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# FATIGUE FAILURE ANALYSIS OF FAN BLADE OF D-18 ENGINE OF "RUSLAN" AIRCRAFT

A.A.Shaniavski

State Centre of Flight Safety of Civil Aviation, Moscow, Russia

## ABSTARCT

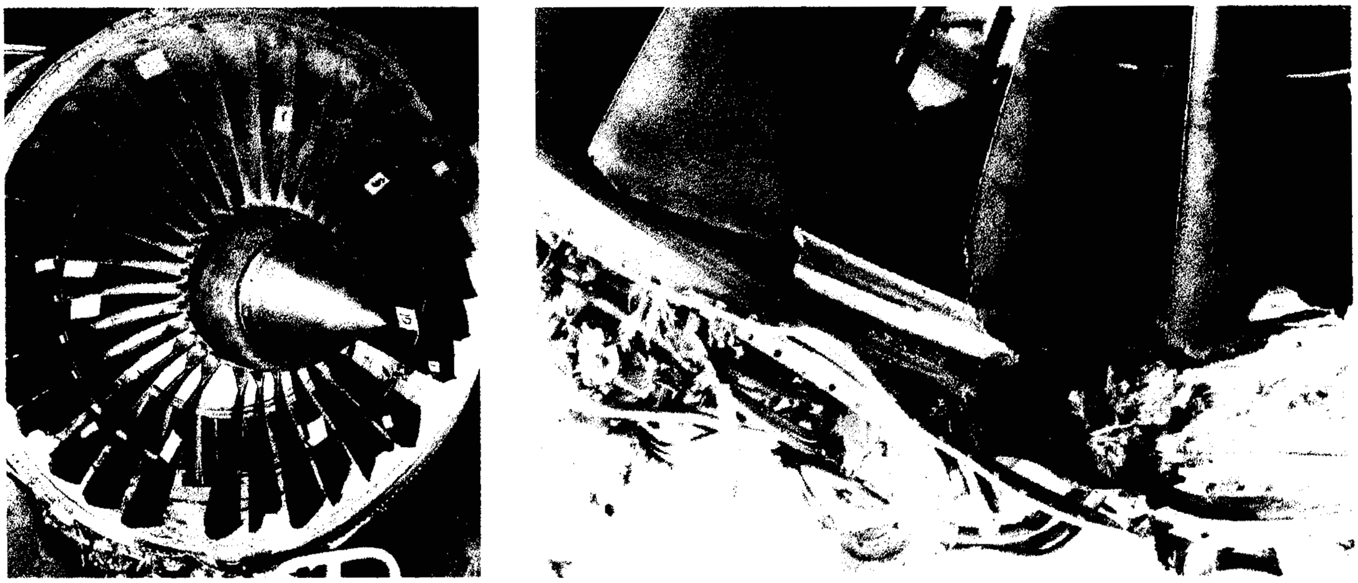
Blades from the Ti-alloy fatigued in service were investigated on the basis of the introduced synergetic approach to fatigue failures analysis. In-flight cascade of events initiated by two fan blades failures was reconstructed from the quantitative fractographic analysis. Good correlation between reconstructed from fractographic analysis and realised in flight situations is demonstrated.

## INTRODUCTION

During flight of an "Ruslan" aircraft two rotor blades of the first fan stage of the D-18 engine have failed. The engine rotation was immediately stopped because of high level of vibration. After the landing it was established that the air intake of one of the aircraft engines is absent and two fan blades of the same engine are broken as a result of fatigue crack development through fir tree tails of these blades. Some fan blades of the same rotor stage had deformations, as shown in Fig.1. Three other blades had cracks through their fir tree tails too.

The flight information from flight recorder have shown an appearance of the discreet signal about deviations of engine's parameters from normal nine seconds before the accident. This information was used to explain blades failures and deformations because of flatter when a very high amplitude of vibration loads can take place. This idea correlated with the fact that the engine's air intake was absent. It could not be found during almost one year after an accident.

