EARLY DETECTION OF FATIGUE CRACKS IN TURBINE AERO-ENGINE ROTOR BLADES DURING FLIGHT

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Abstract

The author shows results of research done in the Air Force Institute of Technology concerning design, development and implementation of modern diagnostic systems for aero-engines. The papers gives brief description of a project of a new advanced monitoring system basing on non-contact blade-vibration measurement.

Recent monitoring systems for engines offer a great potential to reduce the high maintenance costs of aircraft while increasing reliability and safety. These systems check for abnormal engine responses to detect failures, saving lives and reducing secondary damage to the aircraft. A phenomenon of dynamic change of an aero-engine compressor blades natural frequency in course of fatigue cracks propagation in their roots on the example of a Polish turbojet engine is described. On the ground of this phenomenon main working principles of a device, which measures vibrations of turbine engine rotor blades with application of the noninterfering discrete method (MDPh), used for early detection of first stage compressor blade cracks are discussed. Typical representation of the first stage compressor blades vibration during engine acceleration, representation of the first stage blades vibration during engine acceleration and deceleration with steady-state stator distortions, crack initiation and propagation symptoms in blades, comparison of blade vibration spectra of the same blades for different conditions are presented in the paper.

Keywords: *aero-engines, diagnostic of aircraft engine, non-contact blade-vibration measurement, damage to the aircraft*

1. Introduction

Problem of early detection of rotor blades cracks is particularly important for flight safety and economy. An early detection of fatigue root cracks in first stage compressor blades is of a great importance for flight safety due to dimensions and mass of blades. The broken blade causes as a rule the engine break-down, which in case of a one-engine aircraft might be of a formidable danger to the pilot.

Non-contact transducers mounted on the airplane engine, proceeding blade time-of-arrival measurements ceaselessly during flight and ground operation, gather useful information about blade vibrations and the engine health. Special processing of collected data may help preventing from occurrence fatigue cracks in blades or failure of engine.

Some results of investigations and devices which enable a relatively early detection of compressor blade cracks in an airborne engine as well as keeping the crew informed of this fact in order to get safely to the home base are discussed and presented in this paper.

2. Principle of blade vibrations measuring

Investigations of blade vibration dynamics are conducted as a rule by strain gauging which demands previous engine preparation i.e. channels drilling for installation of gauges. Such method is useless for diagnostics of compressor blade vibration state. In this case the noninterference discrete-phase method (MDPh) is very useful (references 1, 2, 3). Considerable development of devices functioning according to MDPh principia is noted down in professional literature. These devices are suitable for investigation of blade vibrations both synchronous and asynchronous with engine rotational speed (e.g. compressor blade vibrations caused by rotating stalls). Their essential advantage is that vibrations of all blades of the stage may be examined simultaneously. Typical vibration picture of compressor first stage blades taken by MDPh device during engine acceleration from idling to $n = n_{max}$ is presented on Fig. 1. At the beginning of engine acceleration vibrations asynchronous with rotational speed are visible (stall blade vibrations).

