disc 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 found. Section 2 (issued February 1981) included:

"(1) Turbine discs associated with combustion equipment defects.

(a) Reject all 3 turbine discs (i.e. H.P., I.P. and L.P.) if evidence of a blockage or a combustion equipment defect has been recorded during service or overhaul.

NOTE- Combustion equipment defects, which cause an uneven or interrupted gas flow, produce vibratory stresses in the turbine discs.

(b) Assess evidence of a blockage or combustion defect as follows:
Reject the discs if testing the fuel burners in 'the as received condition' reveals that the engine has operated for more than 4 hours, with a burner overfuelling or underfuelling by 40 per cent above or below its top or bottom limit respectively."

The engine manufacturer was unable to determine the basis or process by which the above specified criteria (40% over- or underfuelling for more than 4 hours) had been established.

1.18.3.3 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 take-off at SL ISA (Sea Level, International Standard Atmosphere) conditions are given below, except where noted. 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

The gas pressure on the disc was 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.

The predominant steady-state disc loading consisted 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 was estimated to be in excess of 173,000 lbf. Further forces, in the axial direction, resulted from pressure loading on forward and aft faces of the disc and from taper bolt clamping loads. In addition, axial and radial loads could be applied to the tip of the rear seal ring by its contact with the forward seal ring of the IP turbine disc; these would be dependent on 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 would be predominately circumferential, due to aerodynamic loads, but would also have an axial component. The disc could also experience forces from blade bending moments, both in and out of the plane of the disc, and blade torsion. It was thus possible that fluctuation in the blade loading, such as could result from inlet flow distortion, could apply out-of-plane alternating forces to the disc rim.

1.18.3.4 Disc Life

Information from the engine manufacturer indicated that the HP turbine disc design had included considerations of static and fatigue loading strengths. The static strength concerned the magnitude of the steady loading stress that could be withstood without failure. Fatigue strength 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 progress. Low-Cycle Fatigue (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 cycle regime. High-Cycle Fatigue (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 cycle regime.

The capability of a material to withstand combined alternating and steady stresses without fatigue resulting could be determined from a Goodman Diagram for the material, a plot of the envelope of allowable alternating stress against steady stress. A fatigue endurance level could be associated with a given stress condition, where a 100% level represented the maximum alternating stress that could be superimposed on a steady stress without fatigue resulting.

The engine manufacturer had determined a 'Declared Safe Life' (DSL) for the HP turbine 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 scatter experienced in fatigue mechanisms and to provide a safety margin. This was intended to meet applicable airworthiness requirements that the probability of a hazardous failure be Extremely Remote ($<10^{-7}$/flying hour).

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 0.005 inch deep present. A PSL of 20,000 cycles was determined, providing zero margin from the DSL. However, following further specimen testing, the engine manufacturer reported in May 2001 their revised conclusion 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 PN RK40726 HP turbine included consideration of HCF of the blades, but not of the disc. Following G-OJEM's accident, the manufacturer assessed the HCF life of the disc using stress levels estimated from the available information (Para 1.18.3.5). The life 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 relatively 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 HP turbine disc failures (Para 1.18.4), the engine manufacturer conducted a finite element re-analysis of the stress distribution imposed on the HP turbine disc by the steady-state loading at take-off power. Minimum drawing dimensions for the HP turbine 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 to have high maximum stresses and steep local stress gradients. The stresses were higher in the transient than in the heat-soaked condition:

CONDITION	SEAL	RADIUS	TEMPERATURE °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 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 the previous HP turbine disc failures identified the various resonant vibratory modes of the HP turbine 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. These were the most likely of the modes to impose appreciable alternating stresses in the seal ring radius areas, if excited. 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 rim experiencing repeated sinusoidal displacement parallel to the disc rotational axis. 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, nominally at 540 Hz, was only marginally above the 15,000 rpm take-off rating 2nd harmonic (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 resonant frequencies would be above the excitation frequency range (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 Analytical and Test Data – Pre Accident

Prior to G-OJEM's accident, 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 HP turbine 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 configurations 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 14,000-15,500 rpm, depending on the engine configuration, and believed to be due to 1st Family blade flapping. Results included the following:

TURBINE BLADE STANDARD AND STRESS MEASUREMENT POSITION	CONFIGURATION	PEAK-PEAK STRESS tsi	FREQUENCY Hz	SHAFT SPEED rpm	ENGINE ORDER
Pre DRS 611 Disc Rim	Standard Build	0.6	1190	10,200	7
	Burner blanked	2.6	517	15,500	2
	Discharge Nozzle blocked	4.7	467	14,000	2
	Wide Platform Gaps	2.4	1657	14,200	7
	Wide Platform Gaps & Discharge Nozzle blocked	5.5	473	14,200	2
DRS 611 Diaphragm (front face)	Standard Build	3.0	444	13,320	2
	Discharge Nozzle blocked	3.8	479	14,370	2
DRS 611 Diaphragm (rear face)	Standard Build	2.3	444	13,320	2
	Discharge Nozzle blocked	2.9	492	14,760	2

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. None was available to indicate the effects on the diaphragm of fuel burner blockage and/or excessive blade shroud and platform gaps. 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.

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 1F,2D Fourier Index of 1.2. However, direct test data on the effect on diaphragm stresses of either burner asymmetry or wide blade gaps had not been obtained.

1.18.3.8 Analysis – Post Accident

Following G-OJEM's accident, until further testing had been completed (see below), the engine manufacturer concluded that this disc fatigue failure had also resulted from excessive burner flow asymmetry, possibly in combination with excessive blade gaps. 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. Some results of the analysis were made available to the investigation after an extended period.

In addition to these influences (excessive burner flow asymmetry and excessive blade gaps), the investigation identified the possibility that the degree of nip between the HP and IP turbine seal rings could possibly affect the vibratory characteristics of the disc, and hence the alternating stresses experienced by the diaphragm. It appeared that a gap between the rings could significantly reduce the frequency of the HP turbine disc 1F,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 HP seal ring was constrained by contact with the IP ring, to around 90% without this constraint.

It was predicted that with the disc resonating freely the axial movement of the rear seal arm part of the disc would be approximately $\pm 1.3 \times 10^{-3}$ inch. The minimum gap necessary for IP seal ring constraint to be absent from a resonating HP disc would thus be the half-amplitude of 1.3×10^{-3} inch. The maximum HP-IP seal ring gap allowable by build tolerances was 0.5×10^{-3} inch at room temperature. Any gap present could be increased by wear of the seal abutment faces or loss of preload in the turbine assembly clamping bolts. It was predicted that the gap present at room temperature would be reduced by the combined effects of pressure, temperature and centrifugal loading when the engine was operating, by between $0.3\text{-}1.6 \times 10^{-3}$ inch at take-off power in ISA conditions. Thus a minimum room-temperature gap of 1.6-2.9x10^{-3} inch could allow the HP disc to resonate without contact between the HP and IP seal rings resulting.

The investigation identified no further factors that were likely to affect the HP turbine 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-OJEM's HP turbine disc failure (given the apparent absence of NGV blockage) were:

Excessive burner flow asymmetry
Excessive HP turbine blade platform and/or shroud gaps
Degree of HP-IP turbine seal ring nip

1.18.3.9 Test Data – Post 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 an operating test engine similar to Engine 334. 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 ring abutment clearance. 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 made available to the investigation in mid 2000 and further information that provided the first consistent explanation for G-OJEM's failure became available in May 2001.

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 Engine 334	Standard	Standard	Minimal change	Minimal change
Standard	Standard	0.003	2.2	5.8
As for Engine 334	Standard	0.003	1.7	4.3
Standard	Worn	0.003	2.1	6.3

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 $x10^{-3}$ inch HP-IP seal ring gap caused estimated alternating stress levels to increase by a factor of approximately 4-6. This was consistent with the effects of a reduction in the frequency of the 1F,2D/2EO vibration mode to within the normal operating speed range of the engine. The effect of other configurations, without corrosion present, were:

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) %
0.002-0.003 nip	30	108.5
0.002 gap	70	98

With a 5 $x10^{-3}$ inch corrosion pit present and a seal ring gap the fatigue endurance level reportedly exceeded 100%. The stresses with corrosion present but with seal interference were acceptable.

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

The testing also demonstrated high alternating stresses in the HP turbine disc rim, 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 HP turbine blade platform gaps. The second was a 1F,7D/7EO resonance, excitable at an engine speed of 10,000 rpm and sensitive to excessive HP turbine blade shroud gaps.

1.18.4 Other Dart HP Turbine Disc Failures

1.18.4.1 General

A 1991 report by the engine manufacturer indicated that at that time the PN RK49121 type HP turbine disc had suffered 27 cases of failure where part of the disc had detached. All had been attributed to HCF. 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. No further cases were reported at the time of G-OJEM's accident. An additional Dart HP turbine disc failure occurred in June 2001. No cases were known of any Dart IP or LP turbine 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 8 fir trees, or 6% of the circumference. In 6 cases debris had penetrated the engine cowl. Some of the failures had occurred in the climb or cruise, but most were during take-off. 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, where the electrical wires to the feathering pump had been severed by the debris, preventing the propeller from being completely feathered and causing the continued feed of engine oil to an engine bay fire, the aircraft had crashed. The outcome in the other cases was unknown.

Eleven of the rim failures had occurred on the RDa7 and 10 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 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, 9 on the RDa7 and 4 on the RDa10. LCF cracking in the bucket grooves was common in HP turbine discs that had accumulated appreciable operating time, but allowable limits were specified in the Overhaul Manual and this had not led to any disc separation failures.

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, causing overheat and failure of the HP turbine disc. Modification action aimed at preventing recurrence had been taken.

1.18.4.4 Diaphragm Failures

The remaining 4 previous cases and the subsequent case were 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 5 cases, 4 were on RDa7 engines and 1 on the RDa10. The available information from the manufacturer's investigations of the previous cases and preliminary information on the subsequent case, 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 take-off 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 10x4 inch portion of the HP turbine 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 take-off 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 9x2.5 inch portion of the HP turbine 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. YS11, Dart 542-10J, Caribbean, 25 June 1989:

As power was being set for take-off and the engine was accelerating through approximately 10,000 rpm a loud bang was heard, followed by an engine bay fire warning. A 17x3.75 inch portion of the HP turbine disc had detached, 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 0.001 inch deep corrosion pit. Rig testing showed 5 of the burners to have satisfactory flow rates; no information was available on the other 2, 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 for G-OJEM's Engine 334.

4. Fokker F27, Dart 532-7, Columbia, 20 August 1991:

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 HP turbine disc, 8x3 inch and 3.5x3 inch 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 0.003 inch deep corrosion pit in the inner blend radius between the diaphragm and the front seal arm. One HP turbine 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.

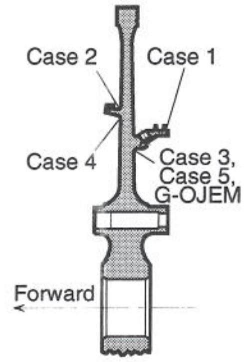
5. Fokker F27, Dart 532-7, Jersey, 5 June 2001:

While climbing through approximately 400 feet agl after take-off a loud bang was heard, followed by an engine bay fire warning. The HP turbine disc departed the engine. The fuselage was not struck and the aircraft returned and landed. A recovered piece of the disc appeared very similar to G-OJEM's Piece B and initial examination suggested that the disc failure modes had probably been similar.

Diaphragm Failure Case Summary:

In all of the above previous cases of HP turbine 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 it had been considered that these represented sites that had been exploited by the mechanism, rather than a cause of the cracking. Fracture locations are shown on the right. Evidence that there had been abnormal turbine inlet flow distortion was found in 3 of the previous cases, in 2 of these due to gross burner fuel flow asymmetry. In the remaining previous case the suspected failure cause was not established; the engine manufacturer was unable to flow check two of the burners.

Fracture Origin Locations



HP Turbine Disc Section

The engine casing and the cowl were holed in 4 of the 5 other known cases and part or all of the HP disc exited. The fuselage sustained damage from debris impact in 3 of the cases and in one of these the main disc parts only narrowly missed the fuselage.

In the 4 previous diaphragm failure cases the operating service accumulated by the discs at the time of failure varied between approximately 7,500-13,000 hours/6,200-15,000 cycles from new and 154-5,000 hours since overhaul. No other cases were known of an HP turbine disc having been 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.

1.18.4.5 Failure Reassessments

The engine manufacturer had attempted to identify any common causal factors for the HP turbine disc failures and in 1991 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.

It was concluded that no changes had been made to the design of the HP turbine 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 '1st Life' engines (ie service between engine construction and its first overhaul or repair), but 5 of the cases had occurred on 1st Life HP turbine 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.

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 timescale for the incorporation of design changes to the disc and anticipated severe operator resistance to any costly 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 in 1994 that the resultant catastrophic risk rate was sufficiently low that corrective action was not necessary.

1.18.5 Burner Service Experience

Following G-OJEM's accident the engine manufacturer issued an Alert Service Bulletin (ASB) (No Da79-A87) in May 1998, applicable to Dart RDa7 and RDa10 Series engines, specifying measures aimed at addressing the suspected causes of the failure. The CAA issued an Airworthiness Directive mandating the incorporation of the ASB. It required the burners to be removed and flow checked and the results passed to the manufacturer. In addition, any burners with a flow rate 10% or more from the requirement were to be returned to the manufacturer for investigation. The allowable period for embodiment of the ASB on a particular engine was dependent on its operating hours since overhaul and on whether DRS 611 had previously been carried out. The schedule called for the ASB to be incorporated on all engines by 31 December 2001.