The main idea about the blades failures cause was based on the very high speed of fatigue crack growth because of very high level of vibration loads during flatter. The maximum length of fatigue fracture zone for one of the failed blade was approx. 40mm. It can be seen that the average crack growth rate to achieve this length during 9s must be near to 4.4mm/s. In flight fan blades withstand two peaks of maximum stresses: one from resonant vibrations with frequency near to 100 Hz and second from nonresonant vibrations with frequency near 200Hz. During 9s the possible crack growth period for the maximum value of frequency for blades is 200x9=1,800 cycles even not taking into account that at this frequency the vibrations are nonresonant. Therefore the average crack growth rate would be $2 \times 10^{-3}$ mm/cycle. It is evident that this growth rate is very high and correlates to the area of low cycle fatigue (LCF). But for the LCF, in contrary, the average growth rate of this level correlates to the maximum crack length 2mm Ivanova [1]. This contradiction would have to be decided from the quantitative fractographic analysis of in-service failed blades.



a) b)

**Figure 1:** Damaged blades (a) in the rotor with (b) the place near the fatigued blade

## INVESTIGATIONS DETAILS

### *Blades material*

Blades were made from Ti-base alloy VT3-1 (Ti-6Al- Mo- Si) and had typical two phase $(\alpha+\beta)$ globular structure which is recommended for titanium blades of aircraft engines. Mechanical properties of the failed blades were such as recommended for practice.

### *Fractographic analysis*

The scanning electron microscope CDS-50 was used to analyse fracture surfaces features. Details of quantitative fractographic analysis for faceted pattern features can be taken from paper of Shaniavski [2].

## RESULTS OF INVESTIGATION

### *Cracks initiation and propagation*

Both failed blades had origins of main cracks placed near the corner between free surface and the blade tail tooth base, as shown in Fig.2. The tooth's surfaces had fretting damage appeared in service during engine operation. Borders of the fretting zones where near the origins of the fatigue surfaces but at some distance from them.

Intensive analysis of fracture surfaces has shown cascades of the secondary cracks on the blades tooth's surface at the distance near to 10 mm from the main origin with their small development in some distance, as shown in Fig.2 for one of the failed blade. Secondary cracks pierced the blade in the perpendicular direction to the tooth surface and had coalescence with main crack.

One of the two fatigued blades had the strongly expressed border between fatigue and fast fracture zones at the distance from the origin near to 20mm. It was a sign that during its development a crack in this blade had not achieved a critical length in spite of complete fracture of the blade. To understand causes of this situation this blade was used for quantitative fractographic analysis with the use of scanning electron microscope.