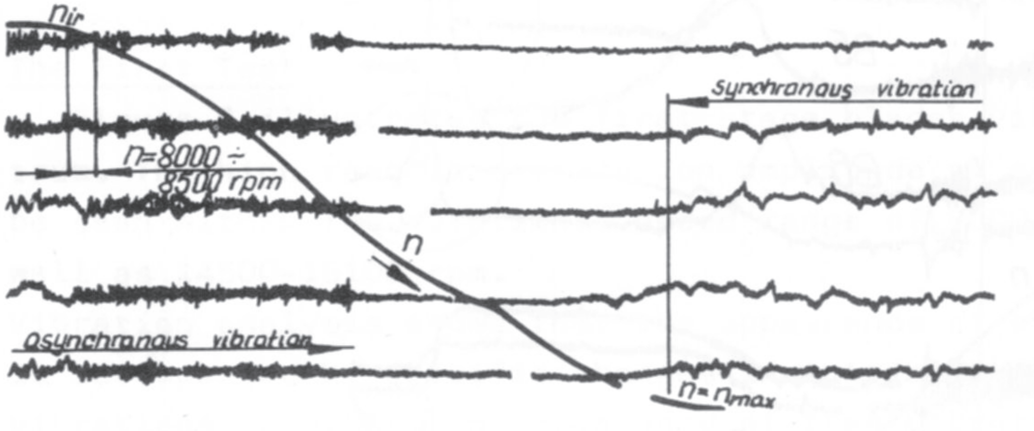

Fig. 1. Typical representation of the first stage compressor blades vibration during engine acceleration

Within the speed range from 8000 to 8500 rpm blade resonance caused by third harmonic of rotational speed is visible. At the end of acceleration phase synchronous blade vibrations (of the first flexural mode) caused by second harmonic of rotational speed can be seen; within the range from 11000 to 14500 rpm blades vibrations are minimal.

It was investigated that gentle blade vibrations (and stresses) picture observed within the whole speed range is disturbed by ingested foreign objects found on the stator blades (e.g. bird remains). In this case a part of the blades which has previously had only weak resonance is subjected to heavy fluctuation of forces (synchronous with rotational speed). The blades resonance vibrations are heavily amplified (Fig. 2). Stresses within blades are increased along with the amplitude of vibrations which during long flight may lead to initiation and propagation of fatigue cracks in the blades roots.

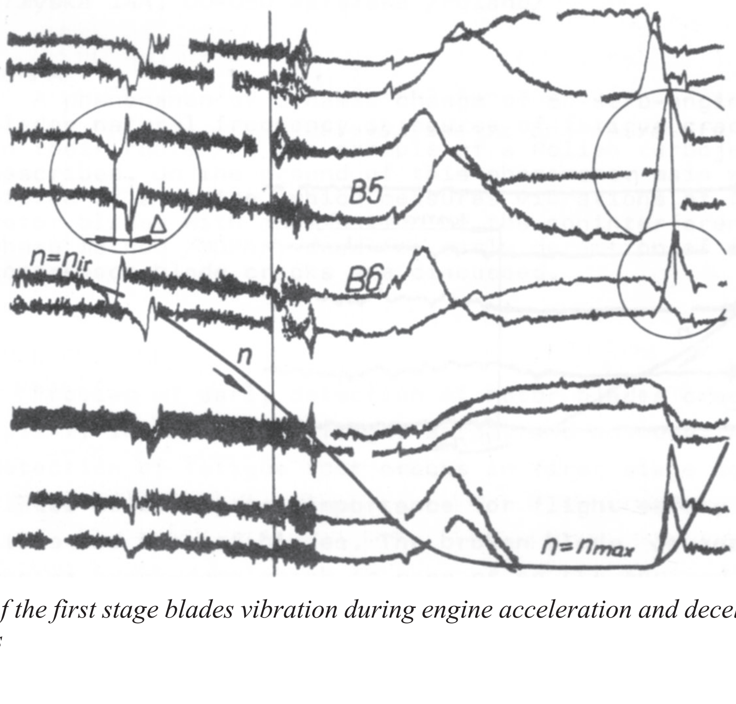

Fig. 2. Representation of the first stage blades vibration during engine acceleration and deceleration with steady-state stator distortions

3. Investigation of first stage compressor blades vibration spectrum changes in respect of initiation and propagation of fatigue cracks

Two turbine aero-engines were tested on the ground test stand up to the moment of their failure (as a result of breakdown of compressor 1-st stage blades). Strong resonance blade vibrations were excited in both engines by attaching of foreign object (a piece of felt) on the 1-st stage stator blades.

In the first case a piece of felt remained in the inlet during all engine test time i.e. from the moment of crack initiation up to the blades break down. For this engine the picture of blades vibrations spectrum changes (in condition of extremal vibrations and stresses) as a function of fatigue cracks initiation and propagation has been found.

