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Research on the Propagation of AE Signals in Helicopter Structural Components

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***Abstract* – The paper contains results of experimental research carried out helicopter bench. In order to create an attenuation chart for AE signal amplitude in helicopter fuselage, a number of experiments were performed on the frame and stringers, inside the fuselage. Later helicopter test bench was used to develop defect localization methodology of helicopter structure fatigue damage technical diagnostics. Analysing helicopter structural defects for different helicopters types it is concluded that the joint elements of helicopter tail boom are still exposed to fatigue crack formation. AE method shows highly effective results predicting fracture of helicopter joint elements.**

***Keywords* – Acoustic emission testing, bolts, fatigue cracks, fractographic research, helicopter structure.**

I. Introduction

Acoustic emission (AE) is one of the most perspective diagnostic methods for determining aircraft damages and fatigue life, which can be used in the process of operation in real-time conditions. Acoustic emission is a process of elastic wave radiation from a material. This process causes the local dynamic rearrangement of the material’s internal structure. In comparison with other non-destructive inspection and diagnostic methods, the AE method has such advantages as high sensitivity, possibility to determine the coordinates of the defect during real-time control, etc. [2]–[5]. The aim of this research is to diagnose defects in helicopter structures during bench tests and develop criterions for the method’s approbation for operation; for this purpose, an acoustic chart of AE signal amplitude has been created.

II. Development of an Attenuation Chart for AE Signal Amplitude in Helicopter Fuselage

In order to create an attenuation chart for AE signal amplitude in helicopter fuselage, a number of experiments were performed on the frame and stringers, inside the fuselage. The experiments involve the use of 4 AE sensors. One sensor was placed on stringer No. 6 between frames No. 9 and No. 10. One sensor was placed on stringer No. 10 between frames No. 6 and No. 7. Sensors No. 3 and No. 4 were placed on the fuselage between stringers No. 7 and No. 8, frames No. 9 and No. 10, from inside and outside respectively.

Fig. 1. Location of AE sensors on helicopter fuselage.

The signal was imitated on the helicopter fuselage between stringers No. 1 and No. 22:

- Between frames No. 9 and No. 10;
- Between frames No. 10 and No. 11;
- Between frames No. 11 and No. 12.

The signal was also imitated on the helicopter fuselage on frames No. 10, No. 11, No. 12 between the stringers.

During the imitation of AE signals on the fuselage between frames No. 9 and No. 10, the arrival of signals was registered on all 4 sensors (Fig. 1). The highest amplitudes were registered in close proximity of the respective sensors. In general, the lowest amplitudes were obtained from the sensors placed on frames. All the sensors registered signals initiated by stringers No. 16 and No. 17; their amplitude level was above a set threshold of 35 dB.

During the imitation of AE signals on the fuselage between frames No. 10 and No. 11, all 4 sensors registered signals coming from stringers No. 4 to No. 12. The highest amplitudes were registered in the areas of those stringers on which the AE sensors were placed. The amplitudes of signal arrival were higher on the sensors inside the fuselage rather than on the sensors outside the fuselage at the same conditions.

During the imitation of AE signals on the stringers, from stringer No. 1 to stringer No. 22, all 4 sensors registered signals. The highest amplitudes were registered on the fuselage and stringer (Fig. 2). It should be highlighted that the sensor placed outside the fuselage at first registered higher amplitude signals than other sensors. It can be explained by the method of fixing the stringer to the fuselage – riveting. Thus, the emitted wave propagates through the rivet to the outer layer of the fuselage where it is registered by the sensor.

