Transportation Safety Board of Canada

Bureau de la sécurité des transports du Canada

AVIATION INVESTIGATION REPORT
A04P0142

IN-FLIGHT POWER LOSS

VANCOUVER ISLAND HELICOPTERS LTD.
BELL 206L C-FVIX (HELICOPTER)
TASU CREEK, BRITISH COLUMBIA
28 APRIL 2004

Canada

Aviation Investigation Report

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Summary

The Bell 206L helicopter (registration C-FVIX, serial number 45139) was in cruise flight at an altitude of about 700 feet above sea level when the pilot heard a sudden unusual noise and subsequently experienced an engine power loss. He lowered the collective and checked the instruments while scanning the area for a landing spot. The engine was still running; however, the turbine outlet temperature was climbing very rapidly and quickly exceeded the range of the gauge.

The pilot subsequently raised the collective slowly, but the main rotor started to droop. He advised the two passengers of an engine failure and entered auto-rotation. While initiating a flared landing, he pulled the collective and confirmed no power from the engine as the low rotor horn sounded. The helicopter landed on a logging road near Tasu Creek, Queen Charlotte Islands, in the Sandspit area at 0829 Pacific daylight time. The pilot shut down the engine immediately on landing. There were no injuries and no airframe damage.

Ce rapport est également disponible en français.

Other Factual Information

The helicopter was equipped with a Rolls-Royce Allison 250-C20R engine, serial number CAE 295208. At the site, a visual inspection revealed a large crack, about four inches long, in the lower weld joint of the ignitor and fuel nozzle mount at the back of the combustion section. The left-hand exhaust duct was also cracked at the weld on the forward side of the outer combustion case, about six inches down from the top. The duct also had impact damage which appeared to have originated from inside the engine. There was no visual damage from the substantial turbine outlet temperature (TOT)-indicated over-temperature that preceded the autorotation. There was a grinding noise heard from the engine when the blades were rotated backwards.

Fuel and oil samples were retained; no obvious anomalies were noted. The lower chip detector was clean, and the oil did not contain obvious metallic particles. The upper chip detector and the freewheel chip detector were removed, and small amounts of metal were found. The compressor showed no signs of foreign object damage, but it could not be turned by hand.

The engine had a total time since new (TTSN) of 6694.4 hours. The times in service for the modular components are provided in Appendix A, with a summary of recent engine maintenance and repair. The technical records contained an entry on 05 April 2004, at 10 817.0 airframe hours, about 45 engine hours before the occurrence, indicating an engine discrepancy related to a fuel-control unit adjustment due to hot starting. Another entry, dated 29 November 2003, indicated that the engine oil press/temp gauge was for a C20B model engine and, therefore, had incorrect markings. Replacement parts were ordered, but there was no subsequent entry indicating that remedial action had been taken. Another entry, on 30 September 2003 at 10 550 airframe hours, indicated that the TOT harness was replaced due to a reading of 0.03 ohms below minimum; the TOT gauge was also replaced due to it reading 15°C low.

Following an engine teardown, several component parts were forwarded to the Transportation Safety Board Engineering Branch Laboratory for further examination and analysis. It was determined that the initiating event that led to the engine failure was the fracture of one blade on the second stage turbine wheel. The metallic debris from this fracture caused secondary engine damage as described below. For a more detailed account of the engine condition at examination, see LP 068/04. A schematic of the turbine section of the engine is shown in Appendix B.

A visual examination of the first-stage turbine wheel revealed many type A and approximately four type B cracks in the blade rim. Type A cracks are cracks in the platform, the surface between the blades. Type B cracks are those that extend over the edge of the wheel rim and onto the face of the rim, but not more than 0.060 inches. If cracking extends beyond 0.060 inches into the rim, the wheel is rejected. It is considered normal to find cracks in the rim area. No cracks are allowed in the blades. The largest type B crack observed was approximately 0.030 inches on the leading-edge rim face, and no type B cracks were observed on trailing edge rim face. Rub damage was observed on several blade tips covering an arc of approximately 180°. Likewise, rub damage was observed on the inner stage seal, covering an arc of 180° as well. Some fretting wear was noted on the curvic coupling.

