Transportation Safety Board of Canada

Bureau de la sécurité des transports du Canada

AVIATION INVESTIGATION REPORT
A06O0150

ENGINE FAILURE – COLLISION WITH TERRAIN

EXPEDITION HELICOPTERS
BELL B206L (HELICOPTER) C-GSMZ
SMOOTH ROCK FALLS, ONTARIO
21 JUNE 2006

Canada

Aviation Investigation Report

Engine Failure – Collision with Terrain

Expedition Helicopters
Bell B206L (Helicopter) C-GSMZ
Smooth Rock Falls, Ontario
21 June 2006

Report Number A06O0150

Summary

At approximately 1000 eastern daylight time, the Expedition Helicopters Bell B206L helicopter (registration C-GSMZ, serial number 46604) departed Smooth Rock Falls, Ontario, destined for a mineral exploration site located north of the community. The aircraft was in a shallow climb in a northbound direction when there was a bang. The engine (Allison/Rolls-Royce 250-C20R) lost power and the pilot carried out an emergency autorotation. The aircraft landed hard, the tail boom separated from the fuselage, and the aircraft rolled over onto its right side. The pilot and front-seat passenger evacuated through the broken front windshield area without difficulty. The rear-seat passenger exited through the left rear window exit. The pilot sustained minor injuries, and the passengers were not injured. A post-impact fire destroyed the aircraft and ignited a substantial forest fire, requiring fire suppression crews.

Ce rapport est également disponible en français.

Other Factual Information

History of the Flight

On the morning of the occurrence, the helicopter departed the Expedition Helicopters facility in Cochrane, Ontario, approximately 27 nautical miles (nm) to the southeast of Smooth Rock Falls. The pilot, the sole occupant of the aircraft, landed in a cleared area near the highway on the outskirts of Smooth Rock Falls. The engine was left in idle, and the main rotor blades were turning as the two passengers boarded the aircraft. After the departure, a cruise climb was performed at approximately 80 to 100 mph.

The aircraft was climbing through approximately 500 feet above ground level (agl) when the power-loss event occurred. The helicopter was over rugged terrain and the glide was stretched to a suitable landing area, which depleted some of the main rotor rpm. During the latter stages of the autorotation, there was insufficient rpm available to arrest the aircraft's rate of descent, resulting in the hard landing. After the aircraft came to rest and the occupants evacuated the aircraft, smoke and a small flame could be seen in the area of the engine bay. The aircraft was eventually consumed by the fire, which caused a forest fire. The emergency locator transmitter (ELT) was not recovered and therefore was not examined. A cellular telephone was used to call the company facility in Cochrane and another helicopter was dispatched to the occurrence site.

Weather

Aviation routine weather reports (METARs) are not available for Smooth Rock Falls. The nearest reporting station is in Timmins, 45 nm to the south. The Timmins weather at 1000 was reported as follows: winds 220° true (T) at 7 knots, visibility 15 statute miles, ceiling 1200 feet broken. Similar conditions were present at Smooth Rock Falls; however, the wind was reported as light.

Flight Crew

The pilot had accumulated over 3700 hours of flight time, including over 3500 hours in the B206 model helicopter. Records indicate that the pilot was certified and qualified in accordance with existing regulations. The pilot was the president and owner of the company. He was well rested before commencing his duty day at 0830 on the morning of the occurrence, and was described as an experienced and skillful pilot.

Aircraft

The helicopter was manufactured in 1978 and had accumulated 14 374 total hours since new. Records indicate that the aircraft was certified, equipped, and maintained in accordance with existing regulations and approved procedures, and there were no known deficiencies before the flight. The helicopter was being operated within its load and centre of gravity limits.

Engine

The Allison/Rolls-Royce 250-C20R engine is a reverse–flow, turbo-shaft engine. The compressor assembly is at the front of the engine and houses a four-stage axial and a single-stage centrifugal rotor system. The assembly compresses the ambient air at a pressure ratio of 7.9:1. The compressed air is then routed to the combustion section at the aft end of the engine where a fuel nozzle and a spark igniter are located. The combusted air then enters the turbine section. The turbine section is mounted between the combustion section and the accessory gearbox. The turbine section consists of a two-stage gas producer turbine, which drives the compressor and accessory gear train, and a two-stage power turbine, which furnishes the output power of the engine. The output power of the turbine is transferred to the main rotor and tail rotor system through the power and accessory gearbox. The engine was manufactured in 1988 and, at the time of the occurrence, had accumulated approximately 4497 hours total time since new.