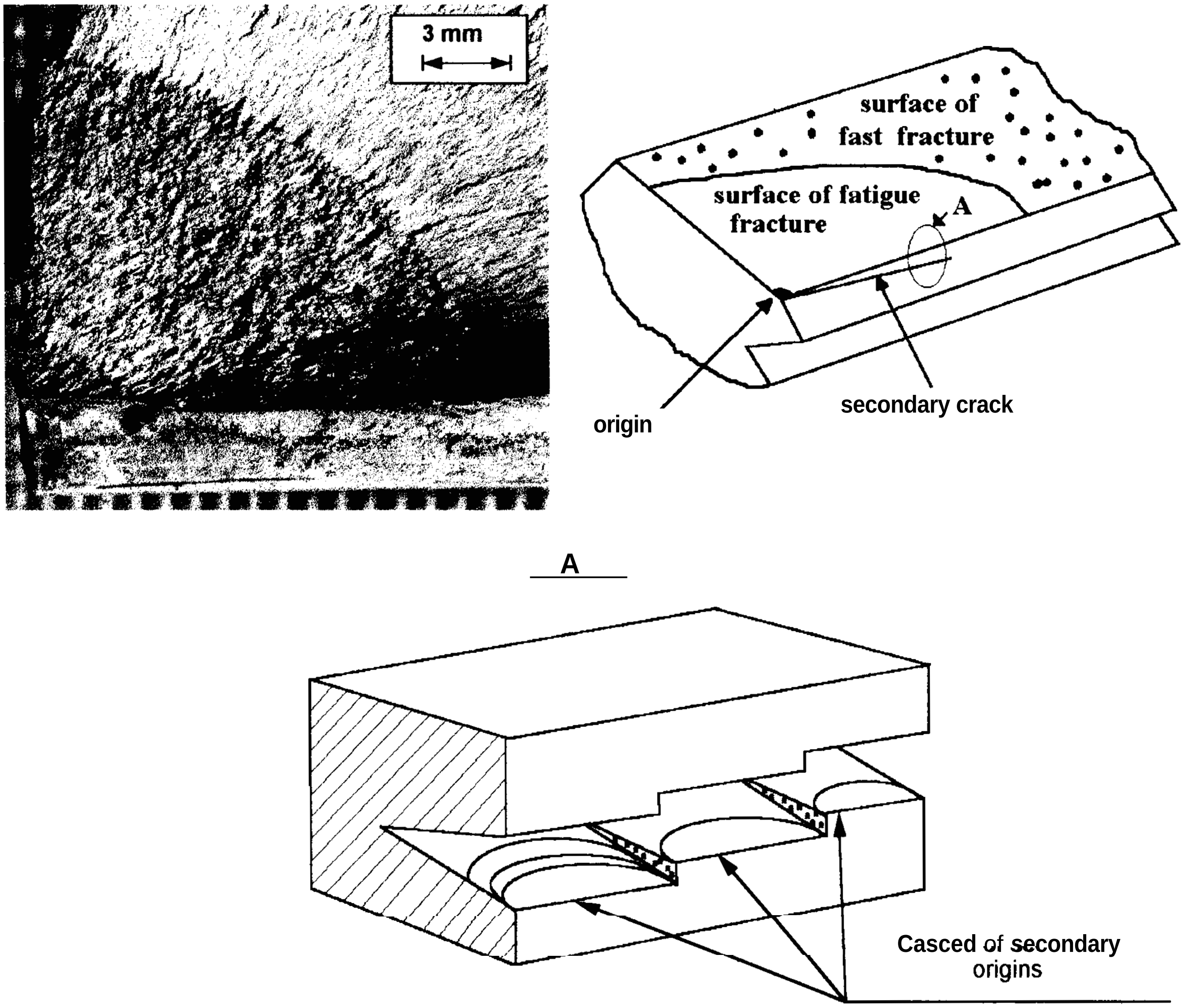



**Figure 2:** Fracture surface (a) of the secondary failed blade and (b) its schematised presentation

The performed analysis have shown regular surface pattern formation as a result of transgranular fatigue crack growth with very low rates without fatigue striations, as shown in Fig.3. The discovered pattern is typical for Ti-alloys with two-phase globular structure fatigued in area of high cycle fatigue (HCF) with rates of crack growth less than $5x10^{-8}$ m/cycle [3]. This fracture relief have not change in the crack growth direction from origin to the border of the discreet transition to the dimpled relief of the blade fast fracture. No place with fatigue striations was seen on the fatigue surface. The border has typical sign of the stretched zone, as shown in Fig.3, that reflected the dramatic increase of external loads during progressive crack development under regular cyclic loads. ***So,*** the maximum length was not reached for one blade and it was reached for another one. Consequently one of the blade was fatigued first and its crack have reached a critical length under regular cyclic loads. Then this blade have failed and its separated part crashed into the second blade which to this moment had the fatigue crack with the length near to 20mm and broke it down. The artificially broken blade was fixed inside the engine but the first blade have gone out.

Because of this cascade of events the secondary broken blade was used to estimate minimum value of the crack growth period and to calculate stress equivalent under which the fatigue crack have grown.

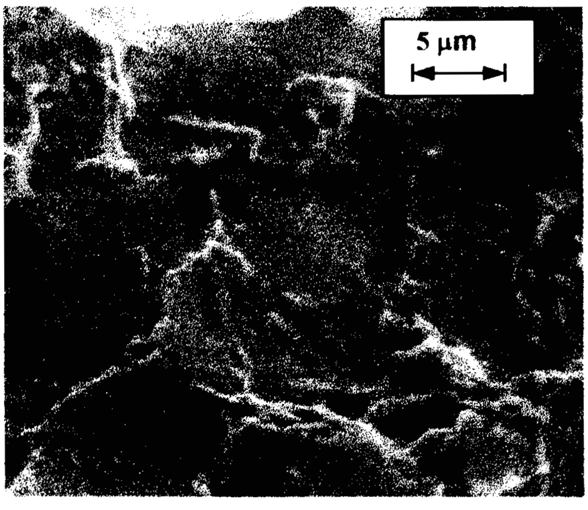
***Crack growth period and stress equivalent***

According to introduced by Shaniavski [4] synergetical approach to fatigue fracture processes analyses, a metal with a growing crack represents an open dynamic system, which is far from equilibrium. As cyclic loading of a construction element is continuing, mechanisms of damage accumulation replace one another sequentially, each starting and keeping on for a certain time. Then, the contribution of a new mechanism may alternatively grow or cease. ***An*** open system evolves by passing through the critical states, referred to as the bifurcation points, to, alternatively, a stability or instability condition. **As** long as the system experiences fluctuations, it cannot avoid instability immediately before a bifurcation point. The newly activated processes of damage accumulation develop or, alternatively, die out, depending on whether the system is able to the self-organized absorption of energy in the ways that shift the construction element toward a greater stability, i.e. longer life times. The general principle of self-organization is that altering the parameters of the extrinsic (applied) influence (the kind or way of cyclic loading) results not in the occurrence of the structure hierarchy but rather in switching on the fracture mechanisms operative in the system. Depending on the conditions of cyclic loading the mechanisms may be complicated to a greater or smaller extent, and the system, change from the lower to higher level of self-organization.