As of mid 1999 the manufacturer had received and analysed data from flow checks on 265 burner sets. Approximately 90% were air-washed burners and 10% fuel-washed burners. 19 sets (7%) were found to have a flow distortion 2D Fourier Index greater than the 4.3 value found for G-OJEM's failed engine. All of the high index sets were air-washed type burners. The Index was generally less than 6, but was in the range 10-12 in 3 cases and in 1 case exceeded 13. The high indices were generally due to appreciable underflowing of one or two burners in a set, or to appreciable overflowing of a single burner. It was noted that no cases were found with appreciable overflowing of two consecutive burners, as was the case for G-OJEM. Checks of the HP turbine discs associated with the 19 burner sets revealed no evidence of cracking. Subsequent summary information from the manufacturer on the analysis of a further 185 burner sets suggested broadly similar results.

No evidence was found to suggest that the burner flow rates had not been correctly checked before they had been fitted and the cause of the apparent flow rate variation with time was not established. No relationship between the Fourier Index and the burner operating time since overhaul was apparent. While it could be envisaged that a reduction in flow in service could result from carbon deposits or other debris partially blocking burner flow paths, the possible causes of an increase in flow were not apparent. In attempts to determine the cause, the manufacturer tried to obtain the return of burners with abnormal flow rates for examination and commenced rig testing to study the effects on typical burners of pressure and pressure cycling. No reason for a used burner to suffer a flow increase in service was established.

1.18.6 Turbine Blade Gap Service Experience

A further Service Bulletin (NMSB No Da72-530), issued in May 1998, required inspection of HP turbine blades and the return of information on the results to the manufacturer, together with rework of the blades in accordance with the latest standard of DRS 611. The SB was mandated by the CAA. It was applicable to Dart RDa7 and RDa10 Series engines that had not previously had the post-October 1992 standard of DRS 611 carried out. Engines that had been inspected and repaired in accordance with the previous standard of DRS 611 (pre-October 1992, where shroud inspection was not specified) were deemed to be pre-DRS 611 standard for the purposes of the Bulletin.

Initial results were as follows:

	WEAR RATE -10^{-3} Inch/1,000 hour						NO OF SETS
	PLATFORM			SHROUD			
	Mean	Min	Max	Mean	Min	Max	
Pre-DRS 611	0.55	0.05	1.80	1.13	0.11	6.67	18
Post-DRS 611	0.36	0.04	1.36	0.56	0.00	1.83	13
New	0.30	0.02	1.02	0.35	0.03	1.33	6

The results suggested that the average wear rate was greater for post-DRS 611 than for new blades, and considerably greater again for used pre-DRS 611 blades, but that there was a large amount of scatter in the wear rates.

1.18.7 HP-IP Turbine Disc Seal Ring Abutment

There were no specific manufacturer requirements or recommendations to inspect for wear of the seal abutment faces during turbine overhaul. The engine manufacturer began checking for wear in this area in the course of G-OJEM's investigation. As of mid 2001, appreciable seal abutment face wear, up to 0.002 inch, had been found on 5 HP and 2 IP turbine discs.

Other information indicated that some wear of the seal abutment faces was commonly found at overhaul, generally greater on the HP-IP than the IP-LP mating faces. HP-IP seal wear, of the order of 0.002 inch and greater, was found on occasion and this was frequently associated with turbine assemblies that were found to have a substandard fit between the tapered clamping bolts and the reamed holes in the discs. In some cases checks at turbine assembly of the bolt draw were apparently carried out on the stacked discs, rather than on each disc individually, and in this case inadequate fit of a bolt in an individual disc could remain undetected.

1.18.8 Possible Turbine Disc Modifications

In the course of various failure investigation studies the engine manufacturer had assessed possible modifications aimed at reducing the susceptibility of the HP turbine disc to HCF. Measures applicable to existing discs included shot peening to reduce residual 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.

Following the engine rig testing after G-OJEM's accident, the manufacturer issued Dart Service Bulletin Da72-533 (Modification 1946), Revision 1, on 12 April 2001 aimed at providing a positive nip of 2-3x10^{-3} inch between HP-IP turbine seal rings and improved corrosion protection for the HP turbine disc. 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. Additional mandatory limits on incorporation of the SB, dependent on turbine assembly cycles since overhaul, were imposed following the June 2001 failure (Para 1.18.4.4)

1.18.8 HP Turbine Disc Failure Consequences

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. Previous experience of gas turbine engine non-containments 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 HS748, with these assumptions, substantial non-contained HP turbine debris would not strike the wings but could strike the fuselage. The fuselage subtends an arc of approximately 43° from the engine positions and the probability of a fuselage strike by a released 12 inch square portion of disc, for example, was estimated to be in the order of 12%. All primary and secondary flight controls of the HS748 are operated by mechanical systems that cross the turbine plane in runs situated just beneath the cabin floor. There would be a small possibility of all of the flight control runs lying in the path of a substantial portion of non-contained disc.

2. Analysis

2.1 Flight

2.1.1 The Engine Failure

Examination of the aircraft and the accident site made it clear that the accident had been precipitated by the abrupt failure and non-contained release of the HP turbine from the No 2 engine. There had been only a few seconds warning of the impending failure, in the form of sparks seen emanating from the No 2 engine jet pipe by passengers, and this information was not available to the crew. There was no evidence of any indications, flight deck or otherwise, that could have warned the crew of abnormalities with the engine before the non-containment. There was therefore no reason for the crew to abort the take-off.

Evidence from witnesses, the FDR record and the distribution of the wreckage showed that the failure had occurred around 4 seconds after lift-off, when the aircraft was in the order of 60 feet agl. The failure caused an abrupt loss of power from the engine and immediately initiated a substantial fire around the engine nacelle. It also caused the automatic selection of full wet power on the No 1 engine, which assured best climb performance but also resulted in maximum power asymmetry, and the FDR record shows that the aircraft initially yawed 11° right as a result.

The evidence from the wreckage indicated that the violent disruption of the engine associated with the turbine failure initiated a substantial overboard fuel leak from the fuel heater assembly forming part of the LP fuel supply line in the nacelle fire zone.

2.1.2 The Crew Handling of the Emergency

In the classic engine failure/fire at or after V_1 scenario for this type of aircraft, it would be appropriate for the handling pilot to continue the take-off and, when safely established in the climb, the non-handling pilot would carry out the memory items of the Engine Fire/Failure Check List. The aircraft would then be positioned for an approach at the departure airfield or a suitable alternate.

The sequence of events in this accident did not match the classic scenario. The first indication that anything was amiss came shortly after the aircraft became airborne when the flight deck crew heard a noise, which was identified in the course of the investigation as being made by the HP turbine disc leaving the engine. It soon became evident that the right engine had failed. The shouts of the passengers were audible from the flight deck and the crew was made aware of a substantial fire in the area of the right wing around the engine. It was at about the time when the No 1 told him that the right engine was on fire that the Commander took control. Fearing for the structural integrity of the aircraft and aware that a considerable amount of runway remained ahead of the aircraft, he made the decision to re-land and reduced power on the left engine. There was only a short time during which this option was available to him and once made the decision was irrevocable. It was after this that the engine fire warning bell sounded, some 12 seconds after the failure.

Despite the fact that the yaw was increased momentarily by the water methanol system activating on the live engine, the Commander managed to place the aircraft on the runway with about 448 m of paved surface remaining, 27 seconds after lift-off.

Although the Operations Manual did contain appropriate but ambiguous advice, the Commander still has the ultimate discretion. The decision to re-land had to be made rapidly with the information available to him at that instant. This decision was sensible in the circumstances, as was his decision to take control from the competent but far less experienced First Officer. It would not have been possible for the Commander to have communicated his decision to re-land to the First Officer quickly enough to allow him to fly the demanding handling manoeuvre in the time available.

2.1.3 Emergency actions

Although drills for Engine Fire in the Air/on the Ground, and Imminent Overrun were available to the crew, the timescale and nature of the event was such that it is unreasonable to expect them to have carried out any formal drill until the aircraft had come to a halt.

Had there been an opportunity to carry out the Engine Fire in the Air/on the Ground and Imminent Overrun drills, the LP cock would have been selected to OFF, and this should have stopped the flow of fuel to the fire. From this point, the combustible material potentially available to the fire would have consisted of the fuel contained in the fuel system downstream of the cock, together with an unknown quantity of released engine oil. Insufficient evidence was available to assess the likely persistence, progression, intensity and effects of the nacelle fire had the aircraft continued for a circuit and landing, but it appeared possible that significant structural fire damage to the aircraft could have resulted. In at least one previous case of HP turbine disc failure, albeit with particular damage characteristics having resulted and the detailed causes unknown, control had apparently been lost and the aircraft had crashed catastrophically.

Immediately following the engine failure, the No 1 cabin attendant had to release her seat belt on two occasions, while the aircraft was manoeuvring, in order first to reach down the PA/Interphone handset and then to access the controls to select interphone. The action was necessitated by the location of the handset socket and controls in relation to the crew seat, 4.3 feet above the seat pan and 1.3 feet left of its centre. It was clearly undesirable that the attendant should have been unrestrained at this point with the aircraft manoeuvring and about to execute a potentially firm emergency landing. The aircraft manufacturer noted that the layout of cabin equipment could have been changed in service. It issued an All Operator Message on 19 May 1998 describing the problem and the requirement to meet current national standards. However, it is uncertain that this will lead to the rectification of a deficiency that was already readily apparent.

It has therefore been recommended that:-

> The CAA require modification of the PA/Interphone system on HS 748 aircraft at the rear cabin attendant position to render the handset and its controls accessible to an attendant strapped into the aft cabin crew seat. [Safety Recommendation No 2001-8]

2.1.4 The Landing

The aircraft track at touchdown was somewhat to the right of the runway heading, causing the aircraft to begin closing on the centreline before turning left and tracking parallel to the centreline down the left side of the runway to the end. There were no suggestions of directional control problems and the crew's action in not attempting to gain the runway centre while attempting to stop the aircraft as rapidly as possible was reasonable. Use of heavy braking on the runway was evident but the FFPS was not withdrawn to allow the No 1 engine to enter ground fine pitch and increase the retardation. This may have been due to concern that the resultant swing towards the No 1 engine could have added directional control difficulties, particularly with the aircraft relatively close to the left side of the runway.

The aircraft ran off the end of the runway 9 seconds after touchdown and this was an insufficient period for the crew to action the Imminent Overrun Drills.

2.1.5 The Nose Landing Gear Collapse

G-OJEM ran off the runway end at an estimated speed of 62 kt and it was reasonable to have expected it to come to a halt on the flat grass overrun area without appreciable further damage, except possibly due to collision with the approach lights in the area. As it happened, the wheels missed all the lights and the aircraft escaped further damage until encountering the old airport access road. The road surface was very slightly elevated above the surrounding ground surface and tyre and ground markings showed that the ramping effect caused a slight bounce as G-OJEM crossed the road, at relatively low speed. It was possible that the aircraft response to the ramping forces applied in sequence to the nosewheels and main wheels had tended to generate a degree of nose-down aircraft pitch during the bounce. The forces generated when the nosewheels re-landed on the softer ground on the far side of the road were sufficient to cause overload failure of the nose landing gear mounting structure.

The damage associated with the nose landing gear failure, its detachment from the aircraft and the resultant short slide of the forward fuselage on the ground was localised and did not directly hazard the occupants. However, collateral damage to the battery and battery control earth circuits rendered the aircraft electrically dead, once the No 1 engine had been shutdown. This disabled the LP fuel cocks and engine bay fire extinguishers, among other services.

2.1.6 The Evacuation Drill

Having come to rest, 19 seconds after touchdown, the crew acted to evacuate all the occupants as rapidly as possible and to carry out a rudimentary shut down check. The AFS had been alerted but it had not occurred to either pilot to action the Evacuation Drill, possibly because it appears that little or no emphasis was given to it in training and as a consequence neither had been required to practice it recently, even as a touch drill. Additionally, at this point the aircraft had suffered a sudden traumatic engine failure, an overrun in the dark, a nose landing gear failure and consequent unknown airframe damage, and a loss of all electrical services apart from emergency lighting. A substantial fire had burnt around the right engine and the adjacent wing for an appreciable period and was continuing, and the crew was understandably concerned that an explosion could occur at any time. In these circumstances the immediate evacuation of the aircraft was the priority. Had the crew actioned the full Evacuation Drill at this point, selection of the LP cocks and fire extinguishers would have had no effect because the aircraft would have been electrically dead almost immediately after the No 1 engine HP cock had been selected to feather.

In the absence of the drill actions it was understandable that the LP cock switch was not operated. As with many other aircraft types, the LP cock is very rarely operated by flight crews and in training little emphasis is placed on its function and the importance of its role in an emergency drill. Experience of previous accidents has indicated that the importance of this action in combating an engine bay fire is commonly not fully appreciated by crews. This can be particularly relevant in the case of aircraft types not provided with an engine fire handle, which closes HP and LP fuel cocks with a single action. It appears that there can frequently be a misapprehension that the predominately important measures following the initiation of an engine bay fire are to get the HP fuel cock closed and to operate the engine bay fire extinguisher, having feathered the propeller, if applicable. While HP fuel cock closure may be essential for some aircraft types, in many cases the closure of the LP fuel cock will have a similar effect in stopping the engine, albeit with a short delay. The additional very important function of the LP cock would be to isolate the engine bay from the main source of flammable fluid, the fuel in the aircraft tanks.

