TECHNICAL ANALYSIS REPORT No: 05/02

OCCURRENCE No: 200104983

REFERENCE No: BE/200100024

Examination of failed components from a Garrett TPE331-11U turboprop engine

Fairchild Industries Inc. SA227-AC VH-VEH

EXAMINATION OF FAILED COMPONENTS FROM A GARRETT TPE331-11U TURBOPROP ENGINE

FAIRCHILD INDUSTRIES INC. SA227-AC, VH-VEH

1. FACTUAL INFORMATION

1.1 Introduction

While on descent for a landing at Melbourne airport, the crew heard a loud 'bang' from the left engine and noted the aircraft yaw to the left. Observing that the left engine output torque had dropped to 15 percent and all other indications were normal, the pilot elected to leave the engine operating at low power and continued the flight to Melbourne.

Initial investigation by maintenance staff found many damaged and missing blades within the turbine assembly of the left engine. To conduct an investigation into the engine failure, the ATSB subsequently obtained the first stage turbine rotor disk and all segments from the first stage inlet guide vane assembly.

1.2 Component history[2]

At the time of failure, the Garrett TPE331-11U-611G turboprop engine (s/n P44460C) had accumulated 21,657 hours and 32,365 cycles since entering service in March 1986. 2,328 hours and 2,473 cycles had elapsed since the last major overhaul in late 1996.
The stage-one turbine wheel (s/n 203501-2113) was installed during this overhaul and had a cumulative service life of 9,901 hours and 13,600 cycles when fitted. All 36 stage-one turbine blades were replaced with new items at this time.

The stage-one inlet guide vane segments were not identified as life-limited components by the manufacturer and thus did not have a documented service history. The segments are inspected periodically and removed from service if cracking or other 'metal loss' is found[3].

1.3 Visual examination and fractography

1.3.1 Stage-one turbine disk assembly

The stage-one turbine assembly comprised thirty-six air-cooled blades, retained within the disk by a tapered firtree connection (figures 1 & 2). While most of the turbine blades showed severe mechanical breakage immediately above the base platform, a single blade had broken away from beneath the platform (figure 3). Occurring at the top of the firtree root, the blade fracture was characterised by a stained and discoloured surface that presented several distinct regions (figure 4). The most dominant of these was a zone of fatigue crack progression-marks extending into the section from a small region on the outer surface of the root (concave blade side, figure 5). The total area of fatigue cracking represented around half of the total cross-section at the point of failure; the remainder showing features consistent with ductile overload fracture.

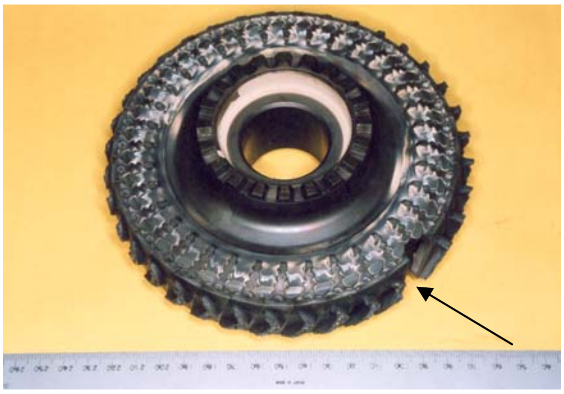


Fig. 1. Forward face of the stage-one turbine wheel – missing blade indicated.

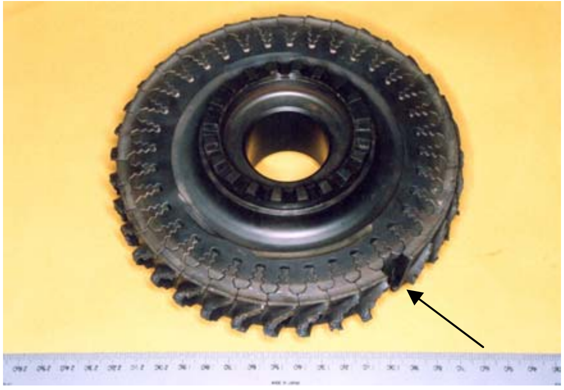
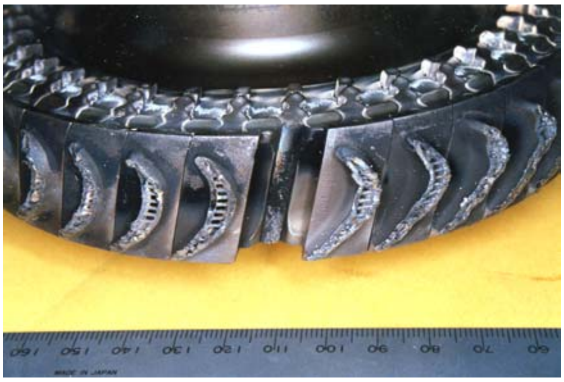


Fig. 2. Rear face of the stage-one turbine wheel – missing blade indicated.



