Applied and Computational Mechanics 19 (2025) 87–96

Surface fatigue microcracks in turbocompressor blades of a turboshaft engine

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Received 30 January 2025; accepted 24 June 2025

Abstract

Rescue and military helicopters are sometimes exposed to extraordinary operating conditions. Such loading can cause non-standard fatigue defects with very specific types of failure. Failure of a turbocompressor blade during a flight results in an inoperative power unit, leading to a very probable catastrophic scenario. This paper deals with fatigue microcracks of one of the widely used nickel alloys, ZS6K, operated in helicopter turboshaft power units. Despite its widespread use, fracture mechanics and the crack growth rate data are not well known. This knowledge gap exists for historical reasons, as early design practices were based largely on extensive ground and flight testing.

Fatigue microcracks observed on turbocompressor blades of a turboshaft engine, made of ZS6K substrate material covered with a protective aluminum layer, after operation under high load, have not been described yet. The novelty of this article is to gather and describe the newly observed initiated fatigue cracks of turboshaft engines and their specific conditions under which they occur. This is the first important step for further research focused on an in-depth description of fracture mechanics, experimental investigation of short crack growth, and mathematical modeling of crack growth in this specific material composition under high-load spectrum.

The obtained results fill a gap in operational and material data necessary for the durability and damage tolerance assessment. This paper ultimately contributes to the broader goal of increasing the operational safety of turboshaft power units.

Keywords: turboshaft engines, turbocompressor blades, fatigue microcracks, nickel alloy, protective layer

1. Introduction

This paper deals with a problem of initial fatigue microcracks observed in blades of turbocompressor of helicopter turboshaft engines during inspection periods. These parts are very critical. Any structural damage to even one component during operation can lead to failure of the entire powerplant and potentially damage the fuselage. Then, the helicopter becomes partially or completely inoperative.

Certification specifications for engines are described in the European Aviation Safety Agency (EASA) standards CS-E or Federal Aviation Administration (FAR) Part 33 standards. These standards cannot cover all non-standard conditions, e.g., rescue or military operations. Laboratory testing of small parts, such as blades, provides only substrate material data. Ground testing of engines is only an approximate simulation of real operating conditions and can bring only some very simplified examples of real operating conditions.

In the past, there have been several crashes of military training and rescue helicopters caused by the damage of turbocompressor blades of the turboshaft engine. Clear causes have not yet been found. However, during regular inspections using non-destructive testing (NDT) methods,

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https://doi.org/10.24132/acm.2025.960

damage of the protective alitation blade surface layer and sometimes also of the substrate material, including the formation of fatigue microcracks, has been observed on the blades. Several studies dealing with crashes caused by these phenomena were published [12, 13, 17]. Usually, the turboshaft engines are loaded similarly to turboprop and jet engines. The big difference are the missions and taxiing where the helicopter must withstand special operating, e.g., hovering. The influence of high temperature and material degradation on turbine blades is described in articles [1, 5, 7, 10, 11, 15, 16, 19, 22, 23]. New methods for testing miniature specimens have been developed that can also be used for small turbine blade testing [6, 14].

One of the widely used materials for turbine blades is nickel alloy ZS6K. The composition of this material, protective layer and material degradation processes were studied in [2–4, 8, 21]. Specific operating, general overhauling and their impact on the observed microcracks in blades, made from this alloy, are not well known and described. This situation has led to an unclear conclusion in investigations of some air accidents. Other nickel-based superalloys with and without the protective layer are described in detail in [18] and [20].

In the first part of this paper, the methodology and experimental procedure based on real operating conditions are defined. Then the observed degradation effects were described and summarized. The goal of this study is to extend previous research by addressing the knowledge gap related to the analysis of fatigue microcracks.

2. Experimental procedure

This section describes an experimental procedure focused on a chosen demonstrator. The turbocompressor blade of the helicopter turboshaft engine TV3-117 was chosen as a demonstrator. Operation and maintenance procedure of the demonstrator and material composition of the blade will be described in the following paragraphs.

