Australian Government

Australian Transport Safety Bureau

ATSB TRANSPORT SAFETY INVESTIGATION REPORT

Additional Analysis of Left Engine Failure VH-LQH

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Introduction

VH-LQH was involved in an accident during takeoff from Toowoomba Airport on 27 November 2001. It was evident from witness reports that the aircraft lost thrust from the left propulsion system. The propulsion system consists of a Hartzell three-bladed propeller driven by a Pratt and Whitney Canada PT6A-20A gas turbine engine.

Examination of the left propulsion system after the accident revealed that there had been a structural failure in the turbine, or hot, section of the engine. The turbine section comprises two stages: one stage to drive the compressor (compressor turbine) and the other to drive the propeller (power turbine). Both stages had been damaged extensively. The event which initiated the sequence of structural failure throughout the compressor turbine and power turbine stages was not obvious.

Identification of the event that initiated structural failure in the turbine section

There are two distinct sources of evidence that may assist in the determination of the event that initiated the sequence of structural failure in the turbine section of the left engine: evidence of engine performance prior to the accident flight and physical evidence remaining after the collision with the ground.

Engine performance prior to the accident

The deterioration of engine performance can be monitored by recording key engine parameters under known operating conditions and load. The regular monitoring of changes in these parameters (trending) can provide an early indication of component deterioration.

Physical evidence remaining after ground impact

Component deterioration and ultimate failure occur by a number of physical/chemical mechanisms. Specific physical features are created by the action of each mechanism. Examination of the physical features of engine components after engine failure may allow the mechanism of component failure to be determined.

Examination of the left engine physical evidence must be conducted with the knowledge that following the initial failure event engine rotation and combustion did not cease immediately. Continued rotation of the compressor and power turbines and the continuation of combustion created damage which is secondary to the initiating event. In addition, structural damage was caused by ground impact.

An additional impediment to the establishment of the nature of the initiating event is the incomplete recovery of all remnants of fractured components.

Compressor turbine and power turbine failure modes and maintenance actions

The physical evidence associated with the compressor and power turbines needs to be evaluated against the known modes of failure and the maintenance actions designed to prevent these modes of failure occurring during engine operation.

Gas turbine engines do not have an infinite life. The components of the engine are subjected to various processes of deterioration during operation. The known modes of deterioration are related to the processes of wear, deformation under the influence of stress and elevated temperatures (creep), reactions between the materials used to form the engine components and chemical species in the operating environment, and fracture arising from the presence of propagating cracks. These are progressive modes of failure that develop as a result of increasing numbers of engine cycles and total time in operation.

In addition, there are modes of deterioration that may be classified as accidental failure modes, exceeding operating limits (turbine temperature and speed), turbine blade tip/shroud interference, impact damage through contact with a foreign object.

The prevention of structural failure and detrimental performance loss is achieved through a limit on the total time in service for components subjected to cyclic stresses and ongoing maintenance actions, such as, inspections and measurements. The specification of limitations, inspections and measurements form a system of maintenance for the engine. The system of maintenance defines the actions required to be undertaken, when each action must be carried out and when the actions must be repeated.

Actions – PT6A-20A

Component total time in-service limit

There is a life limit on the 1st stage compressor hub, 2nd stage compressor disc, 3rd stage compressor disc, impeller, compressor turbine disc, power turbine disc. These components are required to be replaced when they have been subjected to a specified number of engine cycles to prevent the development of fatigue cracking. The PT6A-20A engine compressor turbine disc and power turbine disc must be replaced after the component has been exposed to 18,000 engine cycles (PW&C SB No 1002R24 *Turboprop engine rotor components – service life*).

The lives of the engine main bearings, the 1st stage planet gears and sun gear in the reduction gearbox, and the power turbine blades are limited to 12,000 hours total time in service (PW&C SB No. 1803R1 *Turboprop engine operating time between overhauls and hot section inspection frequency*).

Compliance with the limitations on the lives of rotating components requires knowledge of the history of each component in terms of total time in-service and the number of engine cycles accumulated during this time in service.

Overhaul

The engine is disassembled to allow the detailed inspection and measurement of components. Component deterioration is assessed against allowable limits. Components may be repaired or replaced.

The standard interval between overhauls is 3,600 hours in service. This interval may be extended by incorporating suitable changes to the maintenance system. The changes required are detailed in engine manufacturer's SB 1803R1 and the Australian Regulatory Authority's AD/ENG/5 Amdt 9 Appendix A 'PT6A Enhanced Maintenance Program'. AD/ENG/5 allows the interval between overhaul to be extended to 5000 hours through the implementation of more frequent inspections directed at identifying the initial stages of component deterioration, allowing corrective action to be taken before significant operational or performance problems develop.

There are other proprietary maintenance systems such as MORE (Maintenance Of Reliable Engines) that can be applied to PT6 engines.

The compressor turbine disc and blades must be sent for overhaul inspection (includes non-destructive testing and measurement of each blade) after 5,000 hours in service if a new set of blades was installed at the last overhaul or after 3,000 hours if a full or partial set of used blades had been installed at the last overhaul. (SB 1803R1)

Hot section inspection

The engine is partially disassembled to allow the combustion chamber, compressor turbine section and power turbine section to be examined in detail. (SB 1803R1)

The normal frequency of hot section inspections is 1,800 hours in service. This interval does not increase as the overhaul interval increases (SB 1803R1). Unscheduled hot section inspections are performed in response to problems associated with starting, operation or performance.

The hot section inspection interval is reduced in the maintenance system specified in AD/ENG/5. Other maintenance systems may allow hot stage inspections to be triggered by the results of engine performance monitoring.

The hot section inspection is a detailed examination of the combustion chamber components, the compressor turbine components and the power turbine components. The inspection of compressor turbine components includes an examination of the clearance between the blades and shroud, and examination for evidence of blade tip rub, sulphidation, erosion, impact damage, coating loss, cracks and movement in the blade to disc connection. The inspection of power turbine components includes an examination for evidence of impact damage, erosion, corrosion and cracks in the turbine blades. The trailing edge of the power turbine blades is identified as a critical location for the development of cracks. Once cracks in the power turbine blades are detected they must be removed from service.

Borescope inspection

Borescope inspection allows the compressor turbine blades and nozzle guide vanes in addition to parts of the combustion chamber to be inspected while the engine is still installed in the aircraft. Detailed optical inspection is directed to: the compressor turbine blade roots, the compressor turbine blade tips, the leading and trailing edges of the compressor turbine nozzle guide vanes, and the combustion chamber liner cooling ring.

The power turbine stage is not examined during the standard PT6 borescope inspection.

Preventative maintenance actions

There are a number of actions designed to prevent deterioration of components and engine performance. Compressor washing using a specified solution is designed to remove surface contaminants from compressor blades and vanes. Washes are carried out on a regular basis with their frequency being determined by the nature of the operating environment. In addition to compressor washing, the turbine stage may be rinsed to remove corrosive residues from compressor washing and to remove sulphate deposits. The cleaning and checking of fuel nozzles is aimed at preventing the development of irregularities in the combustion process.

Checks of oil filters and metal chip detectors in the engine lubricating system provide an early warning of deterioration in engine bearings.

Maintenance system applied to the left engine, VH-LQH

The left engine was being maintained under the maintenance system specified in AD/ENG/5.

The system of maintenance requires compliance with the life limitations specified by the engine manufacturer for various components (rotors, discs, gears and power turbine blades).

The system of maintenance also requires:

- the engine to be overhauled at intervals of 5,000 hours
- hot section inspections to be performed at intervals of 1,250 hours (or 1750 hours if borescope inspections are conducted at intervals of 610 hours)
- fuel nozzles to be cleaned and flow checked at intervals not to exceed 460 hours
- a check of the oil filter for contamination, a check of the reduction gearbox chip detector, an inspection of the 1st stage compressor for foreign object damage are conducted at intervals not to exceed 220 hours

- a power recovery wash to be performed in response to trend monitoring deviations or at intervals not to exceed 220 hours.

