Australian Transport Safety Bureau

TECHNICAL ANALYSIS REPORT

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Examination of an RB211-524G-T Turbofan Engine Compressor Failure

Boeing 747-438, VH-OJU, 15 December 2002

EXECUTIVE SUMMARY

An Australian registered Boeing 747-438 aircraft operating a regular passenger transport flight sustained the failure of an engine shortly after takeoff from Los Angeles, USA. The engine was subsequently shut-down and the aircraft returned for an uneventful landing.

The failed engine was a Rolls Royce RB211-524G2-T model. Preliminary inspection by the operator's maintenance personnel found evidence of extensive internal mechanical damage within the high-pressure compressor section of the engine and as a result, returned the engine to Australia for inspection and overhaul.

The Australian Transport Safety Bureau examined the engine following its disassembly into primary modules. The engine had failed as a result of the liberation of a single blade from the first-stage high-pressure compressor section. That failure subsequently precipitated a titanium metal fire within the compressor, extensively damaging the following stages and rendering the engine inoperative.

The engine manufacturer has attributed three previous failures of RB211 high-pressure compressors to the loss of blades from the first-stage rotor. The blade losses were all associated with fatigue cracking of the dovetail root connection. The manufacturer identified uneven centrifugal loads on the blade roots as a significant factor in the development of blade cracking; possibly exacerbated by 'patchy' root friction and minor mechanical imperfections in the critical blade root transition region.

Evidence from the current investigation indicated the nature of the failure to be very similar to the previously reported events.

NR Blyth
Senior Transport Safety Investigator
Technical Analysis

1. FACTUAL INFORMATION

1.1. Introduction

A Boeing 747-438 aircraft operating a high-capacity regular passenger transport flight from Los Angeles, USA to New York, USA sustained the in-flight failure of the number-one engine while climbing through approximately 6,000 feet on departure from Los Angeles. The failure was characterised by an initial airframe jolt and a subsequent escalation in the engine exhaust gas temperature. Passengers also reported seeing flame emanating from the number-1 engine tail pipe. The engine was subsequently shut down and the aircraft returned for an uneventful landing.

1.2. Engine information

The engine was a Rolls-Royce RB211-524G2-T model, serial number 13762 (general arrangement figure 1). The engine had operated for a total of 13,922 hours and through 1,395 cycles at the time of failure. The engine was on its first operational life since manufacture, and as such had no prior overhaul or maintenance workshop visits. An initial inspection by ground engineering personnel found evidence of a mechanical failure within the high-pressure compressor stages of the engine. As a result, the engine was removed from the aircraft and returned to the operator's maintenance facilities in Australia for more detailed examination.