The turbocompressor consists of two stages blade assembly, where hot gases from the gas chamber are released. The second stage rotor blade was chosen as a sample of this research and it is shown in Fig. 1. The stage consists of 101 rotor blades. Structural design, operational and maintenance data of TV3-117 are summarized in [9].

The basic operational engine data are listed in Table 1. The nominal revolution value (100 %) is 19 500 RMP. There are five basic operational regimes with defined maximum allowed RPM and temperature values.

Fig. 1. The demonstrator – turbocompressor blade

Table 1. Operational regimes, revolution and temperatures of the turbocompressor

Regime	Maximum allowed revolutions [% of 19 500 RPM]	Maximum allowed engine sensor temperature [°C]
Emergency	101.15	990
Take off	101.15	990
Nominal	99	955
Cruise 1	97.5	910
Cruise 2	95.5	870

The research methodology is based on data collection and classification from 60 TV3-117 turboshaft engines, corresponding to 6 060 turbocompressor blades that were exposed to real operation in civil and military training mode. Monitored data includes fatigue crack lengths longer than 0.05 mm, the area of fatigue cracks occurrence, the absence of protective layer and the oxidated area. The engines TV3-117 were disassembled and the turbocompressor blades were maintained as defined in the general overhaul manual.

Each blade was washed and subjected to visual inspection and NDT fluorescent penetrant inspection (FPI) according to the mandatory maintenance procedure. The blades with fatigue cracks longer than 0.05 mm were subjected to a detailed scanning electron microscope (SEM) analysis of the cross section additionally to the mandatory maintenance procedure. The blade cut was performed in the half of the blade length. Then the gathered data was checked and a continuous data classification and data summary were performed. The estimated error of this procedure is given by the used methods, especially NDT, which is defined after the long maintenance experience by a value not exceeding 1 %.

2.1. Material composition

The blade consists of two different materials. The substrate material is a polycrystalline (equiaxed) nickel-based superalloy ZS6K, composition of which is summarized in Table 2, and the aluminium protection layer as shown in Fig. 2. There is a transition layer between the substrate material and the protective layer.

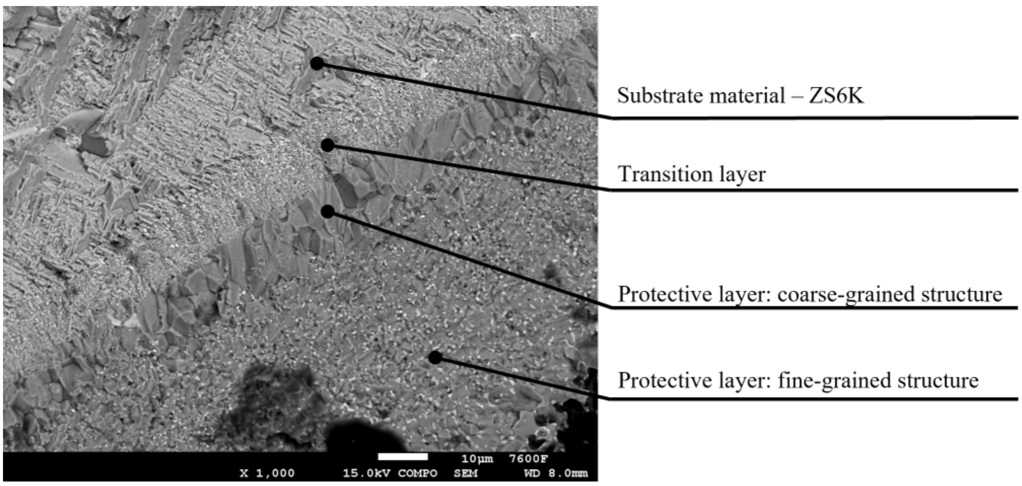


Fig. 2. Surface layer composition of the turbocompressor blade

Table 2. Chemical composition in [%] of nickel alloy ZS6K [2]

C	Cr	Al	Ti	Mo	Co	W
0.13–0.2	9.5–12	5–6	2.5–3.2	3.5–4.8	4–5.5	4.5–5.5
Fe	MnSi	PS	Ce	B	Pb	Bi
2	0.4	max 0.015	max 0.015	max 0.02	max 0.005	max 0.015

Nickel alloy ZS6K combines unique physical properties, capable to withstand the aerodynamic pressure and the centrifugal force. It is a nickel superalloy with low Fe content and high Cr, W, Mo and Co content with Al and Ti elements promoting the formation of the γ' phase.

