

Transportation Safety Board  
of Canada



Bureau de la sécurité des transports  
du Canada

## **AVIATION INVESTIGATION REPORT**

**A04P0206**

**(AMENDED REPORT)**



### **ENGINE POWER LOSS**

**QUANTUM HELICOPTERS LTD.**

**MD HELICOPTER (HUGHES) 369D C-GWPQ  
BOB QUINN AIRSTRIP, BRITISH COLUMBIA**

**11 JUNE 2004**

**Canada**

The Transportation Safety Board of Canada (TSB) investigated this occurrence for the purpose of advancing transportation safety. It is not the function of the Board to assign fault or determine civil or criminal liability.

## Aviation Investigation Report

### Engine Power Loss

Quantum Helicopters Ltd.  
MD Helicopter (Hughes) 369D C-GWPQ  
Bob Quinn Airstrip, British Columbia  
11 June 2004

Report Number A04P0206

### *Summary*

On 11 June 2004, the MD Helicopter (Hughes) 369D (registration C-GWPQ, serial number 700755) was lifting a 900-pound sling load when there was a loud bang accompanied by a partial engine power loss. The pilot performed a forced landing, and the aircraft hit the ground and rolled onto its right side with the main rotor blades still turning. The engine continued to operate on the ground and was shut down by the pilot. There was no post-impact fire. The pilot experienced accident-related health issues some time after the occurrence.

*Ce rapport est également disponible en français.*

## *Other Factual Information*

The helicopter was equipped with a modular-type engine, Rolls Royce Allison 250-C20B (S/N CAE-836346). The compressor assembly (S/N CAC-35032) had accumulated 7832.8 hours of total time since new (TTSN) and 891.2 hours of time since overhaul (TSO). It had been installed on this engine for about 108 hours. It was previously installed on a Bell 206B helicopter and operated with a bleed air centrifugal particle separator. The accident helicopter was fitted with a FDC/aerofilter inlet barrier filtration system. There was unremarkable damage discovered on a compressor, first-stage, rotor-blade leading edge during the latest engine installation. However, the damage was considered acceptable by the aircraft maintenance engineer (AME).

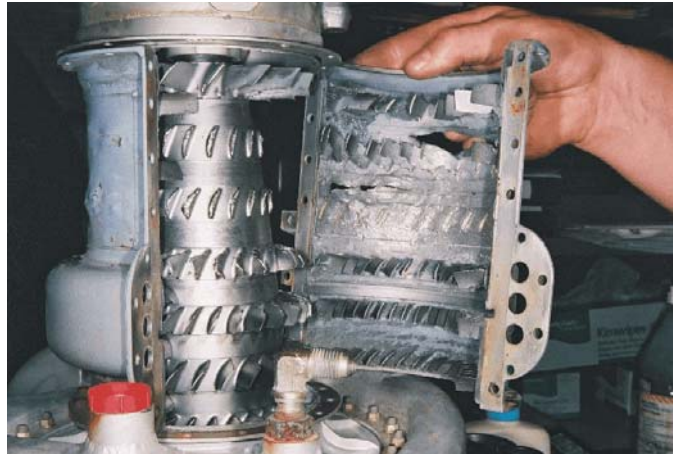
The engine was recovered by Quantum Helicopters Ltd. and shipped to a local repair and overhaul facility for teardown and examination under the control of the TSB investigator. The teardown inspection revealed that the compressor (P/N 6890550, S/N 35032) had shed all the 2<sup>nd</sup> and 3<sup>rd</sup> stage rotor blades, most of which had departed through the case halves and were recovered in the engine compartment, along with other debris.

The failed compressor and related components were forwarded to the TSB Engineering Laboratory for further examination and failure analysis (see Appendix A for the list of components). The primary components of interest were the compressor case halves and the compressor rotor shown in Photo 1. Photo 2 shows the compressor inlet, which was examined for possible foreign object damage.

After the TSB investigation was completed, the engine was released to Quantum Helicopters Ltd. and it was further examined by R.J. Waldron & Company (1987) Ltd.