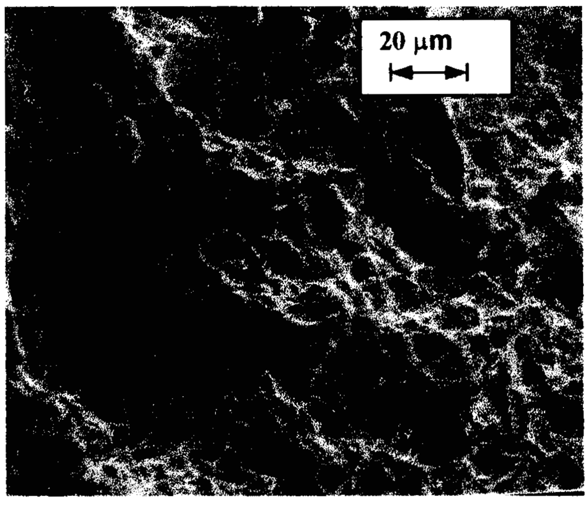
There may be various causes, say, change of the of loading system (e.g.,by transition from uniaxial to biaxial loading), level, frequency, ***etc.,*** for a fracture mechanism to get altered by bifurcation. Yet whichever deviation of the loading conditions, the resultant features of the fracture surface morphology, reflective of the self-organization steps, are fully intrinsic to a material, i.e., not extrinsically formed. Non of the material volumes is originally informed on the conditions to be experienced next and, therefore on the kind of sequence of the forthcoming fracture mechanisms. Yet it strictly follows the intrinsic mechanism of crack-growth, permitent to a certain range of the crack growth rates. This concept is methodological ground for ascertaining fractographically the level of the equivalent stresses in the elements of aviation constructions, Shaniavski [5].

Whatever the conditions or mode of cyclic loading, the achievement of definite crack growth rate, *viz.* $5x10^{-8}$ m/cycle and $2.1x10^{-7}$ m/cycle, sequentially controlled the transitions from Stage I to Stage II and from the Substage II*a* to Substage IIb, respectively. It may happen that the in-service loading conditions prevent passing from Stage I to Stage II development of fracture. This situation is found typical for investigated blades made of the titanium alloy VT3-1. It was the case of high stress ratio, R, cycling: fatigue striations do not form, Shaniavski [3], with R greater than 0.8, and the faceted pattern only forms indicating to the predominance of slip, as shown in Fig.3.

The value of $5x10^{-8}$ m/cycle was used in calculations as the maximum value of growth rate during all period of in- service fatigue crack growth notion about unified description of fatigue cracks growth for Ti-, Fe-, Al-




a)




b)

**Figure 3:** Faceted pattern (a) fracture surface and (b) area of the border between fatigue and fast fracture zones of the fatigued blade

based alloys on the basis of synergetic approach, Shaniavski [4,5]. Supposing that real growth rates were less than this growth rate value, the crack growth period in a number of cycles can be estimated by the relation:

$$N_p = \Delta a / S_{max} = 0.02/5\text{x}10^{-8} = 400{,}000 \text{ cycles} \quad (1)$$

In Eq. 1, $N_p$ - crack growth period, Aa - crack growth length, $\delta_{max}$ - maximum value of the crack growth rate. Last flight of the aircraft was performed during 12 min. For the maximum frequency of the blade of 200 **Hz** the duration of blade stressing under cyclic loads correlated to 144,000 cycles. Therefore the maximum number of cycles that blade could not exceed during last flight is three times less than the number of cycles needed for crack development with constant maximum growth rate value of $5\text{x}10^{-8}$ m/cycle.

The performed estimation of the fatigue crack growth period was based on the supposition that the maximum frequency of cyclic loads was realised during all time of engine operation in last flight from start to finish. In reality engine operation have variations in a number of rotations per minute. Therefore the crack growth can be developed only at some intervals of engine operation when the maximum frequency take place in flights. ***So*** in last flight the real period of crack growth was even much shorter and absolutely not comparable with that required for observed crack length. Hence it is evident that the fatigue crack growth in each blade was performed during long time and take place not in last flight but in several flights.

