



Australian Government

Australian Transport Safety Bureau

ATSB TRANSPORT SAFETY REPORT

Aviation Occurrence Investigation – AO-2008-008

Final

Engine failure – Jabiru, NT – 11 February 2008

VH-VAZ

Beech Aircraft Corporation 1900



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Figure 1: Courtesy of the operator.

Abstract

On 11 February 2008, at about 0720 Central Standard Time, following takeoff from runway 27 at Jabiru Airport, NT, a Beech Aircraft Corporation 1900D, registered VH-VAZ, sustained an auto-feather of the left propeller and subsequent left engine failure.

The aircraft was being operated on a charter flight to Darwin with two pilots and a passenger on board. The pilots reported that, following the engine failure, they completed a single-engine circuit and landing at Jabiru. Subsequent examination of the left engine revealed catastrophic internal damage to the power section of the engine. The initiator of the damage was the release of a power turbine second-stage blade. Metallurgical examination determined that the failure of the second-stage turbine blade had occurred as a consequence of the initiation and growth of a high-cycle fatigue cracking mechanism from the downstream trailing corner of the blade fir-tree root post. At the time of blade fracture, approximately 25% of the root cross-section had been compromised by fatigue cracking.

The investigation found that during the most recent overhaul of the engine, the overhaul facility did not comply with the engine manufacturer's service bulletin regarding second-stage turbine blade replacement. Consequently, outdated blades were installed.

THE AUSTRALIAN TRANSPORT SAFETY BUREAU

The Australian Transport Safety Bureau (ATSB) is an operationally independent multi-modal bureau within the Australian Government Department of Infrastructure, Transport, Regional Development and Local Government. ATSB investigations are independent of regulatory, operator or other external organisations.

The ATSB is responsible for investigating accidents and other transport safety matters involving civil aviation, marine and rail operations in Australia that fall within Commonwealth jurisdiction, as well as participating in overseas investigations involving Australian registered aircraft and ships. A primary concern is the safety of commercial transport, with particular regard to fare-paying passenger operations.

The ATSB performs its functions in accordance with the provisions of the *Transport Safety Investigation Act 2003* and Regulations and, where applicable, relevant international agreements.

Purpose of safety investigations

The object of a safety investigation is to enhance safety. To reduce safety-related risk, ATSB investigations determine and communicate the safety factors related to the transport safety matter being investigated.

It is not the object of an investigation to determine blame or liability. However, an investigation report must include factual material of sufficient weight to support the analysis and findings. At all times the ATSB endeavours to balance the use of material that could imply adverse comment with the need to properly explain what happened, and why, in a fair and unbiased manner.

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Central to the ATSB's investigation of transport safety matters is the early identification of safety issues in the transport environment. The ATSB prefers to encourage the relevant organisation(s) to proactively initiate safety action rather than release formal recommendations. However, depending on the level of risk associated with a safety issue and the extent of corrective action undertaken by the relevant organisation, a recommendation may be issued either during or at the end of an investigation.

The ATSB has decided that when safety recommendations are issued, they will focus on clearly describing the safety issue of concern, rather than providing instructions or opinions on the method of corrective action. As with equivalent overseas organisations, the ATSB has no power to implement its recommendations. It is a matter for the body to which an ATSB recommendation is directed (for example the relevant regulator in consultation with industry) to assess the costs and benefits of any particular means of addressing a safety issue.

About ATSB investigation reports: How investigation reports are organised and definitions of terms used in ATSB reports, such as safety factor, contributing safety factor and safety issue, are provided on the ATSB web site www.atsb.gov.au

FACTUAL INFORMATION

History of the flight

On 11 February 2008, at about 0720 Central Standard Time¹, following takeoff from runway 27 at Jabiru Airport, NT, a Beech Aircraft Corporation 1900D, registered VH-VAZ, sustained an auto-feather of the left propeller and subsequent left engine failure.

The aircraft was being operated on a charter flight to Darwin with two pilots and a passenger on board. The pilots reported that, following the engine failure, they completed a single-engine circuit and landing at Jabiru.

The passenger reported to the flight crew that debris, which was described as being 'white chunks of metal', was coming out of the exhaust of the left engine. Observers on the ground reported seeing a puff of smoke, followed by flames coming from the left engine. Ground personnel reported that, on inspection, there was visual evidence in the engine exhaust of catastrophic damage to the power, or hot section of the engine (Figure 1).

Figure 1: View looking into the exhaust of the left engine



The flight crew later reported that, at the time of the engine failure, the aircraft's landing gear was retracted and that the engine was in the TAKEOFF POWER configuration. They further reported that the engine failure occurred shortly after

¹ The 24-hour clock is used in this report to describe the local time of day, Central Standard Time (CST), as particular events occurred. Central Standard Time was Coordinated Universal Time (UTC) + 9.5 hours.

selecting the engine bleed air OPEN, and that it was preceded by a loud ‘banging’ noise, followed by a left yaw of the aircraft.

Subsequent analysis of the aircraft’s flight recorder data indicated that the engine failure occurred about 20 seconds after takeoff, at about 600 ft above ground level and at an indicated airspeed of 169 kts. The data indicated normal operation of the engine prior to the occurrence.

The left engine was removed by the operator’s personnel and shipped to an approved engine overhaul facility for disassembly and examination under the supervision of the Australian Transport Safety Bureau (ATSB).

Engine disassembly

The engine disassembly, examination and inspection confirmed catastrophic damage to the power section of the engine including the:

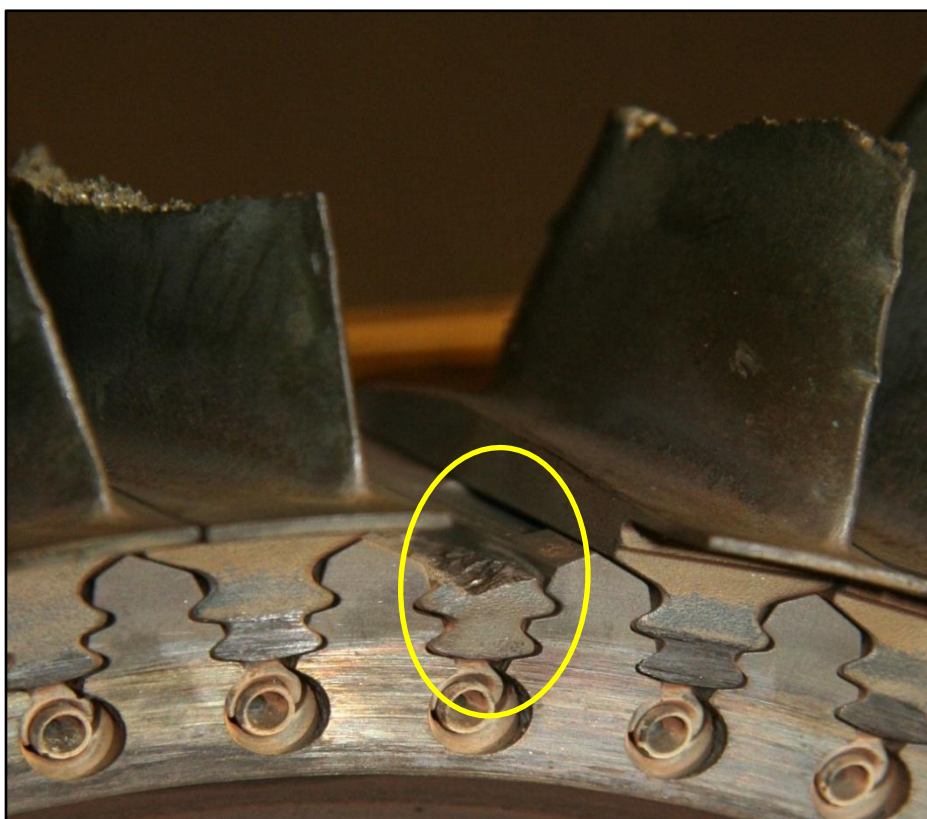
- exhaust duct
- hot section shroud housing
- hot section shroud segments
- power turbine (PT) housing
- PT first and second-stage vane rings
- PT first-stage disk blades
- PT second-stage disk blades (Figure 2).

Figure 2: Power turbine second-stage disk



The compressor turbine disk (CT) sustained minor impact-related damage to the aft side (power turbine side) of the blades. There was minor splatter of molten metal on, and minor damage to, the leading edges of the CT disk blades. One PT second-stage blade had separated from the assembly below the blade platform (Figure 3).

Figure 3: Power turbine second-stage turbine blade close up



The components of the power section were sent to the ATSB's technical facilities for further examination. For further details concerning the metallurgical examination and findings of that examination, refer to Attachment A.

Testing

The engine fuel pump and fuel control unit (FCU) were sent to the engine manufacturer for testing under the supervision of the Transportation Safety Board of Canada. That testing confirmed the normal operation of the fuel pump and FCU.

Engine history

The Pratt and Whitney Canada PT6A-67D engine, serial number 114239, was installed in the aircraft in May 2007. The last overhaul of the power section was completed by an overseas overhaul facility on 24 May 2005. At that time, the power section had accumulated 18,058 hrs total time in service (TTIS), with 26,100 cycles since new (CSN). On 8 February 2007, the time of the last maintenance on the engine, the power section had accumulated 21,466.2 hrs TTIS, 29,054 CSN, 3,408 hrs time since overhaul (TSO) and 2,954 cycles since overhaul (CSO).