### *Compressor Case Halves*

Examination of the compressor case halves showed burst-type tear damage in the skins of both halves, adjacent to the 2<sup>nd</sup> stage and 3<sup>rd</sup> stage rotor paths. Penetration through the skin was from the inside in all instances. The most significant tear was a large hole through the skin of one of the



**Photo 1.** Compressor case half removed exposing damage to blades and vanes leading and trailing edges



**Photo 2.** Inlet front support guide vanes and first-stage rotor blades leading edges showing minimal foreign object damage

case halves in the 2<sup>nd</sup> stage rotor path, approximately 70 mm long by 13 mm at its widest opening. It was concluded that most or all of the 2<sup>nd</sup> stage rotor blades exited the compressor at this location.

The 1<sup>st</sup>, 4<sup>th</sup>, 5<sup>th</sup>, and 6<sup>th</sup> stage compressor vanes were intact, but showed impact damage on both the leading and trailing edges of the vanes. The damage observed was consistent with the failure debris passing through the compressor while it continued running for a short period following the failure. Five of the 2<sup>nd</sup> stage stator vanes had fractured at or near the root. The vane stubs were bent over in the direction of rotation, and a sixth vane had completely pulled out of its slot in the vane band. All of the vanes from the 3<sup>rd</sup> stage stator had also fractured at or near the root with the vane stubs bent over in the direction of rotation. Optical microscopic examination of the vane fractures revealed predominantly impact and rub damage in the direction of compressor rotation, which had obliterated most of the original fracture surface details. However, the vane fracture profiles and damage patterns were consistent with overload failures and secondary damage. The leading and trailing edges of the vanes in all stages were also examined for erosion; no measurable erosion was observed.

A 3<sup>rd</sup> stage stator vane row band was removed from one of the case halves to examine the condition of the brazed joint from the back of the vane band. All joints showed the washers intact and good braze penetration in the joint. The slot in the 2<sup>nd</sup> stage stator vane row, where the vane had pulled out, likewise showed complete braze material penetration in the joint. Cross sections through the vanes on either side of the vane pull out location were mounted to examine their joint condition. There were some minor porosities observed, but, in general, the joints were good with complete penetration of the braze material and well-developed fillet radii between the vanes and the vane band.

### *Compressor Rotor*

The leading edges of the compressor front support vanes and 1<sup>st</sup> stage rotor blades were examined in detail for any indication of foreign object damage (FOD). On the leading edge of the front support vane, located at about the one o'clock position, there were two small indentations (Photo 2). The original surface of the vane at each mark was still visible with some oxidation and was not disturbed by any material smearing, indicating that the marks had been present for quite some time. The leading edges of the 1<sup>st</sup> stage rotor blades showed some tip damage on a few blades, and a relatively minor impact mark near the root of one blade that warranted a more detailed examination.

Optical examination of the impact mark showed that the leading edge material of the blade had been disturbed by contact with a small piece of foreign material or debris, and there appeared to be a small particle of material trapped within the indentation. The indentation was examined at higher magnifications in the scanning electron microscope (SEM). Elemental analysis confirmed the presence of a small aluminum-rich particle that was foreign to the surrounding material of the blade (Appendix C, photos 9 and 10). The material debris that had been collected from the engine cowl area during recovery and the compressor rotor blade pieces were examined, and four pieces of what appeared to be broken rivets were identified. An analysis for material composition showed material spectra containing a predominant nickel peak for sample number 1, predominant titanium and cadmium peaks for sample number 2, and predominant

iron and chromium peaks for sample number 3. Sample number 4 was almost wholly aluminum with small amounts of magnesium, which is consistent with a standard aluminum rivet material.

All of the blades from the 2<sup>nd</sup> and 3<sup>rd</sup> stage rotors had failed at or near the root. All the 4<sup>th</sup> stage rotor blades showed severe impact damage and tip rub, and were bent opposite to the direction of rotation. Three of these blades had fractured at or slightly below  $\frac{1}{3}$  span and two blades had fractured at mid-span. The 5<sup>th</sup> and 6<sup>th</sup> stage rotor blades showed secondary impact marks, predominantly along the leading edges, with blade damage progressively diminishing rearward.