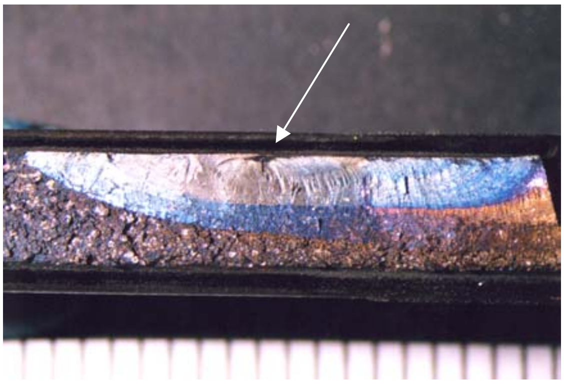


Fig. 3 (Top left). Failure of the turbine blade from beneath the platform base.

Fig. 4 (Above). Blade fracture surface (as-received).

Fig. 5 (Left). Blade fracture surface following cleaning – area of fatigue progression markings clearly visible. Fatigue origin indicated at arrow.

Inspection at low magnification (approx 40X) found the fatigue cracking had initiated from the surface of the root section on the high-pressure (concave) side of the blade. The initial areas of fatigue propagation were characterised as featureless, faceted areas with no visible progression markings and fracture following defined crystallographic planes (figures 6 & 7). Such areas are typical of Stage-I fatigue cracking in nickel alloys. Similar features were observed for some distance along the edge of the fracture (figures 8 & 9). Located at 9.5 millimetres from the forward edge of the blade, the largest faceted area formed the base of several characteristic 'ratchet' marks often associated with fatigue crack origins (figure 10). At the higher resolution offered by the scanning electron microscope, the progression of Stage-II fatigue cracking from the base of the faceted area was clearly observed, with regular arrest marks identified (figure 11).

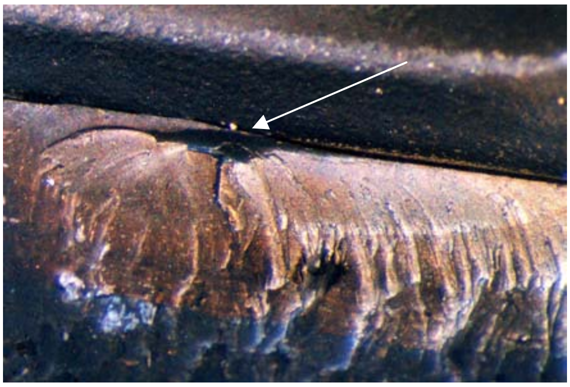


Fig. 6. Close optical view of the fatigue origin (arrowed).

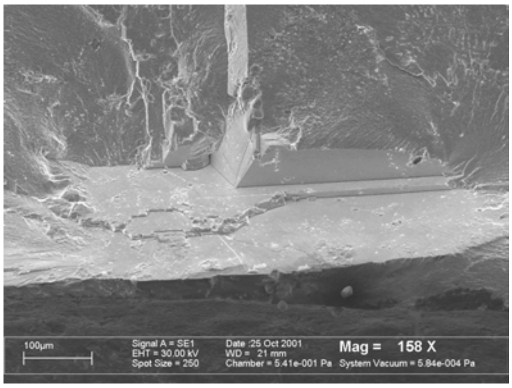
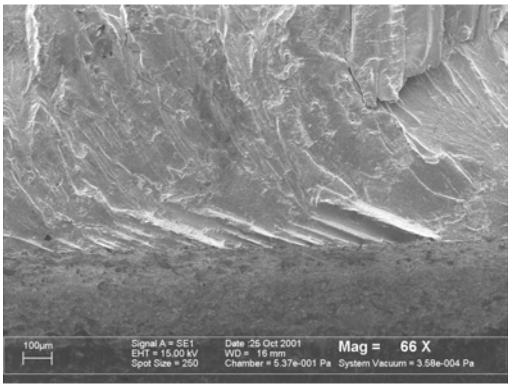
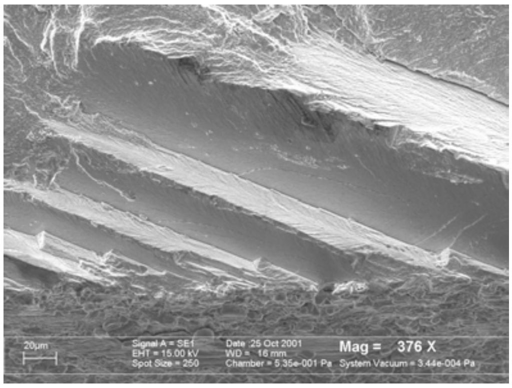


Fig. 7. Faceted appearance of the fatigue origin – typical of Stage-I fatigue crack propagation.