During the last overhaul of the power section on 24 May 2005, the part number (PN) 3118563-01 PT second-stage blades were removed. Second-stage blades with

part number PN 3118353-01² were installed. The PN 3118353-01 blades had a 12,000 hour life limit. An entry by the overhaul facility in the engine maintenance documentation incorrectly annotated compliance with Service Bulletin (SB) 14172R1, with their entry noting that there were 8,951 CSN and 6,049 cycles remaining on the PN 3037313 serial number A00087CM disk at that time.

The engine manufacturer authorised the use of Avgas for a maximum of 150 hrs between engine overhaul periods. The operator was asked about any possible use of diesel or Avgas in the engine and responded:

Prior to entering ops with VA [operator abbreviation], no record existed of such usage. From January 1st 2007 to 19th December 2007 the aircraft flew a combined total of 4.5 hrs on Avgas according to our records, following supply failure at one of the remote strips.

Significant engine maintenance items are noted below (Table 1).

Table 1: Significant engine logbook maintenance items³

Date	Hours TTIS ⁴	Maintenance/inspections completed
8 February 2008	21,466 hrs	50/200 hr inspection, foreign object damage (FOD) inspection and compressor wash.
14 December 2007	21,266 hrs	50/200 hr inspection, compressor FOD inspection and wash.
22 October 2007	21,069 hrs	50/200 hr inspection, compressor FOD inspection and wash.
16 July 2007	20,688 hrs	50/200 hr inspection, compressor FOD inspection and wash.
9 May 2007	20,494 hrs	Fuel nozzle replacement and borescope.
12 January 2007	20,211 hrs	50/200 hr inspection, compressor FOD inspection and wash, fuel nozzle replacement and borescope.
23 November 2006	20,019 hrs	Hot end inspection of CT vane assembly and disk, calibration of engine temperature and torque indicating systems.

The investigation could not determine the number of engines that may be in service with the post-SB power turbine second-stage blades installed and compliance with SB 14172R1 incorrectly annotated the affected engines' maintenance documentation.

² The engine manufacturer advised that the pre SB14172 blades, PN 3118353-01, had not been available for procurement since 1 September 1993.

³ All required 50/100 and 200 hourly intervals inspections were completed as required.

⁴ Power section total time in service.

Engine manufacturers advisories

The engine manufacturer's SB14172R1⁵ listed the replacement of PT rotor balancing assemblies. It noted that the purpose of the bulletin was to reduce shroud thickness variation, increase blade strength and improve blade tip clearances by installing PN 3118563-01 PT second-stage blades. The new blades were made from a different material, which had an allowance for final blade tip machining at the disk assembly level and a cast to size shroud. Engine overhaul shops were directed to re-identify the second-stage PT disks to PN 3118447-01 upon replacement. Compliance with the SB was required when the engine was disassembled and access was available to the necessary subassembly.

In September 1994, the engine manufacturer implemented a fleet retrofit program to monetarily credit customers for the turbine blade replacement. The program was in effect until 31 December 1998. The manufacturer implemented another program in 2004 for the same purpose. The engine manufacturer advised that the engine on this aircraft was manufactured with post SB14172R1 PT second-stage blades installed (PN 3118563-01 blades).

Engine manufacturer's comments

The engine manufacturer was contacted and requested to provide information on the application of SB14172R1 as outlined below.

- No mention of 'backwards modification' (reverting back to the older PN 3118353-01 blades, and whether or not it was authorised)

Response- There is no such instruction in the SB 14172 to revert back to pre-configuration since [the manufacturer] introduced a product improvement to increase engine reliability which requires embodiment of the SB upon access. The pre SB14172 are required with recurrent periodic inspection per Maintenance Manual (72-00-00 periodic inspection) to start as the blades reach 1,500 hrs as opposed to starting at 4,000 hrs for the post SB14172. [the manufacturer] has no records showing that the operator did such inspection from 1,500 hrs.

- Clarify the engine manufacturers' position regarding the issues of replacing post-SB14172 blades into the PT disk during overhaul and the retirement times of that particular blade.

Response-The requirement to replace is not just for Overhaul but rather any time access is made available. SB 14003 doesn't make reference to the P/N 3118353-01 (Pre 14172) since the SB is required to be embodied due to its code 5 embodiment requirement.

According to the manufacturer, a code 5 embodiment or category 5⁶ was action required when the engine was disassembled and access was available to the necessary subassemblies (i.e. modules, accessories, components or build groups).

⁵ Original issue date 8 October 1993, revised 12 April 1996 and superseded by SB14259 under revision No.4, 9 March 2005.

⁶ Included categories 1 through 10 with 1 being the highest priority and compliance before the next flight required.

Engine condition trend monitoring

The operator was maintaining the engine using the engine manufacturer approved engine condition trend monitoring (ECTM) system. The ECTM system required pilots to fly the aircraft in cruise flight at stabilised speeds/temperatures and to either manually record the engine parameters, or to have them recorded electronically using a data logger. The aircraft did not have a data logger installed⁷.

The investigation reviewed the ECTM data from the operator for the engine from 1 July 2007 to 29 January 2008. The data indicated that on 24 January 2008, all parameters were reading lower than the original baseline⁸ with a notation 'need more trend data'. The data recorded indicated that from 17 December 2007 to 4 January 2008 the engine 'delta'⁹ fuel flow and inter-turbine temperature (ITT) decreased below baseline values with the delta Ng¹⁰ above the baseline. After 4 January 2008, the delta ITT increased, with the other recorded values remaining stable. A notation made on the records stated 'same comments as last week will re-base after torque gauge cleaned at 31 January 2008'. None of the notations regarding ECTM data was recorded in the aircraft engine logbook.

Engine Condition Trend Monitoring raw data was also sourced from 3 March 2006 to 1 February 2008 and indicated that the data was taken at greatly varying altitudes¹¹, outside air temperatures and airspeeds¹². The ECTM data did not note any obvious anomalies of the engine parameters that would have indicated impending failure.

Regarding large variations in altitude, the engine manufacturer's training manual recommended that:

Keeping the same flight profile minimizes scatter on the trend graph, it is preferred to remain within the same altitude band (5,000' band) from day to day. Flight at many different altitudes (12,000 ft. to 25,000 ft.) will require different engine loads for pressurizing and heating the cabin. These changes in engine loads are not compensated for by the WebECTM program and will cause scatter on the trend graphs.

⁷ Use of a data logger eliminated potential human error when recording information.

⁸ Recorded data was compared to the engine baseline parameters established when the engine was new or recently overhauled. The baseline could be revised when it was clearly established that it was set with incorrect values or flight conditions had changed significantly.

⁹ The difference between the actual engine parameter and the predicted engine mathematical model parameter value.

¹⁰ Gas generator speed.

¹¹ Varying from a minimum of 2,300 ft pressure altitude to a maximum of 25,000 ft pressure altitude.

¹² Indicated airspeeds from a minimum of 160 to 230 kts.

ANALYSIS

The failure of the left engine following takeoff was the result of the catastrophic failure of the engine hot section and specifically the power turbine (PT) section. The flight crew correctly identified the problem engine and took timely and appropriate action to return to the airport and complete an uneventful single-engine landing. There were no indicators to the flight crew or to maintenance personnel of the impending failure of the PT section components.

The engine condition trend monitoring (ECTM) system data that was collected for the aircraft's engines was documented at largely varying pressure altitudes, outside air temperatures and airspeeds, making the data unreliable. Regardless, it did not appear that the impending failure of the PT blade would have been detectable using the ECTM system.

The engine manufacturer advised that the engine was manufactured with post Service Bulletin (SB) 14172R1 power turbine second-stage blades installed (part number (PN) 3118563-01 blades). During the subsequent overhaul of the engine by an overseas overhaul facility, PN 3118353-01 second-stage PT blades were installed, and compliance with SB 14172R1 was incorrectly annotated in the engine's documentation. Advice from the engine manufacturer indicated that the PN 3118353-01 second-stage PT blades should not have been installed in the engine, as they were the subject of an earlier, fleet-wide engine upgrade campaign. The involvement of the overseas overhaul facility contributed to the inability of the investigation to establish why the pre- SB 14172R1 blades were installed during the May 2005 engine overhaul, and the reason for the incorrect annotation in the engine's documentation.

However, the older PN 3118353-01 PT blades, if installed, were subject to a recurrent periodic 1,500 hr inspection. A review of the engine's maintenance documentation did not show any evidence that those recurrent inspections had been carried out. Technicians scheduling engine maintenance subsequent to the May 2005 overhaul may have been misled by the incorrect annotation of the engine's compliance with SB 14172R1. The effect would have been that the technicians would have interpreted that the routine inspection of the blades was not yet required.

FINDINGS

Following takeoff, the aircraft's left engine sustained a catastrophic failure of the power turbine (PT) section resulting in a complete loss of engine power from that engine.

From the evidence available, the following findings are made with respect to the engine failure involving VH-VAZ and should not be read as apportioning blame or liability to any particular organisation or individual.

Contributing safety factors

- During operation, a blade on the second-stage PT disk separated, resulting in substantial damage to the engine hot section.