Engine Teardown and Examination

The engine was sent to the TSB Engineering Laboratory for teardown and examination. Visual inspection of the internal engine components revealed that all of the third- and fourth-stage compressor wheel blades had fractured at their roots. The outer portion of the blades of the first-stage turbine wheel had burnt off, and a crack, approximately ½ inch long, radiated inward from the outer edge of the wheel. Five blades of the second-stage turbine wheel had portions of their tips missing. A total of 28 of 54 compressor blade portions were recovered during the compressor disassembly.

Photo 1. Compressor section

Numerous engine components were selected for a detailed metallurgical examination; however, only the below-mentioned components will be discussed in this report. The TSB Engineering Laboratory report (LP 062/2006) is available upon request.

First-Stage Compressor Wheel

A normal degree of erosion of the leading edge of first-stage compressor wheel blades was noted. The leading edge tips of a few blades were slightly bent in the direction of rotation. However, the uniform surface oxidation observed on the blades indicated that the bent tips were not caused by recent foreign object damage that could have contributed to the engine failure. There was no evidence of tip rub on the first-stage compressor wheel blades. Consequently, it was concluded that the tips had been bent some time before the engine failure, possibly even at the time that the first compressor wheel was installed.

Second-Stage Compressor Wheel

The blades on the second-stage compressor wheel showed a reddish discolouration near the leading edge due to mild erosion consistent with normal service. Rubbing damage was observed at the base of the trailing edge close to the rim on numerous blades; however, it was consistent with damage caused by the failure of the third-stage compressor wheel. There was no evidence of tip rub or impact damages such as bent tips or transferred material observed on these blades.

Third-Stage Compressor Wheel

The examination of the third-stage compressor wheel revealed that 12 of the 29 blade roots were too damaged to determine the failure mode. The remaining 17 had cracks propagating across more than 50 per cent of the individual blade thickness. The progressive cracks were located mid-chord of each blade, and the blades were located uniformly around the circumference of the wheel.

Photo 2. Damaged third-stage compressor wheel

The cracks showed the clamshell shape of typical fatigue cracks. However, beach marks or striations were not observed due to the oxidation of the fracture surfaces caused by the post-impact fire. Consequently, it was not possible to estimate the propagation rate of the fatigue cracks. The fracture surfaces and cross-sections through the fatigue crack origins of selected blades did not reveal any specific cause for the initiation of the fatigue cracks. There was no evidence of foreign object impact damage on the blade surfaces that were available for examination.

The microstructure of the blade material was homogeneous and its composition was in agreement with the specification. No inclusions or porosity that might have initiated a crack were observed. Evidence of corrosion damage was noted on the fracture surfaces at the fatigue crack origin; however, it was determined that this corrosion damage was consistent with internal oxidation of the fracture surface during the post-impact fire and was therefore unrelated to fatigue crack initiation.

Thin axial cracks were observed in the coating on the convex side of some blades adjacent to the fatigue crack origins; however, they did not penetrate into the substrate. These coating cracks were consistent with damage caused by the bending of the blades in the direction of the suction (convex) side during the final overstress fracture.

Fourth-Stage Compressor Wheel

The blade fracture surfaces of the fourth-stage compressor wheel showed smearing in the direction opposite to the wheel rotation. Small areas that were not damaged by smearing showed ductile dimpling, which is typical of failure due to overstress. No evidence of progressive failure was observed. These observations are consistent with overstress failure of the fourth-stage compressor wheel blades caused by impact with blade fragments originating from the third-stage compressor wheel. The fourth compressor wheel material composition was in agreement with the manufacturer's specification.