It has therefore been recommended that:-

> The manufacturer revise and the CAA approve an amendment to the HS748 Flight Manual to emphasise the importance of LP fuel cock closure as a priority action in the event of an engine bay fire. The CAA should consider whether similar measures are necessary for other aircraft types.
> [Safety Recommendation No 2001-9]

It is less easy to understand why the engine bay fire extinguishers were not operated; it is probable that this was a simple omission in the traumatic situation which existed. Fire extinguisher operation may have had some beneficial effect, but no evidence was found to confirm that this would have been the case. Experience of previous accidents has indicated that extinguisher operation may extinguish a nacelle fire in some circumstances, but may well not be effective if flammable fluid continues to be fed to an engine bay fire, particularly where there is an airflow through the engine bay. In some situations the extinguishant can initially knock down the fire but be rapidly purged from the bay by the airflow, allowing fresh supplies of flammable fluid to re-ignite on hot surfaces. However, it was possible that the firing of G-OJEM's No 2 engine bay fire extinguishers could have suppressed the fire and it was undesirable that the system was disabled by the electrical system damage.

2.1.7 The Evacuation

No difficulties were experienced with opening the appropriate exits and the evacuation of the occupants from the aircraft proceeded rapidly. The rear doorway sills were a significant height above the ground but the escape chute(s) could not be deployed because passengers were standing on the stowage hatch. This would seem to be a likely happening in many emergency evacuation situations.

Even had the hatch been accessible, it seemed likely that chute deployment may well have taken an inordinate length of time. It involved manual action, to find and open the stowage hatch, attach the two side panel clips, throw out the chute, brief and despatch two Able Bodied Persons down the rope and wait for them to find the handles and tension the chute. This would have had to be carried out at night, with only emergency lighting, and with the stress promoted by the substantial engine bay fire burning outside the windows. The effectiveness of the procedure would have been in some doubt, as the sill height in this case would have left the chutes at a relatively steep angle. It was likely that the evacuation time was appreciably shortened by the decision to proceed with the evacuation without attempting to deploy the chute(s), albeit with a greater risk of escape injury. Fortunately, no major injury resulted, although the drop to the ground was appreciable.

The inconsistencies between the Operations Manual and G-OJEM's equipment with regard to the colour coding of the chute anchorages is unlikely to have caused difficulties, had the chute(s) been deployed. However, the possibility could not be dismissed and rectification would be simple.

It has been recommended that:-

> The operator ensures that the colour coding of the anchorages for the HS748 emergency escape slides are readily interpretable and are consistent with the descriptions in the aircraft operating and maintenance manuals.
> [Safety Recommendation No 2001-10]

As a means of emergency evacuation for passengers from a public transport aircraft, the type of chute installed is of a very basic and outdated design. There was a likelihood that deployment could be delayed or prevented by passengers standing on the stowage compartment hatch or, if deployed, the hole in the cabin floor formed by the open stowage compartment could itself constitute a hazard. While it can be envisaged that the chutes would be of use in some circumstances, particularly in the case of a premeditated evacuation, it appeared likely that these would be limited and that in many cases attempts to deploy them would significantly slow an evacuation. It is unlikely that such a system would meet passengers' expectations, or that this original design would be thought to be of an adequate standard some 36 years after type certification.

The height of the rear exit sills is relatively modest with the aircraft standing in a normal attitude on all three landing gears but increases considerably with the nose landing gear collapsed. In either case, the drop would not be manageable without injury by passengers of all ages and levels of physical agility and the provision of an effective system is considered to be a reasonable expectation. An automatically inflated slide system had apparently previously been installed.

It has therefore been recommended that:-

> The CAA require the HS748 to be provided with an easily deployable and effective means of descent from both fuselage rear doors in an emergency evacuation situation with any landing gear configuration.
> [Safety Recommendation No 2001-11]
>
> The CAA review the escape systems fitted to other UK public transport aircraft certificated to an earlier standard and require similar action where doubts exist as to their effectiveness.
> [Safety Recommendation No 2001-12]

2.1.8 The Loss of Electrical Power.

The operation both of the No 2 LP fuel cock switch and the No 2 engine bay fire extinguisher switches would have been ineffectual because of the loss of electrical power. Other relevant services such as the PA system would also have been lost.

The power loss was caused by damage to electrical wiring that was attached, close to the nose landing gear mounting points, onto parts of the structure disrupted by the gear failure. The evidence indicated that the gear failure had resulted from predominately rearward overload. Although the aircraft manufacturer had no record of total electrical power loss in 28 previous nose landing gear collapse cases, no reasons were found to suggest that the type of structural damage that occurred in this case was abnormal for such an event. There was thus a possibility that generally similar battery system wiring damage and consequent total electrical system loss could occur as the result of nose landing gear overload failure. Modification of the wiring routes in the vicinity of the nose landing gear mounting structure would make the associated systems less vulnerable in a nose landing gear collapse.

It has therefore been recommended that:-

> The manufacturer and the CAA reassess the crashworthiness of the aircraft electrical system in the event of nose landing gear overload failure and take measures aimed at ensuring that safety-critical services remain operable in such a case. These services should include LP fuel cocks and engine bay fire extinguishers.
> [Safety Recommendation No 2001-13]

2.1.9 The Fuel Leakage.

The sustained overboard fuel leak from the No 2 nacelle resulted from breaches in the fuel heater assembly, located in the engine bay beneath the engine hot section. The leakage occurred because of partial disconnection of two of the assembly's flexible joints, intended to allow for relative movement between the front and rear of the assembly. The evidence indicated that the damage had resulted from displacements that had exceeded the design limits, bottomed out the flexible joints and forced off the circlips intended to limit the telescopic action of the joint assemblies. It was clear that the violent break-up of the HP turbine had imposed considerable shock loads on the engine, which resulted in the gross damage to the nozzle box, the extensive fracturing of the intermediate casing and the rotational displacement of the nozzle box. While the engine mounting structure was apparently undamaged, it appeared that the loads had caused sufficient movement of the forward part of the nacelle relative to the wing to overtravel the fuel heater assembly joints and initiate the leak. It could have been halted by closure of the No 2 LP fuel cock, but this was not achieved.

When the engine failed, it was likely that there were a number of potential ignition sources in the region of the fuel heater assembly. These could have included combustor flame release, hot surfaces associated with engine running, frictional heating and sparking associated with the HP turbine disc break-up and release, and possibly released engine oil that had been ignited. The evidence indicated that the leaking fuel had ignited almost immediately the leak had initiated, had been the major contributor to the fire around the nacelle and had burned until extinguished by the AFS. The leak continued for some hours, until the system was manually isolated. The leakage found during the investigation was gravity fed and was influenced by the nose down pitch attitude of the aircraft; with the aircraft level but with boost pumps operating it was likely that the leakage rate would have been much greater.

While no significant damage resulted from the fire, it undoubtedly represented an appreciable hazard to the aircraft and probably, quite reasonably, influenced the crew in their actions. Had the fire not developed as a result of the engine failure, the event would have been much less of a threat to the occupants. It would be impractical to protect the fuel heater assembly from a direct hit by part of a released HP turbine disc, but it appears possible that adjustments could improve its ability to withstand the displacements likely to be associated with this type of engine failure. Some 260 HS748 aircraft and a large number of similarly configured Dart engines remain in service throughout the world.

It has therefore been recommended that:-

> The CAA require the aircraft manufacturer to consider measures aimed at ensuring that the LP fuel heater assembly in the HS748 engine bay

can tolerate the displacements likely to result from a major failure of the engine rotating assembly without releasing fuel.
[Safety Recommendation No 2001-14]

2.1.10 Fire Fighting

The evidence suggested that the AFS attendance was expeditious, and was effective in verifying that all the aircraft's occupants had escaped, in limiting the fire damage to the aircraft and in securing the accident site. The difficulties experienced by the officer-in-charge in communicating with the fire fighting teams did not appear to have caused major problems in this case but had the potential to be detrimental to fire fighting performance and the likelihood of occupant survival. Communication difficulties in managing an airport fire response resulted in a previous AAIB recommendation

Safety Recommendation 4.14, Aircraft Accident Report 8/88:

A requirement should be introduced for an effective communication system for Rescue and Fire Fighting personnel as part of the licensing requirements for all major airports. That requirement should include provision for communication on the same system by the officer in charge of the units deployed by any local authority fire service having standing arrangements to attend such airports.

It has therefore been recommended that:-

The CAA review, in the light of advances in communication technology, its response to and reconsider Safety Recommendation 4.14, Aircraft Accident Report 8/88.
[Safety Recommendation No 2001-15]

2.2 HP Turbine Failure

2.2.1 The Turbine Disc Failure

The evidence showed that part of the No 2 engine HP turbine 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 both portions of the disc to severe rotational imbalance loads, and signs were found of jamming interference of the disc pieces with the nozzle box casing. The damage found was fully consistent with the large forces generated having caused the turbine clamping bolts to fracture and the main part of the disc to break free, and both disc portions to penetrate the nozzle box casing and the cowl. The engine investigation therefore concentrated on attempting to establish the reasons for the HP disc failure.

2.2.2 The Possible Disc Failure Consequences

The two major portions of the disc exited the nacelle with considerable energy, as evidenced by the appreciable distance that each travelled, and it was fortunate that the exit point from the nozzle box and their trajectory happened to be such that they both missed the cabin. The debris also missed electrical wiring looms and the LP fuel heater in the engine bay. Wiring damage in a previous case had apparently contributed to catastrophic consequences and a direct hit on the fuel heater by a portion of the disc could well cause fuel leakage at a much higher rate than occurred in G-OJEM's case.

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

2.2.3 The Disc Fracture Mode

It was clear from the detailed examination that a fatigue crack had developed in the inner blend radius between the diaphragm of the HP turbine disc and the rear seal ring. The crack had progressed circumferentially for approximately 7 inch, or around 40% of the circumference, and axially through around 80% of the disc thickness. The evidence indicated that at this point the disc was sufficiently weakened that normal operating loads were sufficient to extend the crack radially from either end and to fracture the remaining 20% of the cracked section, causing a substantial portion of the disc to detach, with consequent severe powerplant disruption. There was no doubt that the circumferential cracking of the HP turbine disc had been responsible for the sudden disruptive failure of the disc.

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, apart from surface corrosion. The corrosion pits at which the fatigue had originated, up to 0.004 inch deep, were part of the fairly extensive light corrosion present on the disc surface. The fact that such corrosion was reportedly quite usual indicated that, while the pitting was likely to have reduced the fatigue strength of the disc, it was not the sole cause of the failure.

No signs of gross overstress, overtemperature, or LCF effects on the disc were present and the evidence strongly 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 had resulted from the failure region having experienced fluctuating loads that, superimposed on the steady loads, had exceeded the capability of the material, and that this had resulted from a vibratory condition of the disc.

Prominent striation features associated with fatigue progression on the crack surface were not a characteristic of the disc material, and the striations that were present were rendered more indistinct by corrosion of the surface. Attempts to establish the number of cycles involved in the fatigue crack progression by estimating the total striation count along the surface provided a broad indication of the likely range, in the order of 1-5 million load reversal cycles, but no accurate count could be determined. 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 positive estimate could be made of the number of load cycles that had been involved in producing the crack.

In summary, it was very probable that the HP turbine disc failure had resulted from extensive HCF cracking caused by 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 The Disc Life and Stress Background

The Dart was an early design of gas turbine and the turbine that failed in Engine 334 was of a type that had been 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 HP turbine 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 HP turbine 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. Only a small margin existed for superimposed alternating stresses without fatigue development in these areas being likely.

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 reasons for apparently similar vibratory conditions having caused HCF cracking of the rim on some occasions and the diaphragm on others had not previously been established.

Engine rig testing had established the various frequencies and modes of natural vibration of the disc and suggested that a 1F,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 the elements of the disc. Cyclic variation of the bending stress in the diaphragm/seal ring radii generated by the centrifugal field would inevitably be associated with such diaphragm bending vibration. The frequency was only marginally above the 540 Hz 2nd harmonic of the engine spool speed at take-off power. The engine manufacturer had considered that the mode could be excited by blade flapping loads imposed on the disc rim if sufficient variation in the gas loads on individual blades were experienced as the disc rotated. The possible causes of circumferential gas load variation were blockage of the output from a combustion chamber or of NGV apertures, or excessive burner flow asymmetry. The limited test results available had supported this, and some of the previous HP turbine disc failure investigations had concluded that these mechanisms had been responsible.

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 ring radius areas resulting from these features had been available.

2.2.5 The Causes of the Disc Failure

All of Engine 334's combustion chambers, and their flame tubes, were essentially intact and no evidence was found of foreign objects having entered the engine or of any parts upstream of the HP turbine having separated before the failure. It was concluded that it was unlikely that partial blockage of the gas flow path to the turbine had occurred.