Recorded maintenance actions, left engine VH-LQH

Rotating component lives (discs, rotors) TTIS, unknown cycles

PT blade TTIS, unknown hours

Propeller reduction gears TTIS, unknown hours

Engine main bearings TTIS, unknown hours

Engine total time in service 6,831 hours

Time since overhaul 3,556 hours

Time since hot stage inspection 1,567.1

Time since borescope inspection 402.5 hours

Time since oil filter and metallic chip detector inspection, power recovery wash AD/ENG/5 196.5 hours (7 June 2006).

Characteristics of failure modes

Creep

Creep is a process of continued plastic deformation (change in component dimensions) which occurs at high temperatures as a result of the stresses developed in a component during operation and sustained over a period of time. The rate of deformation (change in dimensions) is a function of temperature, stress and time. For a constant stress, the operational time for a particular change in dimension is decreased with increasing component temperature. For a constant temperature, the time for a particular change in dimension is decreased with increased sustained stress.

If the process of creep is allowed to continue unchecked, deformation becomes localised and the component ruptures (stress rupture) - plastic deformation is localised, and the combination of blade stress distribution and temperature distribution results in necking around the midspan region. Extensive microcracking is visible on the surface in this region. Microstructural examination would reveal voids adjacent to fracture.

Compressor blade life is limited by creep deformation to a specified increase in blade length. It is not based on creep deformation to rupture.

Indicators of creep in the compressor turbine blades of PT6 engines are the loss of material at the trailing edge of blade tips, cracks in the trailing edge, creep demarcation line, necking at the trailing edge, and blade stretch measurements that exceed the limit specified in the maintenance documentation. (P&WC FN #968, PT6A-41)

Because all blades from a compressor turbine are manufactured from the same alloy, are stressed equally, and exposed to the same hot gas stream, there should be little variation in the extent of the creep process from blade to blade in a turbine installed in an engine.

Creep can affect turbine nozzle guide vanes, resulting in distortion and cracking. Damaged nozzle guide vanes can be a cause of resonance and subsequent cracking in turbine blades through the effect of nozzle shape and dimension on the gas velocity through the nozzle.

Fatigue

Fatigue is a process of crack initiation and growth in response to the creation of alternating stresses during operation. There are three distinct modes of alternating stress states that may be encountered in gas turbine operation. Alternating stresses developed in rotating components through the change in rotational speed during engine operation (in particular the engine start/stop cycle). In turbine engine terminology this mode of fatigue cracking is known low cycle fatigue (LCF).

Low cycle fatigue cracks initiate in regions where the stresses developed through rotation are highest, eg the rims of rotors and turbine discs, and blade to disc connections. Crack propagation occurs on a plane perpendicular to the direction of maximum stress.

Low cycle fatigue failure is prevented by retiring parts after a specified number of loading cycles (engine start/stop cycles).

Alternating stresses may be created in turbine blades through the excitation of a resonant state through variations in gas impulse loads. These variations may be created by two abnormal conditions: the creation of a non-uniform temperature and pressure distribution across the face of the turbine through abnormalities in the combustion process, and the creation of gas velocity differences through abnormalities in the shape of individual turbine nozzles. In turbine engine terminology this mode of fatigue cracking is known as high cycle fatigue (HCF).

The effect of the creation of structural resonance is to create bending stresses of high magnitude. The excitation of resonance in compressor turbine blades results in a mode 1 bending response. In this mode the nodal point is at the base of the blade (transition from the airfoil section to the blade platform). High cycle fatigue cracking develops at the location of the flexural node.

Some turbine blades incorporate integral tip shrouds that provide a stiffening ring around the blade tip of a turbine assembly. The presence of a blade tip stiffening ring provides a resistance to airfoil section bending through gas loads or resonance.

High cycle fatigue is prevented by maintenance directed at maintaining uniformity of the combustion process and uniformity of the shape of turbine nozzles.

Alternating stresses may be created when changes in gas temperature result in regions of a component (turbine blade/vane or combustion chamber) being heated or cooled at different rates to other regions of the component. In turbine terminology this mode of fatigue cracking is known as thermal fatigue.

Thermal fatigue cracking occurs in regions of thinnest cross-section in regions where the greatest thermal strains occur. In the case of turbine blades, a rapid reduction in gas temperature results in the thinner trailing-edge of the blade cooling more rapidly than the thicker middle and leading-edge sections. The difference in cooling results in a difference in the extent of thermal contraction between the trailing edge and the remainder of the blade and creation of tensile stresses in the trailing edge.

Fatigue cracking always propagates on a plane perpendicular to the local maximum stress. In the case of thermal fatigue, stress is no longer a function of predictable mechanical loading, but is a function of thermal gradients and local distortions in blade shape (warpage). Thermal fatigue cracking is likely to follow irregular crack paths.

Thermal fatigue failure is prevented by retiring a component after its exposure to a specified number of thermal cycles (engine start/stop cycles) and limiting the severity of thermal strains through operational techniques, such as, slow power changes.

Blade tip wear

The tips of turbine blades are subjected to wear when contact is made with the turbine shroud while the turbine is rotating. Rubbing contact leads to adhesive wear and transfer of material from tip to shroud. Under non-steady state operating conditions, such as takeoff and climb, blades and shrouds are heated at different rates. Thermal expansion of a blade without an equivalent amount of thermal expansion in the shroud may result in rubbing contact between the blade tip and shroud.

Erosion

Turbine blades are exposed to a high velocity hot gas stream. Any fine solid material (dust, grit, carbon) entrained in the gas stream will impact the blades and erode the surfaces. Dust and fine grit may be ingested into the engine from the surrounding environment. Particles of carbon may be formed during combustion.

The erosion resistance of most turbine alloys is inversely proportional to temperature (the higher the temperature the greater the rate of erosion). (*Aircraft Gas Turbine Guide*, General Electric Aircraft Engine Group pp 1-28)

Sulphidation

The presence of chemical species, such as sodium, sulphur, vanadium and chlorides in the hot gases that drive turbine blades can react with the material from which the blades are manufactured. (Hot corrosion in gas turbines, *Engineering Failure Analysis*, vol 9 (2002) pp 31-43)

There are two forms of hot corrosion. At higher temperatures the condensation of alkali metal salts (sodium sulphate) can attack the oxide surface film that protects the underlying alloy. The attack proceeds through the creation of a porous oxide scale and oxidation of the base alloy. There are four distinct stages to the process. In the final stage, the loss of material may cause an imbalanced turbine, or the loss of blade section may be sufficient to result in fracture under the normal operating stresses.

Attack may also occur at a lower temperature. In this case the attack is characterised by pitting of the base alloy through the action of deposited sodium sulphate.

Because the compressor turbine blades are exposed to higher temperatures than the power turbine blades, they are affected by the higher temperature, blistering and scaling, mechanism. Power turbine blades are affected by the lower temperature, pitting, mechanism.

Hot corrosion may be minimised by removal of surface deposits from blades through turbine washing. The progress of sulphidation may be monitored by visual inspection.

Foreign object damage

Occasionally, larger solid particles/objects are ingested into the engine intake or small pieces of the engine structure are liberated and entrained in the gas stream. The collision of these particles of significant mass can cause mechanical damage to turbine blades. Mechanical damage can affect the structural integrity of component directly and it can affect other components through a disruption in the gas flow.

Blade tip/shroud interference

Deformation of the engine structure during abnormal operation or ground impact may result in interference between turbine blade tips and shrouds. If the interference force is high enough, blade bending may result in blade fracture. Blade expansion created by the rapid heating of turbine blades may also create high interference forces between the blade tips and shroud.