Optical microscopic examination of the 4<sup>th</sup> stage rotor blade fractures revealed overload features consistent with secondary damage. However, one of the three 4<sup>th</sup> stage rotor blades that failed closer to the blade root did show what appeared to be macroscopic features of cyclic overload fracture. This blade fracture was sectioned from the rotor for SEM examination at higher magnifications. Examination revealed only overload fracture features and surface rub patterns, but gave the false appearance of cyclic steps when viewed at lower magnifications.

Optical microscopic examination of the blade fractures from the 2<sup>nd</sup> and 3<sup>rd</sup> stage rotors predominantly revealed overload features. All of the blades had failed at or slightly above the blade root and the blade stub remnants were bent opposite to the direction of rotation. However, one blade fracture in the 2<sup>nd</sup> stage was unique in that it showed a flat, transverse fracture surface from the leading edge back towards mid-span that changed to a slanted fracture towards the trailing edge. Areas of the fracture surface towards the leading edge showed faint striations indicating a progressive failure mode. This blade fracture was sectioned from the rotor for detailed SEM examination.

A SEM examination of the 2<sup>nd</sup> stage rotor blade, showing faint striations, confirmed the presence of distinctive fatigue striations beginning near the leading edge of the blade and progressing aft towards approximately mid-span (see Appendix B, photos 3 through 8). The aft section of the blade fracture towards the trailing edge was characteristic of the final overload failure of the blade. All fracture detail in the leading edge region where fatigue initiated seemed completely obliterated by post-fracture rub damage, and only small pockets of resolved fatigue striations remained.

A cross-section was taken through the fracture just aft of the leading edge for metallurgical examination of the blade near the region of fatigue initiation. There were no metallurgical anomalies or any other defects detected at this cross-section. Metallographic assessment of this section showed a uniform grain structure typical of precipitation-hardened stainless steel 17-4 PH. Elemental composition of the compressor rotor blade material was verified by energy dispersive x-ray analysis as a 17-4 PH stainless steel alloy, the material specified for the compressor rotor. The hardness was also uniform and averaged 34.5 HRC, which satisfied the manufacturer's material hardness criteria of 30–38 HRC.

Subsequent re-examination of the damaged region close to the location where the blade was cut for microstructural analysis, about  $\frac{1}{4}$  of the blade length back from the leading edge, revealed fractographic features consistent with porosity (see Appendix D, photos 11 through 14). The contours of fatigue striations, which were resolved aft of the region with porosity, indicated that the direction of fatigue crack propagation was from the porosity toward the interior of the blade.

The details forward of the porosity had been obliterated by post-fracture damage and, therefore, the direction of the striations was not visible in that area.

## *Analysis*

High cycle fatigue fracture was confirmed for one of the 2<sup>nd</sup> stage rotor blades, and separation of this blade is considered to be the lead event in the compressor failure. This is consistent with the damage pattern observed in the compressor, including the shedding of all of the 2<sup>nd</sup> and 3<sup>rd</sup> stage rotor blades, and the major penetration observed in the case halves in the 2<sup>nd</sup> stage rotor path. Failure analysis of the stator vanes did not reveal any indications of premature failure, and the vane failures were all considered to be secondary in nature, resulting from the initial 2<sup>nd</sup> stage rotor blade failure. This was supported by a review of several other compressor failures for this engine type, which showed that, in all cases where a compressor stator vane failure was identified as an initial cause of failure, there was no associated shedding of the compressor rotor blades such as occurred in this incident.