The investigation into the causes of the HP turbine disc failure of necessity relied heavily on information and expertise supplied by the engine manufacturer. For an extended period, prior to the engine rig testing results obtained in 2000, the manufacturer attributed the failure to the effects of excessive fuel burner asymmetry with a probable contribution from excessive HP turbine blade platform and shroud gaps.

Some asymmetry was found with Engine 334's burner set, although the flow deviation from the requirement and the value of the 2D Fourier Index appeared to be relatively modest. However, the greatest flow deviations were in the sense of burner overfuelling, which the manufacturer considered likely to produce more damaging conditions for the disc than underfuelling by the same amount. Also, the appreciable exceedances were associated with two consecutive burners, and the manufacturer considered such a distribution to be particularly damaging. This suggested that the calculated 2D Fourier Index did not fully reflect the potential of burner flow asymmetry to cause disc excitation and cast doubt on its usefulness as an indicator.

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 HP turbine disc to resonance. However, the results of the measurement, in response to G-OJEM's accident, of burner flows for engines returned from service showed that a significant number had an Index appreciably greater than that for Engine 334, without HP turbine HCF damage evident. The inconsistency was even greater for one of the previous diaphragm failure cases in which the Index was considerably less than that for Engine 334. This indicated that the susceptibility of a particular HP turbine to HCF damage was strongly influenced by factors additional to those causing flow disturbance. The engine manufacturer 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 wide gaps was unknown and, where disc failure had occurred, it had not been possible to assess the gaps. This was the case for Engine 334, because of the level of damage. 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 Engine 334's HP turbine.

In the course of the investigation the amount of nip (or gap) between the HP and IP turbine seal rings was identified as a further possible factor that could affect the HP turbine disc resonant frequency. It appeared that the degree of contact between the seals was likely to affect the characteristics of the disc out-of-plane bending vibration, but no evidence to quantify the effect had been available.

Engine rig testing after G-OJEM's accident to quantify the effects of the suspected factors took an extended period. However, the results were crucial to the investigation as they reportedly clearly showed that neither fuel burner flow asymmetry to the degree found on Engine 334, nor excessive HP turbine blade gaps, had a significant effect on alternating stress levels in the HP turbine disc seal radius areas. The evidence suggested that these factors were more relevant to fatigue in the HP disc rim area. The predominant adverse effect on the diaphragm stress levels resulted from the presence of a significant gap between the HP and IP turbine seal rings.

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 rings, from 30% with an interference of $2\text{-}3 \times 10^{-3}$ inch, to 70% with a 2×10^{-3} inch gap. A 4-6 times increase in the stress levels at the rear seal ring inner radius was associated with a 3×10^{-3} inch room-temperature ring gap. Such a gap apparently allowed full amplitude HP disc resonance in the 1F,2D/2EO mode, without constraint by contact between the HP and IP disc seal rings, and had the effect of reducing the natural frequency of the mode sufficiently to put it in the normal operating speed range of the engine spool.

In fact, the analysis and test results indicated that a minimum room-temperature gap of between $1.6\text{-}2.9 \times 10^{-3}$ inch could, after allowance for the effects of pressure, temperature and centrifugal loading at take-off power, allow full-amplitude HP disc resonance. The 0.5×10^{-3} inch maximum build gap that was allowed by design tolerances was less than this but was sufficient to provide a small gap at take-off power. While this would not be sufficient to allow full-amplitude resonance, a partial-amplitude vibration could cause frettage wear of the HP and IP rings, resulting in enlargement of the gap. If this were to proceed to the point where full-amplitude resonance occurred, then the high stress levels in the disc diaphragm would result. It could also be expected that seal ring wear could result from a sub-standard fit of the turbine assembly clamping bolts, which could remain undetected because of the practice in some cases of conducting assembly checks on the disc stack, rather than on individual discs. However, the bolt fit could not be checked for Engine 334 because of accident damage.

The stress levels determined by the rig testing with a 3.0×10^{-3} inch ring gap were somewhat below those required to cause high-cycle fatigue, as assessed from test and analytical results for the material of the disc. However, further work by the manufacturer finally indicated that corrosion did indeed have a significant adverse effect on high-cycle fatigue and that, with similar depth corrosion pits to those present on Engine 334's HP disc and a seal ring gap present, the fatigue endurance level exceeded 100%.

It therefore appeared from the available evidence that a gap between the HP and IP turbine seal rings, that had probably been enlarged by in-service wear, had reduced the disc resonant frequency to within the normal operating speed range of the engine. The combined effects of resonance and disc corrosion had caused high-cycle fatigue cracking of the disc. The manufacturer also considered this to be the case. However, the manufacturer also believed, based on experience in other areas, that excessive fuel burner flow asymmetry and excessive HP turbine blade platform and shroud gaps had been responsible for two of the previous HP turbine disc diaphragm failures and may have contributed to Engine 334's failure.

2.2.6 Fuel Burner Flow

No evidence was found to suggest that Engine 334's fuel burners had not been correctly flow calibrated and assembled before installation and it was considered most likely that the out-of-limits flows found after the accident had resulted from changes that had occurred in service. This conclusion was supported by the appreciable number of burner sets returned from service that were found, after G-OJEM's accident, to include burners with a flow well above the required range.

It appeared possible that the corrosion pitting of all the burner distributor plates and/or swirl chambers found could have compromised the metal-metal seal between these components. Any flow across the inner seal area would bypass the regulating slots in the swirler and lead to an increase in burner output. No other anomalies were apparent on Engine 334's two significantly overflowing burners.

It was clear that the manufacturer had long considered that the consequences to the HP turbine of significant fuel burner flow deviation were potentially serious or even catastrophic. While no appreciable effect due to the level of burner asymmetry found on Engine 334 was identified by the engine rig testing following G-OJEM's accident, it appeared that the possibility that a higher level of asymmetry could excite HP turbine resonance could not be dismissed. It also appeared that excessive asymmetry may lead to HP disc rim fatigue. It has therefore been recommended that:-

> The engine manufacturer undertake further research with the aim of determining the causes of in-service deviation from the installation setting of Dart engine fuel burner flow and developing means of preventing variation beyond acceptable limits.
> [Safety Recommendation 2001-16]

2.3 Dart Engine Turbine Failure Prevention

2.3.1 General

A number of measures aimed at preventing recurrence of Dart HP turbine disc failure were taken after G-OJEM's accident by means of Service Bulletins issued by the engine manufacturer and made mandatory by the CAA. However, a significant number of disc failures had previously occurred over many years, all with potentially serious or catastrophic consequences, without effective measures to prevent further failures having been taken. Attempts were made during the investigation to assess the reasons for this, in order that measures could be recommended aimed at ensuring more prompt and effective action in similar situations in the future.

2.3.2 Inspection Criteria

The Engine Overhaul Manual had included, at least since early 1981, requirements for discs to be scrapped in the event of gross deviation in burner fuel flow. By this time, many years before G-OJEM's accident, a considerable number of cases of failure of the Type RK40726 HP turbine disc had become apparent to the engine manufacturer and the CAA, and it was clear that fuel burner flow asymmetry had been identified as a possible cause. However, the engine manufacturer was unable to say how their specific requirement for the rejection of all three discs if the engine had operated for more than 4 hours with a burner flowing 40% above or below its top or bottom limit respectively had been established. The rationale behind this requirement could therefore not be determined.

The requirement made it clear that the engine manufacturer had concluded that appreciable over or underfuelling could induce damaging vibratory stresses in the turbine discs. However, no requirement or recommendation was made to engine maintenance agencies, including the manufacturer's own overhaul department, to measure the flow rates of as-received burners from engines on inspection, repair or overhaul shop visits until after G-OJEM's accident.

Such measurement was apparently seldom made in practice, and so the flow rate of a burner would not normally be checked until after it had been stripped and overhauled and all information on the burner conditions to which the turbine had been subjected would be lost. While the progression of HCF in the HP turbine disc tended to be relatively rapid, as evidenced by the lack of cases of HCF cracking found during inspection, the initiation stage was very much slower and knowledge of burner flow deviation would have been highly relevant in the circumstances.

Additionally, the lack of as-received burner checks meant that information on any trend in burner flow variation in service was lost. Such information could have alerted the manufacturer before G-OJEM's accident to the appreciable incidence of significant variation that has been revealed by the measures taken as a result of the accident.

In the circumstances the absence of a burner flow check, a minor operation involving very little time or expense, was quite inconsistent.

It has therefore been recommended that:-

> The engine manufacturer include a requirement in the Engine Overhaul Manual for an as-received burner flow check for the Dart engine at all maintenance shop visits. The Manual should also include requirements for any turbine disc action necessitated by the flow check results that fully reflect the latest knowledge on the possible effects of burner flow deviation on the discs.
> [Safety Recommendation No 2001-17]

Difficulties were experienced during G-OJEM's investigation in establishing the repair standard of the HP turbine blade platforms and shrouds of a number of engines of possible relevance to the investigation. This occurred because the implementation of a repair scheme was recorded only in the Workpack generated by a maintenance visit and commonly held by the maintainer, and not in the associated certificate entered into the Engine Log Book. It would be beneficial, in some cases essential, for such information to be readily available to the engine operator and any subsequent maintainer.

It has therefore been recommended that:-

> The CAA review and revise as necessary the requirements for the recording of maintenance actions with the aim of ensuring that information essential to the further operation and maintenance of the aircraft is readily available.
> [Safety Recommendation No 2001-18]

A further difficulty in establishing engine repair standards was caused by the manufacture's substantial change to DRS 611 in 1981, whereby inspection and repair of blade shrouds as well as platforms was required, without any change in the repair scheme identification. Both standards of the scheme were in use concurrently over a period of approximately a year and it was not possible to positively establish which standard had been used for certain engine repairs.

It has therefore been recommended that:-

> The CAA require the engine manufacturer to designate repair schemes in such a way that the standard that has been used on any particular occasion is readily apparent.
> [Safety Recommendation No 2001-19]

2.3.3 The Regulatory System

The evidence indicated that the CAA had been aware of the previous disc failures, and involved with the manufacturer in assessing the continued airworthiness of the engine and judging it to be satisfactory. A program to improve combustor flame tube integrity had probably helped to reduce cases of NGV blockage. The only other action taken before G-OJEM's accident that the manufacturer believed would assist in preventing recurrence was the revision of DRS 611, but the incorporation timescale was such that most engines of the type currently remain without this repair having been accomplished. Thus, little effective action had been taken and the requirements with regard to possible HCF damage to the HP turbine disc due to fuel burner flow deviation had remained inconsistent and incomplete, as described in Para 2.3.2. The evidence to support the previously attributed causes of HP turbine disc failure appeared to have been insufficient in a number of areas for a high level of confidence. The predominant factor, in the case of G-OJEM's accident, of the gap between the HP and IP turbine seal rings had apparently not previously been identified or investigated.

The manufacturer and the CAA had apparently been satisfied with the continuing airworthiness of the engine type on the basis of failure rate statistics. HP turbine disc failures, including diaphragm failures, had continued. G-OJEM's failure had serious, and potentially catastrophic effects. A further case has occurred following G-OJEM's accident.

It has therefore been 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.
> [Safety Recommendation No 2001-20]

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

The likely effectiveness of the improvements to the engine manufacturer's investigation review and risk assessment process made since G-OJEM's accident could not be fully assessed. It has therefore been recommended that:-

> The CAA and the engine manufacturer consider the need for further improvement to their systems intended to ensure effective action to prevent recurrence following potentially catastrophic in-service failures of UK type-certificated equipment used on public transport aircraft.
> [Safety Recommendation No 2001-21]

2.4 Flight Data Recorder Issues

The aircraft FDR system had a number of faults that had not been addressed adequately either at recorder installation or during routine maintenance. These faults rendered a number of important parameters unintelligible or unreliable after the accident.

The first of these faults resulted from a mismatch between the recorder fitted and the particular aircraft installation. The Plessey PV1584 recorder type had a number of variants that created the potential for the confusion that resulted in the Operator installing an incompatible recorder unit. This resulted in a number of important parameters being rendered unintelligible. As an oversight of this nature stems from an absence of appropriate procedures and guidance material.

It has therefore been recommended that:-

> The CAA review the adequacy of their current procedures to ensure that aircraft being entered into the UK Register have DFDR installations that satisfy the requirements of applicable performance specifications.
> [Safety Recommendation No 2001-22]

> The CAA provide adequate guidance material on the subject DFDR installations to all personnel responsible for the surveying and regulation of such installations.
> [Safety Recommendation No 2001-23]

> The CAA, in conjunction with operators, review the DFDR installation of aircraft on the UK Register which are fitted with the type PV1584 DFDR to determine compliance with the applicable minimum performance standards.
> [Safety Recommendation No 2001-24]

As a result of incidents of incompatible recorder/system being found, the airframe manufacturer has issued an All Operators Message to ensure that the correct DFDR variant is fitted to existing aircraft.

The second issue involved the loss of any useful altitude information as a result of a badly worn altitude transducer. Successive annual calibrations of the system failed to reveal that the transducer output was not being recorded within the required accuracy over its normal range.