This conclusion was checked by the stress equivalent, $\sigma_e$, calculation. The simplest relation for this calculation was used in the next form:

$$\sigma_e = K_{fc} / \sqrt{\pi . \Delta a} \quad (2)$$

The maximum value of the stress intensity factor maximum $K_{fc}$ for Ti-alloy VT3-1 is near to 64 $MPa.\sqrt{m}$ at the stress ratio R=0. This value was used to estimate stress equivalent for the secondary failed blade. It was broken artificially. Consequently the estimated stress equivalent will be somewhat more than that which performed the fatigue crack development. ***So,*** the stress equivalent is near to 255 MPa.

The crack development through the blade tail was performed under stress ratio more than R=0.6. In this area of the high stress ratio for Ti-alloys VT3-1 and VT8 the stress amplitude can be estimated from the relation presented by Shaniavski [3]:

$$\Delta\sigma = \sigma_{max}(1 - K) \quad (3)$$

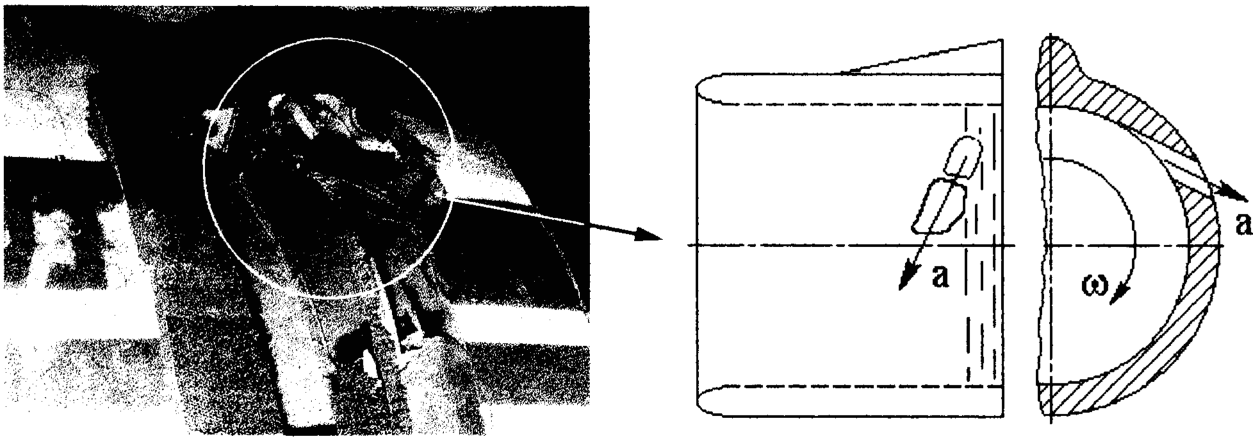
The estimated stress amplitude, $\Delta\sigma$, from the relation *(3)* is near to $100MPa$ for R=0.6 and the maximum stress value is $\sigma_{max} = 255MPa$.

The blades were tested during design to withstand the engine regimes when blades have the maximum cyclic load amplitude. Tests have shown maximum value of $\Delta\sigma / 2 = 50$MPa with the frequency of the blade near to 200Hz. It should be pointed out that the resonance frequency for this type of blades is near to 100**Hz**.

***So,*** the performed quantitative fractographic analysis have shown that failures of fan blades were realised under cyclic loads without violations of engine operation in the last flight. The first blade was broken because critical length of the fatigue crack have been reached in flight. This blade have broken the second blade which had the fatigue crack in its tail at this moment. All events with blades failures were performed in usual conditions of the engine operation.

One year after the accident the air intake of engine was found and investigated to understand the real cascade of events for blades failures.

It was found that a loss of the air intake in flight was a result of fan blades failures, as shown in Fig.4. Air intake has damages only from failed fan blades parts. There were no traces of any foreign objects which may enter the air intake in flight and cause its malfunctioning and disturb an engine operation. **A** loss of air intake




a) b)

**Figure 4:** Place of air intake to engine (a) with a hole and (b) schema of its damage because of the failed blade

in flight was caused by failure of all the bolts which fasten air intake to engine because of very high energy of flying parts of broken fan blades.

Therefore the performed quantitative fractographic analysis with estimations of the fatigue crack growth period and stress amplitude on the basis of the introduced synergetic approach to fatigue failures investigations have shown the realistic results for determination of events cascade with the fan blades of the aircraft engine **D-18.**

To minimize a possibility of such a failure of fan blades of **D-18** in future service it was recommended to avoid during engine operation a continuous use of the regimes with maximum vibration stresses in blades.

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