Other safety factors

- During engine overhaul on 24 May 2005, the part number 3118353-01 second-stage PT blades were installed, contrary to a fleet-wide engine upgrade campaign and Service Bulletin (SB) 14172R1, and maintenance documentation incorrectly annotated as complying with SB 14172R1. As a result, the part number 3118353-01 second-stage PT blades were not inspected at the required periodic interval of 1,500 hrs.
- Engine condition trend monitoring data was being collected or recorded in a manner that was not recommended by the engine manufacturer.

Other key findings

- The flight crew correctly identified the problem engine and took timely and appropriate action to return to the airport and complete a single-engine landing.
- There were no indicators to the flight crew or to maintenance personnel of the impending failure of the PT section components.

APPENDIX A: TECHNICAL ANALYSIS REPORT

ATSB TECHNICAL ANALYSIS REPORT
AO-2008-008

Engineering Failure Analysis of Turboprop Engine Components

Pratt & Whitney Canada, PT6A-67D

Beech Aircraft Corporation 1900D

VH-VAZ, 11 February 2008

Released in accordance with section 25 of the *Transport Safety Investigation Act 2003*

SUMMARY

On 11 February 2008, a Beech 1900D aircraft (registered VH-VAZ), operating a charter flight from Jabiru, NT to Darwin, NT, sustained the in-flight failure of the left engine shortly after departure from Jabiru airport.

The failed engine, a Pratt & Whitney Canada PT6A-67D turboprop, had operated for approximately 21,466 hours since new, and 3,408 hours since the last overhaul. Disassembly and laboratory examination determined that the engine failure was precipitated by the fracture and release of a single blade from the second-stage power turbine rotor; subsequently producing gross mechanical interference within the turbine confines and the forced failure and damage to the adjacent rotating and stationary components.

Metallurgical examination determined that failure of the second-stage turbine blade had occurred as a consequence of the initiation and growth of a high-cycle fatigue cracking mechanism from the downstream trailing corner of the blade fir-tree root post. At the time of blade fracture, approximately 25% of the root cross-section had been compromised by fatigue cracking.

While the fatigue crack origin was clearly identified, the metallurgical examination found no direct evidence of contributory defects or damage at that location. The possibility remains however, that non-metallic inclusions or surface features similar to those that were identified in the origin area, may have been present in some form at the origin itself; thus predisposing the blade to premature fatigue cracking from the location observed. The loss of the corresponding fracture surface with the outer section of the turbine blade hindered any further investigation in that respect.

None of the other blades from the second-stage power turbine rotor showed any external indication of cracking or potentially detrimental physical features.

FACTUAL INFORMATION

Introduction

On 11 February 2008 at about 0720 Central Standard Time (2150 UTC, 10 February), following takeoff from runway 27 at Jabiru Airport, NT, a Beech Aircraft Corporation 1900D, registered VH-VAZ, sustained an in-flight failure of the left engine and the subsequent auto-feathering of the left propeller. The flight crew reported that the failure was characterised by a loud ‘banging’ noise and an associated left yaw of the aircraft. The sole passenger aboard the aircraft described ‘white chunks of metal’ coming from the left engine exhaust, while observers on the ground noted smoke and flames coming from the left engine. Following the engine failure, the flight crew completed a single circuit of the aerodrome and a landing back at Jabiru.

Preliminary inspections of the engine by ground personnel revealed evidence of extensive damage within the engine turbine stages. As a result, the engine was removed from the aircraft and shipped to an approved overhaul facility for disassembly and examination under the supervision of investigators from the Australian Transport Safety Bureau (ATSB).

The engine examination confirmed extensive mechanical damage to the turbine (power) section of the engine, including the:

- exhaust duct
- hot-section shroud housing
- hot-section shroud segments
- power turbine housing
- power turbine first and second stage vane rings
- first-stage power turbine disk blades
- second-stage power turbine disk blades.

Scope of the examination

To assist in the analysis of the damage sustained by the engine and the identification of factors that had contributed to the in-flight failure, the following engine components were received at the ATSB’s Canberra laboratories, after completion of the engine disassembly:

- Power turbine stator housing, with damaged power turbine interstage stator assembly in-situ,
- fragmented power turbine vane ring and air seal,
- compressor turbine disk (P/N: 3040911, S/N: 75A096) and installed blades,
- first-stage power turbine disk (P/N: 3037312 N, S/N: A0008NH7) and installed blades (P/N: 3120112-01),
- second-stage power turbine disk (P/N: 3037313L, S/N: A00087CM) and blades (three removed),

- three blades (P/N: 3118353-01) removed from second-stage power turbine disk.

Figure A1 (Attachment A) presents a cross-sectional view of the PT6A-67D turbine (hot) section, identifying and illustrating the correlation of the components examined.

Engine details

The PWC PT6A-67D turboprop engine was a light-weight, free turbine powerplant, employing a five-stage combined axial/centrifugal compressor, with a single-stage axial compressor turbine and an independent two-stage axial power turbine driving the propeller via a reduction gearbox. The engine was nominally rated at 900 kW / 1,200 shp¹³ maximum continuous output power.

At the time of the last workshop maintenance visit on 8 February 2008, the engine power section module (S/N: PCE 114386) had operated for 21,466 hours since new (TSN) and 3,408 hours since the last overhaul (TSO), which was conducted in May 2005. Overhaul records showed that both first and second-stage power turbine blades (P/N: 3120112-01 and 3118353-01 respectively) were renewed at that time. Histories of other non-lifed¹⁴ turbine section components (e.g. stator assemblies) were not available from the maintenance documentation examined.

The last engine hot-section inspection (HSI¹⁵) was completed on 23 November 2006, at an engine TSN of 20,019 hours and TSO of 1,961 hours. Engine HSI's were scheduled for intervals not exceeding 2,000 hours of operation.

Component examination

Power turbine stator housing and interstage stator vane ring

The power turbine stator housing and interstage stator assembly were received as a unit (Figure 1), with the degree of distortion sustained by the vane ring preventing its free removal from the housing. Internally, the housing presented severe disruption and break-up of the vane ring around the rotational plane of both first and second-stage power turbine disks (Figure 2). The vane ring elements showed prominent impact and metal loss from the trailing edges, together with a pronounced rotational (torsional) and radial distortion of the vane set in the direction of turbine rotation. The stator air seal had separated from the ring inner platform as a result of the vane element distortion. The vane set also showed a degree of axial out-of-plane displacement toward the inner ends – the displacement presenting as a ‘dishing’ of the vane set in the upstream direction. Evidence of localised contact between the vane set inner platform ends and the upstream ends of the second-stage power turbine disk was also noted around much of the diameter – producing a machining metal-loss effect.

Externally, the turbine stator housing rear flange mounting bolt holes showed circumferential distortion – consistent with the torsional forces that produced the

¹³ Shaft Horse Power – power available at the engine output.

¹⁴ Non-lifed components are typically maintained ‘on condition’, whereby their suitability for continued service is assessed during each periodic inspection.

¹⁵ A prescribed periodic inspection of the engine combustion and turbine components.

Figure A1: Power turbine stator housing (down-stream end upward)



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Power turbine stator ring

The first-stage power turbine stator ring had sustained extensive structural break-up, with multiple fractures through the vanes and outer shroud sections (Figure A3). Five main ring segments and many smaller fragments were removed from the engine, with the interstage baffle plates separated from the stator ring bore. Visually and under the stereomicroscope, all principal fracture surfaces exhibited similar irregular and lightly oxidised characteristics – consistent with the fractures occurring during the breakdown event. Some evidence of pre-existing cracking was noted at the recess corners within the outer ring (Figure A4), however none of the cracks showed features suggesting their direct contribution to the break-up of the ring structure. The vane aerofoil sections, although damaged from the breakdown event, did not show any other evidence of advanced service-related degradation.

Figure A3: Main power turbine stator ring sections – as received



Figure A4: Suspected prior cracking of stator ring – between arrows



Compressor turbine disk and blades

The engine compressor turbine disk and blade-set presented comparatively low levels of mechanical damage, when compared against the downstream turbine components. The leading edges and tips of all blades showed only light debris impingement effects and adherent products (Figure A5). The blade trailing edges, around approximately one-half of the disk circumference, showed localised impact, deformation and edge breakage at the mid-span position (Figure A6). The regularity and form of the damage suggested a hard-body foreign object impact mechanism (Figure 7), with the absence of similar leading-edge damage suggesting the ingress of the material from a downstream location.

The downstream face of the compressor turbine disk showed evidence of light rotational contact against the stationary outer periphery of the interstage baffle or the inner rim of the power turbine stator ring.