It has therefore been recommended that:-

> The CAA should require that, during a mandatory DFDR installation calibration, a range of expected data values should be stipulated for every point of a transducer's travel that is tested. These range values should be stated in either the calibration test procedure or the appropriate test results sheet.
> [Safety Recommendation No 2001-25]

The third issue is the failure of successive annual checks of the recording system to reveal that the fact that the sampling rate of certain parameters was to the FAA compliant standard and not at the required CAA rate. This came about because of the industry practice of only selecting the first sample occurring in each one second sub-frame of data. Thus the replay organisation was unable to detect that the normal acceleration parameter was being sampled only 4 times per second rather than the 8 times per second required for CAA compliant systems. Taking into account the design of data acquisition and recording equipment, it is conceivable that there may be instances where, due to a malfunction in the equipment, such failures can remain undetected for a considerable period of time. As the built-in-test facilities within this type of equipment are unlikely to warn of such failures, it is therefore recommended that:

> The CAA should require that, during a mandatory readout of a DFDR, a section of the read out data should be examined to determine that all parameters have been recorded in accordance with the repetition rate specified in the data frame layout and conversion document pertinent to the DFDR installation being assessed.
> [Safety Recommendation No 2001-26]

3. Conclusions

(a) Findings

1. The crew members were properly licensed, medically fit and adequately rested to conduct the flight.

2. The accident was precipitated by the catastrophic failure of the No 2 engine immediately after take-off resulting in a sudden loss of power and an immediate substantial nacelle fire. No prior warning was available to the crew.

3. The No 2 engine failure resulted from the fracture and non-contained release of the HP turbine disc.

4. The shock loading caused by the turbine break-up resulted in significant leakage from the fuel heater installation. This fuel leakage was a major contributor to the fire, which posed a considerable hazard to the aircraft.

5. The Commander's decision to land the aircraft immediately on the runway remaining was sensible in the circumstances.

6. The controls and handset for the No 1 cabin attendant's PA/Interphone system were inaccessible to her while strapped in her seat.

7. The nose-landing gear failed when the aircraft ran into soft ground following a slight bounce induced by crossing an airport road.

8. Structural damage caused by the nose-landing gear failure disabled the LP fuel cocks, engine bay fire extinguisher and other aircraft services.

9. The crew's actions in evacuating the aircraft without fully completing the evacuation drills were understandable in view of the severity of the fire.

10. The operations manual did not emphasise the importance of the LP fuel cock in isolating the fuel supply to the engine bay.

11. The aircraft evacuation was completed rapidly and without major injury in spite of the appreciable height of the rear door sills above the ground.

12. The escape chutes were not used and attempts to deploy them would probably have appreciably delayed the evacuation. The chute system was outdated and inadequate.

13. The deployment of the Airport Fire Services was expeditious and effective. Some communication difficulties were experienced but this did not cause major problems in this case.

14. A number of Flight Data Recorder system faults prevented meaningful information on several parameters from being obtained. Routine checks had failed to detect this.

15. The HP turbine disk failure resulted from extensive high-cycle fatigue cracking. It was likely that the cracking had occurred relatively rapidly.

16. The presence of a gap between the HP and IP turbine seal rings probably reduced the frequency of an HP disc vibratory mode sufficiently for excitation to occur within the normal engine speed range. Substandard fit of the turbine clamping bolts, excessive fuel burner flow asymmetry and excessive turbine blade gaps may have contributed.

17. Surface corrosion had reduced the fatigue strength of the HP turbine disc sufficiently for disc resonance to cause fatigue cracking.

18. The turbine was of an old design and had a relatively low alternating stress margin.

19. Inadequate control of engine manufacturer repair scheme designation and recording of repair scheme implementation made it difficult to establish engine repair standards in some cases.

20. A substantial number of previous HP turbine disc failures had occurred, with catastrophic consequences in at least one case. A further case has occurred susequently.

21. The previous HP turbine disc failures 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 suggested other possible factors.

22. Effective action to address the suspected causes of the previous HP turbine disc failures and prevent recurrence had not been taken.

(b) Causal factors

1. Significant reduction in the fatigue strength of the HP turbine disc due to surface corrosion.

2. Inadequate control of the fit between engine turbine assembly seal members, possibly influenced by inadequate turbine clamping bolt fit, causing sufficient reduction in the natural frequency of an HP turbine disc vibratory mode to allow its excitation within the normal operating speed range and consequent excessive stressing of the disc.

3. Fuel leakage from the engine bay fuel system, resulting in a major nacelle fire.

4. Failure to identify the turbine assembly seal member fit and HP turbine disc corrosion as possible contributors to disc fatigue damage after previous similar failures.

4. Safety Recommendations

The following safety recommendations were made during the course of the investigation:

4.1 **Safety Recommendation No 2001-8** The CAA require modification of the PA/Interphone system on HS 748 aircraft at the rear cabin attendant position to render the handset and its controls accessible to an attendant strapped into the aft cabin crew seat.

4.2. **Safety Recommendation No 2001-9** The manufacturer revise and the CAA approve an amendment to the HS748 Flight Manual to emphasise the importance of LP fuel cock closure as a priority action in the event of an engine bay fire. The CAA should consider whether similar measures are necessary for other aircraft types.

4.3. **Safety Recommendation No 2001-10** The operator ensures that the colour coding of the anchorages for the HS748 emergency escape slides are readily interpretable and are consistent with the descriptions in the aircraft operating and maintenance manuals.

4.4. **Safety Recommendation No 2001-11** The CAA require the HS748 to be provided with an easily deployable and effective means of descent from both fuselage rear doors in an emergency evacuation situation with any landing gear configuration.

4.5. **Safety Recommendation No 2001-12** The CAA review the escape systems fitted to other UK public transport aircraft certificated to an earlier standard and require similar action where doubts exist as to their effectiveness.

4.6. **Safety Recommendation No 2001-13** The manufacturer and the CAA reassess the crashworthiness of the aircraft electrical system in the event of nose landing gear overload failure and take measures aimed at ensuring that safety-critical services remain operable in such a case. These services should include LP fuel cocks and engine bay fire extinguishers.

4.7. **Safety Recommendation No 2001-14** The CAA require the aircraft manufacturer to consider measures aimed at ensuring that the LP fuel heater assembly in the HS748 engine bay can tolerate the displacements likely to result from a major failure of the engine rotating assembly without releasing fuel.

4.8. **Safety Recommendation No 2001-15** The CAA review, in the light of advances in communication technology, its response to and reconsider Safety Recommendation 4.14, Aircraft Accident Report 8/88.

4.9. **Safety Recommendation 2001-16** The engine manufacturer undertake further research with the aim of determining the causes of in-service deviation from the installation setting of Dart engine fuel burner flow and developing means of preventing variation beyond acceptable limits.

4.10. **Safety Recommendation No 2001-17** The engine manufacturer include a requirement in the Engine Overhaul Manual for an as-received burner flow check for the Dart engine at all maintenance shop visits. The Manual should also include requirements for any turbine disc action necessitated by the flow check results that fully reflect the latest knowledge on the possible effects of burner flow deviation on the discs.

4.11. **Safety Recommendation No 2001-18** The CAA review and revise as necessary the requirements for the recording of maintenance actions with the aim of ensuring that information essential to the further operation and maintenance of the aircraft is readily available.

4.12. **Safety Recommendation No 2001-19** The CAA require the engine manufacturer to designate repair schemes in such a way that the standard that has been used on any particular occasion is readily apparent.

4.13. **Safety Recommendation No 2001-20** 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.

4.14. **Safety Recommendation No 2001-21** The CAA and the engine manufacturer consider the need for further improvement to their systems intended to ensure effective action to prevent recurrence following potentially catastrophic in-service failures of UK type-certificated equipment used on public transport aircraft.

4.15. **Safety Recommendation No 2001-22** The CAA review the adequacy of their current procedures to ensure that aircraft being entered into the UK Register have DFDR installations that satisfy the requirements of applicable performance specifications.

4.16. **Safety Recommendation No 2001-23** The CAA provide adequate guidance material on the subject DFDR installations to all personnel responsible for the surveying and regulation of such installations.

4.17. **Safety Recommendation No 2001-24** The CAA, in conjunction with operators, review the DFDR installation of aircraft on the UK Register which are fitted with the type PV1584 DFDR to determine compliance with the applicable minimum performance standards.

4.18. **Safety Recommendation No 2001-25** The CAA should require that, during a mandatory DFDR installation calibration, a range of expected data values should be stipulated for every point of a transducer's travel that is tested. These range values should be stated in either the calibration test procedure or the appropriate test results sheet.

4.19. **Safety Recommendation No 2001-26** The CAA should require that, during a mandatory readout of a DFDR, a section of the read out data should be examined to determine that all parameters have been recorded in accordance with the repetition rate specified in the data frame layout and conversion document pertinent to the DFDR installation being assessed.