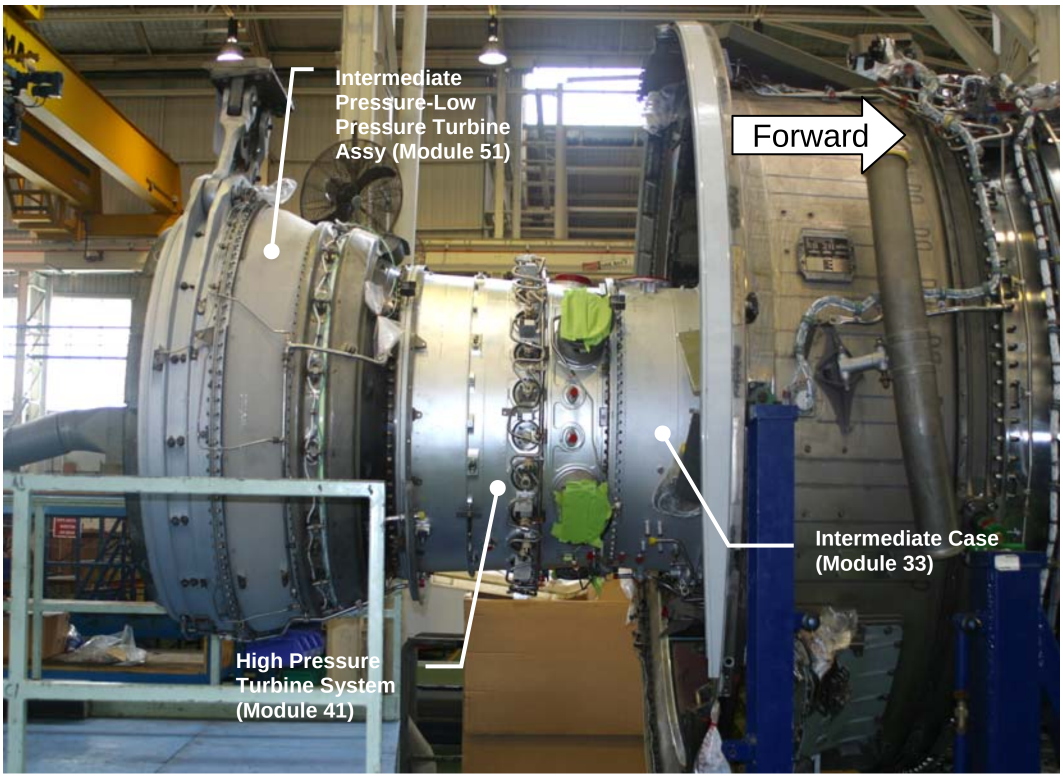


Fig. 1 General assembly – RB211-524G2-T

1.3. Inspection

A representative of the Australian Transport Safety Bureau (the author) examined the engine following its disassembly into modules.

1.3.1. Module 41 (HP System):

Module 41 of the RB211-524 engine core (figure 2) contained the high-pressure compressor (HPC), combustion chamber and high-pressure turbine (HPT) assemblies. Together, the HPC and HPT units made up the engine N3 spool. Inspection of the module centred on the examination of the first-stage high-pressure compressor rotor that was directly visible at the forward end of the module (figure 3). All of the compressor blades had sustained extensive damage to the tips and edges, with characteristic melting and loss of the leading and trailing edge corners (figure 4). Extensive metallisation and mechanical damage of the subsequent stage stator and rotor units was evident, as was the intermittent circumferential burn-through of the compressor case (figure 5).

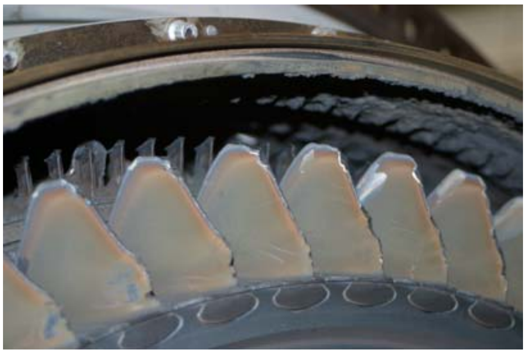


Fig. 2 (Top L) Module 41 assembly as examined. First-stage high-pressure compressor disk is at the left.

Fig. 3 (Top R) First-stage high-pressure compressor rotor, viewed looking aft. Missing blade is arrowed.

Fig. 4 (Above) Extensive metal loss as shown by the remaining HPC first-stage blades. The conical shape and extensive surface metallisation (splattering) is indicative of a titanium fire having occurred within the compressor.

Fig. 5 (Left) Burn-through sustained by the HPC first-stage case as a result of the titanium fire.

A single blade from the first-stage HPC rotor was missing from the disk, having completely liberated from its single-tier dovetail slot (figure 6). The trailing edge of the disk slot showed a prominent ridge of metal where the lost blade had evidently folded over in the slot before being released (figure 7).

Fig. 6 HPC 1 disk with missing blade.

Fig. 7 Metal deformation and burring found on the slot edge.

The root stub of the released HPC first-stage blade (figure 8) was recovered amongst the debris removed from the engine intermediate case (module 33) upon disassembly. While extensively battered and abraded, the stub clearly originated from the HPC first-stage rotor (figure 9). Close inspection of the stub showed evidence of the cracking and loss of the forward trailing edge corner of the blade dovetail root block (figure 10). The relatively smooth, even nature of the fracture surface was consistent with a fatigue cracking mechanism, originating from the edge of bedding area at the top of the dovetail (figure 11). None of the other 58 blades within the HPC first-stage assembly showed any visible evidence of cracking when inspected with the unaided eye.

Fig. 8 HPC first-stage blade stub as-recovered.

Fig. 9 Orientation of the blade stub with respect to the HPC first-stage disk.

Fig. 10 HPC first-stage blade stub, showing missing forward corner (arrowed).

Fig. 11 HPC first-stage blade stub corner fracture with suspected fatigue crack initiation region shown.