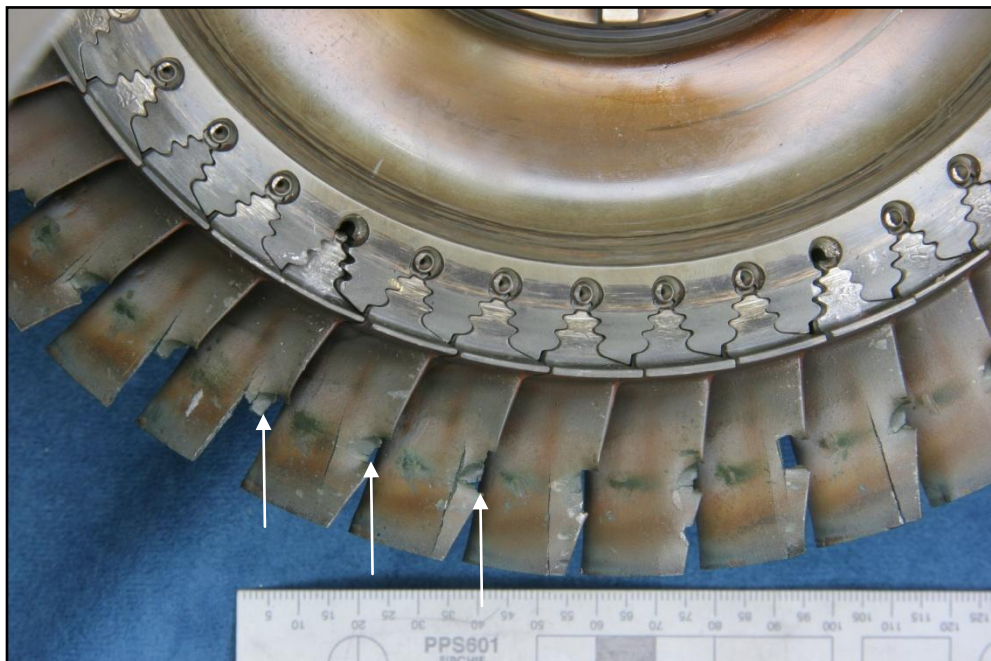
Figure A5:Upstream face of the engine compressor turbine disk



Figure A6:Downstream face – compressor turbine disk



Figure A7: Suspected hard-object impact damage to compressor turbine blade trailing edges (arrowed)



First-stage power turbine disk and blades

As-received, the first-stage power turbine disk assembly presented the fracture of all blades through the aerofoil section at the base of the root platforms (Figure A8). Fracture characteristics were consistent, and typical of forced failure under bending or impact overstress conditions.

The leading edges of the blades at the root platform transition, had sustained extensive rotational contact wear, or ‘machining’ damage, extending for approximately one-third of the blade chord width (Figure A9). The blade platform extensions beyond the upstream and downstream disk faces had also been machined or broken away in many areas. Corresponding wear to the upstream inner circumference of the power turbine stator ring suggested that it was interference between the two components (stator ring and turbine disk) that had produced or contributed to the observed blade damage.

Figure A8: First stage power turbine disk and fracture blade set



Figure A9: Rotational wear damage to the blade leading edge transition (arrowed) – result of interference with the stator ring



Second-stage power turbine disk and blades

The second-stage power turbine disk assembly was received with three blades removed from their fir-tree slots. Approximately half of the remaining blades had fractured through the aerofoil section, at varying positions along the span, while the remainder had failed at the root platform transition (Figure A10). All observable blade aerofoil fractures resembled those of the first-stage blades, with all being typical of forced fracture under localised impact or more general bending overstress conditions.

The inner leading edges of many blades showed a hemispherical notching effect, typical of rotational contact against the inside edges of the power turbine interstage stator assembly. The blade platform ends and the general blade leading edge profile showed similar metal loss and associated bending, edge burring and breakage.

Of the three removed blades, one had fractured transversely through the fir-tree root transition beneath the blade platform (Figure A11). That blade (Figure A12) presented a fracture surface that was characterised by two distinctive regions – a flat and uniformly discoloured zone extending from the trailing edge corner / break-edge of the first fir-tree waist, transitioning sharply to a prominent dendritic morphology for the remainder of the section.

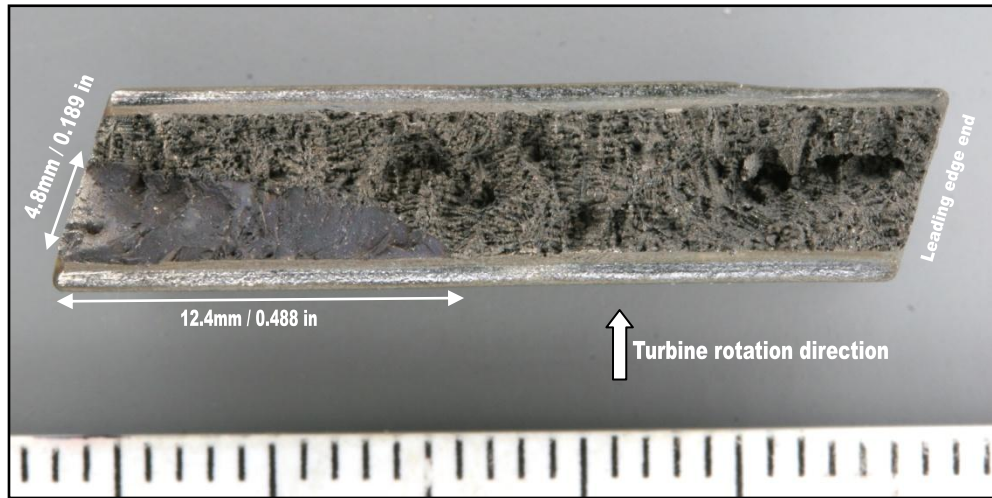
Figure A10: Second-stage power turbine disk – as received (upstream face)



Figure A11: Blades removed from the second-stage power turbine disk



Figure A12: Fracture surface regions from the sub-platform failed blade



Blade examination – second-stage power turbine

The single fractured second-stage power turbine blade that exhibited mixed-mode fracture morphology was the subject of further detailed study to characterise the fracture mechanisms. The liberated sections of the blade above the plane of fracture were not recovered amongst the debris removed from the engine, and as such, the part number of the failed blade could not be verified. However, general visual and dimensional comparisons against the two neighbouring blades showed no evidence that the particular failed blade was of a different form or characteristics that the remainder of the installed items (P/N: 3118353-01).

Fractography

The region of comparatively flat, transverse fracture at the root trailing corner extended for approximately 12.4 mm chord-wise and for 4.8mm in a through-thickness orientation (Figure A12). Although stained and discoloured, the surface detail in the corner region was moderately well-preserved, and under low-power microscopy, presented curved, concentric fatigue crack progression markings. The transition to coarse dendritic fracture occurred abruptly at the limit of fatigue cracking, with the remainder of the fracture being typical of ductile overstress failure in cast nickel-superalloy components.

Close study of the fracture morphology at the section corner revealed a hemispherical surface sub-zone of bright crack progression markings, surrounding an irregular darker feature (Figure A13). The zone was located on the root trailing edge, immediately adjacent to the corner break-edge transition to the blade trailing edge end (Figure A14).

Figure A13: Corner region at the origin of fatigue cracking (arrowed)

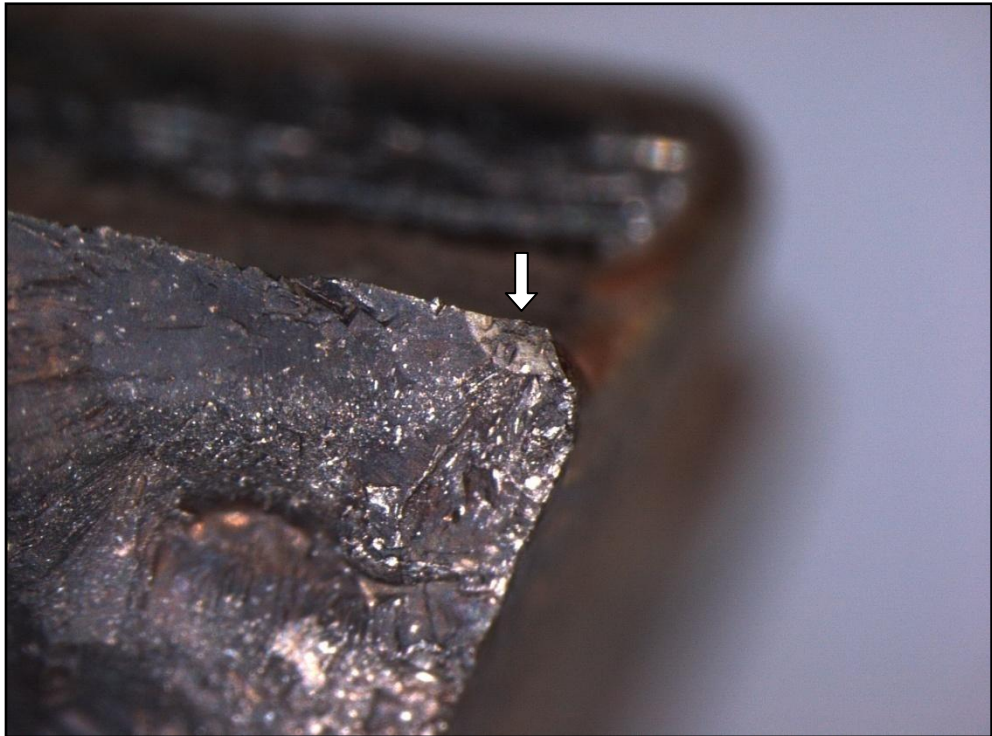
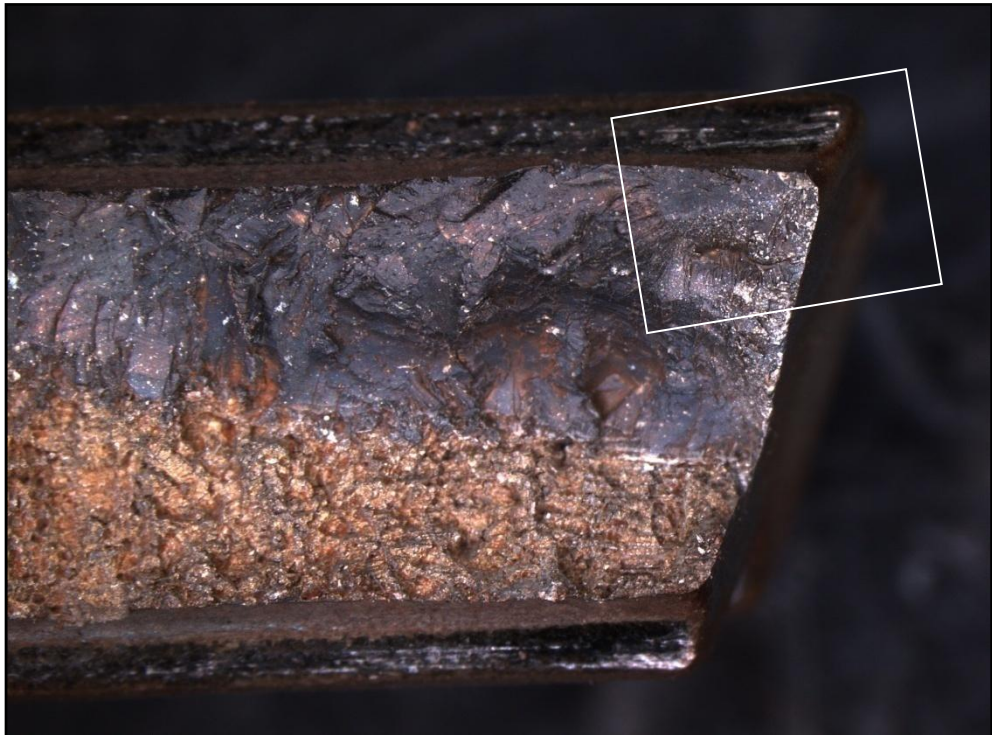