In the second case the testing was performed according to earlier experience but the piece of felt remained in the stator only up to the initiation of .fatigue cracks. At the moment of fatigue crack initiation the felt was removed and the engine was run with nominal vibration and stress levels.

The First Test

Figure 2 illustrates the first stage blades vibration spectrum. The high resonance vibration amplitude of some blades can be seen within the rotational speed range of 8000-8500 rpm as well as 14500-15100 rpm.

Vibration analysis shows that the appearance of cracks in blades is accompanied by specific changes in initial spectrum of their vibrations (Fig. 3). The frequency of free vibrations of the cracking blade decreases. As a result of this the two above mentioned strong resonance ranges drift towards lower values of rotational speed. In consequence, the first resonance range drops below engine idling. The second resonance range moves also into the direction of lower engine rotational speeds (as the blade crack develops). Relatively quickly this resonance is displaced into the rotational speed range of 13500-14000 rpm in which the resonance never occurs if the blades have no cracks. The overall time of crack propagation in tested engine, up to the moment of blade break-down, has totaled 29 minutes.

The Second Test

The resulting blades vibrations spectrum taken in conditions of a partial foreign object presence in the stator is demonstrated on Fig. 4a. The engine was run up to the moment of fatigue cracks initiation within the blades (Fig. 4b) and then the engine was shut down and the felt removed from the inlet. Next the engine was rerun and the blades vibration spectrum was registered (Fig. 4c).

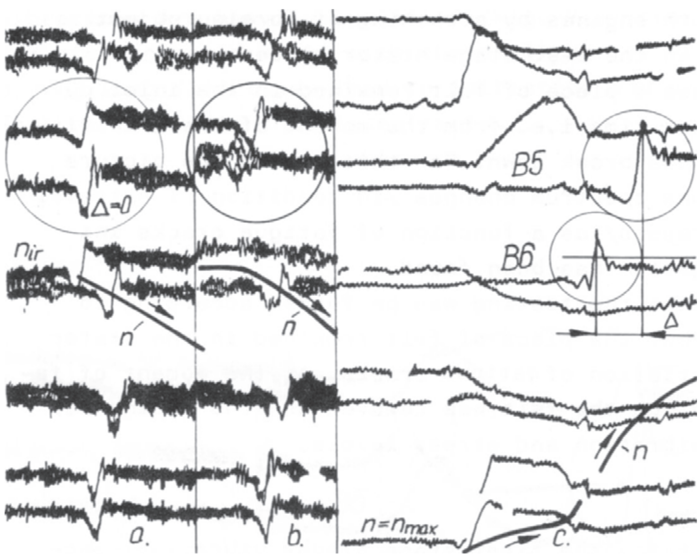

Fig. 3. Crack initiation and propagation symptoms in blades number B5 and 86 (compare with Fig. 2), (a) the resonance bandwidth of B5 is found to be near the B4, (b) displacement of the resonance bandwidth of the B5 blade below the engine idling is visible, (c) the resonance bandwidth shift from the scope of 14500-15100 rpm to 13000-13500 rpm is visible

Comparison of both vibration spectra taken demonstrates that the vibration spectrum of cracked blades essentially differs from that of blades without cracks (like in the first test) though the blades measurements indicated that the cracks affected only about 15% of blade cross-section. In total, engine was run, about 16 hrs. from the moment of cracks appearance on three blades up to the break-down of one of them.