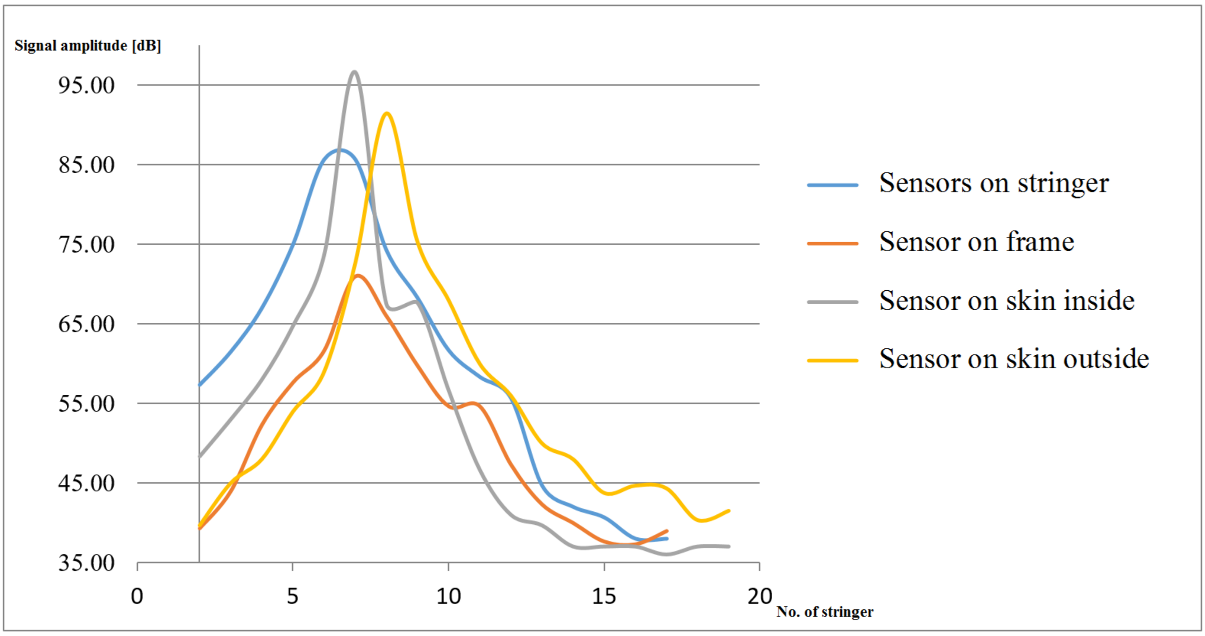


Fig. 2. Imitated AE signal amplitude dB (*y* axis), on stringers (*x* axis) No. 1 to No. 22 between frames No. 9 and No. 10.

During the imitation of AE signals on the stringers, the signal simultaneously arrived at the sensor placed on the stringer. It can be explained by the fact that in both cases the stringers crossed frame No. 10. Therefore, the acoustic wave was more likely to propagate in stringer and fuselage material up to the AE sensor. In the second case, between frames No. 10 and No. 11, the signal also simultaneously arrived at the sensor placed on the frame. This can be explained by the existing technology: stringers are attached to frames in flying direction, thus not creating any additional structural obstructions for the waves. Another aspect that promotes successful wave propagation is location of the sensor on the frame against flying direction.

During the imitation of AE signals on the stringers between frames No. 11 and No. 12, signal amplitude was more even on all sensors with a distance from the AE sensors. Besides, with a distance from the AE sensors, lower and lower amplitude signals were reaching the sensor outside the fuselage. The receiving range above 35 dB was reducing on all sensors, but the sensors on the frame and stringer were still receiving good results of wave registration up to stringer No. 13. Higher amplitude signals from stringers No. 13 and No. 14 were registered on the sensor inside the fuselage.

During the imitation of AE signals on the frames, the AE sensors on the stringer, frame and fuselage skin registered signals initiated on the frames up to stringer No. 23. The sensors inside the fuselage registered signals with an amplitude of above 35 dB up to stringer No. 18. In general, the signals had significantly lower amplitudes both on the sensor outside the fuselage and the stringer than on the sensors placed on the frame and inside the fuselage.

During the imitation of AE signals on frame No. 11, there were several linear stages with a distance from the AE sensors, like in the case described above.

The attenuation charts of AE signal amplitude were created after summarizing signal reception amplitudes (Fig. 3). They illustrate the amplitude of registered AE signals from the source located at a particular distance; moreover, only amplitudes above the 35 dB threshold have been taken into account.