The protective aluminium layer covers the substrate material against the effects of high-temperature corrosion caused by high temperature and exhaust gases. A blade designed in such a way can withstand all requested operational conditions. The protective layer consists of fine-grained and coarse-grained structures, Fig. 2.

The protective layer to the substrate material is applied by a heat treatment process which is called alitation. This process is performed by spraying aluminum powder on the surface of the substrate material followed by diffusion annealing. The layer should be homogeneous, non-porous and uniform over the entire surface of the blade. The thickness of the protective layer should range from 0.02 to 0.05 mm. This interval is actually very difficult to achieve due to the high complexity of the diffusion process. Therefore, in some specific cases, the thickness of the protective layer may be outside the acceptable interval.

2.2. *Specific operation conditions*

Turbine blades of turboshaft engines are loaded very similarly to the turbine blades of turboprop or jet engines. The big difference is the mission and training procedures when the helicopter must withstand flights at very small airspeeds or hovering. The helicopter must be able, contrary to the aircraft, to fly in backward or sideward direction.

The air speed of the helicopter engine intake thus takes on lower values than the air speed of the aircraft engine intake. The helicopter engine is not cooled naturally, and the temperature of engine hot parts increases very fast.

Also, the landing phase is different in the case of a helicopter and an aircraft in terms of engine cooling. During the final approach, the aircraft engine power is about 30 % of the maximum power and after the landing, the aircraft is taxiing. During this phase, the aircraft engine and its blades are cooled proportionally within a few minutes. The helicopter turboshaft engine must be operable at maximum speed and land very fast. The turboshaft engine blades are coolled much faster.

Another aspect of military helicopters is the ability to operate at a very low altitude while following the contours of the terrain. This type of flying can mix airflow with ground debris, sand or dust. All these operational specifics cause increased stresses and the most unfavourable conditions for resistance to thermal and fatigue damage.

Summarized facts about the monitored helicopter engine operation are described in the following points:

1. No engine was subjected to a load higher than the maximal allowed level.
2. All engines were operated within allowed limits. This means the maximum RPM, temperature, pressure or vibration were not exceeded.

3. All engines were operated in training regimes (no special missions).
4. All engines were fully operatable and no destruction was recorded.
5. All engines were operated in the interval of 1 300–1 500 flying hours.

2.3. Maintenance procedures

To ensure the safe operation of the helicopter and continued airworthiness of the engine, prescribed maintenance practices must be followed. The turboshaft engine critical parts must be subjected to regular service procedures – the general overhaul. The criterium for applying this procedure is the number of flying hours, flights or time period.

The engine is disassembled, cleaned and several procedures must be done, e.g., a defectoscopic test, then a new protective layer is applied on blades surfaces and checked. Then the whole engine is subjected to a ground test, flight test and after that the helicopter's airworthiness can be restored.

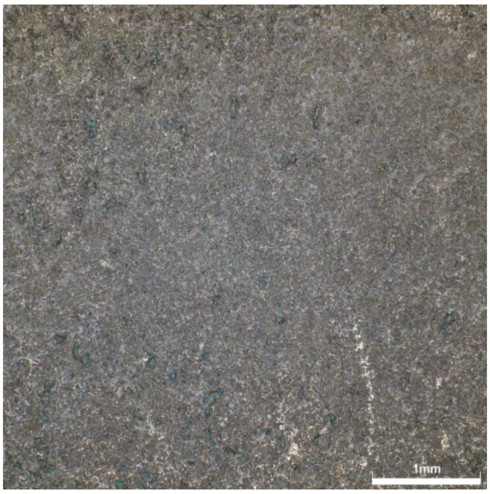
3. Observation and results

This section describes the identified blade defects before and after the general overhaul and summarizes the results and reasons for the initiation of these defects.