The possibility of FOD as a precursor to the compressor rotor blade failure was considered. However, for a number of reasons, it could not be concluded positively that FOD was a precursor. On this installation, an inlet barrier filtration system was installed; however, there appears to have been impingement by a small foreign particle on one of the 1<sup>st</sup> stage rotor blades, as evidenced by what appeared to be a recent impact mark and a particle of aluminum metal transfer. The engine had continued to run for a short time after ground impact, and it is possible that something was ingested into the compressor at that time, leaving the mark observed on the 1<sup>st</sup> stage blade. On the other hand, the mark on the leading edge of the front support vane was oxidized, indicating that it had been present for some time. Since the debris in which the piece of aluminum rivet was found was collected among general debris found in the engine cowl area after the accident, it cannot be positively linked to the presence of the small particle of aluminum in the impact mark. Given these observations, FOD cannot be ruled out. However, the presence of a porosity pocket near the fatigue-initiation zone makes FOD a lesser candidate as a probable fatigue crack initiator.

The detection of a porosity pocket on the fracture surface in the damaged area near the leading edge suggests a cause for the fatigue crack initiation. Fatigue striation contours in the vicinity are such that the direction of crack propagation was in line with the porosity pocket location.

The following TSB Engineering Laboratory report was completed and then amended 29 November 2005:

LP 082/2004 – Compressor Failure

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

## *Findings as to Causes and Contributing Factors*

1. The compressor failure resulted from the separation of a 2<sup>nd</sup> stage rotor blade due to a high cycle fatigue fracture, which had initiated near the leading edge of the blade.
2. The material of the blade met the desired hardness; however, the damaged area of the blade near the leading edge contained a porosity pocket.
3. While post-fracture mechanical damage in the leading edge area prevented determination of the fatigue crack propagation direction forward of the detected porosity, it is possible that the fatigue crack initiated at the porosity pocket.

## *Other Finding*

1. Foreign object damage could not be linked to the fracture initiation mechanism.

*This report concludes the Transportation Safety Board's investigation into this occurrence. Consequently, the Board authorized the release of this report on 06 April 2006.*

*Visit the Transportation Safety Board's Web site ([www.tsb.gc.ca](http://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.*

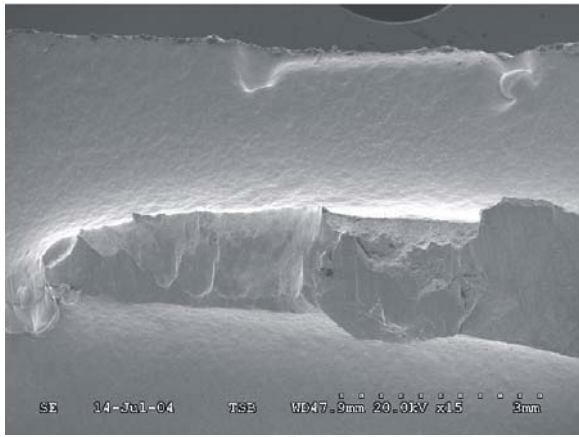
## *Appendix A – Compressor Components Forwarded for Analysis*

Rolls Royce Allison 250-C20B  
Engine S/N CAE-836346

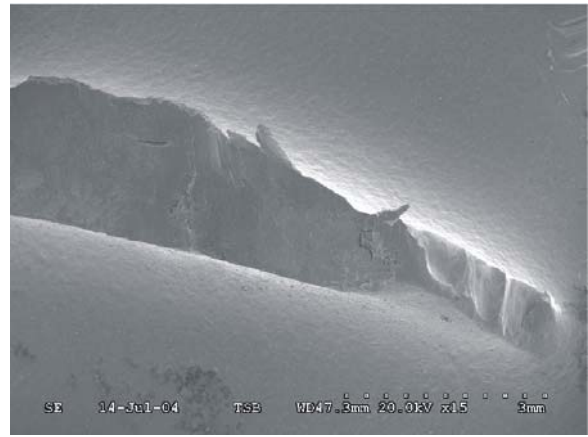
<b>Component</b>	<b>Part Number</b>	<b>Serial Number</b>
Scroll	6851574F	AP18581
Front Support	6890530-B	22977
Rear Support	6851430	ER22658
Case Half Set	23057142	26792
Front Diffuser	6870811	ER-29772
Vane Diffuser	6870813	
Spur Adapter Gearshaft	23031922-C	CG124630
Bearing Housing	6893617-C260	17
Miscellaneous bolts and nuts	N/A	N/A
#1 Bearing	6898607	HAZ101
#2 Bearing	6889093	HAQ799
Carbon Seal Mating Ring	23034513B	63005SOCN
Compressor Rotor 1 <sup>st</sup> Stage	6890501	KR69611
Compressor Rotor 2 <sup>nd</sup> and 3 <sup>rd</sup> Stage	23060422	C36140
Compressor Rotor 4 <sup>th</sup> Stage	23060404	C34556
Compressor Rotor 5 <sup>th</sup> Stage	23060405	ER41944
Compressor Rotor 6 <sup>th</sup> Stage	23060406	ER41335
Impeller	3058147-C	KR99876
Rotor Blade pieces and debris	N/A	N/A



