SHORT FATIGUE CRACKS OF IN-SERVICE FATIGUED TURBINE BLADES

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ABSTRACT

It will be discussed features of the in-service short crack growth (SCG) through the foil base of the turbine blades manufactured from the superalloy GS6K. There were several cases of in-flight fatigue failures turbine blades at the airplane have flown (500-1500) hours. The blade frequency during flight under the biaxial cyclic loads of bending-torsion is approx. 4kHz. So, the material in-flight fatigue failure took place in VHCF (very-high-cycle-fatigue) in the range of (0.7-2.0) x10^{10} cycles. There was during flight temperature neat to the 500^0C around the blade volume where the fatigue fracture process was performed.

The paper-reviewed cases studied of the blade fatigue failures and discussed the SCG rate in a number of flights. The short and long crack growth rates dependences on the crack length were analyzed and the unified description of the fatigue crack growth was demonstrated. The stress equivalent for the crack resistance in VCHF area was calculated on the basis of the well-known Murakami's equation and relations introduced by McEvely and Endo for the case of biaxial material stress-state.

Introduction

During the operating period of an engine, growing fatigue cracks of different origins are frequent in turbine and compressor blades. These cracks may arise from the sites of damage—dents, tears, and bends—inflicted on an airfoil by a extraneous matter. They can affect the resonant oscillations of thus damaged blade. For a short period, the blade oscillates with a frequency typical of transient operation conditions of the engine. Once a site of highly concentrated stress exists in a blade, it helps to faster initiation and propagation of a fatigue crack. A situation like that can be the case in the different periods of an engine service.

Nearly normal dispersion pattern is typical of the fatigue lives as of the crack-initiation periods for aviation-structure components in case that they are damaged naturally—by routinely applied cyclic loading. Here, we identify a single loading cycle with a complex of loads applied in a period between the starting and shutting down of the engine. Hence, the single loading cycle involves a complex of loads that differ in the deviation amplitudes and average levels. However, being aware of the chance that the crack can grow to the critical size, one should never describe a crack-initiation period in terms of a traditional statistic approach in case that cracking was initiated by an occasional artificial damage in service. Instead, growth period of a fatigue crack may only be used as a flight-safety criterion so that the critical state of a cracked blade is never achieved in flight.

They measure the growth period of a fatigue crack in blades by the number of flights or of the engine. Each cycle leaves behind the macroscopic or mesoscopic fatigue lines, depending on the nature of damage that the loads applied during the flight produce [1]. A crack only propagates in a blade during a short part of a start-and-stop cycle of the engine. Then it does not advance during the flight unless the blade enters the resonance again. This long-term rest of a crack is marked by a step on the fracture surface, which shows itself as a macroscopic or mesoscopic line, depending on the degree of damage. A line itself with a plateau between this and the next adjacent line is characteristic of the damage caused by a one-flight cycle.

The blade fracture can exhibit a block of beach marks instead of a single beach mark. The number the beach marks depend on how many times in a flight the blades experience resonance. The number and formation pattern of the lines repeat along the crack path. The line blocks are self-similar geometrically, which makes it easy to express the crack-growth period in the number of one-flight loading cycles as equal to the number of the fatigue-line blocks.

In service, the turbine blades typically experience damage with greater permanence than the compressor blades do. In fact, the turbine blades are permanently heated and statically stretched by the centrifugal forces, controlled by the revolution speed of the rotor. Under such conditions, thermally activated creep-fatigue effects become possible within a one-flight loading cycle. Each of the mechanisms controlling the blade durability operates during its own finite period of time. This means that the strength criteria, controlling the failure, will differ in nature, depending on the service-life stage at that a blade fails. Consequently, the service-life dispersion can reveal several maximums (instead of one), each maximum corresponding to a certain dominating fracture mechanism sequentially replacing one another during the service period.

The blades of compressors and turbines of the gas-turbine engines (GTE) normally experience dynamic tensile stresses, caused by rotor revolutions, together with dynamic bending and torsion effects of the gas flow. The blades oscillate with a frequency that varies along the blade to create biaxial stressed states. The blades of different stages differ in their natural oscillation frequencies. The latter change from several hindered oscillations per second for the first-stage fan blades to several thousand, for the last-stage blades of a compressor.

Heating of the blade material is a factor of special importance as long as in this case the fracture mechanisms controlling damage and subsequent crack propagation may both vary depending on temperature.