Figure A14: Location and orientation of fatigue crack origin (enclosed area represented in Figure A13)

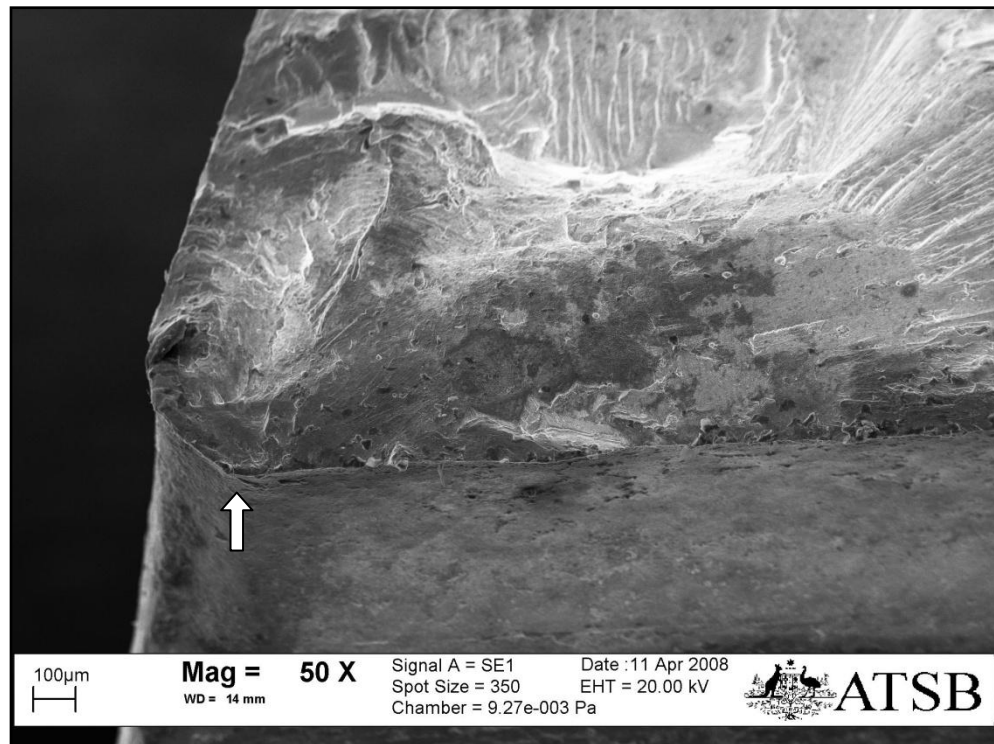


Electron microscopy

Scanning electron microscopy (SEM) techniques were employed to further examine and characterise the fatigue crack origin and surrounds. Although somewhat smeared and damaged, the surfaces around the origin showed evidence of the typical concentric radiating striations associated with high-cycle fatigue (HCF¹⁷) crack growth (Figure A15). Multiple intersecting planes of cracking were evident across the surface – likely representing the underlying grain structure of the blade material.

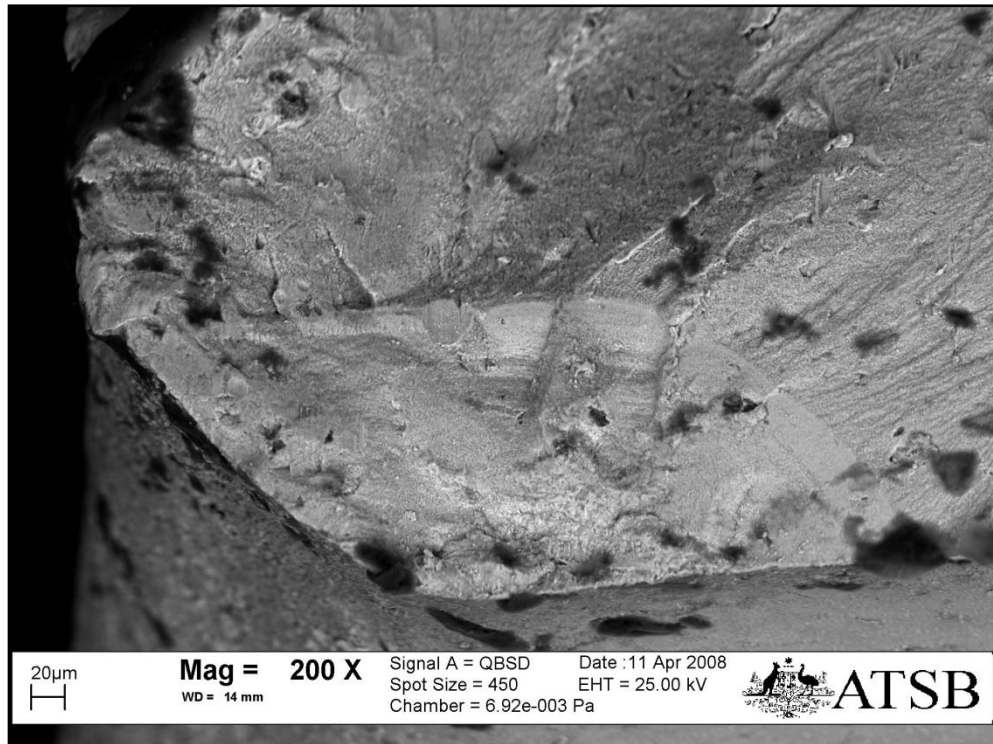
The atomic number contrast provided by back-scattered electron imaging (BSE) of the crack origin (Figure A16) showed no marked compositional variations within the region, aside from random surface accumulations of foreign light-element matter. The electron microscopy did not show any evidence of mechanical damage, tool marks or other localised physical features associated with, or adjacent to the origin region.

Figure A15: Low magnification SEM image of the fatigue crack origin (arrowed) and surrounds



¹⁷ A mode of fatigue crack propagation where the dominant cyclic stress environment is a high-frequency vibration, oscillation or resonance.

Figure A16: Fatigue crack origin – BSE image



X-ray imagery

Energy-dispersive x-ray mapping was carried out across the origin region depicted in Figure A16. Aside from the visually apparent accumulations of surface matter (characterised as silicon or carbon based contaminants), there was no clear evidence of any entrained material with a compositional base that was significantly different to the bulk blade alloy.

Metallography

A sample removed from the fractured blade root was mounted and prepared to allow the examination of multiple parallel sections inward from the end face towards the centre of the fatigue origin region. Prior to mounting, the origin area was measured under the SEM to provide a datum against which the position of the prepared surfaces could be related (Figure A17).

At low magnifications, the blade root material displayed a coarse dendritic microstructure, characterised by a semi-continuous network of an interdendritic second phase (Figure A18). At higher optical and SEM magnifications, the interdendritic network showed a skeletal, plate like morphology, typical of topologically close-packed (TCP) phases such as *Laves* phase (Figure A19), surrounding the dendrites of very fine cuboidal gamma-prime (γ') precipitate within a solid-solution gamma (γ) matrix (Figure 20).