1.3.2. Module 51 (IP & LP Turbines):

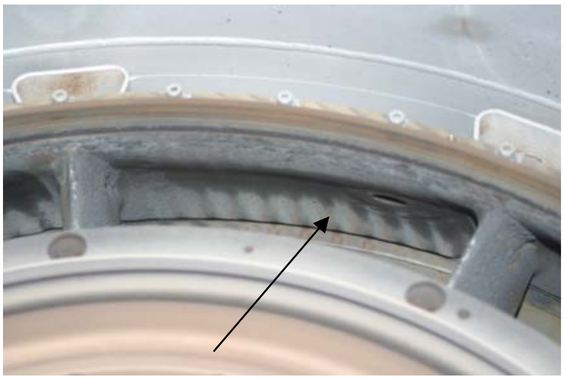
The inlet of the module 51 turbine assembly (figure 12) showed only minor, low level abrasion to the leading edges of the first stage intermediate pressure (IP) turbine inlet guide vanes (figure 13). Light metallisation and unusual gas flow patterns were evident through the nozzle guide vane (NGV) unit, however gross overheating or metal loss from the NGV leading edges had not occurred.

Fig. 12 Inlet to IP/LP turbine (module 51)

Fig. 13 Minor damage to the leading edges of the IP turbine inlet guide vanes.

1.3.3. Module 33 (Intermediate case)

The surfaces of the intermediate case immediately forward of the HPC inlet showed heavy metallisation and evidence of forward gas flow (figure 14), typical of a compressor stall/surge event. Where the HPC first-stage case had perforated, the inside walls of the intermediate case were sprayed with oxidised metal and exhibited a degree of overheating effects (figure 15). A single case strut showed localised perforation at the juncture to the outer case, with the puncture subsequently being oversprayed with the oxidised metal from the HPC (figure 16). Indentation marks adjacent to the puncture suggested the perforation might have resulted from the released HPC first-stage blade impacting the strut base.



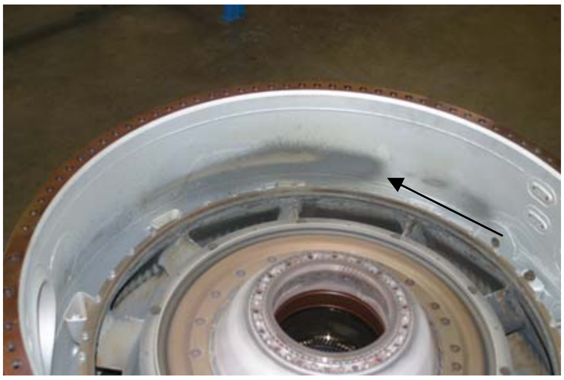


Fig. 14 (Top L) Module 33 intermediate case looking forward. Evidence of forward gas flow and combustion.

Fig 15 (Top R) Intermediate case with metallisation spray on the internal walls from the burn-through of the HPC first-stage casing.

Fig 16 (L) Intermediate case strut showing puncture and metallisation overspray.

1.4. Blade cracking:

The engine manufacturer was aware of a mechanism whereby high cycle fatigue cracking developed from uneven load distribution across the bedding surfaces of the blade dovetail. ‘Patchy friction’ between the contact surfaces resulting from a breakdown of the dry lubricant within the joint has been considered to be a significant factor, however evidence also exists to suggest that a stress-raising influence such as possible root flank mechanical damage is also contributory. Analysis by the manufacturer has shown the fundamental blade root direct centrifugal stresses to be low.

The engine manufacturer has attributed three prior failures of RB211-524G/H-T engines to this mechanism.

2. ANALYSIS

2.1. Failure mechanism:

Evidence gathered from the inspection of the engine modules indicated that the VH-OJU number-one engine failure was a result of the liberation of a single blade from the first stage high-pressure compressor rotor. Friction from the subsequent blade impacts with its neighbours produced conditions suitable for the onset and development of a titanium fire within the HPC assembly, which produced melting of the blade edges and the downstream compressor components. The characteristic pointed blade profile and the extensive covering of components with an oxidised metal spray (metallisation) was typical of this phenomenon. Disruption of the normal operation of the high compressor also produced engine airflow reversal or ‘surging’ which carried the metallisation and titanium combustion products forward into the intermediate casing.

The liberation of the HPC first-stage blade was attributed to the development of high-cycle fatigue (HCF) cracking across the forward corner of the blade dovetail block. Eventual loss of this material allowed the operational centrifugal and bending loads to fold the blade forward, breaking it free from the slot and producing the mechanical slot-edge burr as noted.

The evidence gathered during the examination of the subject engine (serial number 13762) suggested that the failure was a recurrence of a mechanism known by the engine manufacturer to have contributed to several previous high-pressure compressor failures in RB211 engines. That mechanism is understood to be the subject of ongoing investigation to determine a satisfactory solution.