Overtemperature exposure

Turbine temperature limits are specified for each engine model for each phase of flight, these limits are specified in the aircraft flight manual, see for example Appendix A. When temperature limitations are exceeded, a graduated series of actions is detailed in the engine maintenance manual. The nature of the actions depends on the maximum temperature reached and the duration at this temperature. The first level action is: determine and correct the cause of overtemperature, inspect the trailing edges of the power turbine blades (by looking through the exhaust port), record in engine log book. The second level of action is to perform a hot section inspection. The third level of action is to perform a hot section inspection and return the compressor turbine disc and blade assembly to an overhaul facility for blade stretch check and disc fluorescent penetrant inspection. Finally, if the temperature exceeds 850 ˚C for more than 2 seconds the engine must be returned to an overhaul facility, the compressor turbine and power turbine blades must be discarded, and both turbine discs must be measured for stretch and crack checked by fluorescent penetrant inspection. (P&WC Maintenance manual p/n 3015442, *Overtemperature limits, all conditions except starting,* figure 2-6-3, page 2-6-10)

The effects of exposure to temperature above operating limits range include an increased rate of material strength change (precipitate coarsening) with an accompanying effect on creep rate. Short higher temperature excursions can cause warpage of blades and buckling of sheet metal components.

Hot stage inspection notes (P&WC maintenance manual p/n 3017042, *Power turbine blades – inspection* page 658) alert the inspector to check the trailing edge of power turbine blades for 'rippling'. The trailing edge of a turbine blade is placed under compression as the trailing edge heats faster and expands against cooler thicker sections of the blade. Buckling occurs if the yield strength, at the temperature of the trailing edge, is exceeded.

Very high temperatures can cause a dramatic change in precipitation hardened nickel superalloys through the re-solution of the precipitates. In addition, the mechanical properties of the alloy are reduced so that yielding and tensile fracture may occur under the stresses of normal operation. For example, alloy IN100 at 760˚C (1400˚F), tensile strength 10,70MPa, yield strength 860 MPa, at 1090˚C (2,000˚F) tensile strength 380 MPa, yield strength 240 MPa. (*ASM Metals Handbook*, vol 3, 9th ed. p 243). An increase in temperature from 760˚C to 1,090˚C results in a reduction in tensile and yield strength by over 60%.

Manufacturing flaws

It is possible, though rare, that turbine blades may contain manufacturing flaws. These flaws may act as initiators for fatigue cracking and reduce the time to fatigue crack propagation. If manufacturing flaws were present, crack growth and fracture would be expected to occur relatively early in the life of the component.

Engine performance prior to the accident - engine condition monitoring

A gas turbine engine operating cycle consists of the continuous combustion of a mixture of compressed air and fuel, and the controlled expansion of the hot high-pressure combustion gases to create a stream of high velocity gas directed at the blades of turbine wheels (a series of blades fitted to the rim of a disc).

One stage in the turbine section (compressor turbine) drives the compressor itself. The remaining stage in the turbine section drives the load, in this case a propeller.

The efficiency of the engine is determined by how hard the engine must work to produce an amount of torque at the propeller drive shaft.

The efficiency of conversion of energy from the combustion of fuel and air to a drive shaft torque deteriorates with continued operation. The deterioration is a result of changes in the gas path (increased clearances, changes in surfaces, changes in combustion changes in functionality of components). Deterioration may be reversible or non-reversible (reversible – the removal of dirt and surface deposits from compressor blades, non reversible – turbine blade/vane wear, erosion, distortion and cracking).

P&WC engine condition trend monitoring

The P&WC engine condition trend monitoring (ECTM) is based on a thermodynamic model of the energy conversion process for particular engines. The PT6 gas generator parameters of inter-turbine temperature (ITT), gas generator (compressor turbine and compressor) speed (Ng) and fuel flow (Wf) are predictable for a particular load (torque produced at the propeller drive shaft by the power turbine) under known atmospheric conditions. Changes in the gas generator parameters, with continued engine operation, occur with changes or deterioration in the components that form the gas path in the engine. The recording of gas generator parameters and the examination of trends in the changes in these parameters provides a basis for maintenance actions to restore engine performance or correct component deterioration before more extensive damage occurs.

The nature of performance recovery maintenance action is based on a judgement as to whether the trend deviation is normal or not (P&WC Maintenance manual p/n 3015442, *Engine Performance Fault Isolation Chart* Figure 2-7-3). If the deviations occur over several thousand hours of engine operation (P&WC ECTM ver. 5.07 manual p104) they are considered normal and preventative maintenance actions, such as a power recovery wash, fuel nozzle cleaning, power check and, possibly, borescope examination, are performed. Abrupt changes or an increase in the rate of change over an operational period of several hundred hours are considered to be an indication of compressor and/or turbine component deterioration. Maintenance actions involve a structured approach to the elimination of instrumentation errors, compressor air bleed functions, air inlet partial blockages, compressor foreign object damage, compressor dirt buildup, and engine overtemperature during starting, before undertaking a hot section inspection.

Because no vibration monitoring is included in the P&WC ECTM there is no indication of out of balance conditions that may be developing in the engine rotating components. For example, crack growth in a rotating component may not be detected by ECTM. In addition, it has been reported that sulphidation attack on turbine blades cannot be detected by the P&WC ECTM process (P&WC ECTM manual, p 105).

The trend data for the left engine VH-LQH, 'all parameters up' (Interstage turbine temperature, ITT up, fuel flow, Wf up, compressor speed, Ng up), indicates that there may have been a fault in the compressor stage of the gas generator section or the power turbine section (PW&C maintenance manual p/n 3015442, Table 2-7-1 *ECTM Shift Fault Isolation*).

Physical evidence

Compressor turbine

Each compressor turbine blade had fractured during the engine failure sequence. Only the blade sections still attached to the turbine disc were recovered. It is evident that blade fracture occurred at varying locations along the blade span, the angle of the fracture plane varied with respect to the blade axis (it was not consistently perpendicular to the blade axis) and the macroscopic fracture surface features were dendritic in nature.

Dendritic fracture surface features in cast nickel based superalloys are created when a component is fractured by a single overstress condition. The preferred fracture path reflects the underlying dendritic nature of the microstructure created during casting.

Figure 1: Compressor turbine, upstream side, left engine VH-LQH

Blade failure by the mechanisms of creep, high cycle fatigue (HCF), and sulphidation can be eliminated on the basis of fracture location and physical features not present at the fracture locations.

Creep failure (stress rupture) occurs at a blade location that coincides with maximum temperature and increasing mechanical stress – typically around mid span. Failure is preceded by localised plastic deformation (necking) and surface microcracking. If all blades are manufactured from the same alloy, in the same heat treated condition, then all blades will show evidence of the final stages of creep preceding failure. There are sufficient blade remnants that allow the midspan region to be examined.

Figure 2: Composite photograph - compressor turbine, left engine VH-LQH, superimposed over an intact compressor turbine

The known location for high cycle fatigue in turbine blades with no tip shrouds is at the base of the blade. No evidence of fracture surface features consistent with any stage of crack growth by the mechanism of fatigue was present.

No evidence of gross loss of blade cross-section through sulphidation was present.

The location of blade fracture, angle of fracture plane to the blade axis, and fracture surface features indicates that the blades were fractured as a result of a varying blade tip/shroud interference force.

Power turbine

Only about one quarter of the power turbine blades were recovered from the accident site. Each blade had fractured and exhibited extensive leading edge wear through rotational contact with the displaced interstage baffle.