Figure A17: Dimensional position of the fatigue crack origin region with reference to the downstream (trailing) end face of the blade

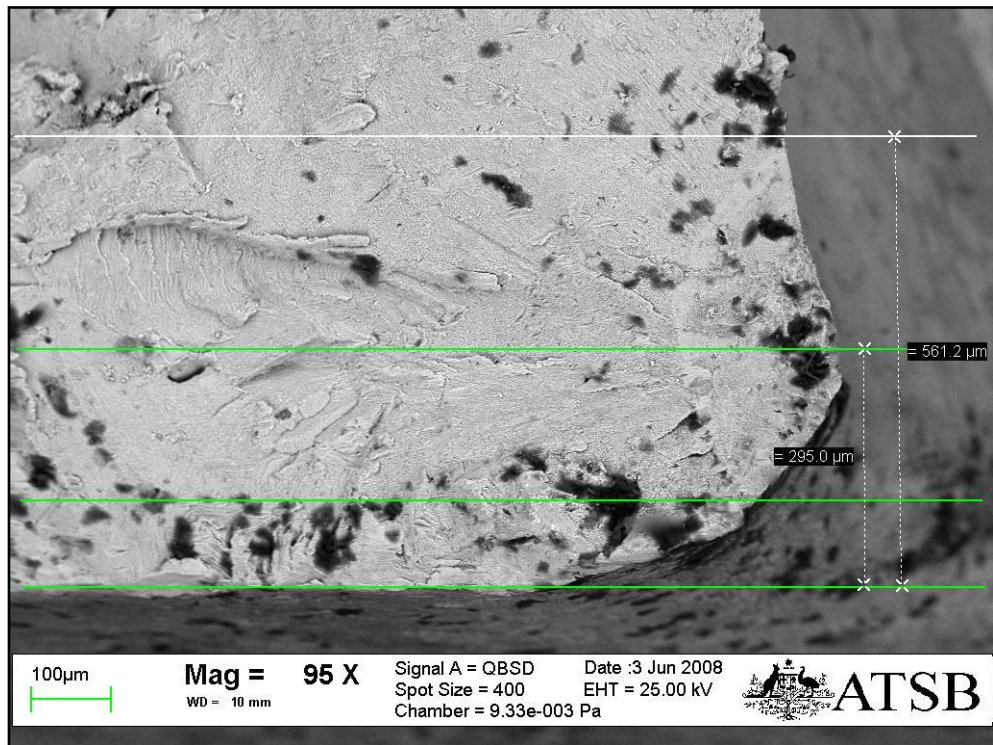


Figure A18: Blade root microstructure – interdendritic network of Laves phase, within a gamma, gamma-prime matrix

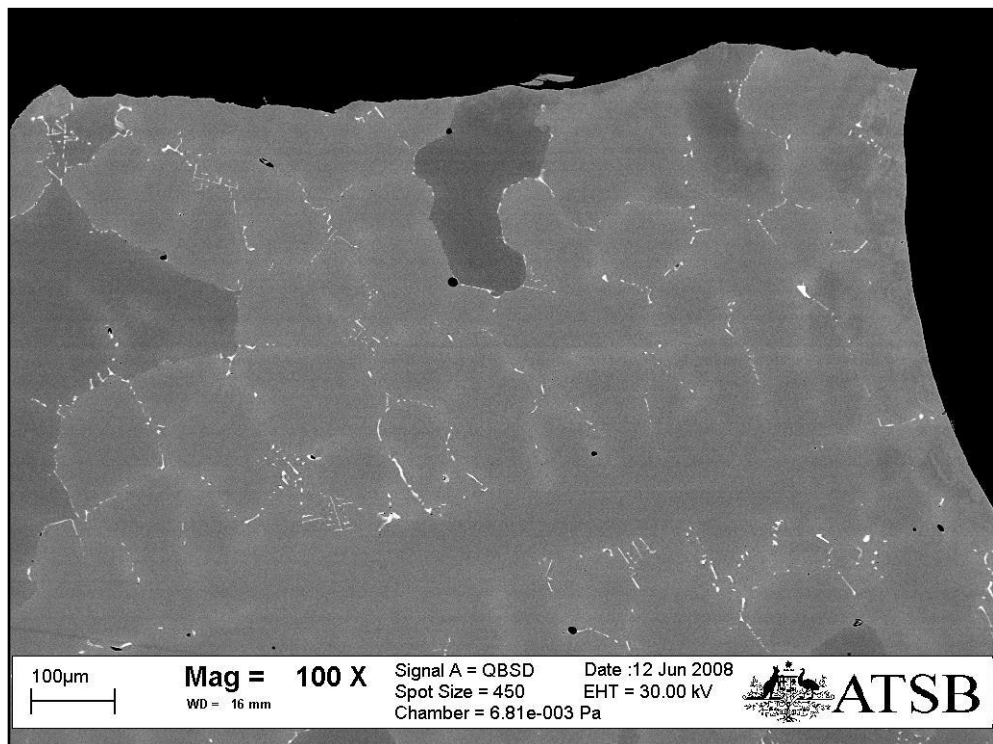


Figure A19: Interdendritic platelets of TCP phase (Laves) – light material

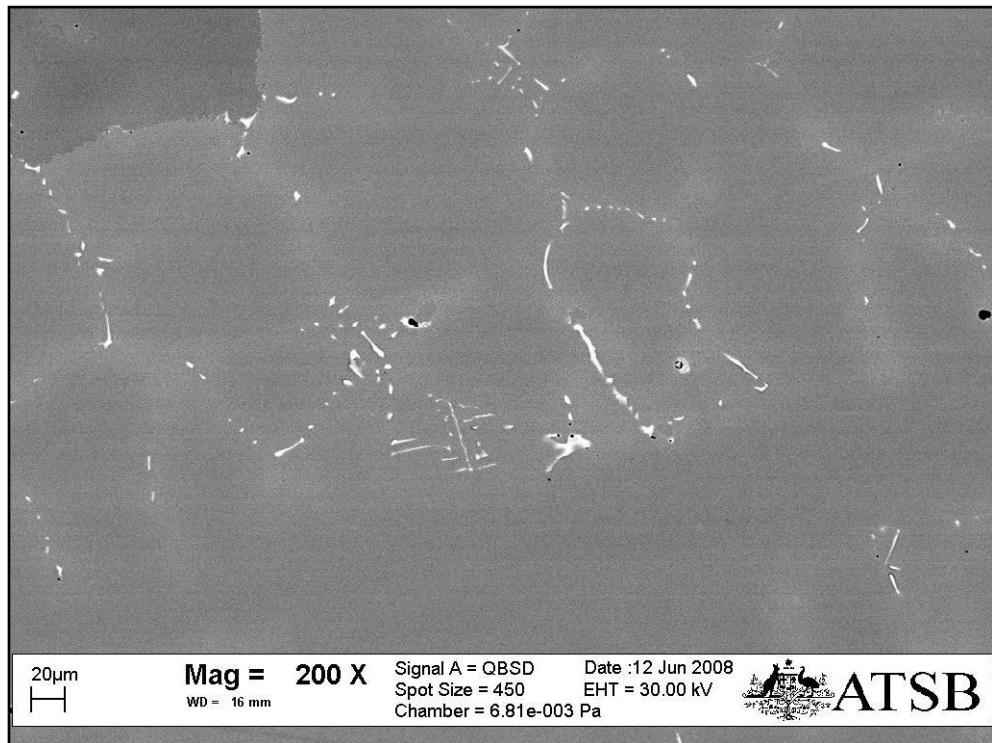
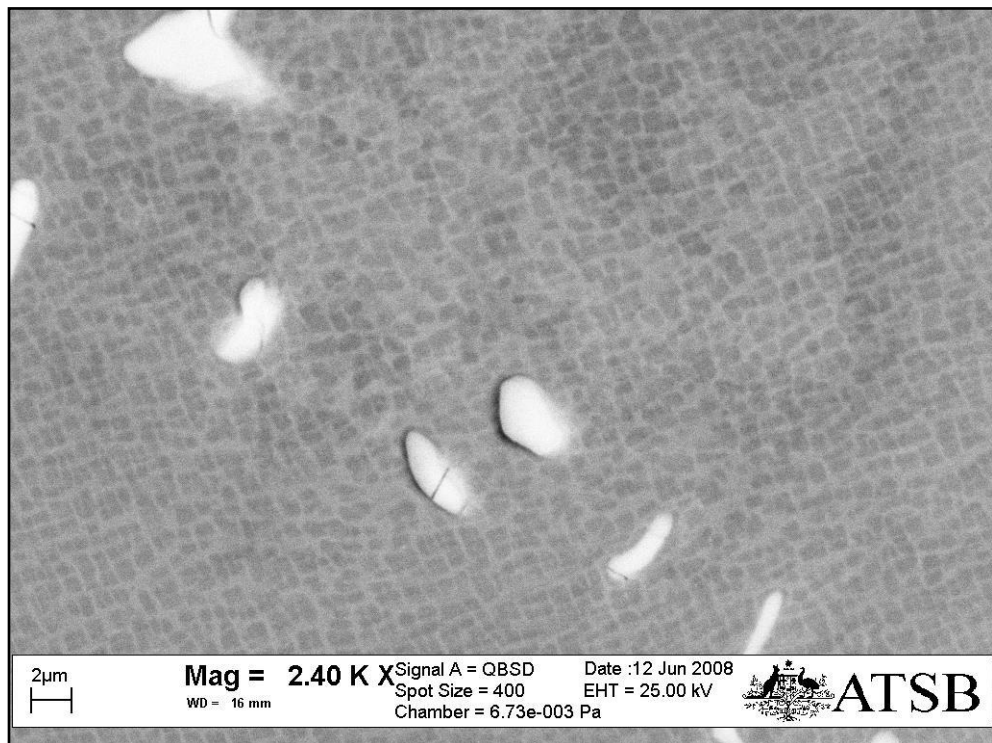