Figures. 8 & 9. Other regions of faceted crack formation – again typical of Stage-I fatigue.

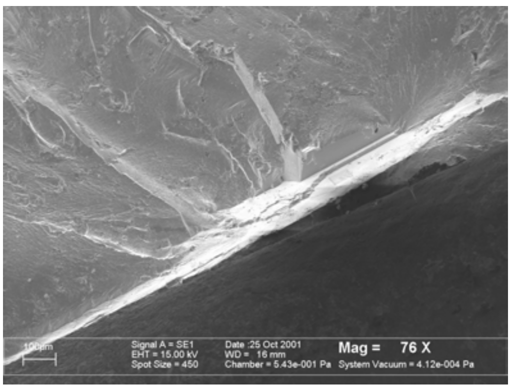


Fig. 10. Electron microscope image of the transition from Stage-I to Stage-II fatigue cracking.

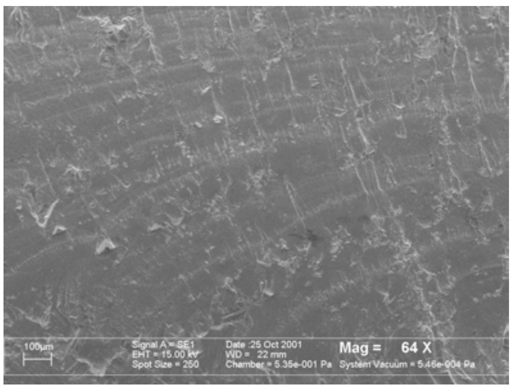


Fig. 11. Electron microscope image of the outer regions of fatigue cracking – regular arrest marks clearly shown.

Examination of the various fracture areas failed to reveal any evidence of physical defects, discontinuities or other anomalous features that were likely to have contributed to the failure. As far as could be determined visually, the seating of the fractured blade within the disk firtree slot was uniform and stable. Galling, fretting and other evidence of localised movement was not observed in any of the stage-one blades or their respective disk slots. Visual and low-power stereomicroscopic examination also failed to reveal any further evidence of cracking or defects within the roots of the remaining 35 stage-one turbine blades.

The fractured blade carried the following identification on the root underside.

89193 P 23151-6053
3108125-1

1.3.2 Turbine inlet guide vane segments

The TPE331-11U turbine inlet guide vane (IGV) assembly comprised of eleven discrete circumferential segments, each containing two guide vanes (figures 12 & 13). As an assembly, the IGV had sustained appreciable mechanical damage during the engine failure – mostly on the outlet side adjacent to the first-stage turbine wheel (figure 14). Many of the guide vanes contained transverse cracking within the leading edge and in some cases, this cracking had developed into appreciable wide-mouthed notching (figure 15). In most areas, cracking of this nature remained confined to a band between six to eight millimetres from the inside platform edge. In the most severe cases, the notching had penetrated the internal air cooling galleries (figure 16).

It was apparent from early inspections that one of the eleven IGV segments had sustained the complete 'burn-through' failure of a single vane (figure 17). The centre of the affected area was coincident with the location of the leading edge notches found on the other vanes. Consistent with overheating and oxidation, the edges of the affected area were thinned and friable and showed a pronounced discolouration of the surface oxide. The failed segment carried the following identification:

99193   3103820-111
3103820-2   GTE   0707
53468 80999R1   A.E.???   [Note: '*???*' denotes illegible markings]
SV2194

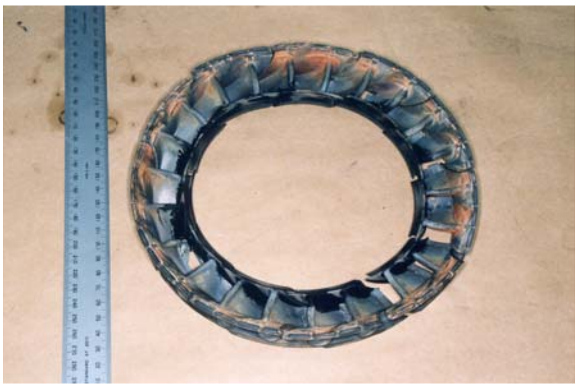


Fig. 12. Turbine inlet guide vane segments – forward faces.

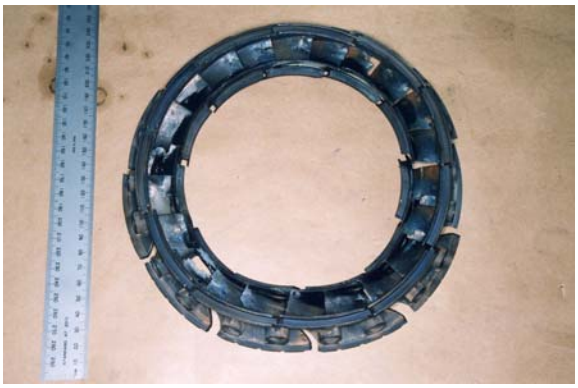


Fig. 13. Turbine inlet guide vane segments – rear faces.