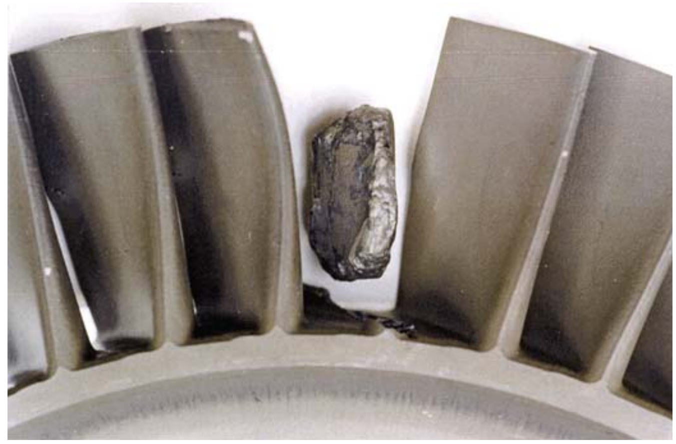
Photo 1. Remains of the failed blade

Examination of the second-stage turbine wheel revealed one failed blade (see Photo 1). Records indicate that the second-stage turbine wheel had accumulated 375.8 hours TTSN, out of an allowable service life of 1775 hours. An optical examination indicated blade failure as a result of fatigue cracking, with multiple initiation sites in the fillet area on the convex side of the blade. Fatigue cracking accounted for approximately 75 per cent of the fracture, with the remainder being an instantaneous overstress rupture. While the majority of the fracture face was normal to the blade axis, the origin area was almost parallel to the blade axis or radial to the turbine wheel. The blade following the failed blade had been bent opposite to the direction of rotation. Bending of this blade effectively reduced its overall length, preserving the pre-event tip condition. Further examination of the bent blade revealed minimal tip rub, and no blueing or mushrooming. Significant tip rub was observed on 75 per cent of the blades, from the failed blade over an arc of approximately 270° clockwise from a leading-edge view. Rubbing on these blades had produced smearing damage to the tips of the blades with blued and mushroomed material at the outer diameter on the cambered face. The balance piston seal (large labyrinth seal) also showed rub damage around approximately 270° of arc, consistent with the blade tip rub. Additionally, rub damage was noted on approximately 80 per cent, or 300° of arc, of the inner stage seal, again consistent with the blade tip rub.

Secondary smearing damage was observed on the trailing edge of several blades, consistent with interaction with the liberated piece of failed blade. The first- and second-stage wheels were mated at the curvic coupling, according to the reference marks vibro-etched on the hubs. Comparison of the blade tip rub showed both wheels were affected in the same general area, with the worst rub damage approximately 90° ahead of the failed blade location.

A scanning electron microscope examination of the second-stage turbine wheel blade fracture revealed multiple adjacent origin areas. Examination of the surface of the crack showed an oxidation layer that served to delineate a clear underlying pattern of fatigue striations. The fracture topography at the initiation site was consistent with low-cycle fatigue cracking as the initial mode of failure, with the observation of beach marks and widely spaced striations. The low-cycle fatigue cracking progressed radially inward toward the hub and then turned normal to the blade axis, progressing in a high-cycle mode. The blade eventually failed in the overstress extension of the high-cycle fatigue cracking. Secondary cracking was also observed in the fillet area, below the fracture plane. Energy dispersive x-ray analysis indicated the blade material was consistent semi-quantitatively with IN 713C, the specified material.

A transverse section was taken through the fracture origin of the second stage turbine wheel blade and mounted for metallurgical analysis. Two secondary cracks were observed in the fillet below the fracture. Chemical etching revealed a typical micro-structure with a visible gamma prime phase and well-defined grain boundaries. Additional sections taken at adjacent blade roots showed similar cracks in the fillet radius. The uniform distribution of the gamma prime phase indicated that there had been no overheating. Direct Rockwell hardness testing averaged 40 Rockwell "C" (HRC) (equivalent ultimate tensile strength of 182 Kpsi), which is within the manufacturer's specified maximum of 42 HRC. No metallurgical anomalies were observed.