Figure A20: High magnification microstructural view, showing the gamma-prime precipitate (packed cuboids) and carbides / Laves phase (white areas)

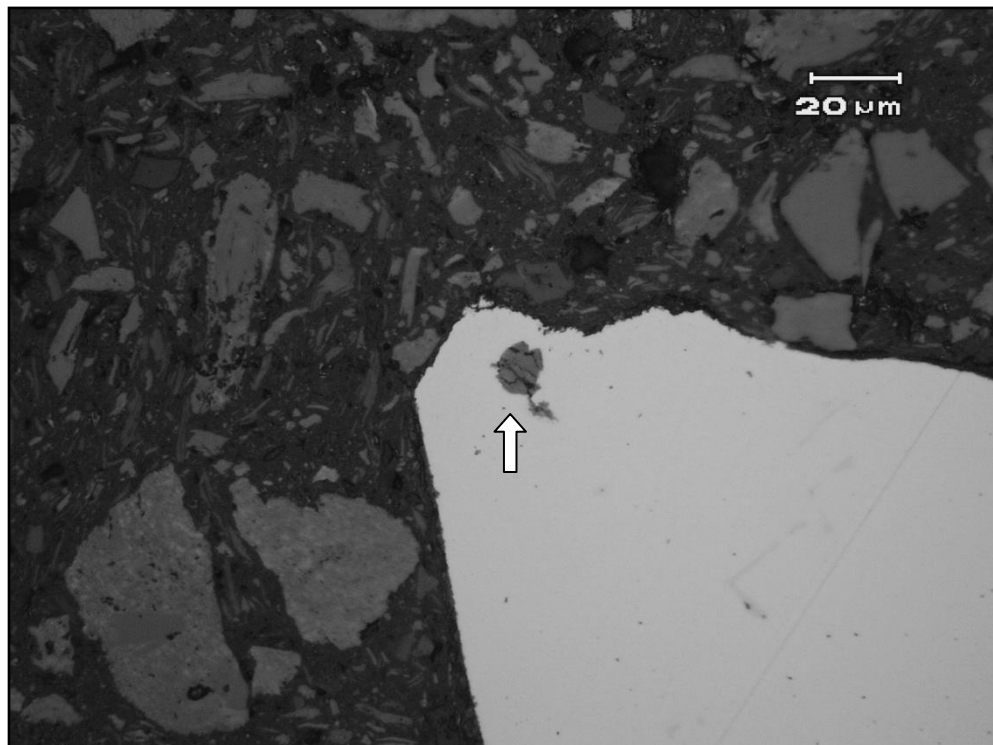


In total, three metallographic planes extending across the origin region were prepared and examined in a serial fashion. While no specific metallographic feature

was identified directly associated with the fracture origin, the examination did identify the presence of an isolated spherical non-metallic inclusion located adjacent to the fracture surface (Figure A21). The inclusion measured around 12 μm in diameter and was located at an axial depth of around 0.295 mm from the blade end face. Subsequent sections showed no evidence of this, or other similar inclusions within the bulk material microstructure.

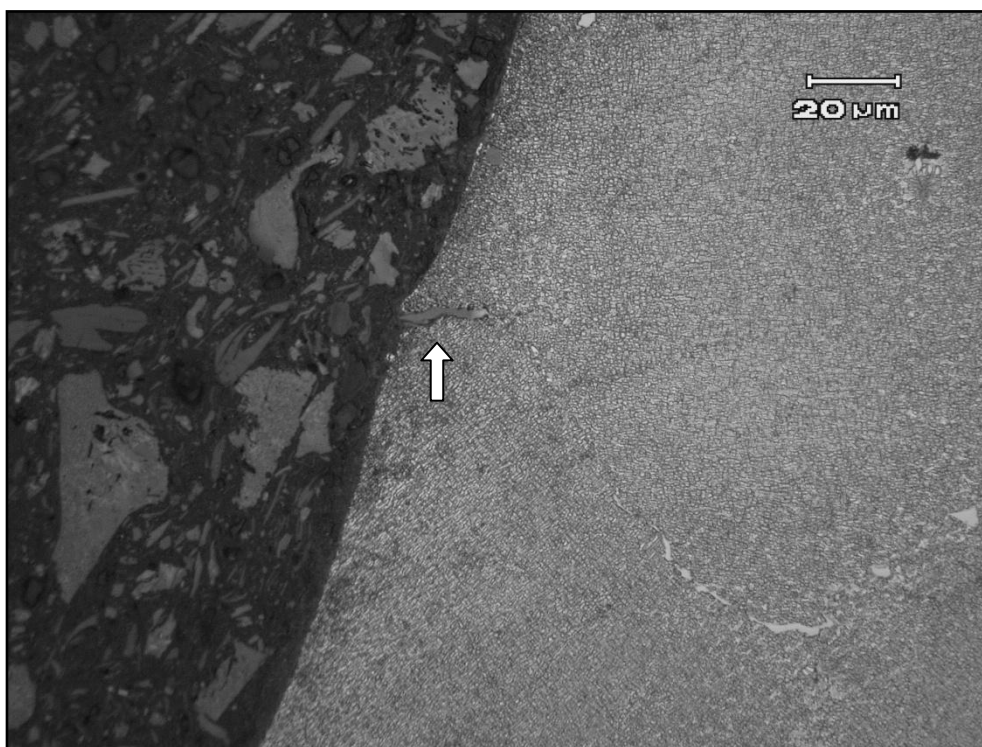
Along the external surfaces of the metallographic sections, several high aspect ratio¹⁸ 'intrusions' were observed – extending to a maximum depth of around 20 μm (Figure A22). In most regions, the intrusions were associated with areas where the interdendritic phase broke the external surface – the intrusions presenting as preferentially oxidised areas of that phase.

Figure A21: Spherical sub-surface non-metallic inclusion (arrowed) adjacent to the fatigue origin region (not etched)



¹⁸ Numerical ratio of the feature depth to its width.

Figure A22: Intrusive surface feature (arrowed) – appears as oxidised second phase (etched in Kalling's reagent)



Micro-analysis

Energy-dispersive x-ray spectroscopy (EDS), conducted on the prepared metallographic sections under the SEM, provided a qualitative indication of the elemental composition of the base metal alloy and the non-metallic inclusion observed microscopically. The base blade alloy presented as a nickel-based superalloy of chromium, molybdenum, aluminium and titanium (Figure A23). The inclusion was predominantly niobium rich, with nickel, oxygen, aluminium, magnesium and zirconium sub-peaks (Figure A24).

Figure A23: EDS spectrum of the blade base metal alloy

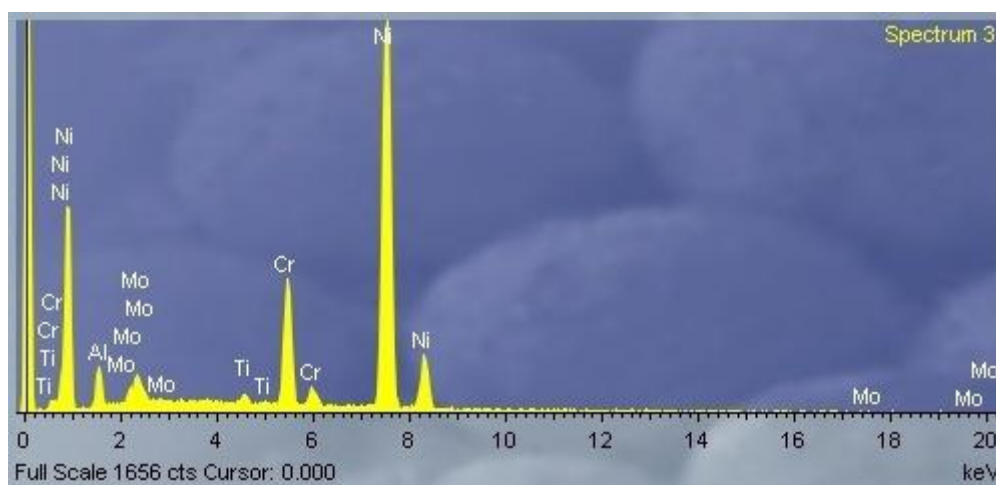
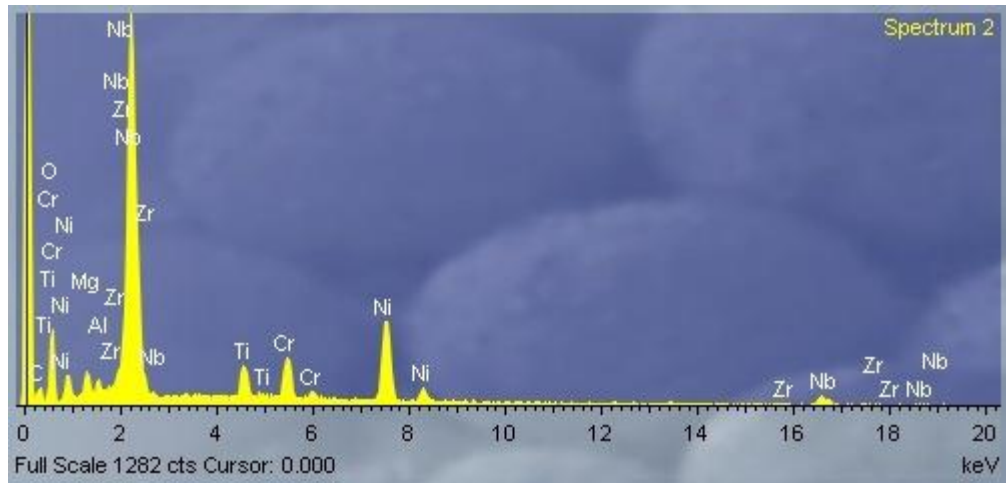


Figure A24: EDS spectrum of the spherical inclusion



Blade inspection

The remaining blades from the second-stage power turbine disk were removed from the disk, cleaned and the root surfaces examined for evidence of defects, damage or cracking – particularly in the areas around the trailing edge corners of the root post.