3.1. Data observed before the general overhaul

It has been found that some blade surfaces are seriously damaged during operating. The difference between the surface quality before and after the general overhaul is shown in detail in Fig. 3. The blade after the general overhaul (before operating) has very smooth surface with no major geometrical differences. The blade surface after the operation time between two overhauls is full of non-homogeneous crater-shaped areas. The usual depth of these craters is in the interval of 3–10 μm and their usual diameter is 10–20 μm. In some places, these areas connect and form a larger area and the protective layer is missing.

Non-homogenous surface leads to material degradation, which can initiate first fatigue cracks. Several very short initial cracks were detected. The protective layer is seriously damaged. Short fatigue cracks initiated from the non-homogenous protective layer are penetrating



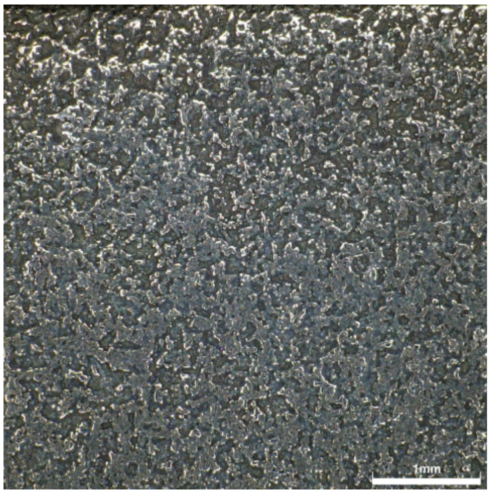


Fig. 3. Blade surface after general overhaul (before operation – *left*) and before next general overhaul (after operation – *right*)

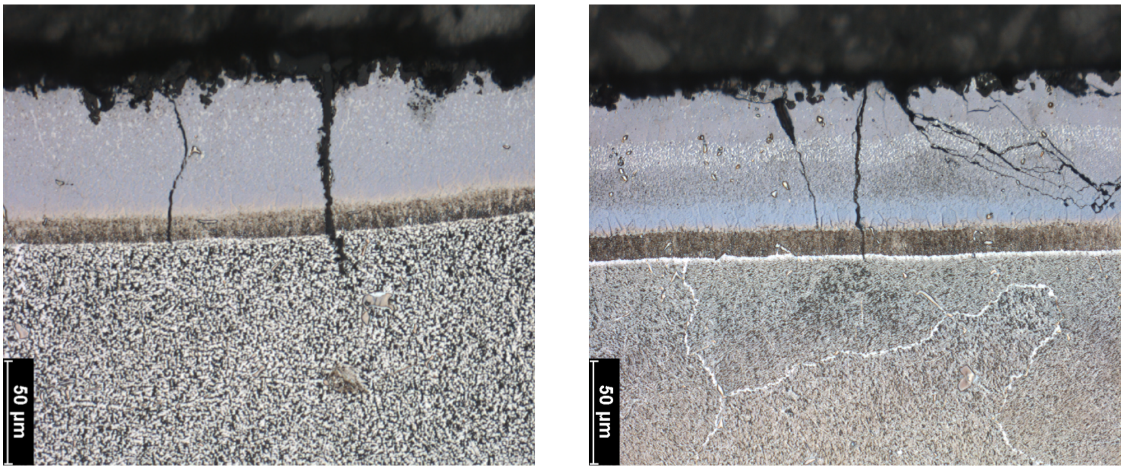


Fig. 4. Microcracks initiated in protective layer surface with the propagation to the substrate material

the substrate material. It is not possible to detect these defects during the operation or standard maintenance procedures requested by the design organization. Standard methods requested by the design organization must find fatigue cracks of the blade (without the protective layer) longer than 50 μm. This crack length is much longer than the value of the found microcracks. These cracks can growth in the substrate material and cause blade destruction. The observed microcracks, Fig. 4, were initiated in porous area of the protective layer. The geometrical non-homogeneity forms the natural micro-notch with a stress concentration.