In service of the engine M-601 were seen in-flight fatigue failures of turbine blades of superalloy GS6K (Figure1). The fatigued blades of various engines had flown in the range of [370-1670] hours at the moment of their failure. The introduced frequency for the blade in-service loading was approx. 4 kHz. That is why the lifetime to failure for the fatigued blades took place in the very-high-cycle-fatigue (VHCF) area: (4000x3600x[370-1670])=(0.53-2.4)x10^{10} cycles.

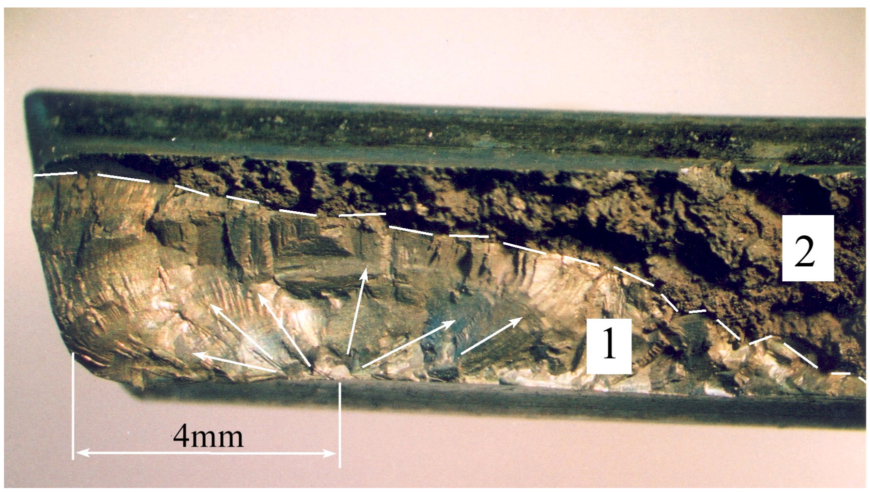

Figure 1. Overview of the fatigue fracture surface one of the investigated blade of the aircraft engine M-601E. The border between (1) fatigue and (2) fast fracture have shown by dashed line.

At the beginning of the blades service their fatigue failures were initiated by the material defects. To decrease probability of defects formation during manufacturing procedure there were introduced improvements in the technological cycle but the rare blades fatigue failure took as formerly place. After the improvements there was not seen defects in the point of the fatigue crack origination of the turbine blades.

That is why the fatigue failure intensive analysis was performed to answer the question about the real cause of the crack origination in the blade after the technological improvements, and to estimate period of the crack propagation. Below more detailed analysis of the crack path was performed for the in-service fatigued blade that had flown 1493 hours or 1084 flights.

Fatigue crack origination and short crack growth

The point of the fatigue crack origination in the investigated blade was under the surface in the distance of approx. 50 μm and has placed at the grain boundary, Figure 2. Inside of the blade was performed the area “1” during first stage of the material cracking from the point of origin.

The first stage of the crack propagation under the surface took place in the distance near to 4mm from the blade out edge. The area “1” was performed because of quasi-cleavage of two grains. The flat facet was produced because the slipping plane broke one grain. The stepped surface was performed because of cracking another grain by the slip bands. Both surfaces were performed as a result of twinning process that schematically shown in Fig 2. There was the rotation process in the blade material inside of the slip bands as the dominant manner of the free surface formation.

The crack growth takes place on border of grains by a natural way without any of influence on this process of flaws of a material. The select of a facet of initial fracture of a material at this or that stage of a loading in limits of natural straggling of its fatigue performances is determined by local inhomogeneity of allocation of chemical devices of an alloy, that follows carried out X-ray spectral analysis. For more homogeneous material the fracture can take place at the greater operating time of a blade at minor removal from 10^{10} cycles.

The second stage of the short crack growth can be seen immediately after the area “1”. Outside of the first area of the origin the crack propagation happened under the cyclic loads first of all predominantly in the deep of the blade section. Thus there was a turn in the direction of the crack growth, so that after the crack propagation to the deep its front was torn so that to become almost in bridge surfaces at the approach to her. Therefore on a small distance from an initial facet of fracture already on 1.0-1.5 mm the crack drastic change of the growth rate is began as a result of jump increase at the moment of an output of the crack front.