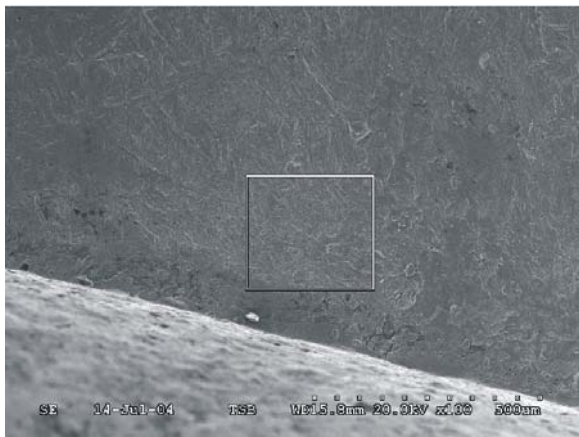
*Appendix B – Photos 3 to 8 Showing 2<sup>nd</sup> Stage Compressor Rotor Blade Fatigue Failure*



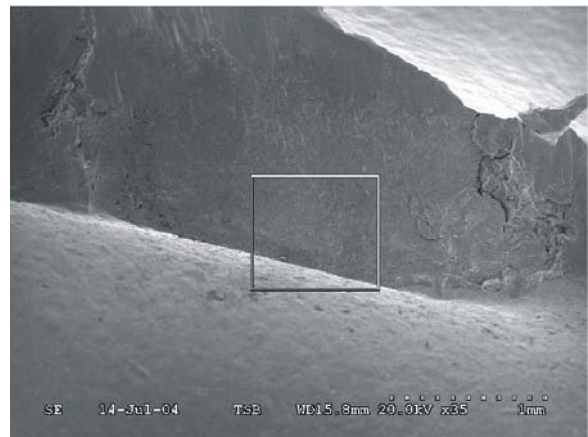
**Photo 3**



**Photo 4**



**Photo 5**



**Photo 6**



**Photo 7. Fracture surface image**



**Photo 8. High cycle fatigue**

## Appendix C – Analysis of Particle on 1<sup>st</sup> Stage