Dimensional analysis of the second-stage nozzle, which had secondary damage, was completed at the manufacturer's facility in Indianapolis, Indiana. The secondary damage to the nozzle primarily affected the outer edge; therefore, the results were still considered meaningful. The analysis produced a blade path diameter that corresponds to a second-stage turbine blade tip clearance of 0.012 inches, within the specification of 0.010 to 0.016 inches. The analysis also showed that, except for the damaged areas, the second stage blade path of the nozzle was within the specification for roundness.

A blade from the first-stage turbine wheel was sectioned longitudinally to examine the gamma prime phase of the micro-structure for signs of overheating. The gamma prime phase was uniformly distributed throughout the blade, suggesting that long-term overheating did not occur.

The third-stage turbine wheel showed significant leading-edge impact damage. The trailing edge was unremarkable and the curvic coupling was clean with minimum fretting wear. Minimum wear was observed on the inner stage seal.

The power turbine support showed that both the inner and outer balance piston seals had significant rub damage around the entire circumference. The third-stage nozzle shield was deformed as a result of impact damage and could not be easily removed. The third-stage nozzle shield saddle had a major through-thickness gouge, at the nine o'clock position from a leading-edge view, consistent with impact from the failed second-stage blade.

The outer combustion case showed a four-inch gaping crack along the boss weld below the fuel nozzle and ignitor ports. The fracture exhibited features consistent with fresh, low-cycle fatigue, considered to be the result of engine vibration after the second-stage turbine wheel blade failure.

The *Rolls Royce 250–C20R Series Operation and Maintenance Manual,* Table 4—Measured Gas Temperature Limits (TOT) in Appendix C, provides actions to be taken when measured gas turbine outlet temperature (TOT) limits are exceeded during starting and shutdown and/or during power transients. About 45 engine hours before the occurrence, the operator documented that the engine fuel-control unit was adjusted because of hot starts, but the amount by which the temperature was exceeded was not recorded. A turbine special inspection is recommended when TOT limits are exceeded. Reportedly, the TOT would briefly peak in the range of 810°C to 830°C. Hot starting events are not recorded by this helicopter's instrumentation. During these "hot starts," apparently the temperature/time limits that trigger a turbine inspection were never reached, after making allowances for the 0.03 ohms (0.6°C) TOT harness discrepancy and a 15°C error from the incorrect TOT gauge (replaced at 63.1 hours).

Apparently, the ultimate temperature (927°C) for which a turbine inspection is recommended was never reached during the so-called hot starts. The TOT harness may have had a 0.03 ohms discrepancy, which would equate to 0.6°C, and the incorrect TOT gauge was replaced at 63.1 hours due to a 15°C error.

Examination of the rear support and diffuser ring revealed that approximately 25 per cent of the major diameter seal was missing. No corrosion was observed and the coloration indicated that the section of seal may have been gone for some time. Further examination showed that the seal was dis-bonding from the substrate. Dis-bonding was observed at both ends of the seal fracture, consistent with bond failure. Records indicated that the seal was installed about 700 airframe hours prior to the occurrence. The fractured pieces of the soft major diameter seal material would most likely be broken down to fine particles and expelled safely through the rear support vent. A slight loss in engine efficiency would be the expected outcome. Failure of this seal was not considered contributory to the engine failure.

Analysis

Possible fatigue initiation scenarios for the second-stage turbine wheel blade include tip rubbing, the presence of physical or metallurgical defects, high-cycle fatigue cracking due to vibration (blade flutter), and thermal fatigue cracking. The absence of any significant rubbing, blueing or mushrooming of the adjacent blade tip, combined with the acceptable dimensional analysis of the second-stage nozzle, suggests that tip rubbing was not an initiating event. Also, tip rub would initiate as high-cycle fatigue cracking normal to the blade axis. An optical examination showed that the initial fatigue cracking was in a radial direction toward the hub. Metallurgical analysis did not reveal any manufacturing or material anomalies that would have contributed to fatigue initiation. Although obscured somewhat by the presence of an oxidation layer, the fatigue striation spacing and initiation site (that is, the mid-chord as opposed to the trailing edge) would indicate that the initial mode of failure was low-cycle fatigue cracking. Once the cracking had started, its presence served as a stress raiser, so that normal service stresses could now drive the cracking in a high-cycle mode.