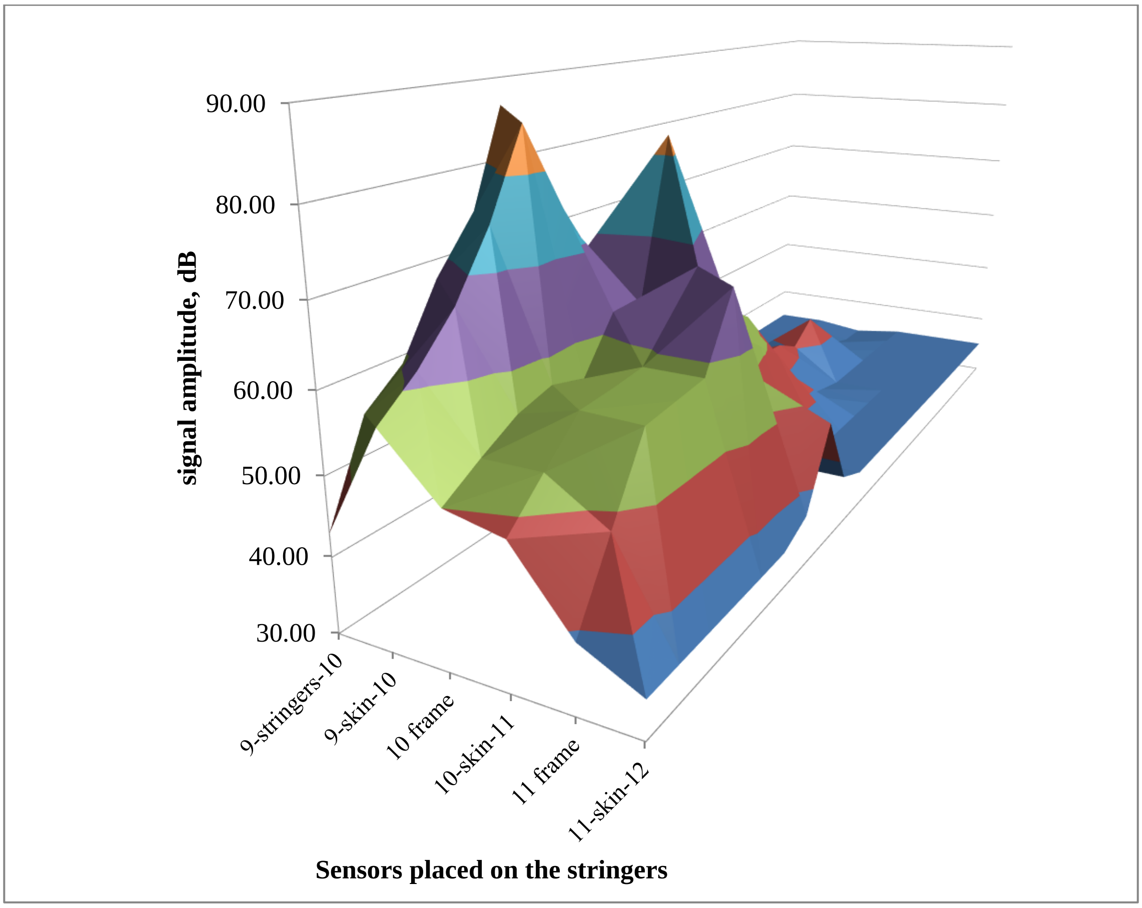


Fig. 3. Registration of AE signal amplitude, dB (*y* axis) from the sensors placed on the stringers (*x* axis).

The results of the experiments with sensors as well as helicopter tail boom and keel were used for creating a diagnostic system.

III. Test Bench for Helicopter Structural Components and Testing Methodology

The test bench used for the experiment is equipped with a system that ensures continuous control of applied loads and automatic registration of an object's stress-strain state during testing.

The load values in the fatigue test program were set on the basis of stress-strain state measurements of a helicopter sample during flight trials [10]. The load was applied to the rotor hub – F_x^H F_y^H F_z^H, stabilizer attachment points – F_y and keel – F_z^k. The frequencies of applied loads are $f_{test} \geq 0.017$ Hz to 0.1 Hz ($T = 10$ s to 60 s).

The acoustic emission measurements were carried out according to the standards and testing program [7], [8], [10]. The sensors were placed on the helicopter structure in the following way: R15α type sensors were installed on the skin of tail booms and keel from the outside (Fig. 4, Fig. 5).