Under low-power stereomicroscopy, none of the blades exhibited any noteworthy features or indications of fatigue cracking similar to that presented by the failed blade.

ANALYSIS

Engine failure

From the observations made during the engine disassembly and subsequent laboratory examination, it was evident that the mechanical breakdown and failure of the engine had been precipitated by the fracture and separation of a single blade from the engine's second-stage power turbine rotor. The gross mechanical interference caused by the blade releasing into the confines of the turbine section, together with the resultant imbalance of the turbine wheel, had produced the subsequent forced failure of many other second-stage turbine blades and the generation of a large quantity of energetic debris. Although unusual in the sense that it required material movement against the gas flow path, it was probable that some of that debris had moved upstream into the rotational space occupied by the stage-one power turbine rotor, where it initiated the cascading forced failure of the blades from that stage. Upstream displacement of the interstage stator components to a point where interference occurred with the first-stage power turbine rotor was also a potential contributor. Failure and break-up of the power turbine stator assembly was consistent with the loads sustained during the multiple likely impacts with the stage-one turbine blade debris.

Second-stage power turbine blade failure

Fracture of the stage-two power turbine blade in question had occurred from beneath the transition platform between aerofoil and fir-tree root sections. A large and well defined area of prior cracking attested to the initiation and growth of a high-cycle fatigue cracking mechanism from the downstream trailing corner of the root post. Fatigue cracking had transitioned to overstress rupture at a loss of approximately 25% of the original cross-section.

Optical and electron microscopy confirmed the origin of the blade fatigue cracking to lie adjacent to the corner break-edge region, with a small, semi-elliptical region of crack development radiating into the section from a point approximately 0.5 mm from the blade end face datum. The origin exhibited no evidence of mechanical damage or other discontinuity, and metallography, EDS micro-analysis and x-ray mapping failed to identify the presence of any anomalous metallurgical features that could be directly associated with the point of crack initiation. A single isolated non-metallic inclusion, probably entrained during the blade casting process, was identified near the fatigue origin, however there was no evidence to associate that feature with the cracking sustained.

Microstructurally, the bulk material presented the expected γ - γ' constituents typical of high-temperature superalloys. There was no evidence of creep voiding or microstructural degradation associated with sustained or transient over-temperature operation. The presence of an interdendritic TCP second phase (suspected as Laves) was, however, considered suboptimal from a fatigue and oxidation resistance perspective.

FINDINGS

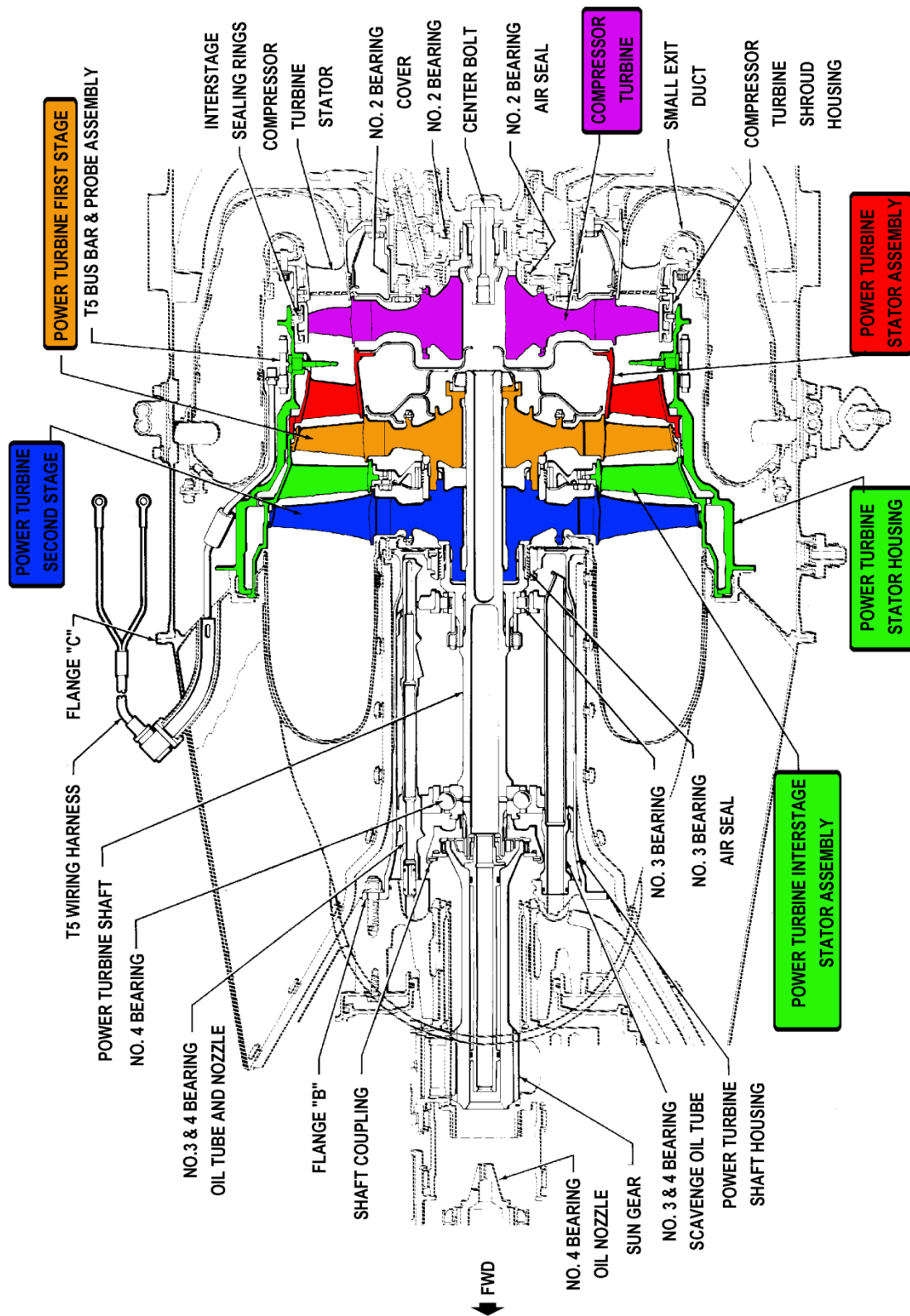
The following statements are a summary of the substantiated findings made during the progress of the engine and component examination.

- Failure of the PT6A-67D engine was consistent with the mechanical disruption and breakage of componentry with the power-turbine section of the engine.
- The turbine section damage had been initiated by the fracture and liberation of a single blade from the second-stage power turbine rotor.
- Fracture of the turbine blade had resulted from the propagation of a high-cycle fatigue cracking mechanism through approximately 25% of the blade fir-tree root post, followed by the overstress rupture of the remaining section under service dynamic loads.
- Fatigue cracking originated from a point adjacent to the downstream trailing root post corner.
- There was no evidence of gross manufacturing deficiencies within the blade microstructure at the fatigue crack origin.
- There was no evidence of tool marks, machining damage or other mechanical damage at the fatigue crack origin.
- The second-stage power turbine blades were replaced with new items during the last engine overhaul.
- The second-stage power turbine blades had operated for approximately 3,400 hours since replacement.
- The last engine hot-section inspection had been carried out approximately 1,450 operating hours prior to the engine failure.

While the examination did not identify any direct evidence of a mechanism or feature that had produced or contributed to the initiation of fatigue cracking from the blade root, it remains possible that an isolated casting anomaly similar to the observed inclusion or intrusion features, had acted to reduce the fatigue endurance of the failed component to a point where it became susceptible to failure under normal operating conditions and timeframes.

ATTACHMENT A

Figure A1A: Cross-sectional illustration of the PT6A-67D turbine section



APPENDIX B : SOURCES AND SUBMISSIONS

Sources of information

The sources of information for the investigation included the:

- flight crew of VH-VAZ
- operator of VH-VAZ
- owner of VH-VAZ
- engine manufacturer
- aircraft and engine historical documentation.

Submissions

Under Part 4, Division 2 (Investigation Reports), Section 26 of the Transport Safety Investigation Act 2003, the Executive Director may provide a draft report, on a confidential basis, to any person whom the Executive Director considers appropriate. Section 26 (1) (a) of the Act allows a person receiving a draft report to make submissions to the Executive Director about the draft report.

A draft of this report was provided to the Civil Aviation Safety Authority, the South African Civil Aviation Authority, the Transportation Safety Board of Canada, the National Transportation Safety Board, the engine overhaul facility, the aircraft operator, the aircraft owner and the engine manufacturer.

A submissions was received from the engine overhaul facility. The submission was reviewed and where considered appropriate, the text of the report was amended accordingly.