Figure 3: Power turbine, left engine VH-LQH

Despite the considerable amount of damage created by continued rotation after the initiating event one distinctive fracture feature was present in a number of blades. It was evident that crack growth had occurred from the blade trailing edge, around the midspan location, by a mechanism that created a ‘faceted’, ‘crystallographic’ fracture surface. This fracture surface feature contrasted with the dendritic features of all other fractures in the power turbine blades.

There are consistent references in the literature on crack growth and fracture of cast nickel-based superalloys to the formation of faceted, crystallographic crack surface features as a result of fatigue crack growth under conditions of low alternating stress intensity.

‘faceted nature caused by crack growth on specific crystallographic planes’. ASM Metals Handbook ‘Fractography’, vol 12, ninth ed.., fig. 846, p393

‘faceted fracture surface crack growth at low alternating stress intensity’ - nickel super alloy fatigue. ‘Application of Fracture Mechanics for Selection of Metallic Structural Materials’, Campbell, Gerberich, Underwood, American Society for Metals, fig 8.19, p281

Fracture surfaces of Inconel 718 ‘cleavage like faceted growth behaviour in low alternating stress intensity regime’ – nickel superalloy fatigue. ‘An Atlas of Metal Damage’, L Engel, H Klingele, Wolf Science Books, London 1981, pp 82 – 83, (see Appendix B).

The evidence of crack growth extending from the trailing edge of several power turbine blades by a mechanism that creates a ‘faceted’, ‘crystallographic’ surface, and a crack path that is not perpendicular to the blade axis indicates that crack growth has occurred by the mechanism of thermal fatigue (see Appendices C and D).

Out of balance rotation

The loss of a small fragment from a power turbine blade will result in vibration (small out of balance). However, the loss of several blades will result in a large out of balance force and significant lateral displacement.

Because the structure housing the turbine shrouds and nozzle guide vane outer rings is continuous and not rigidly supported (see Appendix E), it is possible that lateral movement of the power turbine shroud housing created through the out of balance rotation of the power turbine may be transmitted to the compressor turbine shroud. Lateral movement of the compressor turbine shroud can lead to compressor turbine blade tip/shroud interference. Because of the very small clearance between the compressor turbine blades and shroud, small movements of the shroud would be sufficient to create blade tip/shroud interference and, possibly, blade fracture.

Power turbine blades are designed with a tip shroud. The purpose of the shroud is to create a stiffening ring to support the blades in a power turbine assembly. In the case of the fracture of one blade, the integrity of the shroud ring is compromised and blades adjacent to the fractured blade will be subjected to greater bending forces and a greater risk of failure under high load conditions, such as takeoff.

Summary

The lives of rotating parts in the PT6 engine are limited to prevent fatigue failure. The specified retirement lives apply to all maintenance systems that may be applied to PT6 engines. The total times in service of the rotating components fitted to the left engine VH-LQH are not known.

Both compressor turbine and power turbine blades are rotating components. However, compressor turbine blades are limited, in practice, by creep to a specified strain limit (blade elongation). Power turbine blades are not limited by creep and are retired after 12,000 hours inservice.

The deviations of the parameters monitored by the engine condition monitoring program, interstage turbine temperature, gas generator speed, and fuel flow were all increasing from the established baseline. An increasing deviation in the three parameters is an indicator of problems in the compressor section of the gas generator, as distinct from problems in the compressor turbine section (indicated by increased temperature and fuel flow, but decreased compressor speed). However, an increasing deviation in the three parameters is not an exclusive indicator of compressor problems. Inefficiencies in the conversion of energy from the hot gas stream to propeller shaft torque by the power turbine may also result in increasing deviations in all three parameters (PW&C maintenance manual p/n 3015442, Table 2-7-1 *ECTM Shift Fault Isolation*).

Physical evidence of thermal fatigue cracking is present on several power turbine blades. Crack growth from the trailing edge of power turbine blades is a known mode of blade deterioration. The maintenance actions of power turbine blade life limitation, and hot stage inspection are designed to prevent power turbine blade cracking affecting the integrity of the engine. Borescope inspection is not directed at the power turbine section of the engine. It will not detect cracking in power turbine blades.

It is possible that thermal fatigue cracking may result in a change in form of the trailing edge or the loss of small fragments from the blade trailing edge. These changes may create a change in the gas flow path of sufficient magnitude to cause a deviation in engine performance parameters.

The fracture of several blades in one section of the power turbine disc will result in the lateral movement of the turbine assembly during rotation through the loss of assembly rotational balance. If this lateral movement is transmitted to the power turbine shroud housing it is possible that consequential lateral movement of the compressor turbine shroud could lead to compressor turbine blade/shroud interference and compressor turbine blade fracture.

Creep, fatigue (high cycle) and sulphidation have been eliminated as mechanisms for compressor turbine failure.

Appendix A: An example of maintenance actions following an overtemperature Event

PRATT & WHITNEY CANADA
MAINTENANCE MANUAL
MANUAL PART NO. 3015442

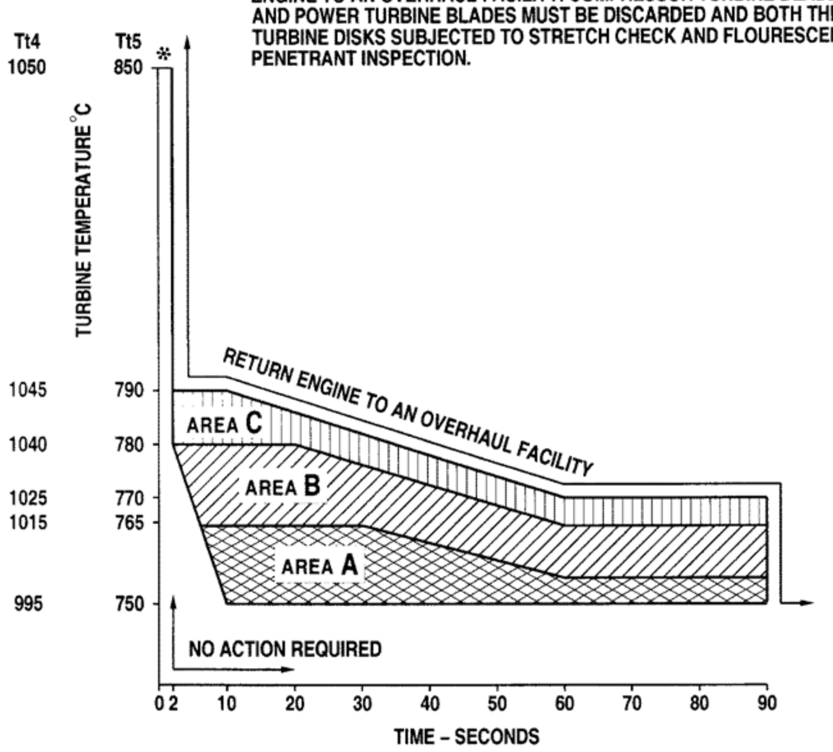
ACTION

AREA **A**
1. DETERMINE AND CORRECT CAUSE OF OVERTEMPERATURE.
2. VISUAL INSPECTION THROUGH EXHAUST PORTS OF POWER TURBINE BLADES AND EXHAUST DUCT TURNING VANES (IF FITTED).
3. RECORD IN ENGINE LOG BOOK.

AREA **B**
1. PERFORM HOT SECTION INSPECTION.

AREA **C**
1. PERFORM HOT SECTION INSPECTION.
2. RETURN COMPRESSOR TURBINE DISK AND BLADE ASSEMBLY TO AN OVERHAUL FACILITY FOR STRETCH CHECK AND FLOURESCENT PENETRANT INSPECTION.

* IF TEMPERATURE EXCEEDS THIS VALUE FOR 2 SECONDS RETURN ENGINE TO AN OVERHAUL FACILITY. COMPRESSOR TURBINE BLADES AND POWER TURBINE BLADES MUST BE DISCARDED AND BOTH THE TURBINE DISKS SUBJECTED TO STRETCH CHECK AND FLOURESCENT PENETRANT INSPECTION.