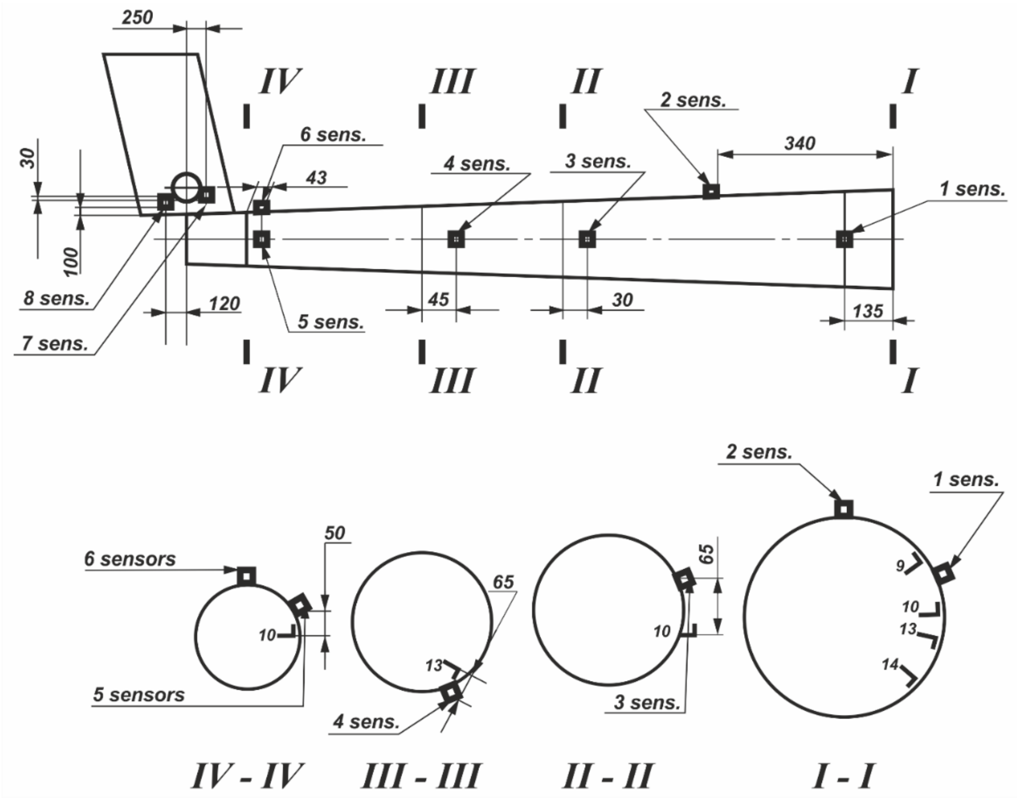


Fig. 4. Location of AE sensors on the tail boom and keel.

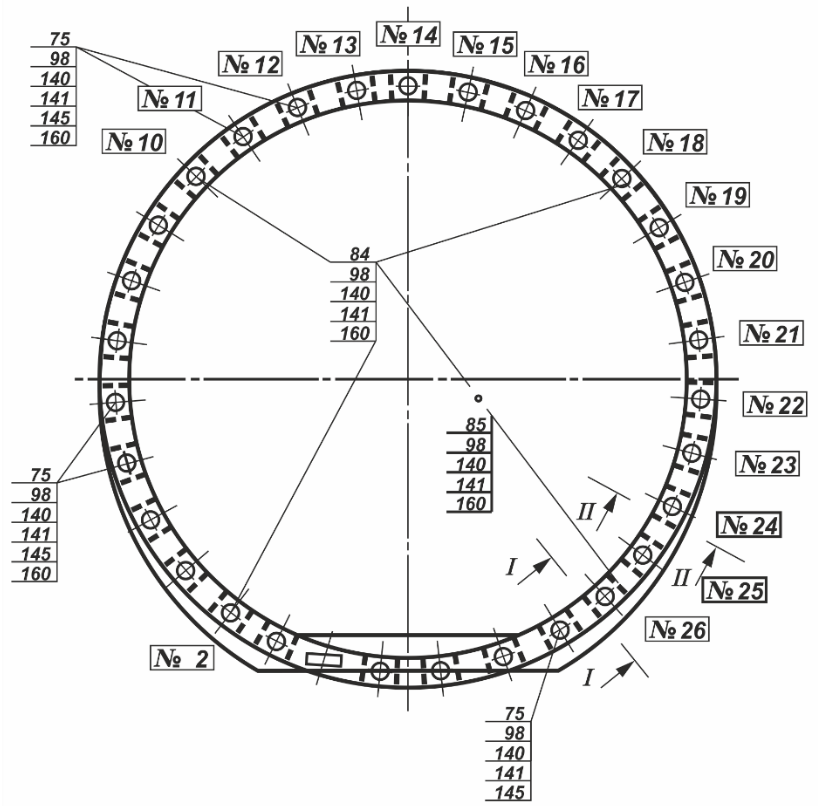
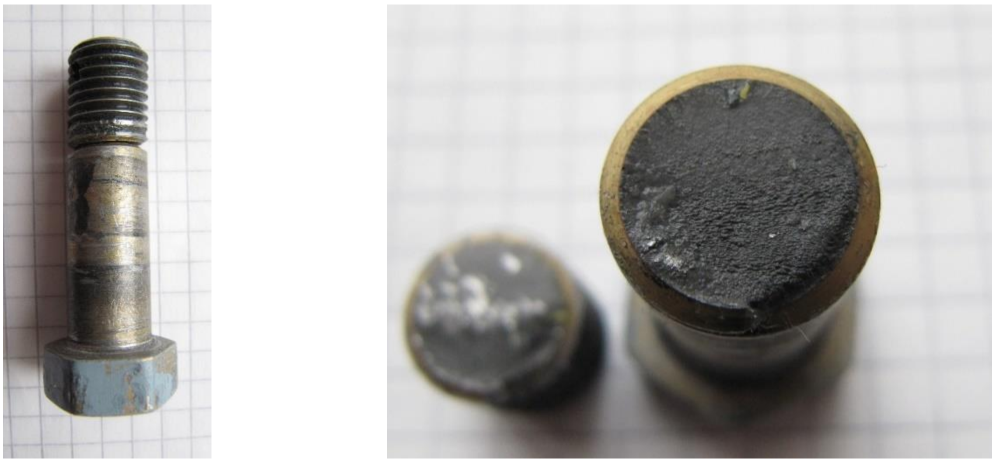


