FRACTOGRAPHIC ANALYSIS OF AIRCRAFT ENGINES COMPRESSOR DISKS FROM Ti-ALLOYS FATIGED IN SERVICE.

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Three different structural failures of the compressor disks are presented. In each case, the role of cracking by fatigue, initiated inclusions, residual stresses, and fretting-wear damage of disk surface, is discussed. The crack growth period and the stress equivalent were calculated.

INTRODUCTION

Fractographic investigations of compressor disks, made from two-phase ($\alpha+\beta$) titanium alloys, show that there are two main sources of the relief morfology wich can be observed in the region of low-cycle fatigue during the maintenance of civil aircraft engines (Ivanova (1), NTSB (2), Shanyavsky (3), Stepanov (4): fatigue striations and the two-phase lamellar structure (LS) of the titaniun alloy. Because the block loads fly-by-fly (FF) is the main source of damage for aircraft components that correlation should be known for different FF loading codition. The correlation 1/3 between FF-block and the fatigue striations was used from reference (4).

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DISKS FROM TITANIUM ALLOY Ti-6.5Al-3.3Mo-0.3Si.

The investigation was conducted on engine fan disks of the titanium alloy, for which the specificati-specification composition is as follows (%wt):(6.0-7.3)Al- (2.8-3.8)Mo - (0.2-0.4)Si - others remainder titanium. The material's microstructure consist of primary plus the transformed ($\alpha+\beta$) lamellar structure.

Two fan disks failure took place during maintenance. The disk "A" nad 1,870 flights and had flown 4,422 hours since it was new. It had 630 flights since the last visual and dye penetrant inspection. A low cycle fatigue crack was initiated at an inclusion. The initiation defect was near the surface. It was a semi-elliptical, 2c=20mm in length and near a=10mm in depth, cluster of nitride titanium inclusion. The fan disk "B" had 4,912 flights and had flown 10,148 hours since it was new. It had 372 flights since the last visual and dye penetrant inspection. The low cycle fatigue crack was initiated near the surface in the same section that it was for the disk "A". Some initiated zones were placed in the depth near a=0.2mm from the surface of the disk with a length of 2c=mm. Crack growth proceeded by low cycle fatigue from this initiated site. Any kind of defects was not registered in the disk s material.

In the crack growth direction the mixed striations and SL relief morfology were developed. The proportion of the fracture surface with fatigue striations did not exceed 50%. The striation spacing dependencies on the crack growth length, c, is shown in Fig. 1. The dependencies have difference one from another one, but the critical crack length is near the same. The calculated crack growth life in a number of striations (S) and fligths (F) for disks "A" and "B" were:7,000(S) -1,400(F) and 4,800(S)-1,000(F).

To explain the difference between those striation spacing dependencies, principles of fracture mechanics were used in this failure analysis. The theortical tool was used in the calculation of a stress equivalent σ_e, explanation can be took from reference (1), and critical flaw size which determine the inspection limits for reliable quality control inspections. The developed calculations shown the critical value $(\sigma_e)_{max}$ that was the same for the both disks and near the stress that must load disk during operation in service. Disks loading conditions were the same during all they life. Therefore the crack growth difference was a result of the high residual stresses in the disk "B". there was very intensive fatigue crack growth at the initiated site because the residual stress equivalent was $(1-0.5)(\sigma_e)_{max}$ in the crack length "c" range 2.5-5mm (See reference (3)). That stress decreased in the crack growth direction.

For both disks the microstructural and forging technologies defects initiated a low cycle fatigue crack growth. The inclusion was missed during inspection by the forging vendor and by the engine manufacturer. The fatigue cracks in both disks were also missed the operator of the airplane. The main conclusions are that tight control of ingot melting and forging technologies are required to avoid micro-structural defects which initiate low cycle fatigue cracks.

The proof test intervals are estimated to be of the order of 300 flights which is feasible for the economic life of aircraft.

DISK FROM TITANIUM ALLOY Ti-6Al-3Mo-2Cr.

The investigation was conducted on engine fan disks of the titanium alloy, for the specification composition is as follows (%wt):(5.5-7.0)Al-(2.0-3.0)Mo -(1.0-2.5)Cr-(0.15-0.4)Si - others remainder titanium. The material s microstructure consist of primary α plus the transformed (α +β) lamellar structure.

The fan disk "C" which on its 7,219 flights had flown 11,930 hours. It had 1,029 cycles since the last visual and dye penetrant inspection. The fatigue crack which was initiated near the bore edge of a fan disk led to fan disk failure. There was the fretting wear zone on the surface near the bore, and a typical fretting-fatigue crack was developed from it. The characteristic cracking behavior commonly observed under fretting suggests that mixed mode (Mode I and II) growth occurs at an early stage. Fretting cosiderably accelerate nucleation and the initial growth of cracks. That is why the crack nucleation was considerably shorter than that predicted from calculations for the lifetime by the low cycle load.