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Overtemperature Limits (All Conditions except Starting)
Figure 2-6-3

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Appendix B: Nickel based superalloy fatigue fracture surface features from 'An Atlas Of Metal Damage', L Engel, H Klingele, Wolf Science Books, London 1981, pp 82 – 83

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111. The fatigue fracture of a nickel alloy turbine blade commenced on the obliquely inclined flat surfaces (Stage I) and subsequently propagated as a transcrystalline fatigue crack (Stage II) (shown at the left-hand side of the picture). (260:1)

112. Hot gas corrosion and fatigue fracture in a nickel-based alloy turbine blade (Nimonic 105). The hot gases attacked the whole surface, but particularly the grain boundaries. This led to the formation of a fatigue crack whose initiation region showed the cleavage-like facets which typically form in nickel-based alloys (Stage I propagation). The continued transcrystalline fracture propagation is no longer crystallographically oriented. (120:1)

113. A fatigue fracture which occurred in a nickel-based alloy operating at high temperature: Stage I of fatigue fracture. (280:1)

114. Cleavage-like, crystallographically oriented fatigue fracture in a laboratory tested specimen of a nickel-based casting alloy (Nimocast 713V). The test was carried out at high temperatures under conditions of pulsating tensile fatigue testing (0 to 0.2kN/mm²) Stage I of fatigue fracture, showing the typical fracture surface found in nickel-based alloys. (525:1)

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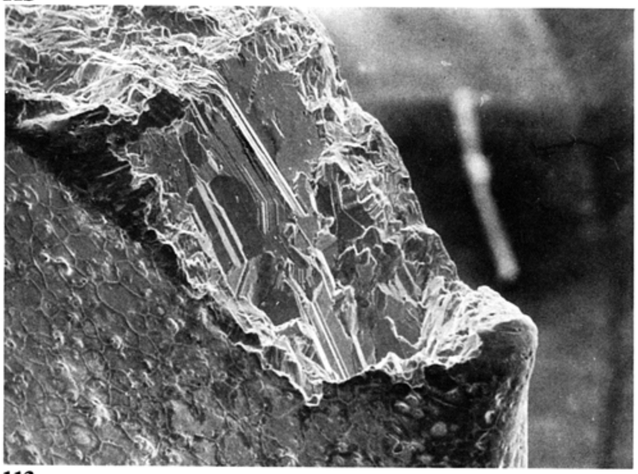

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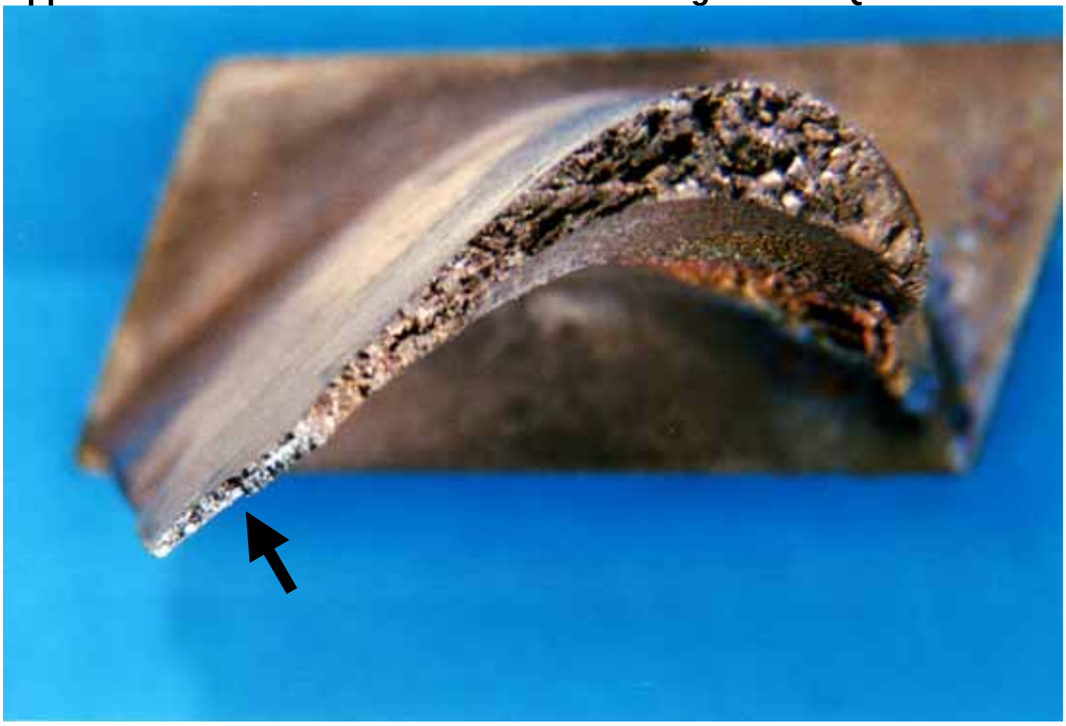
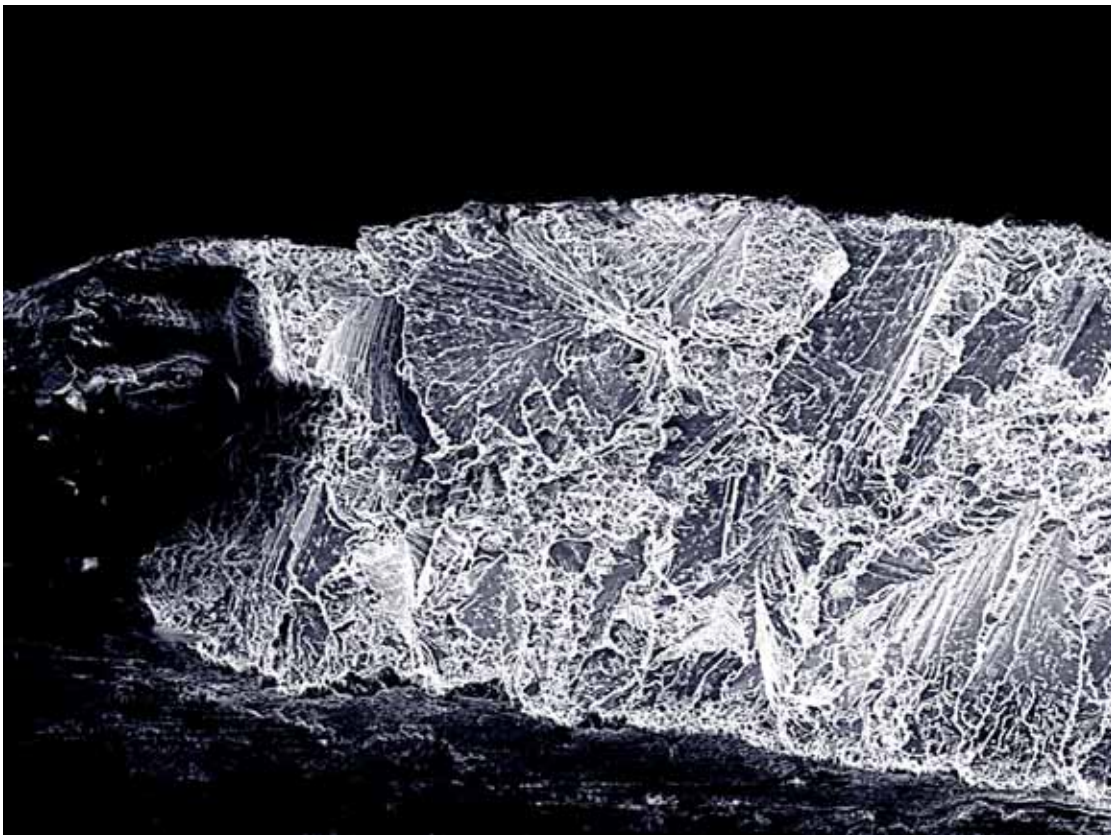
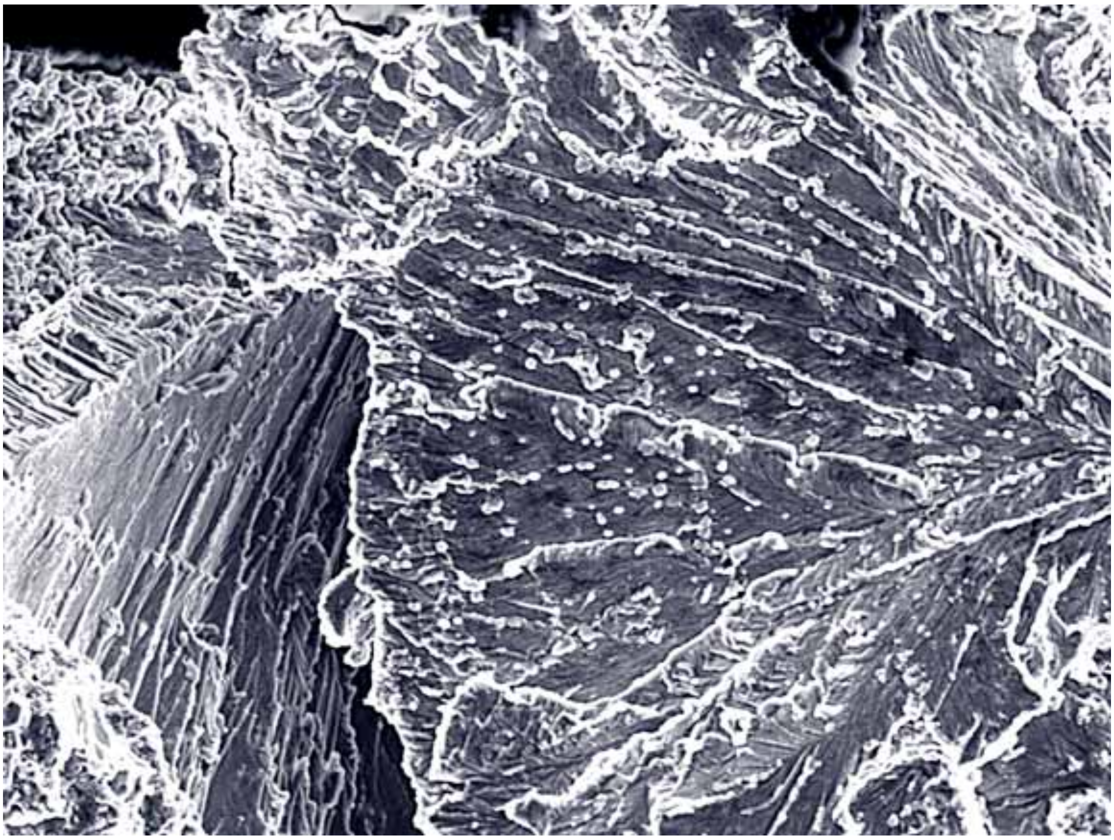
Appendix C: Power turbine blade from left engine VH-LQH