Fig. 5. Joint bolts of tail boom frame No. 1 and fuselage in flying direction.

IV. Analysis of Test Results Prior to the Failure of Bolt No. 15

In the process of inspection in 78 603 cycles, some loosening of attachment bolt No. 15 in the area of stringer No. 1 along the right side was found. The failure was detected on the basis of AE monitoring data [6] in 78 600 load cycles. To remove the bolt, the helicopter tail boom weight was relieved. The bolt appeared to be destroyed and came out without any effort.

The fracture (Fig. 7 b) has a fatigue nature [1], [4], [9]. The nucleus area of the crack is in the base of the first (from the cylindrical part of the bolt) thread turn profile at a distance of 28 mm from the head (Fig. 7 a). By the direction of lettering on the bolt head and marking on the end of its threaded part it was stated that the nucleus area of the crack is on the outer side of the bolt relative to the tail boom axis (Fig. 6).

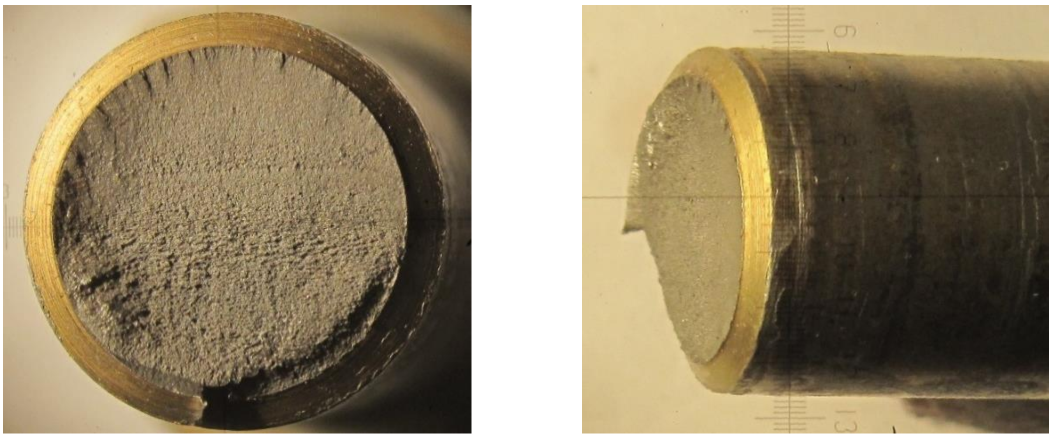


(a) (b)

Fig. 6. Bolt No. 15. (a) Overall view. (b) Fatigue fracture.

The fracture fatigue zone occupies about 95 % of the area and consists of a fine-grained site and a coarse-grained site (Fig. 7).

On a radius of 3.5 mm from the nucleus area of the crack (bolt diameter in fracture – 10 mm) a fatigue relief zone can be seen (Fig. 7).



(a) (b)

Fig. 7. (a) General and (b) side views of the bolt fatigue fracture, bolt No. 15.