As shown in Fig. 5, the observed high temperature oxidation of the protective layer with combination of porosity caused the crack initiation. Also, the oxidated area is a natural notch with higher stress values during the operation.

The facts about the turbocompressor blades after the operation and before the general overhaul can be summarized as follows:

1. 13 % of blades inspected had several cracks longer than 0.05 mm.
2. 85 % of blades were detected with cracks in the area of the leading edge.

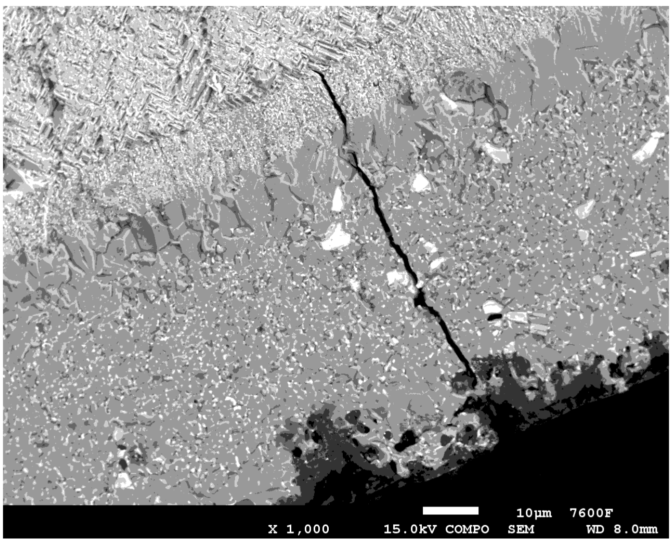


Fig. 5. Microcracks initiated from the high temperature oxidated area

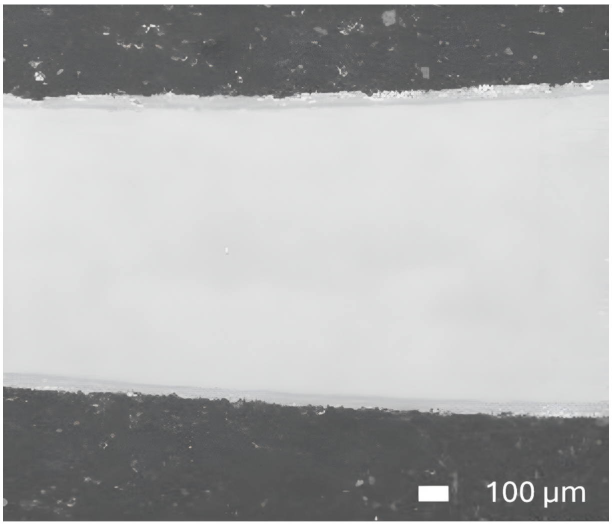


Fig. 6. Examples of a new non-homogenous protective layer

3. 100 % of blades had a seriously damaged protective layer.
4. 100 % of blades had oxidated areas.
5. 15 % of all blades were seriously damaged in substrate material and it was not possible to put them back into operation.

3.2. Data observed after the general overhaul

The purpose of the general overhaul is to restore the blade's protective layer to its original condition. The protective layer has a variable thickness, Fig. 6. The protective layer with unacceptably low thickness is not able to withstand hot gases and burns away. Then the substrate material degrades. The protective layer with unacceptably high thickness is also separates from the substrate material after only a few hours. These occurrences can cause high temperature oxidation process leading to material degradation, notches and first fatigue cracks, Fig. 7.