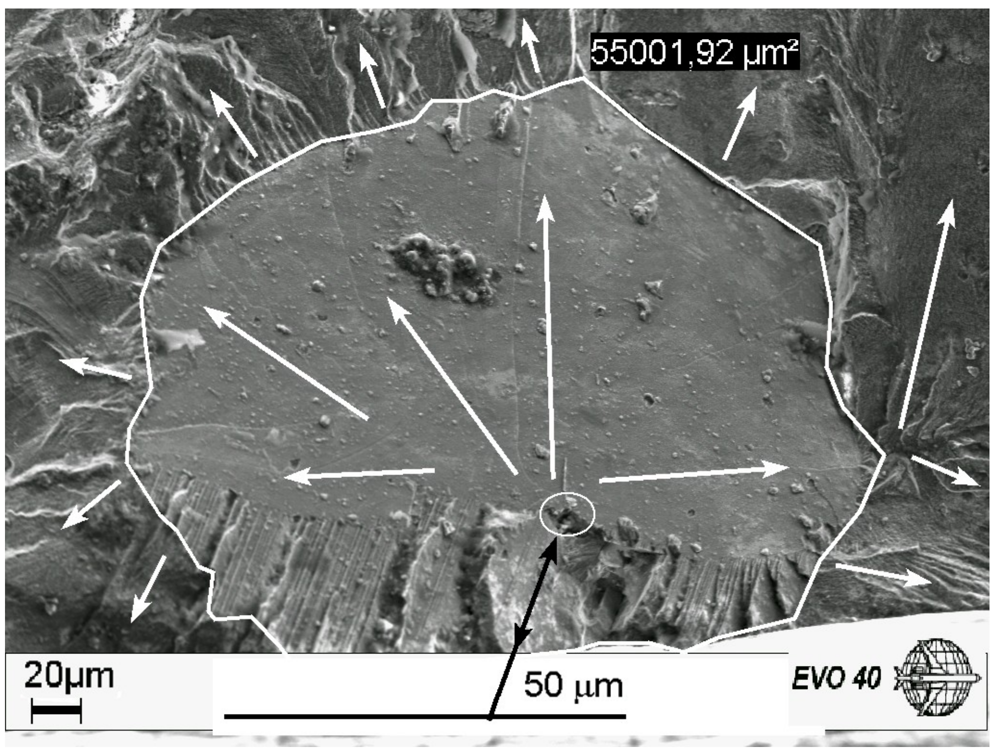

Figure 2. The area “1” of the fatigue crack origination under the surface of the blade which have shown in Fig.1.

The well-known pattern as meso-beach marks were performed on the fracture surface because of material simultaneously cracking under mode I and II, Figure 3. The meso-tunneling process was dominant manner of the material cracking and meso-lines were performed inside of the tunnels. The meso-beach marks spacing had monotonic increase in the crack growth direction along the blade surface in the left and the right hand from the area “1”, as shown in Figure 4.
The meso-line formation process had so regular manner that the several meso-beach marks in a number of 8-12 can be systematised as one block (see Figure 3). The block meso-lines reflects the blade material reaction on the cyclic loads variation during fatigue crack propagation in one flight [1].

The long crack growth period

This regularity can be seen on the basis of the results of the simultaneously estimated fatigue cracking process for the short and the long crack propagation, for instance, in the right hand from the area “1”, Figure 5. Each meso-beach marks spacing was measured only for the short crack. The only big spacing value for the long crack, as it indicated in Figure 5 by white points, was measured in the distance more than 2mm. Then the ten divided the measured spacing value and the calculated by this manner the meso-beach marks spacing was used for the dependence shown in Figure 5. This operation was used because the block of the small meso-beach marks was not seen in many places on the fracture surface for a long crack. As a result of the measurement and calculations the unified dependence of the small meso-beach marks spacing on the crack length was discovered. The ratio one to one between number of small meso-beach marks blocks and number of the blade flights can be introduced from the discovered dependence.