The moment of failure of bolt No. 15 was registered by all the AE sensors installed in the area of tail boom and fuselage joint. The failure was detected in 78 800 load cycles. The accelerated growth of the crack in the material of bolt No. 15 was detected by the growth of AE signal amplitude from

55 dB to 62 dB (Fig. 8) in a range of 76 500 to 76 700 load cycles, i.e. in 2100 to 2300 cycles up to a complete failure, which corresponds to 42–44 flying hours.

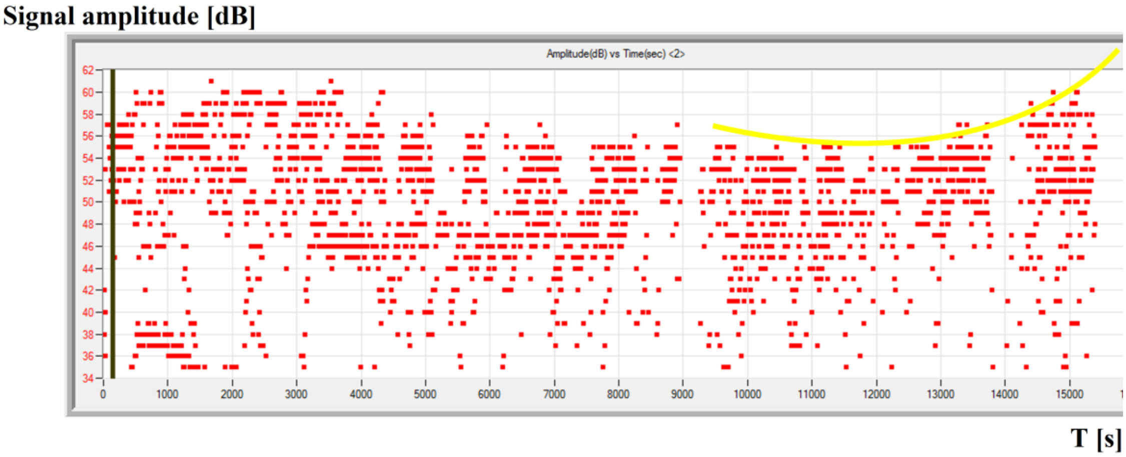


Fig. 8. Time-dependent behaviour of AE signal amplitude, dB (*y* axis) during the accelerated growth of the crack, s (*x* axis), bolt No. 15.

The behaviour of total AE signals (Fig. 9) is also an evidence of the beginning of accelerated crack growth in the material of bolt No. 15.