First-Stage Turbine Wheel

All of the first-stage turbine wheel blades were damaged and exhibited similar fracture surfaces. Examination with the scanning electron microscope (SEM) of the blade fracture surfaces showed no indications of progressive failure. Furthermore, SEM examination of an axial cross-section through a blade indicated that it had been exposed to elevated temperatures above approximately 1150°C (2100°F). The first-stage turbine wheel contained a large rim crack that extended into the wheel flange and 18 smaller rim cracks in addition to the large rim crack. The fracture surface of the large rim crack was examined in the SEM. It was composed of a thermal fatigue region showing fatigue striations and a larger region of dimples and cleavage facets, which are typical of overstress failure. During the failure of the third-stage compressor wheel, the first-stage turbine wheel was subjected to a sudden temperature increase. This severe thermal shock produced high stresses in the turbine wheel rim and resulted in the extension of existing thermal fatigue cracks in the radial direction by an overstress mechanism. The failures of the first-stage turbine wheel blades and formation of the large radial rim crack were consistent with exposure to a severe thermal shock as a result of the compressor failure.

Compressor Case and Blades

The compressor case was examined for evidence of unbalanced operation or tip rub. The blade paths of the first- and second-stage compressor wheels were evenly worn without evidence of tip rub, except for very faint markings in the first compressor wheel blade path. The third- and fourth-stage compressor wheel blade paths also did not show any indications of blade tip rub.

Damaged guide vanes and gouges in the blade path lining, resulting from impact by separated blades, were observed at several locations on the third- and fourth-stage compressor wheel blade paths. Although 28 of the 54 separated third- and fourth-stage compressor blades were recovered during the engine disassembly, the blade fragments were too heavily damaged to determine their failure mode or whether the blade tips had suffered any rubbing before failure.

Compressor Overhaul Information

The last overhaul of the compressor section was performed on 23 July 2004. At the time of overhaul, the third-stage compressor wheel had accumulated approximately 3590 hours since new, and 2310 hours since the previous overhaul. On the date of the occurrence, the engine had accumulated approximately 906 hours since overhaul. Records indicate that, during the overhaul, the surface coating on the second-, third-, and fourth-stage compressor wheels was removed by glass bead media blasting, and a fluorescent penetrant inspection (FPI) of the wheels was performed. The surface was recoated with an intermetallic diffused aluminide coating and a conversion top coating, which provides corrosion protection to cast airfoil surfaces operating in corrosive environments. This procedure was accomplished in accordance with Service Bulletin EPS 10702 Revision A and the Allison/Rolls-Royce 250-C20R series overhaul manual. The first-stage wheel was replaced due to foreign object damage. According to the engine records, the replacement wheel had a total of 1515 total component hours and 2040 cycles when installed. The life limits for the compressor wheels are 7500 hours and 15 000 cycles. The total time allowed between overhauls is 3500 hours.

Component	Part Number	Serial Number	Hours Since New	Hours Since Overhaul	Cycles Since New
Compressor	23050833	CAC-15107	4496.7	906.3	7190
First-stage wheel	23032621	E77982	2421.4	906.3	3776
Second-stage wheel	23032622	E10729R	3216.4	906.3	5160
Third-stage wheel	23032623	E20692R	4496.7	906.3	7190
Fourth-stage wheel	23032624	KR62635	4496.7	906.3	7190
Impeller	23039065	CD82110	4496.7	906.3	7190

Analysis

The aircraft was maintained in accordance with existing regulations, and there were no defects reported before the occurrence flight. The pilot was certified and experienced for the flight. It was determined that at least 17 of the third-stage compressor wheel blades had progressive cracks. One of these cracks progressed to a point where the blade failed under normal operating loads. The damage to the fourth-stage compressor wheel was consistent with overstress damage caused by the liberation of the third-stage compressor blades. Therefore, this analysis will focus on the failure of the third-stage compressor wheel and the subsequent hard landing and rollover.

Although the third-stage compressor wheel was original to the engine, it was well within the total life time limits of hours and cycles, and the overhaul was done within the prescribed time frame. During the last overhaul of the third-stage compressor wheel, there were no noted discrepancies or defects. The 17 compressor blades that failed due to a progressive type of failure were covered in oxidation from the post-crash fire; therefore, a propagation rate could not be determined. However, if the fatigue cracks were present during the last overhaul, it would be reasonable to assume that they would have been noticed during the FPI of the third-stage wheel. Therefore, it is likely that the progressive failure developed at some point after the last overhaul.