Thermally induced fatigue cracking occurs when rapid expansion in the rim area of the turbine wheel produces large, momentary compressive hoop stresses. Compressive stress develops when the rim tries to expand, but is retrained by the cooler hub material. This compressive stress leads to a localized yielding of the turbine wheel rim material. Subsequent steady state operational temperatures then result in tensile stresses in the rim that initiate fatigue cracking. The location and orientation of the cracks in the subject second-stage turbine wheel are considered to be the result of thermal fatigue.

The engine oil pressure and temperature gauge that had been installed for some time was for a C20B model; therefore, it had incorrect markings. Also, the TOT harness and TOT gauge were replaced about 312 airframe hours before the occurrence, due to erroneous readings. It was not possible to confirm that the ultimate temperature (927°C) for which a turbine inspection is recommended was reached during the so-called “hot starts,” or that a maximum continuous temperature of 810°C during steady state operation was reached. The TOT harness may have had a 0.03 ohms discrepancy, and the incorrect TOT gauge was replaced due to a 15°C indication error; it was installed for only 63.1 hours during which, in the worst case scenario, it may have provided inaccurate engine temperature indicatons. The effects of these erroneous TOT indications and any contribution to the turbine wheel cracking could not be determined within the scope of this investigation; however, it was suggested that their contribution was unlikely because turbine blades that have been subjected to long-term overheating will normally show a change in the distribution of the gamma prime phase, due to partial re-solutioning. The gamma prime phase was uniformly distributed in both the first- and second-stage turbine wheels. This does not preclude the possibility that the short-term overheating of the second-stage turbine wheel occurred. It is possible that the wheels experienced short-term temperature excursions related to hot starting events and/or power transients sufficient to induce thermal cracking, without redistributing the gamma prime phase. Hot starting events are not recorded by this helicopter's instrumentation and may not be recorded accurately by an operator, even if detected. About 45 engine hours before the occurrence, the engine fuel control unit was adjusted due to hot starting. The number of degrees by which the temperature was exceeded was not recorded.

The following Transportation Safety Board Engineering Laboratory project was completed:

LP 068/2004 – Engine Parts Examination

This report is available from the Transportation Safety Board of Canada upon request.

Finding as to Causes and Contributing Factors

1. Thermally induced fatigue cracking initiated radially inward in a low-cycle mode in the blade platform fillet area, then progressed normal to the blade axis in a high-cycle mode, eventually resulting in a blade failure due to overstress rupture when the remaining area could no longer support the applied loads.

Findings as to Risk

1. Hot starting events and/or power transients are not recorded in this type of helicopter and may not be recorded accurately by an operator even if detected. Turbine wheel failures may occur when hot starts and power transients are undetected, or if their effects go unchecked.

2. The first-stage turbine wheel revealed many type A and approximately four type B cracks in the blade rim, and cracks in the fillet radius of blades can lead to turbine failures. There is no prescribed scheduled inspection to detect these cracks, but a turbine special inspection is recommended when turbine outlet temperature limits are exceeded. No cracks in the blades are allowed.

Other Finding

1. Approximately 25 per cent of the major diameter seal was missing from the rear support as a result of dis-bonding due to a bond failure that likely resulted in a slight loss of engine efficiency.

This report concludes the Transportation Safety Board's investigation into this occurrence. Consequently, the Board authorized the release of this report on 9 January 2006.

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Appendix A – Modular Components Times in Service and Recent Engine Maintenance and Repair History

Aircraft Type: Bell 206LR
Registration: C-FVIX
Serial Number (S/N): 45139
Time Since New (TSN): 10862.7 A/F hours

Engine Model: Rolls Royce 250C20R
S/N: CAE-295208
TSN: 6694.4 hours
Install date: April 1995 @ Engine TSN = 530 hours, Airframe time = 4699 hours
Incident date: April 2004 @ Engine TSN = 6694, Airframe time = 10862 hours