Region of crack growth with faceted 'faceted' features, arrowed (delineated by the blue heat tinting)

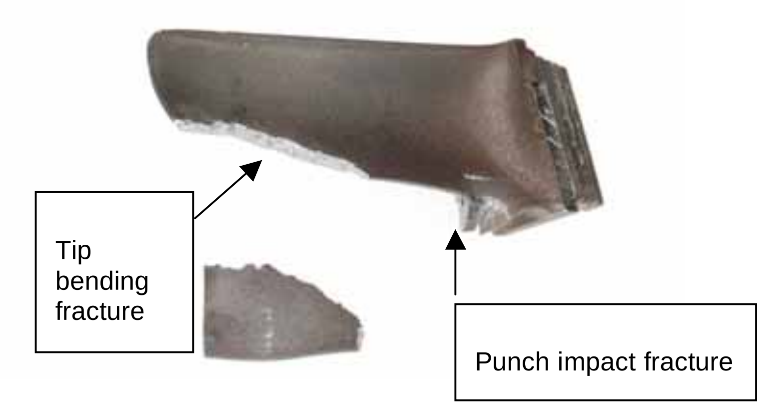


Scanning electron micrograph of the trailing edge crack, magnification 200X

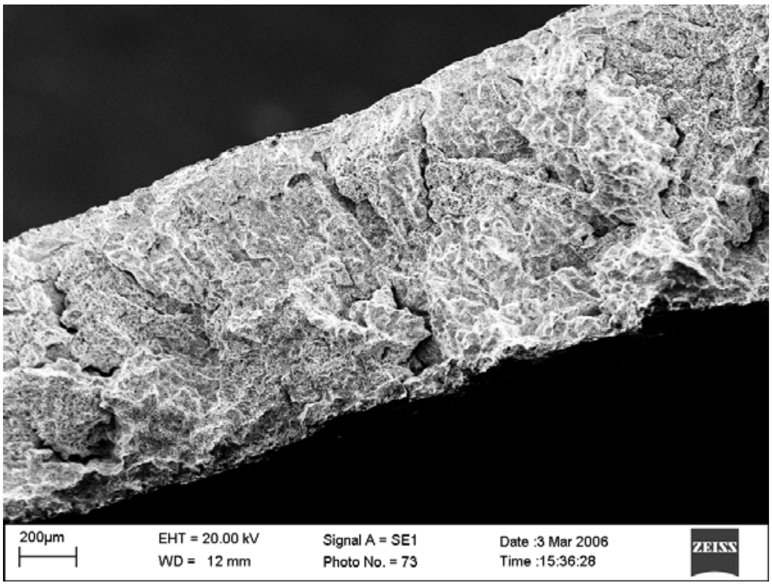


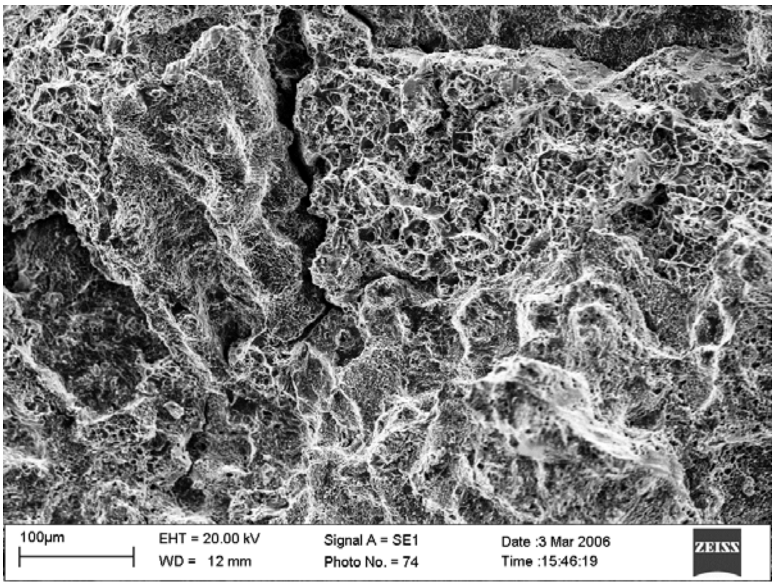
Scanning electron micrograph at higher magnification of trailing edge crack, magnification 350X

Appendix D: Fracture surface features created by applying an excessive stress in the laboratory for comparison