Rotor Blade

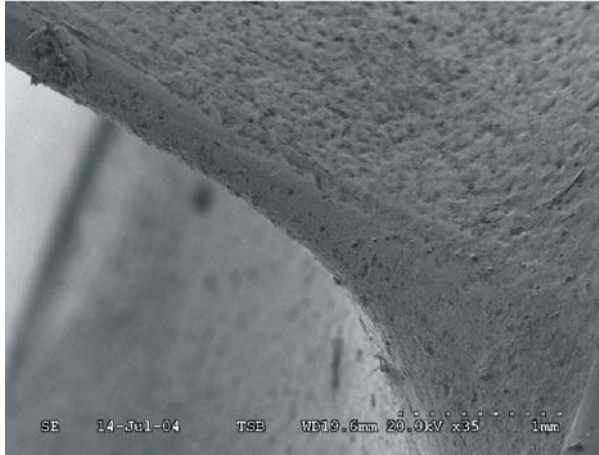


Photo 9

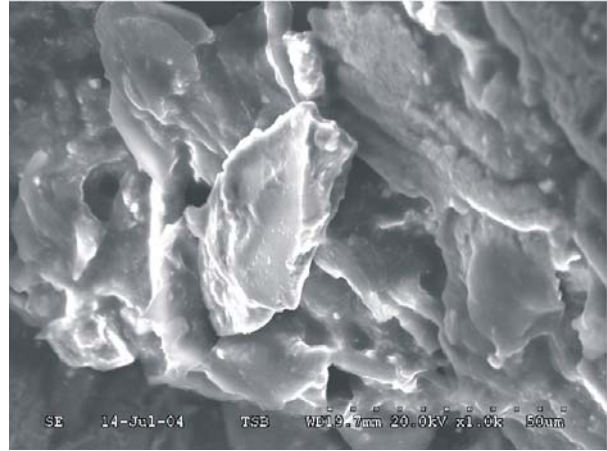


Photo 10

Photos 9 and 10 are scanning electron microscope images of impact mark at 1<sup>st</sup> stage rotor blade leading edge showing the presence of a small imbedded particle.

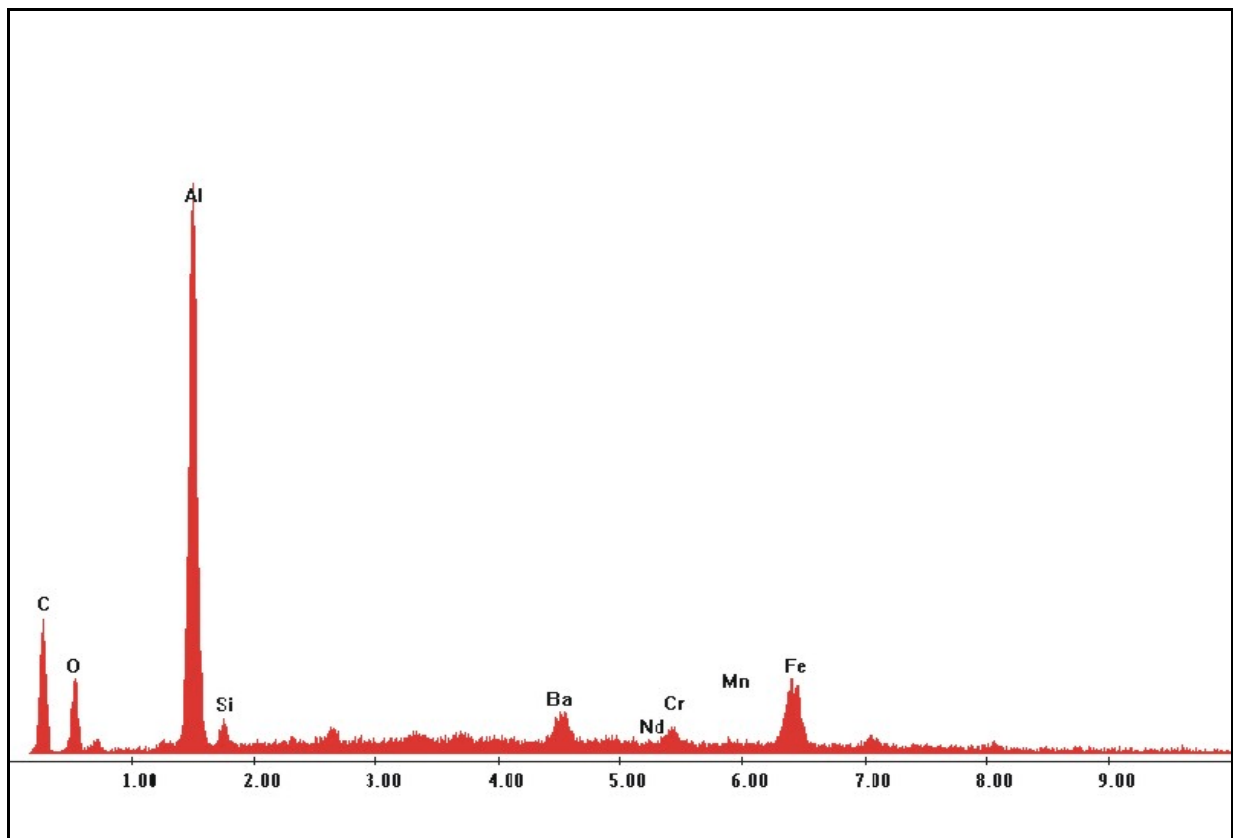
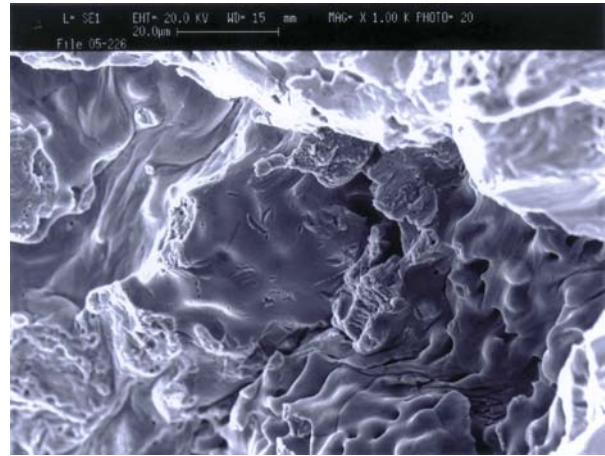