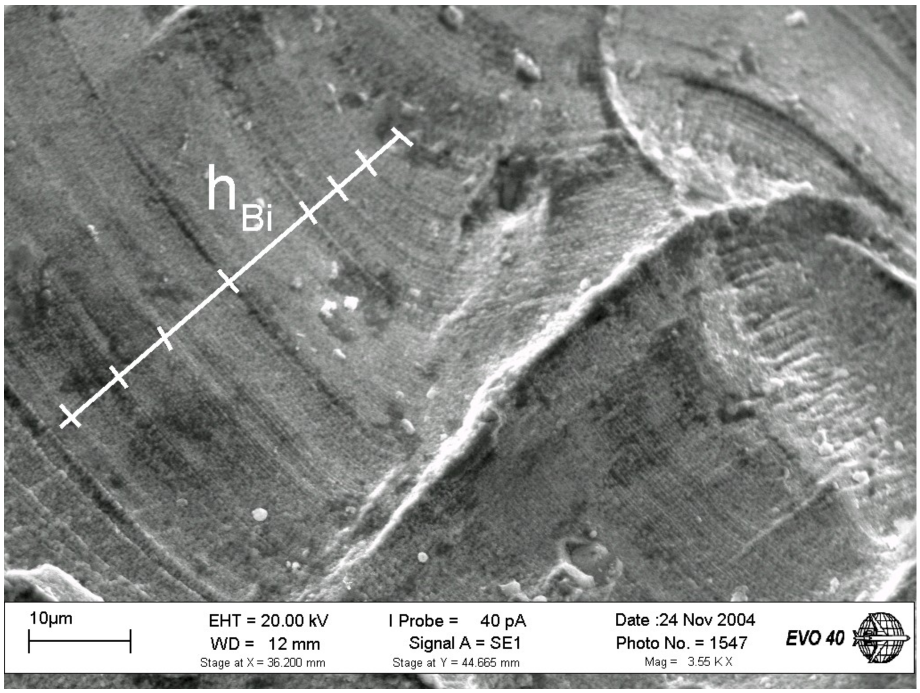

Figure 3. The fracture area with meso-beach-mark spacing placed in the left hand direction in the distance 50 µm from the boundary of the area "1". Block of meso-beach marks is marked by h_B.

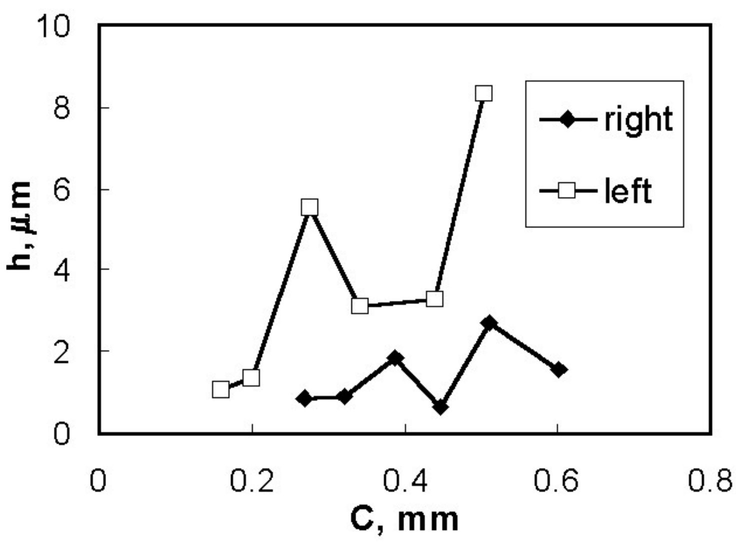

Figure 4. The meso-beach-mark spacing, h, vs. the short crack length for the left and rite hand direction from the semi-elliptical origin border respectively.

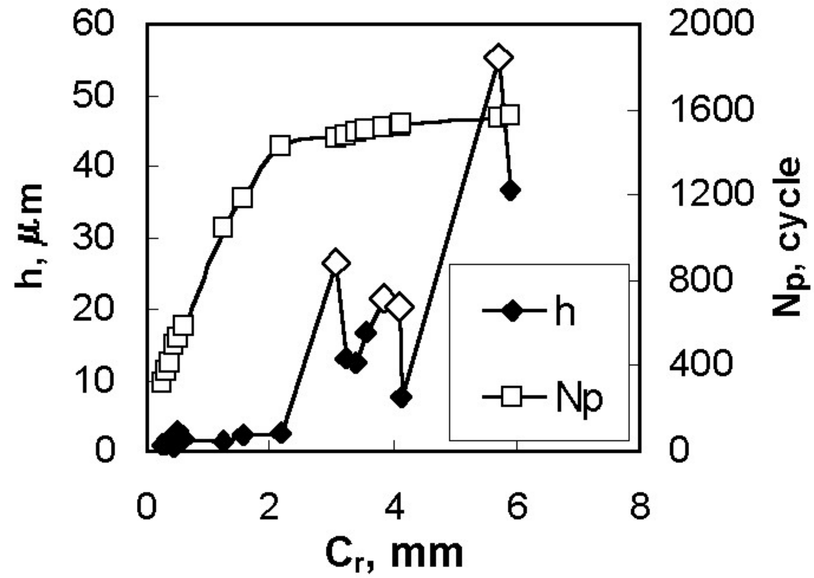

Figure 5. The meso-beach marks spacing, h, vs. the crack length in the rite hand direction from the area «1». White points indicated the measurements of the block beach marks divided by ten.