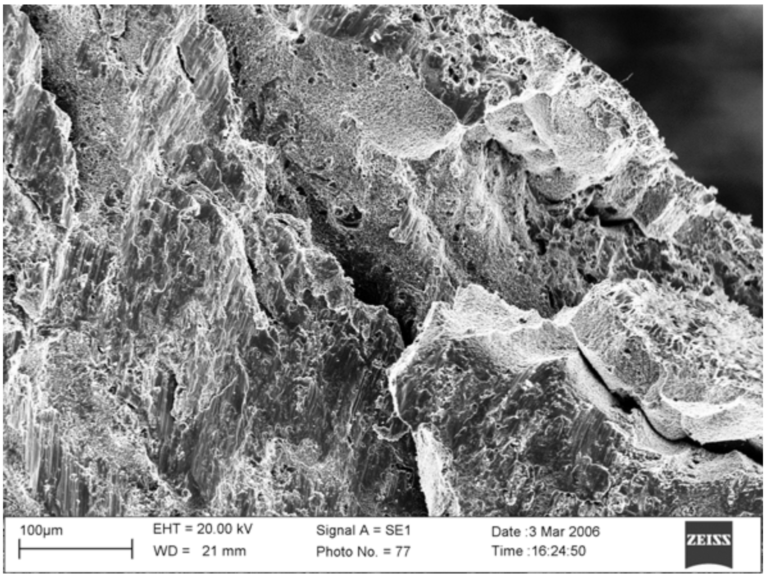


Tip bending fracture features



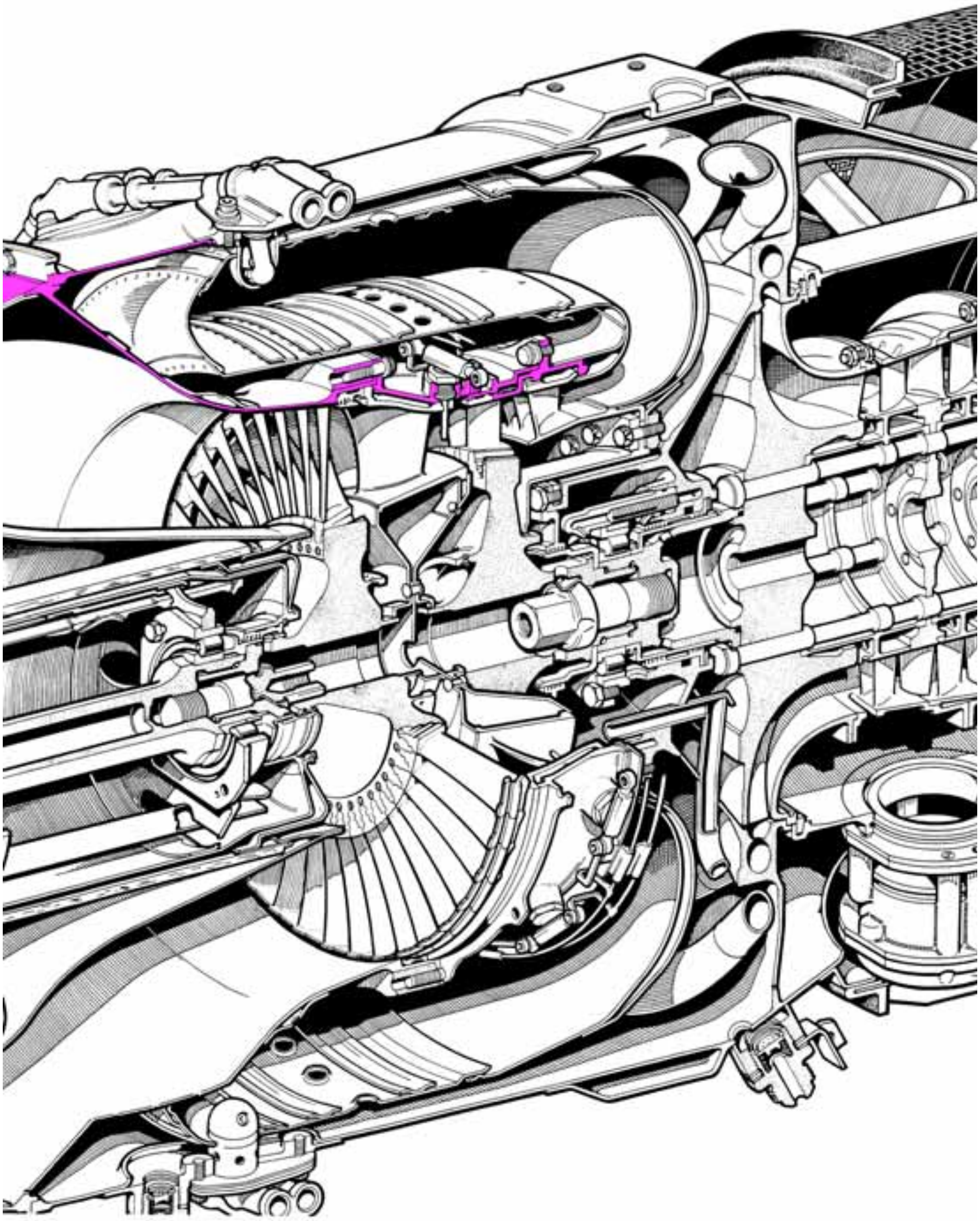


Punch impact fracture

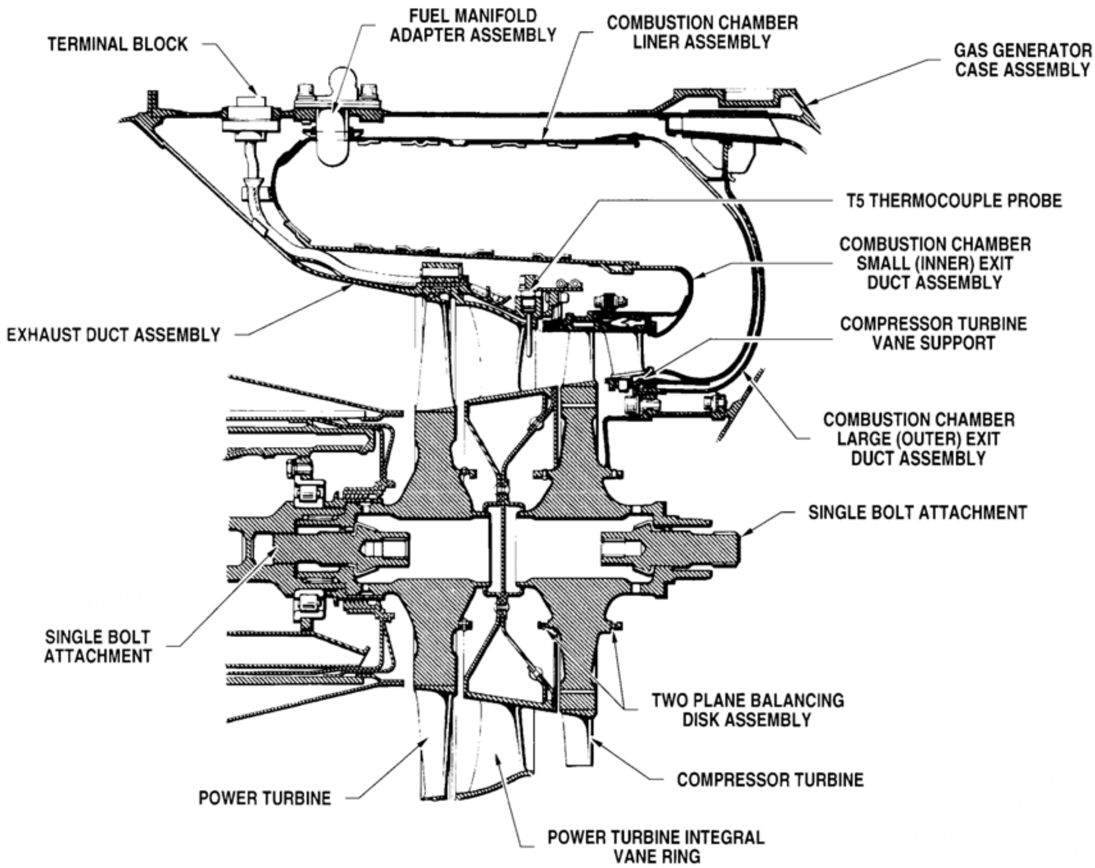


Appendix E: Cross-sectional diagrams of single power turbine stage PT6 engines

Cross-section of turbine section showing the structure of the turbine shroud housings and nozzle guide vane outer rings