4. The basic construction criteria of device designated for inflight early detection of the compressor blades cracks

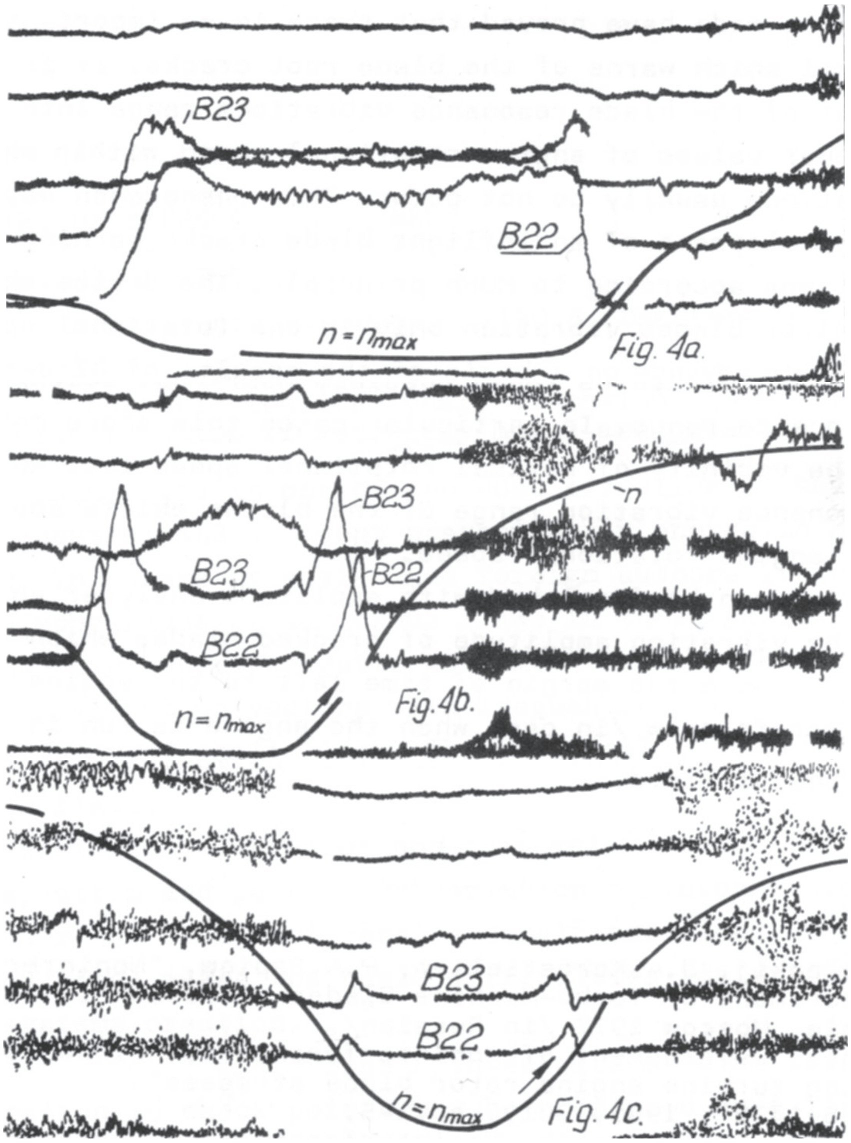

Fig. 4. Comparison of blade vibration spectra of the same blades for different conditions, (a) with steady-state stator distortions. (b) cracked blades (steady-state stator distortions), (c) cracked blades (normal flow conditions)

Investigations made have proved that there is an important diagnostic signal which warns of the blade root cracks. It is the displacement of the blade resonance vibrations range into the scope of lower values of engine rotational speed within which the9e vibrations usually do not occur. This phenomenon may be useful in development of an inflight blade cracks warning device which functions according to MDPh principle. The device should be able to monitor blades vibration only in one rotational speed scope chosen in advance on a basis of the results of blades vibration spectrum investigation. The chosen scope should be free from any blade resonance during operation of an healthy engine.

This scope should also be found possibly near (but below) the normal resonance range. In particular cases this scope could be found in the vicinity of maximal rotational speed i.e. when the normal resonance vibration range of the blades exists above the maximum of engine rotational speed.

The device can be furnished with amplitude analyzer able to estimate the vibration amplitude of cracked blades which will permit to prognose the margin of time left to the engine failure for the aircraft crew (in case when the engine is run in the range of free resonance of cracked blades). The device can also advice the crew how to operate the engine in order to slow down the crack propagation thus creating the possibility of safe return to the airfield.