The carried out analysis of structure meso-beach marks in a median part of section of fracture near an area of the fast fracture of the blade has shown, that there is regular pattern of blocks meso-beach marks of two types. In one case it is series disposed twin meso-beach marks, inside which one more small-sized are hardly appreciable meso-beach marks. In another case it is blocks of several (approx. 10-15) meso-beach marks of a close spacing. So, as stated above, blocks in limits 8-12 meso-beach marks is marked. So, the defined crack growth duration on the basis of the spacing meso-beach marks spacing, h, should be transferred in duration of crack growth on number of flights aircraft components by division it value on 10.

The calculated number of the blade flights was 120 and 160 for the crack propagation from the area "1" in the left and the right hand respectively. The difference of the estimated values can be seen because the crack length in the right hand was longer than in the left hand on the 2mm. The discovered crack growth period for the investigated blade has not principal difference with the earlier established interval 80-200 flights for in-service fatigued blades of superalloys which failures took place at a number of cycles more than 109 flights [1].

The ratio N_p/N_f between crack growth period and lifetime to failure for the investigated blade was [160/1084]x100%= 14.7%. Consequently, in area of the VHCF the ratio N_p/N_f can have approximately the same value that in the HCF area.

Stress equivalent for fatigued blades.

The blade stress state was biaxial tension in the section of the blade fatigue failure with principal stress ratio near to σ_2/σ_1 =0.3. This value was used to estimate the stress equivalent value on the basis of the database taken from the paper [2]. There were introduced ratios between flaw size, a, and the stress equivalent value, $(\sigma_e)_1$, at different biaxial stress ratios in the paper. The investigated blade has the fatigue fracture area "1" under the surface with the size a = 50μm. This size was used to estimate the stress equivalent. The performed estimation of the stress equivalent by the dependencies, which gave the value near to 230MPa.

For evaluating the fatigue limit quantitatively, the $(area)^{1/2}$ parameter model proposed by Murakami et al. [3] and defects are combined in the equation:

$$\sigma_W = 1.56(HV + 120)/(\sqrt{area})^{1/6} \tag{1}$$

The model can be used when $(area)^{1/2}$<1000 μm. In the invested case of the blade fatigue failure parameters of the Equation (1) are Hv=430 and (the fatigue fracture area "1")$^{1/2}$ = 235 μm. That is why calculated value of the equivalent stress is near to

$$\sigma_W = 1.56.(430+120)/(\sqrt{55002})^{1/6} = 299 M\Pi a \quad (2)$$

The same calculation of the stress equivalent by the Equation (1) was performed for other two blades (№2, №3) fatigued in service at 1669 и 370 hours. It was shown that the stress value for blades №2 and №3 is 317 MPa and 345 MPa respectively. The mean value of the stress equivalent for the three fatigued blades was 320 MPa.

The superalloy GC6K has the fatigue limit 200 MPa at the number of cycles 2.108. This value is near to the calculated value 230 MPa, and in 1.5 times less the value that calculated by the Equation (1).

The designed stress equivalent for the blade was near to 48 MPa under the cyclic loads with the introduced frequency 4.1-4.2 kHz. This value is near to 5 times less than calculated by the introduced ratios shown in Figure 5, and is near to 7 times less that the value calculated by the Equation (1).

To understand the contradiction between estimations of the stress equivalent value and designed stress value the fatigue diagram of the GS6K superalloy was analysed to simulate the fatigue resistance of the alloy in the area of lifetime to failure more than 10^8 cycles, Figure6. The new fatigue diagram was constructed on the basis of the known tests data of the smooth and notched specimens (points in Figure 6) at the temperature 950^0C.