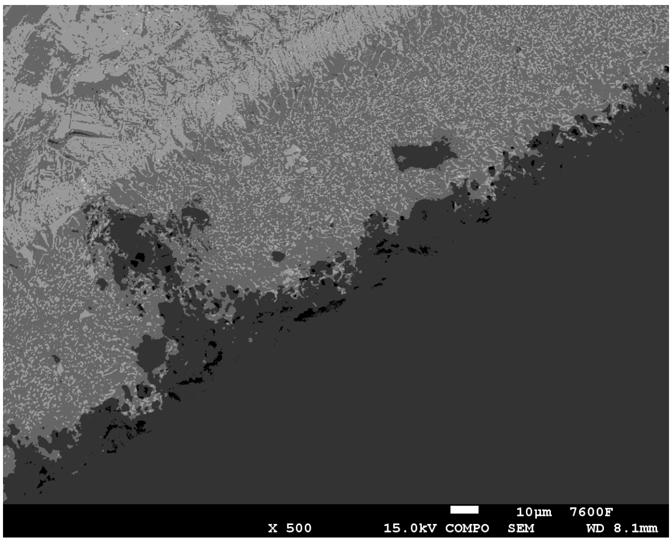


Fig. 7. Examples of the porosity leading to high temperature oxidation

The facts about the blades after the general overhaul and before the operation can be summarized as follows:

1. Some of the blades can have hidden microcracks from the previous operation. These microcracks are very difficult to detect by standard NDT methods. The exact number of these hidden microcracks is not known. The prediction of this effect based on maintenance organization experience is a value less than 5 %.
2. First degradation was already detected after the first ground test during the general overhaul.

The blades of 60 helicopter turbocompressors engine TV3-117 were checked during the performed experiments. Several degradation effects on blade surfaces were detected. There can be several reasons for the initiation of these degradations.

Firstly, the difficulty of the annealing process of the protective layer to the substrate material is the reason of natural notches generating local higher stress values. Thus, first microcracks can be initiated.

Secondly, some of the protective layer inhomogeneities are very difficult to detect. Thus, some blades can be operated with these notches.

Thirdly, the standard NDT methods used in general overhauls are not able to detect cracks smaller than 30 μm. Thus, the overhauled blade with very small cracks from previous operation may be operated.

Lastly, operating on higher load level generally causes higher stress values, especially in the non-homogenous areas of blades.

4. Conclusion and follow-up analysis plan

Degradation processes of the protective layer and substrate material during the time between two overhauls lead to fatigue crack initiation. Very short fatigue cracks were found in blades during operation. Further loading may lead to structural failure if the microcracks are not detected.

The physical reason why fatigue cracks were observed in the leading edge area of 85 % of the blades is that the leading edge has a very small thickness and the rapid cooling due to the immediate engine shutdown causes non-standard deformation and stress distribution where a higher stress peak can occur. Immediate engine shutdowns can occur for a variety of reasons, including training, rescue operations, or military missions. This factor, combined with the thickness of the protective layer below the minimum limit (0.02 mm), causes the protective layer to fail to fulfill its protective function and to burn out. If the thickness of the protective layer is greater than the maximum limit (0.05 mm), small parts of the protective layer peel off from the substrate material. The substrate material then has no protection.

Another physical reason for the appearance of fatigue microcracks can be the roughness of the protective layer. Some fatigue microcracks appear at the location of surface geometric irregularities of the protective layer, Fig. 4. In some cases, these irregularities act as notches. Achieving a reduction in roughness is technologically very difficult.

To eliminate mentioned problems, the operators should be able to identify the number of shutdowns and the maintenance organization should perform in-depth inspection if this situation happens. This recommendation will help determine the possible occurrence of short fatigue microcracks mentioned above. The significance of this recommendation and research work lies in the potential increase in the current level of aviation safety. A deep understanding of fatigue cracks in turbocompressor blades will help predict fatigue crack growth and define the associated necessary maintenance and operation procedures.

The industrial aspects of this research will help describe and simulate newly observed fatigue damage and reduce the number of future helicopter accidents. The gathered data will help operators better predict fatigue aspects of their operational spectrum.

Future steps will be focused on an in-depth description of fracture mechanics, simulation of immediate engine shutdowns, experimental investigation of short crack growth in the observed material and analytical and numerical methods for predicting their growth. A corrective action plan will be designed to ensure sufficient safety of the blade operation after the overhaul. Safety will also be assessed using a probabilistic approach, taking into account the variances of input variables.

Acknowledgement

This publication was supported by the Czech Technical University in Prague, Faculty of Mechanical Engineering, Czech Republic.

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