Metallurgical examination could not determine the initiating event that led to the progressive cracks on a number of the third-stage compressor blade roots. However, when a progressive failure crack reaches a critical size, the remaining blade section fails due to overstress at normal operating loads. It is likely that the third-stage compressor wheel experienced this type of blade failure, causing destruction of the third- and fourth-stage compressor wheels.

The engine power loss occurred while the helicopter was flying over rugged terrain, which was not suitable for landing. Stretching the glide in an attempt to reach a suitable landing area resulted in a loss of main rotor rpm, which subsequently resulted in a hard landing.

The following TSB Engineering Laboratory report was completed:

LP 062/2006 – Engine Teardown and Examination Bell 206L, C-GSMZ

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

Findings as to Causes and Contributing Factors

1. The engine power loss event was a result of the failure of the third-stage compressor wheel.

2. Numerous blades on the third-stage compressor wheel had progressive cracks, which had propagated through approximately 50 per cent of the blade thickness. The liberated compressor blades caused severe internal damage to the third- and fourth-stage compressor wheels.

Finding as to Risk

1. The glide of the helicopter was extended to avoid landing in a wooded area. However, depleting some of the main rotor rpm to extend the glide presents the risk that there may be insufficient rpm available for a safe autorotation and landing.

Safety Action Taken

On 31 May 2007, the TSB released Aviation Safety Information Letter A06W0182-D1-L1. The letter outlines two other recent occurrences involving Allison/Rolls-Royce 250-C20R turbo-shaft engines and the similar mode of failure of the third-stage compressor wheel.

- On 05 October 2006, a Bell B206L, operated by Great Slave Helicopters Ltd., registration C-GHBY, was on approach to a remote landing site when the engine (Allison/Rolls-Royce 250-C20R, serial number CAE-295218) flamed out. A hard landing ensued and the helicopter was substantially damaged when the main rotor severed the tail boom. One passenger sustained a minor injury. The pilot and three other passengers were uninjured.

Detailed examination of the engine determined that a third-stage compressor blade had liberated from the third-stage compressor wheel, part number 23032623, due to high cycle fatigue cracking. Fatigue cracks were evident on 11 of the 29 blades on the wheel. All of the fatigue failures showed initiation around the mid-chord on the convex (compression) side of the blade and the fatigue cracks had multiple initiation sites. The reason for the fatigue cracking has not been determined.

- On 10 April 2006, a McDonnell Douglas MD520N helicopter, powered by an Allison/Rolls-Royce 250-C20R engine, serial number CAE-295415, was involved in a power loss event near Gmund, Austria. The extent of injuries and damage is unknown. Examination by the engine manufacturer determined that the engine had lost power due to a third-stage compressor blade failure, and that the general fracture morphology in the failed third-stage airfoil was consistent with fatigue progression.

There are approximately 979 Allison/Rolls-Royce 250-C20R engines in service, and several Canadian helicopter operators currently use Allison/Rolls-Royce 250-C20R engines in their fleets.

Rolls-Royce has recently achieved Federal Aviation Administration (FAA) certification of new and improved compressor wheels for the model 250-C20R turbo-shaft engine. The new wheels are manufactured from machined wrought stock rather than from cast stock. There is no evidence to indicate that the improved compressor wheels are a response to the recent compressor blade failures; however, the new design is expected to increase longevity and reduce operating costs, and may provide better fatigue performance compared to the cast wheels.

In its response to the Aviation Safety Information Letter, Transport Canada stated that it was drafting an Airworthiness Directive that will address this issue in Bell Helicopter Textron Canada (BHTC) model 206B series helicopters.

Safety Concern

The TSB is concerned about the number of recent failures of third-stage compressor blades on Allison/Rolls-Royce 250-C20R engines due to fatigue cracking, with no determination as to why the fatigue cracking is occurring. Failure of third-stage compressor blades will trigger an in-flight loss of engine power, which can result in substantial damage to the helicopter and death or serious injury to the occupants.

This report concludes the Transportation Safety Board's investigation into this occurrence. Consequently, the Board authorized the release of this report on 20 November 2007.

*Visit the Transportation Safety Board's Web site (**www.tsb.gc.ca**) for information about the Transportation Safety Board and its products and services. There you will also find links to other safety organizations and related sites.*