D F KING

Principal Inspector of Air Accidents

Air Accident Investigation Branch

Department for Transport, Local Government and the Regions

December 2001

FLIGHT TRACK PLAN

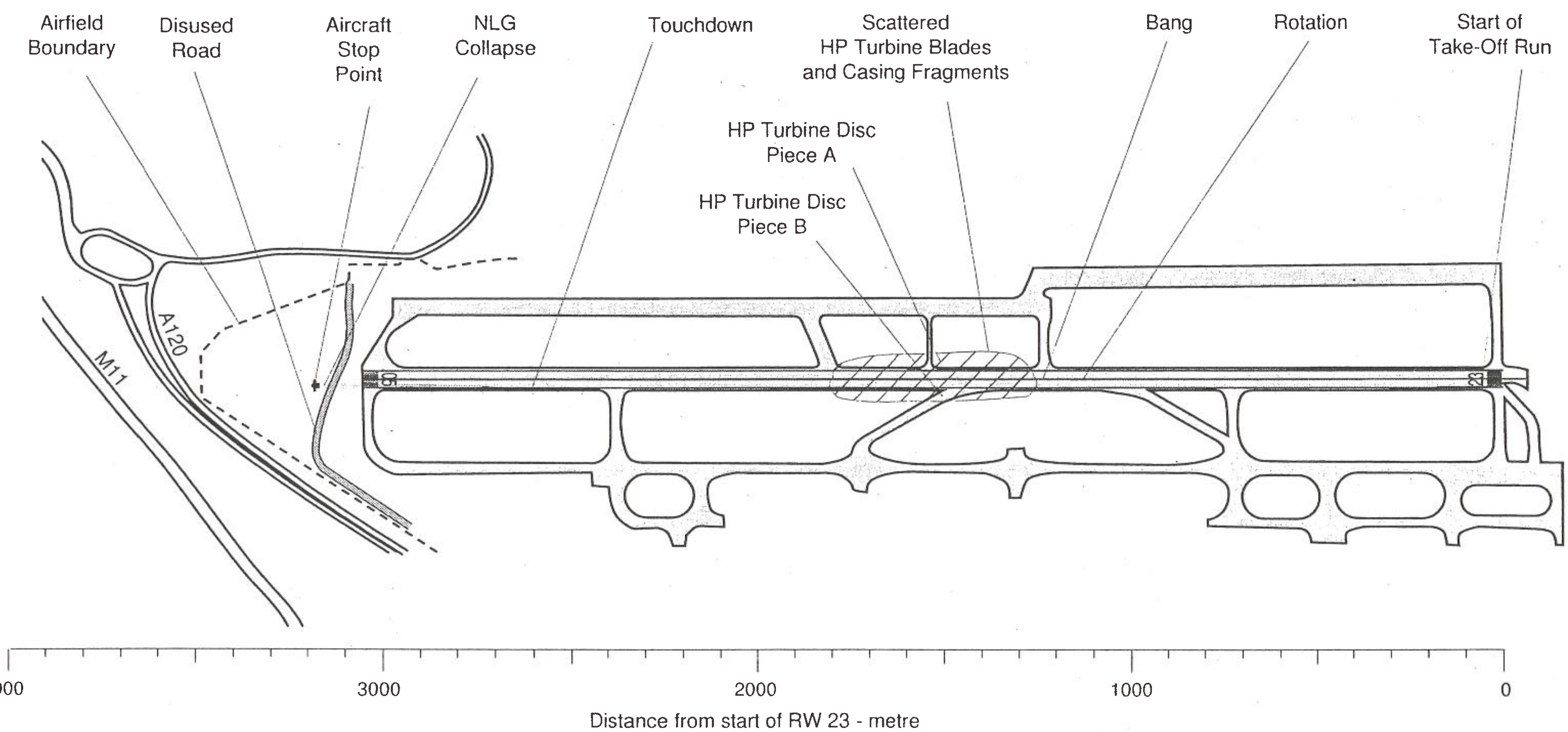


Fig 1

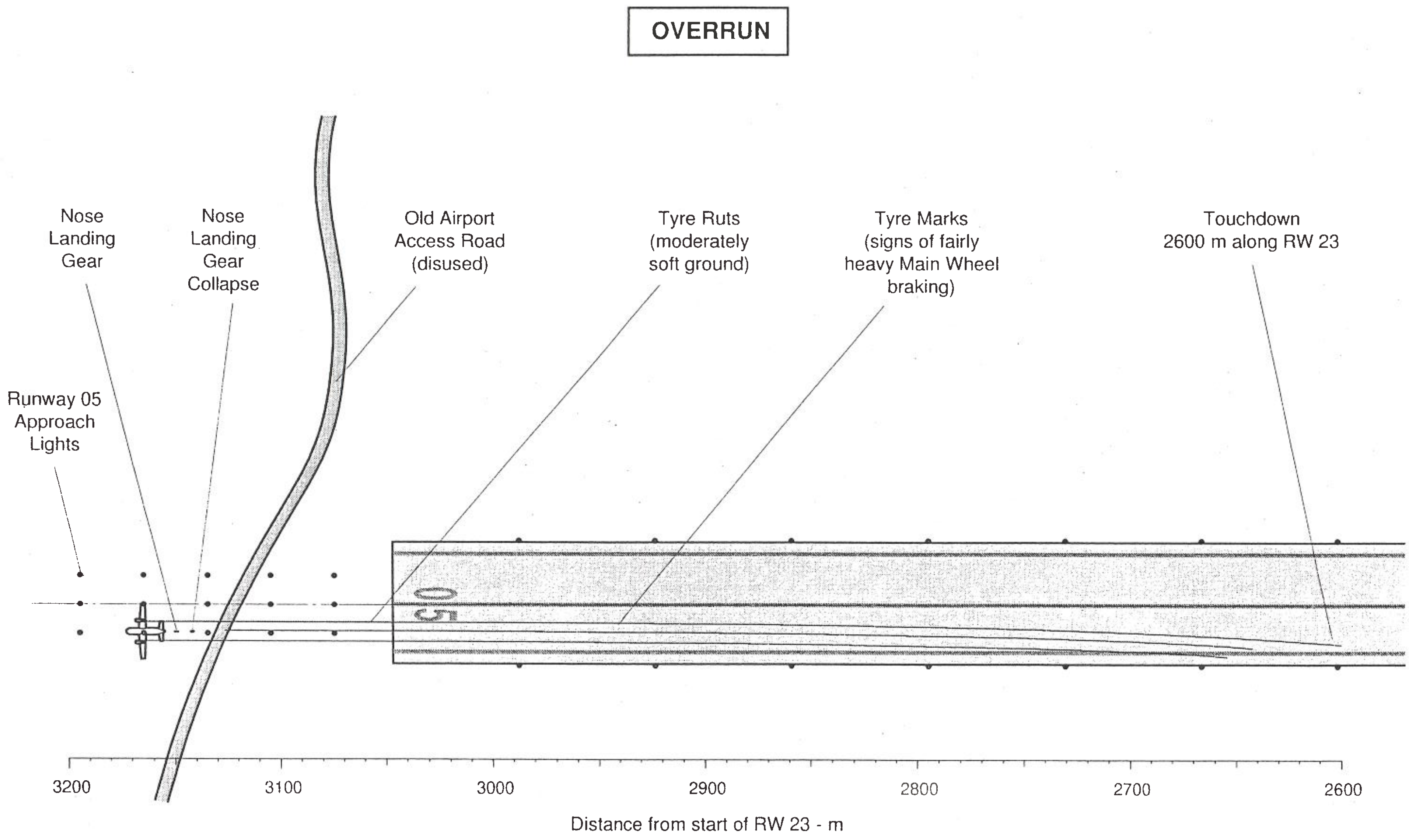

OVERRUN
Nose Landing Gear
Nose Landing Gear Collapse
Old Airport Access Road (disused)
Tyre Ruts (moderately soft ground)
Tyre Marks (signs of fairly heavy Main Wheel braking)
Touchdown 2600 m along RW 23
Runway 05 Approach Lights
05
3200
3100
3000
2900
2800
2700
2600
Distance from start of RW 23 - m