Fig. 14. Mechanical damage to the trailing edges of the inlet guide vanes (typical).

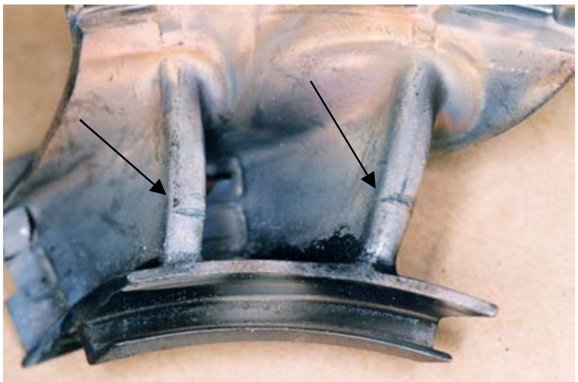


Fig. 15. Thermal fatigue cracking within the leading edges of the guide vanes (arrowed).

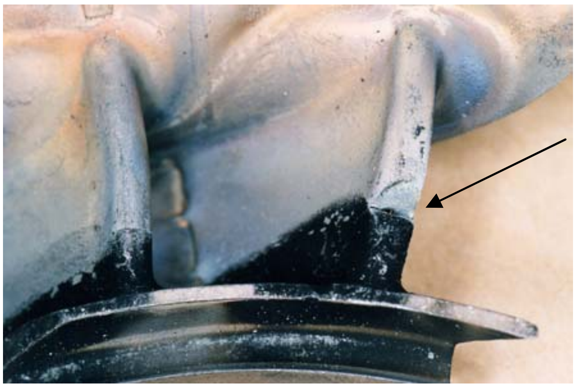


Fig. 16. Advanced thermal fatigue cracking, developed into notches that penetrated the vane cooling galleries.

Fig. 17. Destruction of a guide vane from the effects of overheating and oxidation.

2. ANALYSIS

From the results of the investigation, it was evident that the failure of a stage-one turbine blade within the engine assembly was the principal cause of the engine malfunction observed by the aircraft crew. The blade had failed by bending fatigue cracking that had initiated at the top of the firtree root connection. By design, this location was the area of maximum centrifugal and aerodynamic stresses within the blade. Numerous regular arrest marks within the zone of fatigue cracking attested to the propagation of the cracking over multiple flight cycles. Final overload failure of the remaining section occurred after growth of the fatigue cracking to around half of the total root cross-section. The examination found no evidence of pre-existing defects or conditions that could have predisposed the blade to premature failure. The nature of the fatigue damage, with a large area of faceted Stage-I cracking was consistent with high frequency, low-magnitude driving stresses[1].

The damage to the aerofoil sections of the stage-one IGV assembly was consistent with the effects of thermal fatigue cracking. Expansion and contraction associated with engine temperature changes produces alternating axial stresses in the vanes, which subsequently leads to fatigue crack initiation once sufficient cycles have accumulated.

Growth of these cracks into the aerofoil body leads to the perforation of the internal cooling galleries and the disruption of the circulating airflow. Deprived of cooling, the vanes rapidly overheat, oxidise and fail in the manner observed.

The damage and loss of an inlet guide vane would disturb the flow of high-pressure combustion gasses through the assembly. Given that the stage-one turbine is driven by gas flow through the guide vanes, each turbine blade would experience a once-per-revolution impulse on passing the damaged vane. In this situation, the excitation of resonance or vibration is likely, producing the type of high-cycle, low-magnitude stresses necessary for fatigue crack initiation and growth. The nature of the fatigue cracking experienced and the absence of cracking in any of the other blades suggests the stress level was only marginally above the threshold necessary for fatigue development.

3. SIGNIFICANT FACTORS

The investigation identified the following significant contributory factors in the in-flight failure of the TPE331 engine.

- Development of thermal fatigue cracking in the leading edges of the stage-one turbine inlet guide vanes, leading to the perforation of the internal galleries, loss of cooling airflow and subsequent overheating.

- Overheating failure of a single aerofoil from the stage-one turbine IGV assembly, leading to a disturbance in the combustion gas flow through the turbine and the excitation of downstream blade vibration.

- Development of fatigue cracking within the root of a single blade from the stage-one turbine.

- Liberation of the cracked blade from the turbine rotor.

[1] ASM Handbook, Vol. 19 'Fatigue and Fracture', p50
[2] Engine Certification Log, TPE331-11U-611G, Serial No. P44460
[3] Allied Signal Aerospace Garrett TPE331-11 Maintenance manual, 72-50-07 p265