The data were approximated in the area of VHCF (dashed line in Figure 6). The discovered fatigue curve was shifted in the right hand (uninterrupted line in Figure 6) in area of lifetime to failure in 10 times more. The discovered stress equivalent value by the introduced manner was near to 180 MPa. Therefore, the performed calculation by the three methods has shown approximately the same result. The stress equivalent for the in-service fatigued blades was in the range of 180-200 MPa that in several times is more that was measured in tests for introduce cyclic loads frequency 4.1-4.2 kHz. This result can be explained in the next manner.

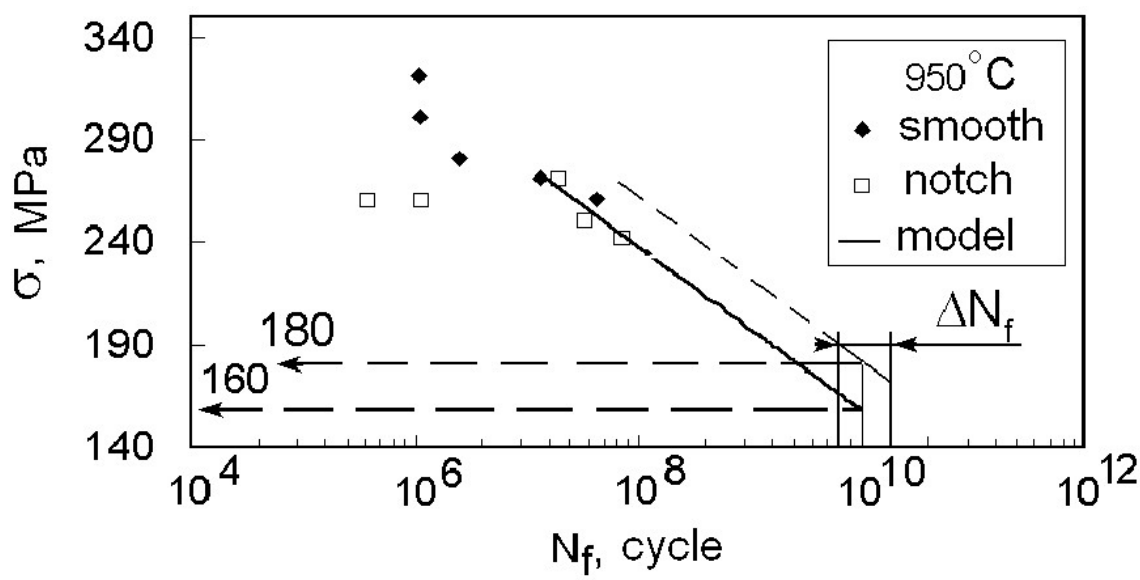

Figure 6. The fatigue lifetime to failure, N_f, against the cyclic stress, σ. Points show the test results for smooth and notched specimens of the superalloy GS6K fatigued at 950^0C. The dark line shows the test results approximation in the VHCF area, and the dashed line shows the corrected test results approximation those location was shifted in the rite hand on the value of $\Delta N_f = 5.N_f$ on the basis of the test results have taken from the review [4]

The in-service fatigued blades subjected to high stress level because of the resonance regime. The main frequency for blades is in the range of 4420-4520 kHr. The in-service blade subjected to cyclic loads in the range of frequency 4050-4200 kHr. Both frequency ranges were discovered from the frequency measurement. The method of the measurement has accuracy not less than 3%. Therefore, there can be seen for blades introduced maximum frequency [4200 + (4200x0.03)]=4326 Hr and the minimum main frequency for blades can be [4420 – (4420x0.03)] = 4288 Hr. Evidently that small part of the in-service blades can be loaded the resonance regime when the cyclic load level can be in several times more than designed value 48 MPa. The introduced and main frequency in-service coincidence is very rare situation for blades. That is why in-service fatigue failure of investigated blades is very rare event.

Conclusions

The in-service fatigue failures of the blades in VHCF area took place because of the introduced and main frequency in-service coincidence. The crack origination in the VHCF area was just below surface with the place for point of origin not far from the 50 μm. The short and long crack propagation regularities for in-service fatigued blades of GS6K superalloy were discovered and the number of cycles during fatigue crack growth period was calculated. It was shown that during the short crack growth period the blade has 8-12 times of resonant regime of cyclic loads in one flight. That is why calculated number of meso-lines was introduced in 160 flights during fatigue crack propagation. The calculated stress equivalent value from the fractographic analysis for the blade resonance regime of cyclic loads was in the range of 180-200 MPa.

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