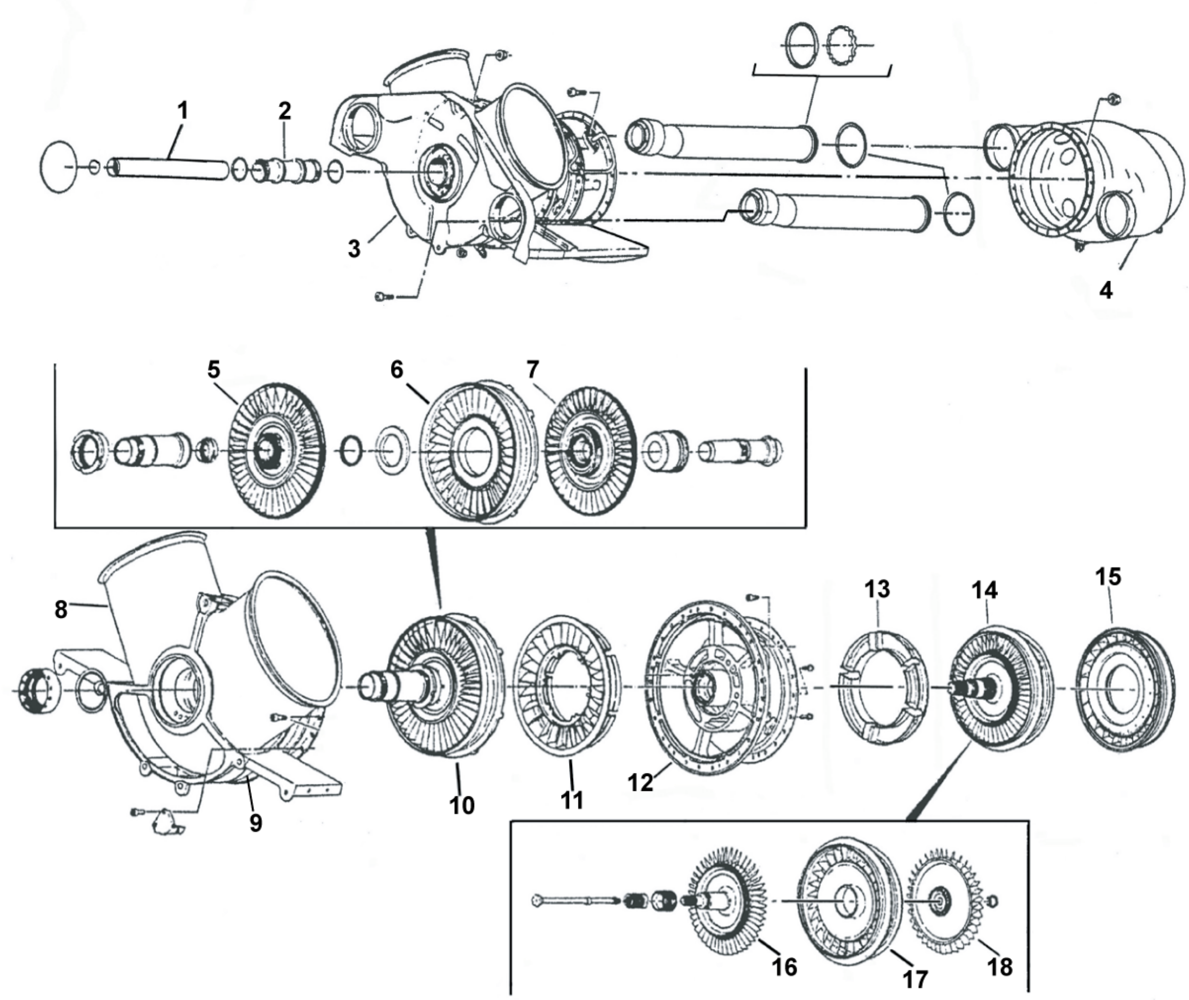
Compressor:
P/N: 23050833
S/N: CAC-15103
TSN: 5724.0 hours
TSO: 2186.6 hours
Time Since Last Repair (TSLR): 789.1 hours
Last repair: January 2003 at ACRO Aerospace under W/O 23-17374
Compressor inspected impeller to shroud clearance IAW GTP5232-3 manual
Installed February 2003 @ 10073.6 A/F time