5. Structure of blade technical status monitoring system

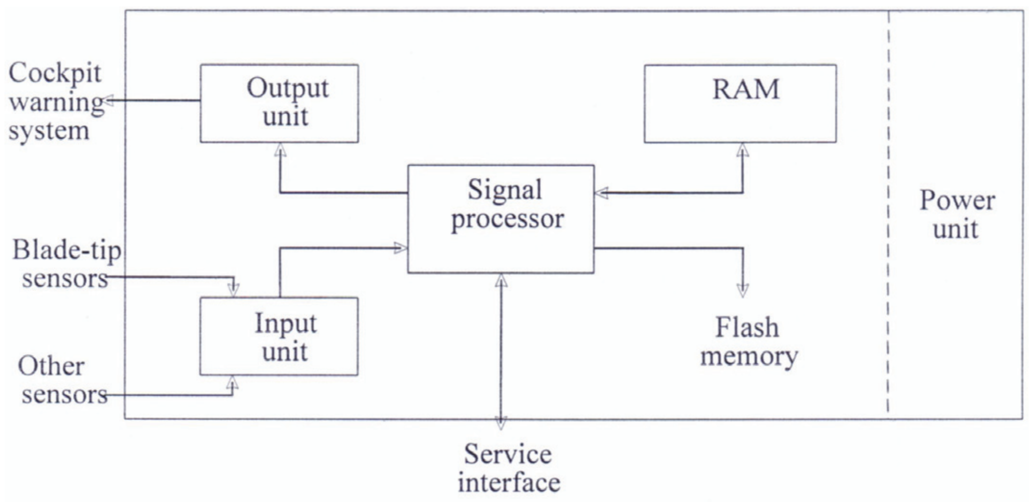

Fig. 5. Diagram of onboard computer

Development of the system requires detailed knowledge on the engine dynamics and aircraft design. The onboard software includes built-in models of the engine and its sub-assemblies and also experimentally set threshold values.

The system inform the pilot only about failures of higher priority and then advise him special procedure. Minor malfunctions and exceedances are reported only to ground personnel. Data from flights are transferred to ground database system. It gathers data generated by onboard computers from the whole fleet. The ground database handles reach engine health and usage information, which could be searched, analyzed. The software looks for suspicious trends and advice maintenance actions.

6. Conclusions

Today health monitoring systems are finding widespread use in aviation for monitoring the health of engines and other flight critical components. Proposed complex monitoring system for the turbojet engine performs numerous functions using only two sensors. It diagnoses engine health and warns against failures of compressor and turbine blades. It could be reliable and cost-effective solution especially for aging aircrafts operating in harsh condition.

References

[1] Zabłockij, I. E., Korostieliew, J. A., Szpiow, R. A., *Noninterferen ce Measurement of turbo-Machine Blades Vibration*, Maszinostrojenie, (in Russian), Moscow 1977.

[2] Oones, H. T., *Development of noninterference technique for measuring turbine engine rotor blade stresses*, AIAA-85-1472, Ouly, pp.1-5, 1985.

[3] Kudelski, R., Szczepanik, R., *Noninterference measurements of turbo-machine blades vibrations*, Proceedings of 6-th Conference – Technology of Turbo-Machines, pp. 373-377, Rzeszow 1988.

[4] Boakes, S., *Definitive HUMS*, www.smiths-aerospace.com.

[5] Szczepanik, R., Kowalski, M., Spychała, J., *Some Aspects of Present Maintenance Philosophy for Helicopters in the Polish Air Force*, Workshop on Fatigue Design of Helicopters, Pisa, 12-13 Sep. 2002.

[6] Witos, M., Szczepanik, R., *Turbine Engine Health/Maintenance Status Monitoring with use of Phase-Discrete Method of Blade Vibration Monitoring,* RTO AVT-121 Symposium on Evaluation, Control and Prevention of High Cycle Fatigue in Gas Turbine Engines for Land, Sea and Air Vehicles, Granada 2005.

[7] *Gas turbine condition monitoring & fault diagnosis. Von Karman Institute*, Rhode-Saint-Genese, Belgium 2004.