First part of a fracture zone was near the surface of the bore with a depth of a=0.5mm and a length 2c= 0.2mm. The fatigue relief was typical for a high cycle fatigue crack. Fatigue striations were not visible in that fatigue surface. Then, the relief s morfology drastically changes to striations and LS-relief took place on the fatigue surface. The striation spacing had 0.13...0.2μm near the border of the initial fracture zone and was typical for the low cycle fatigue crack. Fatigue striations were fragmentary and LS-relief was dominant in the crack growth direction.

A high cycle fatigue crack was initiated from vibrations of a blade which root was communicated with a pivot, placed into the bore. Then a low cycle fatigue crack was dominant. That is why the crack growth direction was moving in accordance to the stress-state of the disk. During the failure investigation, to determine the crack front evolution, it was decided to measure the fatigue striation spacing in all possible places of fatigue surface, and to show the local direction of crack growth. The local direction was used to determine general direction of crack growth for fatigue striation counts. The crack front evolution and the striation spacing dependence on the crack length are shwon in Fig. 2. The crack had quater-elliptic-shape, then through-thickness and semi-elliptic shapes. The crack growth rate was decreased when the transition from through to semi-elliptic shapes of its development took place. The counted striation number was 9,600 for the quater-developing, and 4,500 for the semi-elliptic crack developing. The factor 1.6 must be used to diminish the summarized striations number, because the big variation of the striation value took place in the crack growth direction. Then, the FF number, using the reducing factor 1.6, was near 2900 during lifetime of the fatigue crack growth. Consequently, the fatigue crack was missed by the operator of the airplane.

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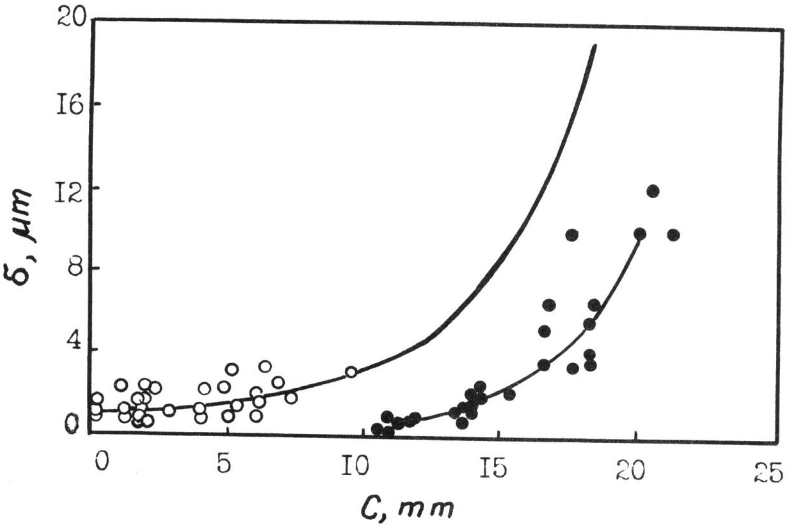

Figure 1 The striation spacing, δ, dependencies on the crack length, c, for the disks "A" (●) and "B" (o).

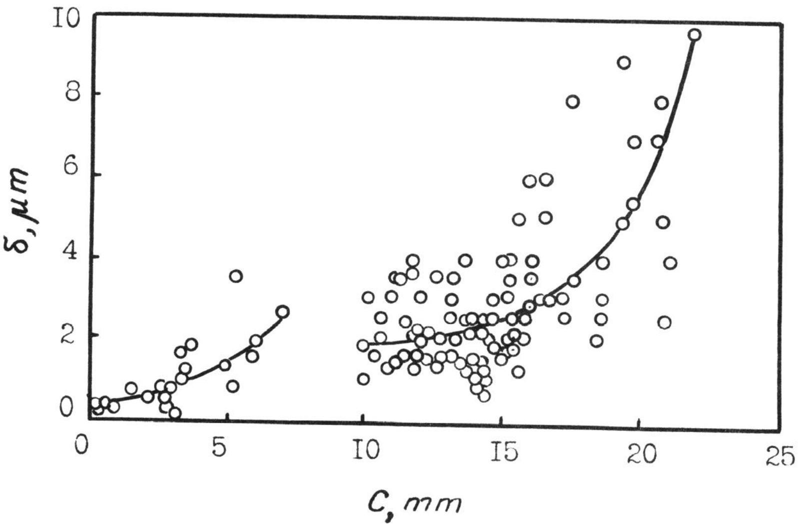

Figure 2 The striation spacing, δ, dependence on the crack length, c, for the disk "C".