Fig 2

ACCIDENT SITE

NO 2 NACELLE VIEWED FROM CABIN WINDOW

Fig 3

HS748 GENERAL LAYOUT

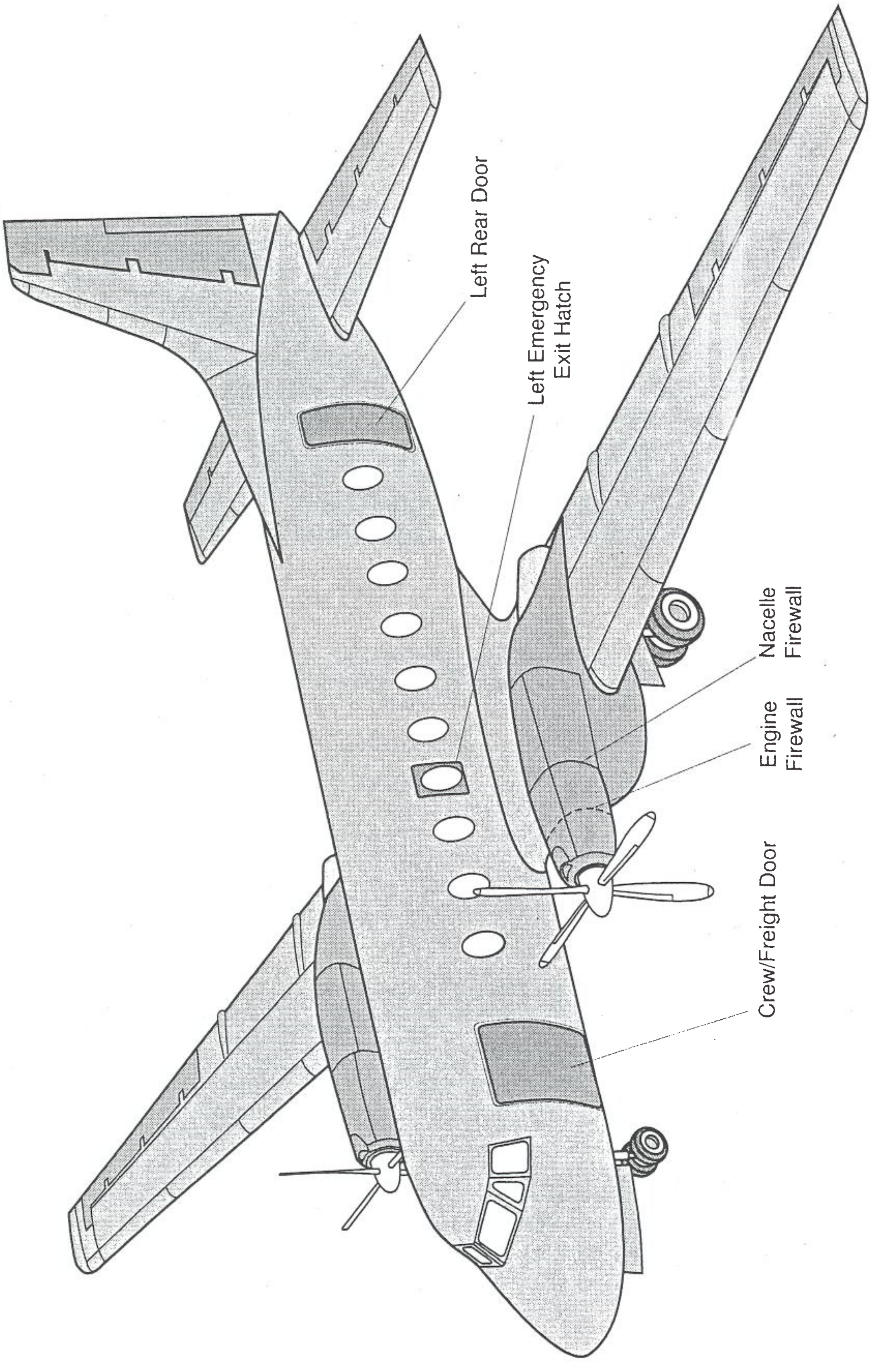


Fig 4

FUEL HEATER

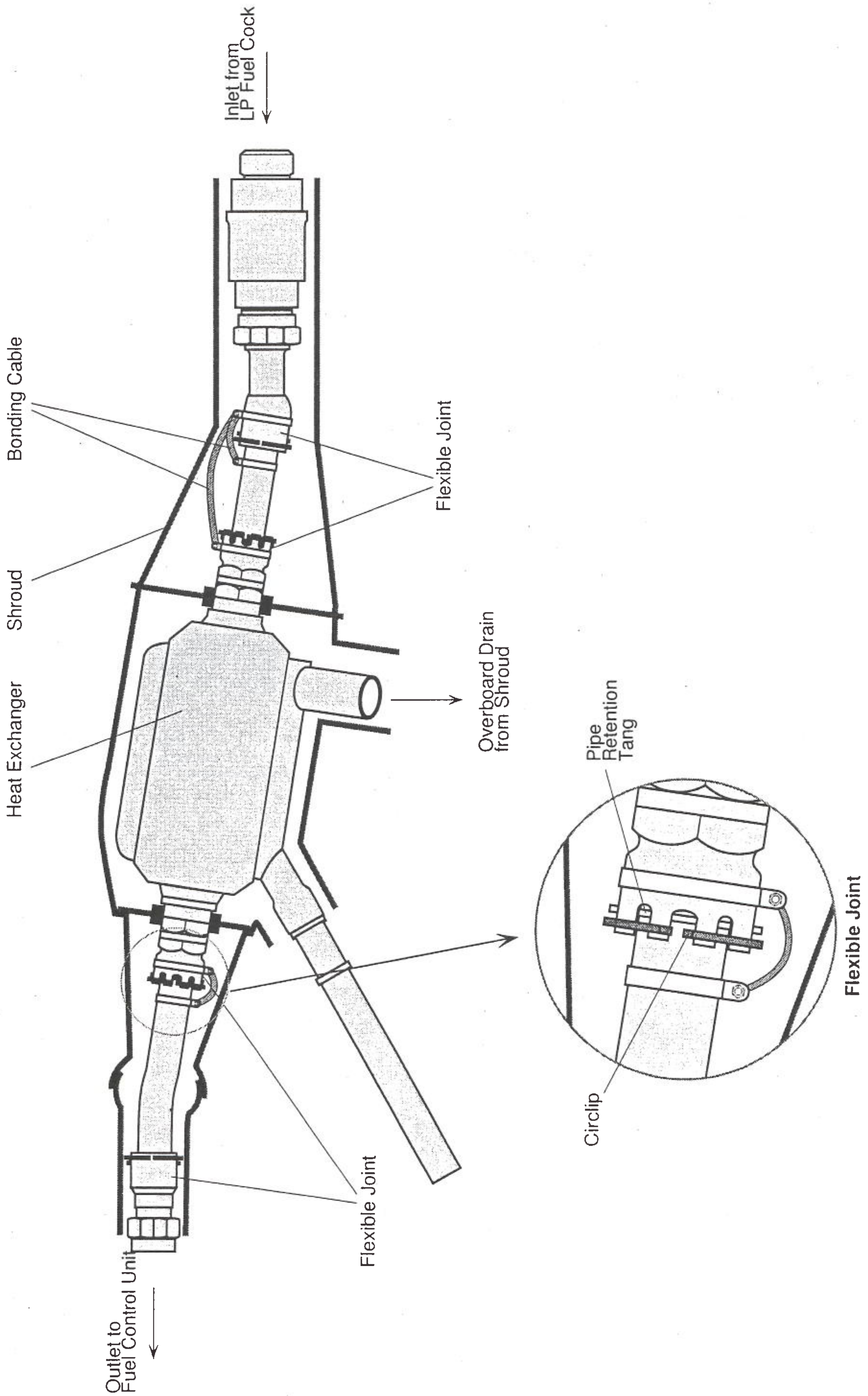


Fig 5

DART ENGINE SCHEMATIC

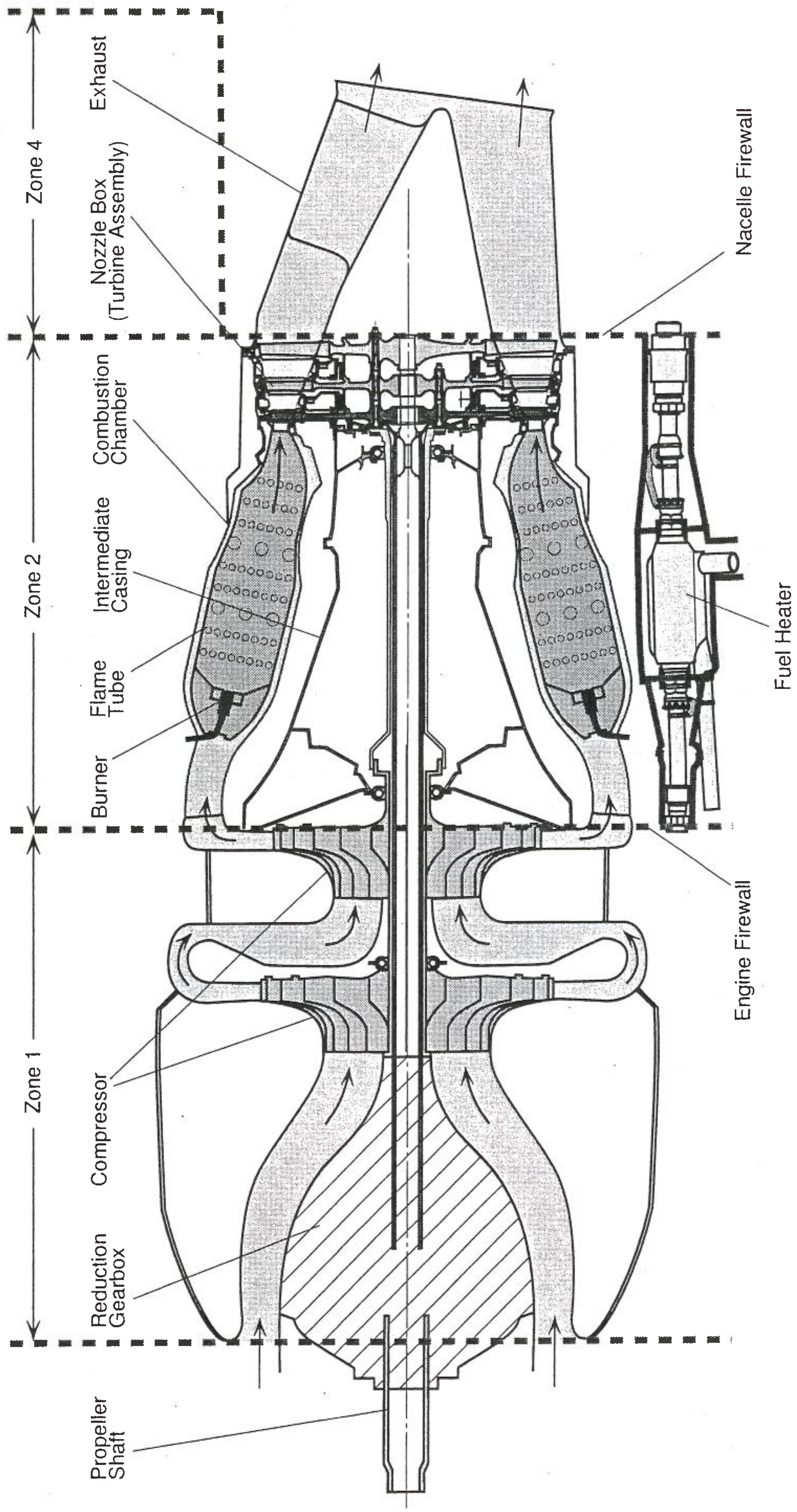


Fig 6

COMBUSTION CHAMBER

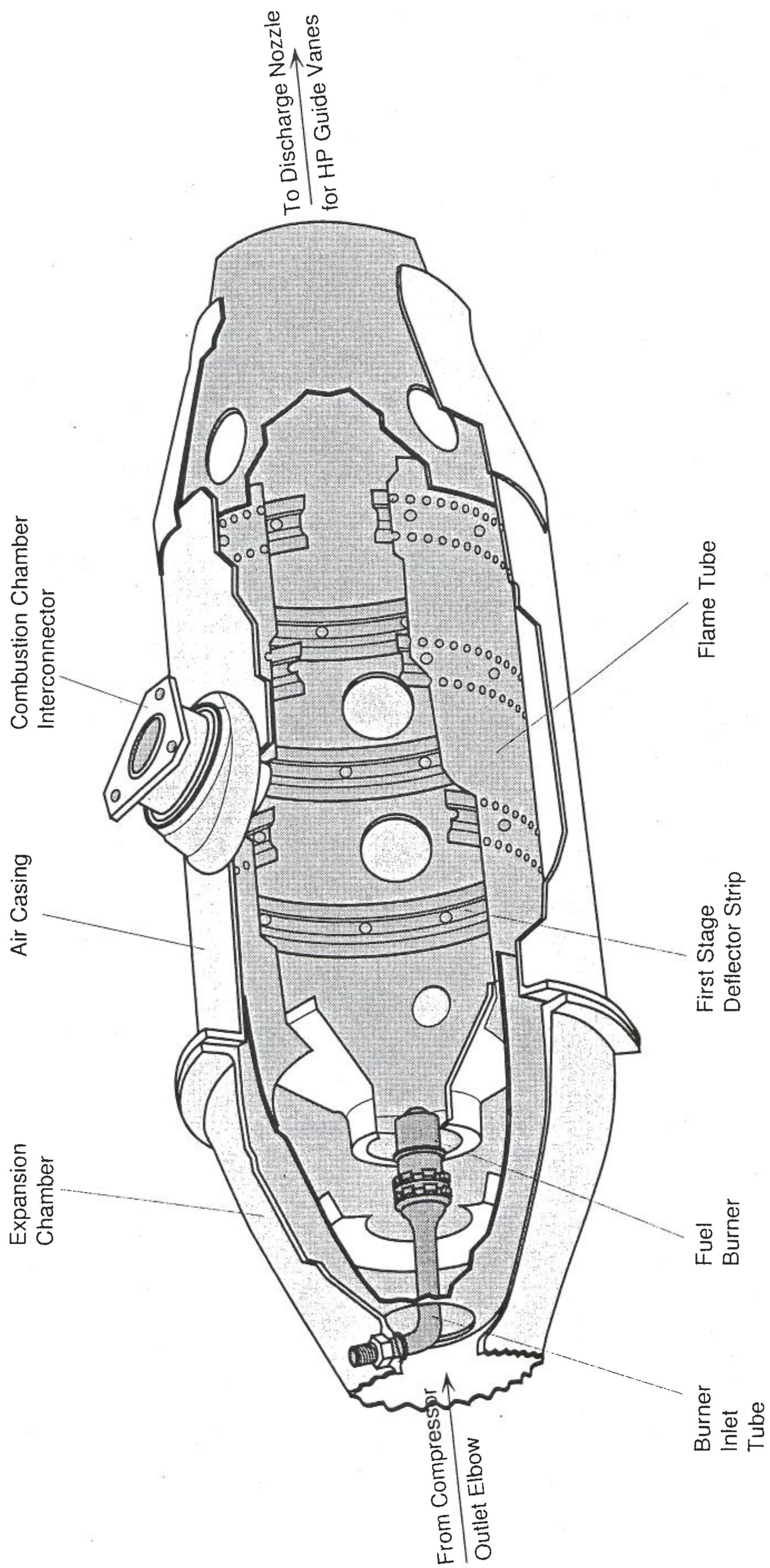
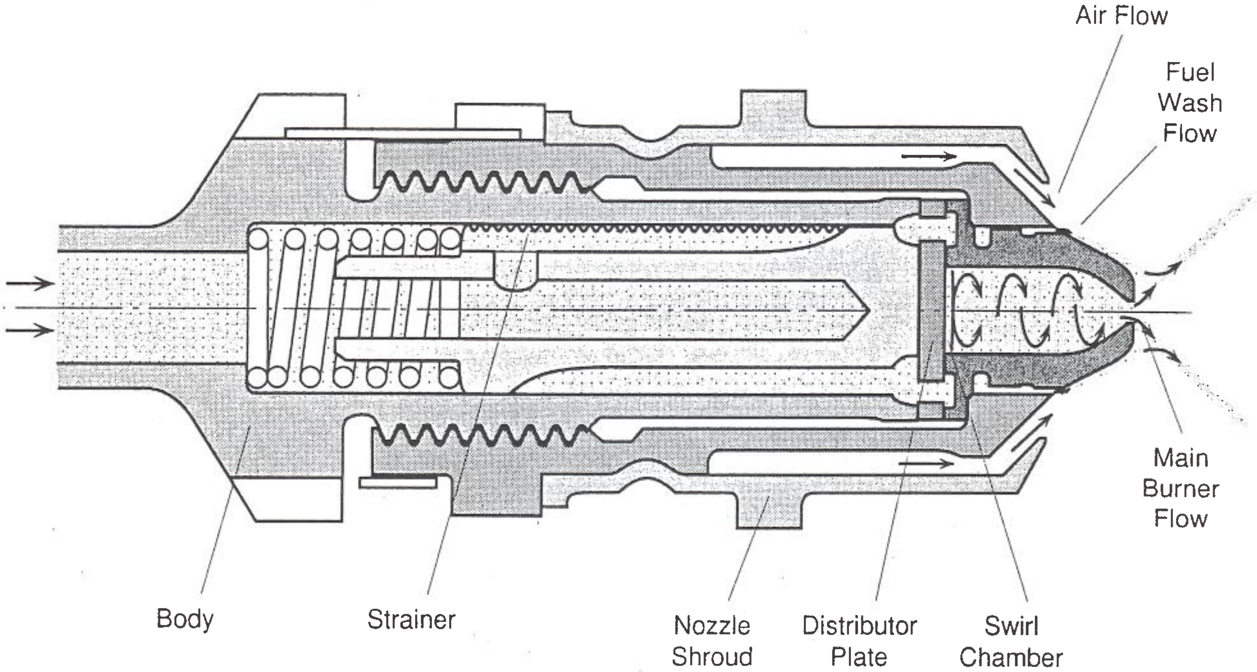
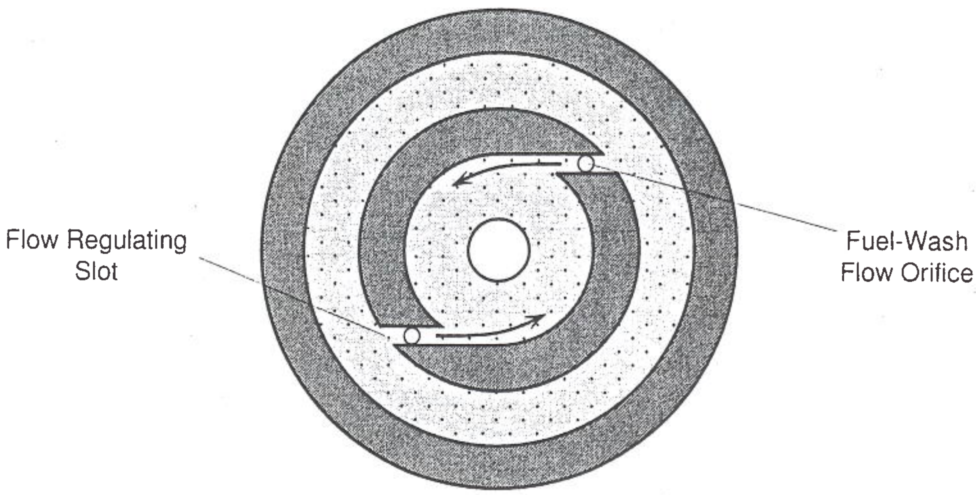