Figure 1. Material analysis spectra of particle revealed a predominantly aluminum composition

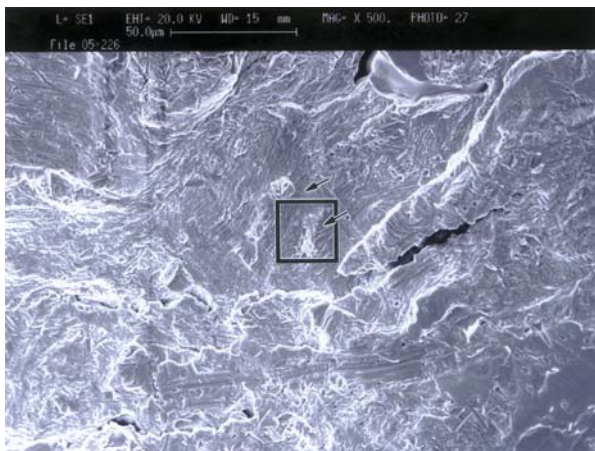
## Appendix D – Re-examination of 2<sup>nd</sup> Stage Rotor



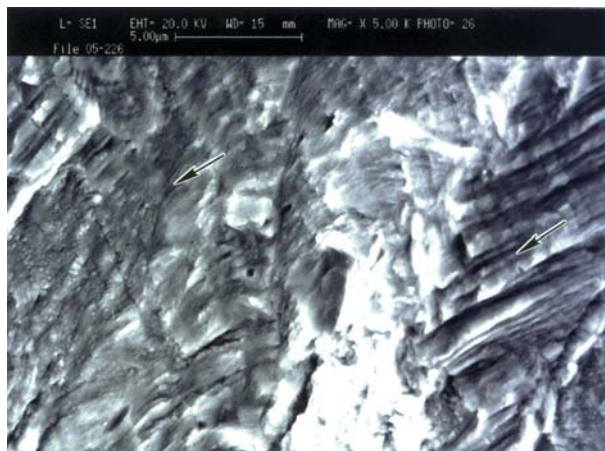
**Photo 11.** Low magnification of the damaged region near the leading edge. The area in the circle shows porosity.



**Photo 12.** Evidence of porosity in the damaged area near the leading edge



**Photo 13.** Area outlined by box in Photo 11. Direction of fatigue crack propagation identified by arrows.



**Photo 14.** Area outlined by box in Photo 13. Fatigue striations indicating the direction of crack propagation from the region with porosity.

Photos 11 through 14 are scanning electron microscope images of the damaged area near the leading edge of the 2<sup>nd</sup> stage rotor blade showing evidence of porosity and direction of fatigue crack propagation.