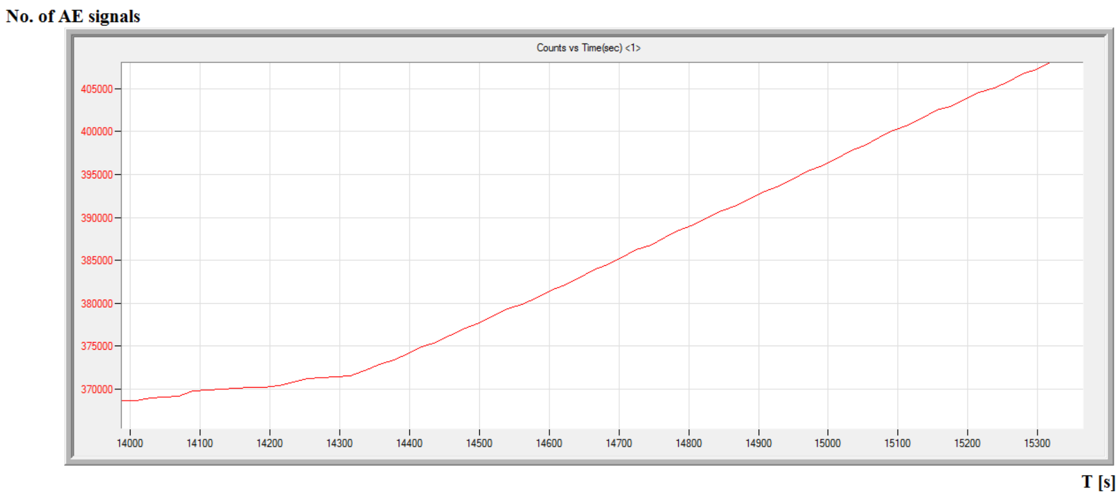
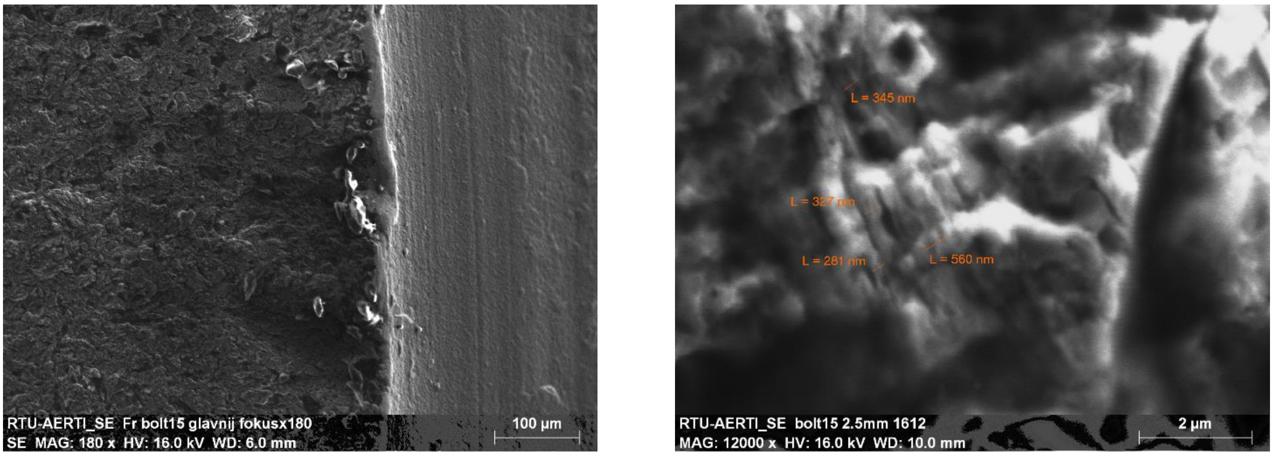


Fig. 9. Time-dependent behaviour of total AE signals (*y* axis) during the accelerated growth of the crack, s (*x* axis), bolt No. 15.

In the material of bolt No. 15, the average size of fatigue mesolines in the area on a radius of up to 3000 μm from the nucleus (Fig. 10 a) is 180 nm (Fig. 10 b). The length of the given site corresponds to 16 666 load cycles.



(a) (b)

Fig. 10. (a) Nucleus area and (b) fatigue mesolines *h* = 177 nm to 186 nm (2.5 mm from the nucleus), bolt No. 15.

Then a characteristic area of crack arrest corresponding to approximately 4000 load cycles was observed on the fracture (Fig. 11). Apparently, crack propagation stopped due to the reduction of load on the bolt as a result of bolt retightening after the failure of bolt No. 14 (at the number of load cycles equal to 69 603 [3]).