Cutaway diagram of a PW&C PT6-27 engine



P&WC Maintenance Manual part no. 3015442, figure 1-4-6, page 1-4-13/14, Feb 01/2002

Attachment B: Transportation Safety Board of Canada supplementary report

Transportation Safety Board of Canada | Bureau de la sécurité des transports du Canada

Alan L. Stray
Deputy Director
Aviation Safety Investigation
Australian Transport Safety Bureau
P.O. Box 967
Civic Square ACT 2608
Australia

Our file Notre référence

A01F0138

January 20, 2006

Re: VH-LQH engines, A01F0138, LP101/02

With reference to your e-mail of December 9, 2005, additional work has been carried out to address your concerns. I will outline what has been done and the results in the same order.

1. Fatigue cracks in power turbine blades

All available power turbine blades were re-examined with particular attention being paid to trailing edge cracks. No trailing edge cracks other than caused by impact damage with flying debris were found. One of the blades with a trailing edge crack undoubtedly caused by impact (Figures 1 and 2) was examined in the scanning electron microscope (SEM). This examination revealed fracture consistent with overstress. The fracture topography at the trailing edge (Figures 3 and 4) was comparable to that described in one of the attachments to your correspondence. It is therefore concluded that the trailing edge crack was a result of overstress situation rather than fatigue propagating under low alternating stress magnitude. The same blade described in the attachment was located and re-examined in the SEM. In our opinion, the fractographic features do not reflect a fatigue mode of crack propagation.

2. Fracture of the power turbine shaft housing

The housing had separated just outside a circumferential weld. The greater portion of the fracture exhibited a slant orientation consistent with crushing (Figure 5). The remainder had a columnar appearance which is believed to reflect the proximity of the weld (Figure 6). There was no evidence of any progressive failure mode.

3. Fracture of power turbine housing attachment studs

Seven out of twelve studs had been cut off with an abrasive disc cutter to facilitate disassembly during the engine teardown at Brisbane, Australia, (Figures 7 and 8). No studs were fractured.

Canada

4. <u>Excessive or above normal operating temperatures</u>

The following paragraph refers to Figures 6 and 7 that appear in the Pratt & Whitney Canada Report No. 01-119. Comparison of microstructures obtained from a section through a representative compressor turbine blade with those of reference microstructures for the applicable alloy indicated that the blade location shown in Figure 6 had reached 2050°F. The location represented by Figure 7 had been exposed to 2100°F. It is entirely possible that the temperature excursion to modify the gamma prime morphology occurred during the failure event. In fact, a loss of efficiency due to compressor blade loss frequently leads to an overheat condition.

In conclusion, the physical evidence (e. g. pronounced leading edge damage to power turbine blades) is consistent with compressor turbine blade failure being the lead event in the left engine power loss. This position was originally stated in conclusion 4.1 of our Engineering Report LP 101/02.

Figure 1 Power turbine blade with trailing edge crack.

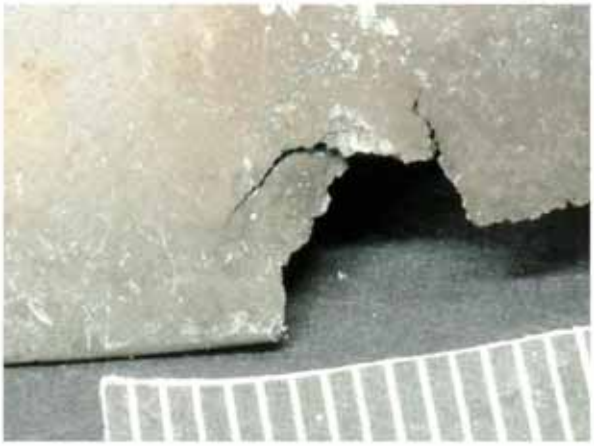


Figure 2 Close-up of the trailing edge crack.

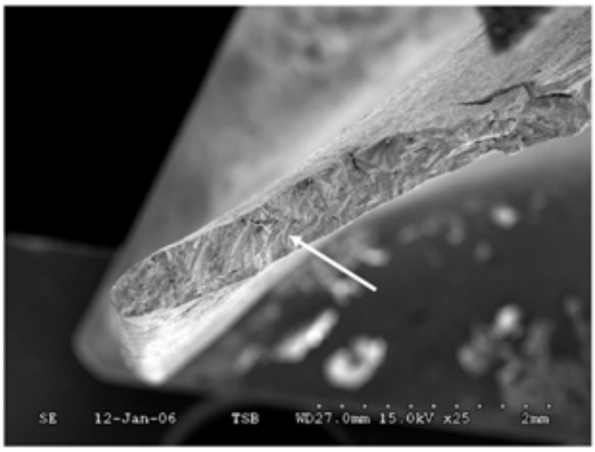


Figure 3 SEM image of the trailing edge fracture. Arrow identifies the area shown at higher magnification in Figure 4.

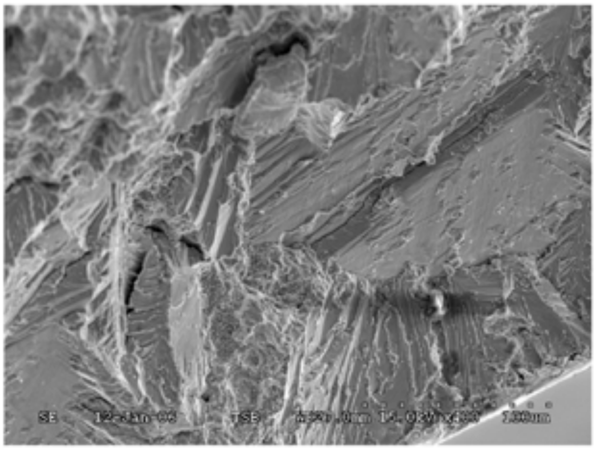


Figure 4 Area identified by arrow in Figure 3.

Figure 5 Separation of the power turbine shaft housing.

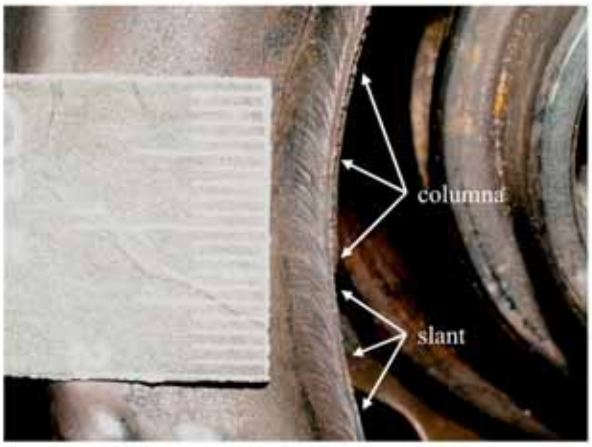


Figure 6 Boundary between columnar and slant topography.

Figure 7 Attachment studs cut off with an abrasive cutter.

Figure 8 Close up of one of the studs.