Gearbox:
P/N: 23035185
S/N: CAG-15111
TSN: 5548.1 hours
TSO: N/P
TSLR: 1142.4 hours
Last repair: July 2002 ACRO W/O 22-9630 Repaired IAW GTP5232-2 manual, 2 ½ bearing replaced
Installed July 2002 @ 9720.3 A/F time

Turbine:
P/N: 23038160
S/N: CAT-38567
TSN: 6426.6 hours
TSO: 1744.6 hours
TSLR: 375.8 hours at (HMI)
Last repair: July 2003 ACRO W/O 23-25802
Turbine repaired IAW GTP5232-3 manual, 1750-hour inspection carried out IAW GTP5232-2 manual and CSL 4035 R5
Installed September 2003 @ 10486.9 A/F time

Scheduled Inspection:
April 17, 2004: 300-hour inspection carried out as per VIH Approved Inspection Program # P-0284, at airframe time 10840.6 hours

Appendix B – Simplified Exploded View of Turbine



1. Turbine-to-Compressor Coupling
2. Turbine-to-Pinion Gear Couling
3. Turbine Assembly
4. Outer Combustion Case
5. Wheel, Turbine, 4th Stage
6. Nozzle Assy., Turbine, 4th Stage
7. Wheel, Turbine, 3rd Stage
8. Support Assy., Turbine & Exhaust Collector
9. Support, Gas Producer
10. Rotor Assy., Power Turbine
11. Nozzle, Turbine, 3rd Stage
12. Support, Power Turbine
13. Shield Assy., Turbine Nozzle, 3rd Stage
14. Rotor Assy., Turbine Gas Producer
15. Nozzle Assy., Turbine, 1st Stage
16. Wheel, Turbine, 2nd Stage
17. Nozzle Assy., turbine, 2nd Stage
18. Wheel, Turbine, 1st Stage

Appendix C – RR250-C20R Measured Gas Temperature Limits and Recommended Maintenance Actions

Table 4
Measured Gas Temperature Limits (TOT)

Temp Range	Time	Maintenance Action
	Steady State	
810°C (1490°F)	Takeoff (5 min.)	If steady state time or condition
810°C (1490°F)	30 Minute Power (1)	limits are exceeded:
810°C (1490°F)	Max. Continuous (2)	Inspect turbine (3)
752°C (1385°F)	Normal Cruise and below	None
	During Starting and Shutdown	
Up to 810°C (1490°F)	No limit	None
810–927°C (1490–1700°F)	Over 10 sec (4)	Inspect turbine (3)
927–999°C (1700–1830°F)	Not allowed (4)	Inspect turbine (3)
Over 999°C (1830°F)	Not allowed	Remove turbine for heavy maintenance or overhaul.
NOTE: Refer to item 9, Table 101 when start temperature consistently exceeds 860°C (1580°F).		
	During Power Transient (All 250–C20R Series Engines) or One Engine Inoperative and/or Emergency Multi–Engine Operation	
Up to 752°C (1385°F)	No limit	None
752–810°C (1385–1490°F)	Until stabilized (5 min max.)	None
810–899°C (1490–1650°F)	0 to 6 sec (intentional use of temperatures in excess of 810°C (1490°F) is not recommended)	None
810–899°C (1490–1650°F)	Over 6 to 12 sec	Max. of 3 occurrences per life of each turbine wheel (3)
810 to 899°C (1490–1650°F)	Over 12 seconds	Remove turbine for heavy maintenance or overhaul
Over 899°C (1650°F)	Not allowed	Remove turbine for heavy maintenance or overhaul.

(1) This limit is applicable only during one–engine–inoperative (O.E.I.) operation of multi–engine aircraft.

(2) This limit is FAA approved for continuous operation; but is authorized by the engine manufacturer only during one–engine–inoperative (O.E.I.) operation of multi–engine aircraft and emergency engine operation.

(3) Refer to Special Inspections, Table 604, 72–00–00, Engine Inspection/Check. Also, record temperature and duration in the Engine Log Book (pink pages, Turbine Assembly, Part IV, Inspection Record).

(4) Momentary peak temperature of 927°C (1700°F) is permitted for no more than one second.

NOTE: The time–at–temperature limits are not additive. The repeated, intentional use of transient temperature limits can result in reduced turbine life and is not recommended.