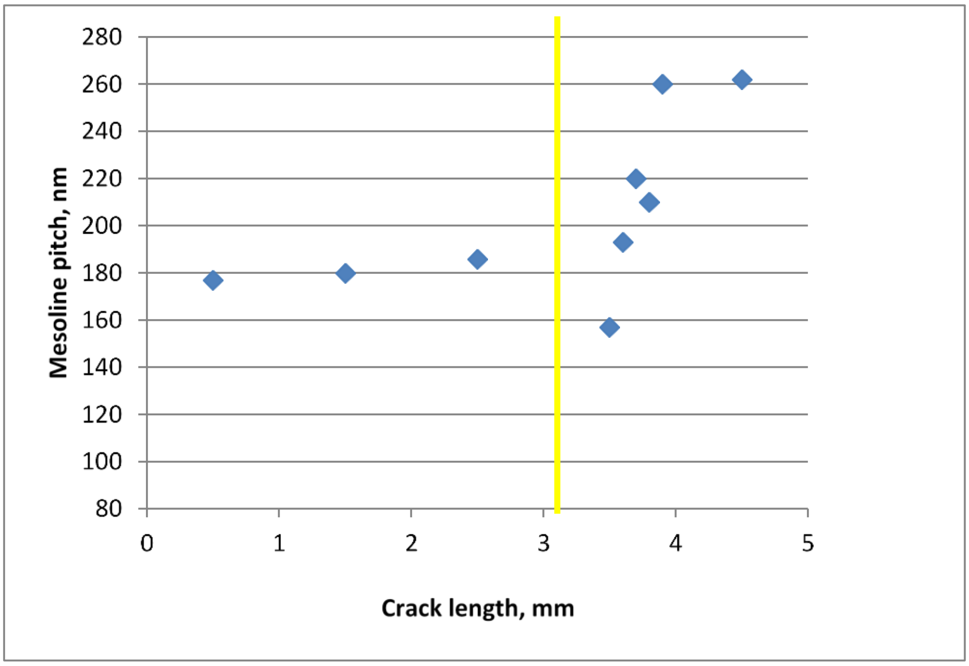
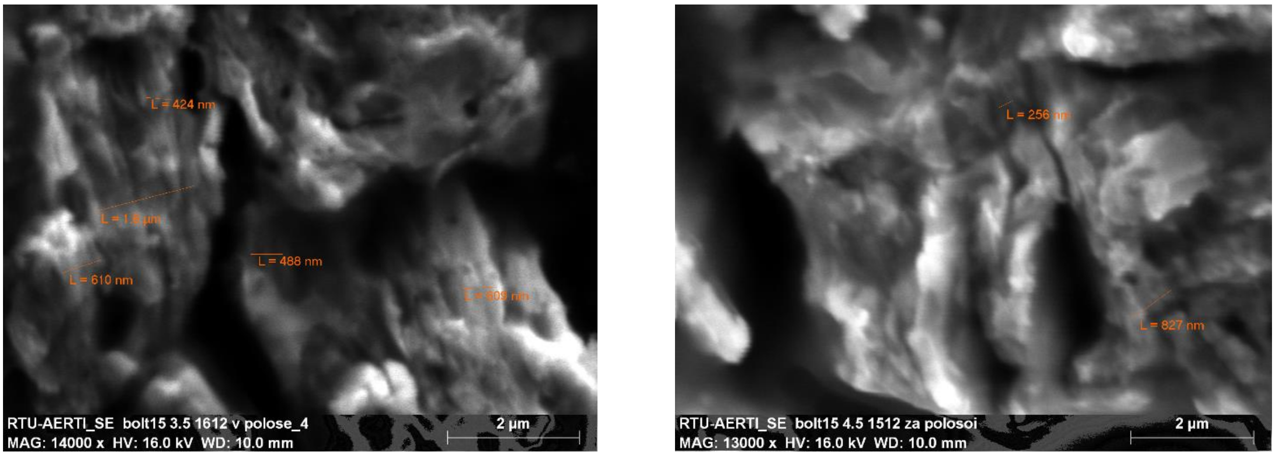


Fig. 11. Dependence of fatigue mesoline pitch increment on the length of the fatigue crack with account of retightening of bolts in the joint between the tail boom and fuselage, bolt No. 15.

Then during 8998 cycles the destruction of the bolt was taking place. At the same time the crack growth rate on a radius of 3 mm to 3.2 mm from the nucleus was on average 150 nm; then on a site with a radius from 3.2 mm to 3.5 mm it was 230 nm (Fig. 12 a), which corresponds to a total of 2637 load cycles. After reaching a radius of 3.5 mm from the nucleus, an accelerated growth of the crack was being observed during 2000 to 2100 load cycles (Fig. 12 b).



(a) (b)

Fig. 12. Fatigue mesolines, bolt No. 15. (a) h = 200 nm to 245 nm (3.5 mm from the nucleus). (b) h = 256 nm to 275 nm (4.5 mm from the nucleus).

V. Conclusion

In particular, the most advisable way of placing sensors for the testing of helicopter structural components is placing them on a certain component, i.e. when controlling the frame, sensors should be placed on the frame. The next step of the methodology is to calculate the amount of sensors depending on the geometrical parameters of the structural component using the obtained attenuation charts of AE signal amplitude. It is not recommended to place sensors outside the fuselage even when performing bench test experiments. The main task of the described experimental research is to determine intervals at which changes in AE signal amplitude occur with account of helicopter structural components.

The fractographic analysis of structural element fatigue was carried out and the correlations of AE parameters with fracture kinetics were identified. The sources of defects were detected and the multi source structure was evaluated based on the fractographic analysis. Small-grained and coarse-grained areas characterizing the crack growth kinetics were observed in bolt fatigue fractures. The crack growth rate of bolt fractures at early stages was on average 180 nm/cycle, but afterwards it reached an average of 230 nm/cycle. The accelerated growth of the crack was predicted with the help of AE equipment 42 to 44 flying hours before the fracture.

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