Fig 7

FUEL BURNER SCHEMATIC





SWIRL CHAMBER

Fig 8

TURBINE ASSEMBLY

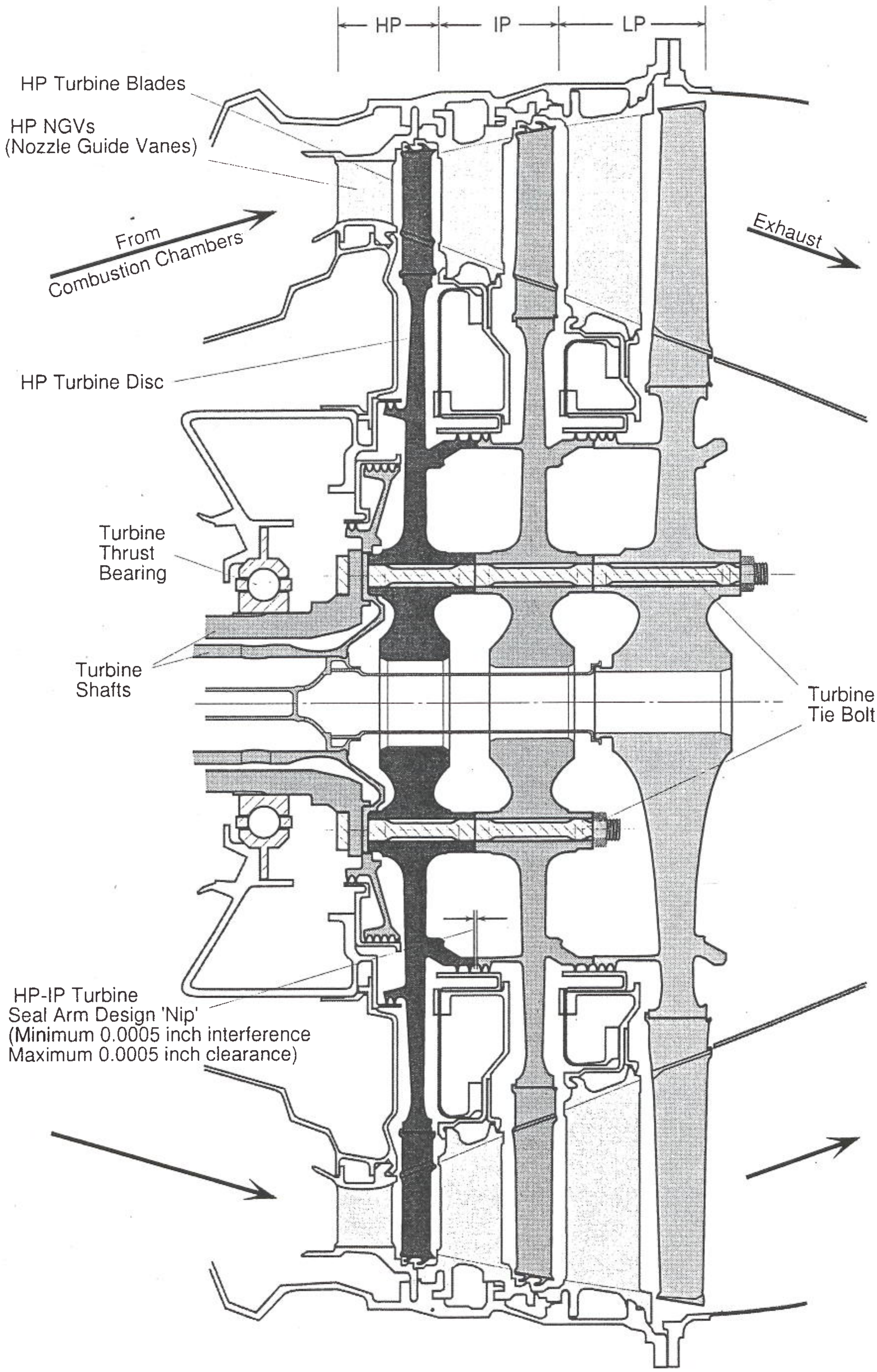
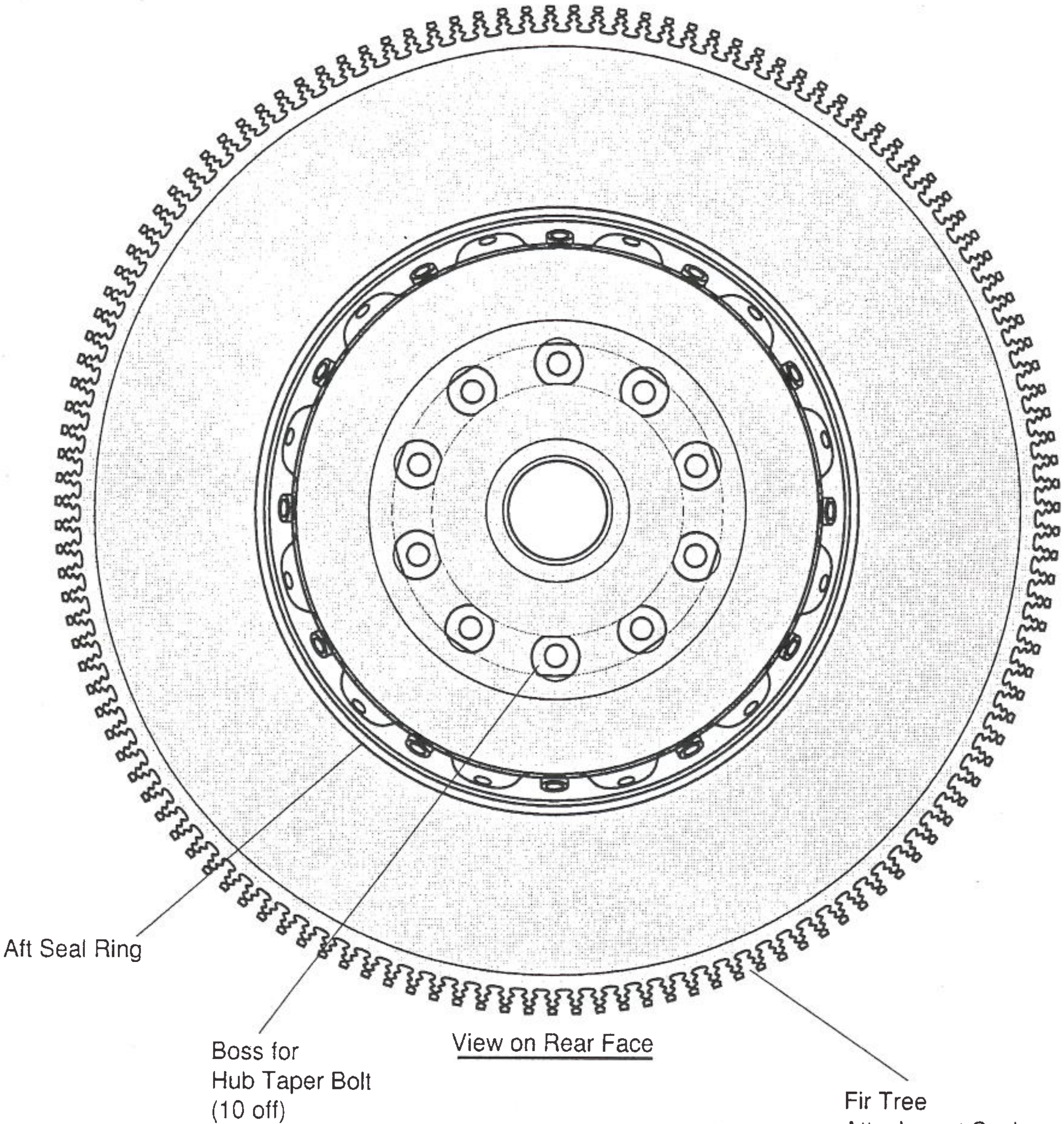
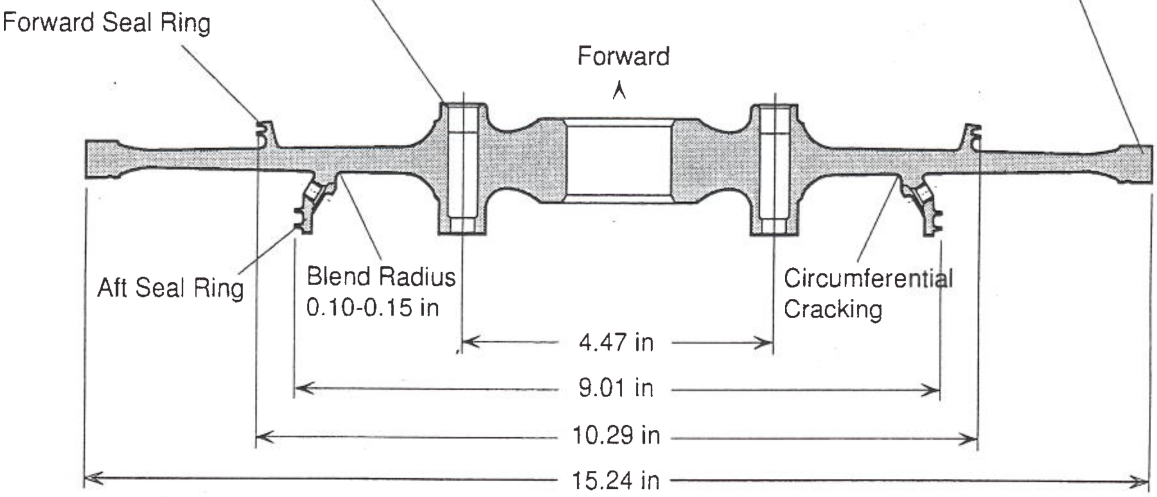


Fig 9

HP TURBINE DISC



View on Rear Face



Sectional View

Fig 10

G-OJEM'S NO 2 ENGINE HP TURBINE DISC PORTION

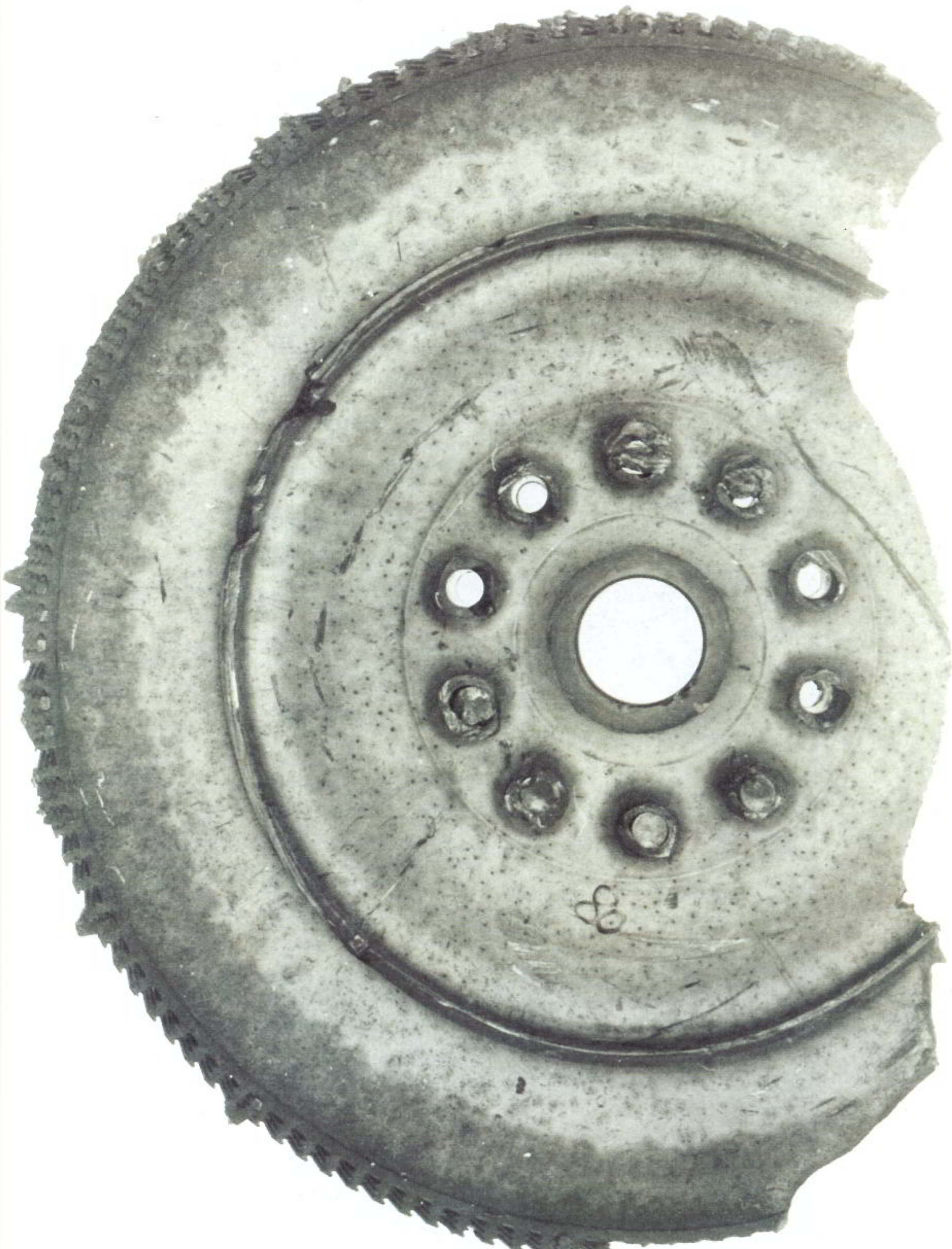


Fig 11

G-OJEM'S NO 2 ENGINE HP TURBINE DISC
FRACTURE SURFACE

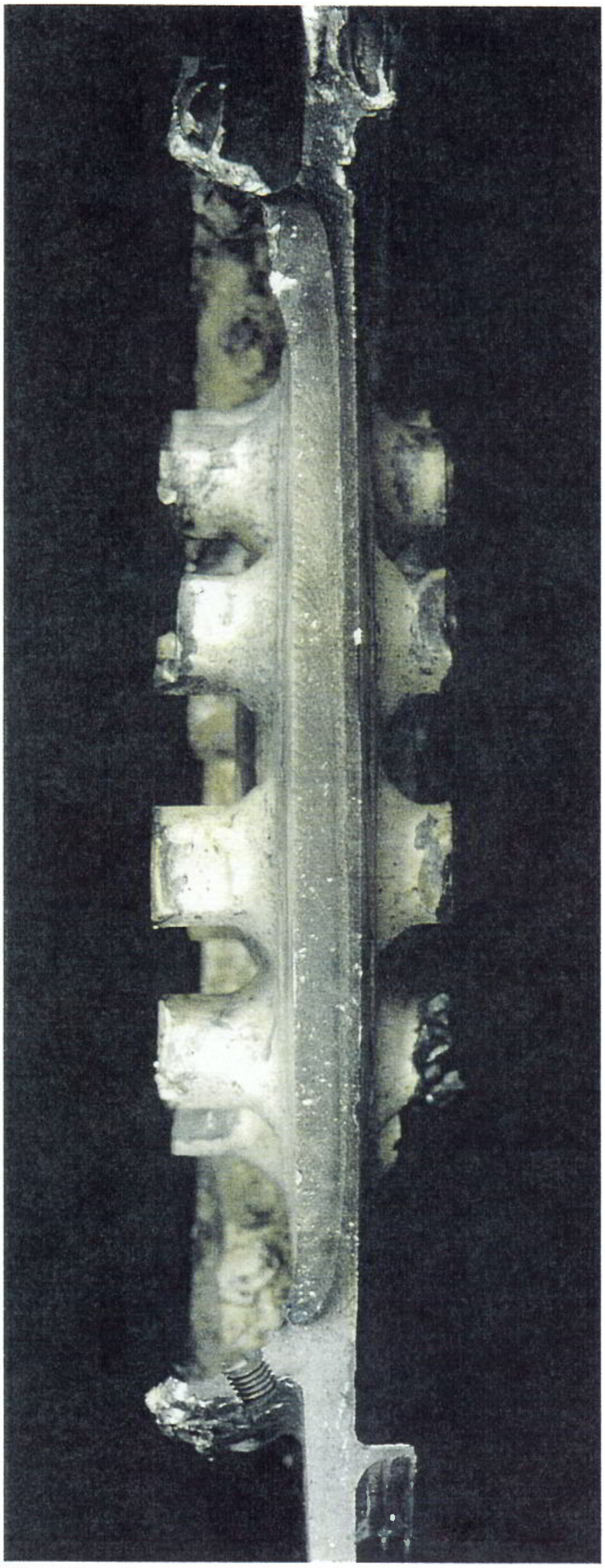


Fig 12