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FATIGUE LIFE ANALYSIS OF CONSTRUCTION ELEMENTS USED IN AVIATION

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Key words: fatigue analysis, durability, aviation construction elements.

Abstract

Reviewed are modern fatigue analysis methods developed and applied to analyze the durability of construction elements used in aviation. An overall characteristic of fatigue life problems is presented, in particular algorithms upon assessing durability until fatigue crack initiation and fatigue propagation. As an example, the results of a fatigue analysis is provided for aircraft engine compressor blades and an aircraft wing and fuselage connection.

ANALIZA TRWAŁOŚCI ZMĘCZENIOWEJ ELEMENTÓW KONSTRUKCJI LOTNICZYCH

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Słowa kluczowe: badania zmęczeniowe, trwałość, elementy konstrukcji lotniczych.

Streszczenie

Dokonano przeglądu współcześnie rozwijanych metod badań zmęczeniowych stosowanych do analiz trwałości elementów konstrukcji lotniczych. Podano ogólną charakterystykę zagadnień wytrzymałości zmęczeniowej, a szczególnie algorytmy postępowania przy ocenie trwałości do zainicjowania pęknięć zmęczeniowych i trwałości propagacyjnej. Przedstawiono wyniki analiz trwałościowych dla łopatki sprężarki silnika lotniczego i połączenia skrzydło-kadłub samolotu.

Introduction

A wide range of methods were developed over the past decades relating to the fatigue reliability of aircraft fleet and aeronautical equipment – in particular with respect to modeling the initiation and propagation of fatigue cracks, nondestructive methods of assessing its state and implementing the fail safe concept in structure design, or probability estimates of operating risk and limit states (YANG 1993, ICAF 1985, AGARD 1975). Fatigue life and reliability issues are increasingly often taken into account in technical requirements placed on new structures. In the specific case of aircraft fleet and aeronautical equipment, the bases for such requirements are generally modern standards or regulations (e.g. BCAR or FAR) as well as special requirements by the user of specific structures. Within this context, properly formulated and experimentally verified test methods gain significance, as do computational models for estimating fatigue strength and life of materials, elements and structures (O'DONOGHUE et al. 1995, ATLURI 1986, MISHNAEVSKY 1997, TRYON et al. 1996, PARK 1995) as well as knowledge on the accuracy of generated estimates. The durability problem of structural components is an interdisciplinary matter, it combines a wide range of science fields. Even if it is only examined within mechanical and strength categories – it is a complex problem, since in simple and standard cases it still requires the analysis of many issues. These include: determining widely understood material and geometric characteristics of the element, assessing the actual load spectrum, estimating durability based on computational models with elements of fracture mechanics as well as experimentally verifying generated results.

Durability analysis methods

Most aviation structures are designed according to safe life principles, but proposed changes to aviation regulation requirements are aiming to also base the qualification of structures on crack growth models. A solid model for assessing the fatigue life of construction elements – adequate for complex requirements with respect to the geometric variety of elements as well as load spectrum – requires the knowledge of material characteristics, material strength properties, mainly the material cyclic strain curve, fatigue curves such as $\varepsilon = f(N)$ or $\sigma = f(N)$ and/or propagation curves such as $a = f(N)$ or $da/dN = f(\Delta K)$.

Durability in the form of the number of cycles to initiate fatigue crack, respective for the given load spectrum, may be determined by the following procedure (KŁYSZ 1991, KŁYSZ 1999, SOBCZYKIEWICZ 1983, BUKOWSKI et al. 1992):

– for ranges $\Delta\sigma_i$ of subsequent load cycles (i is the number of cycles in the spectrum), determining ranges $\Delta\varepsilon_i$ respective for cyclic strain curve of tested material, in the form of:

$$\frac{\Delta\varepsilon}{2}=\frac{\Delta\sigma}{E}+\left(\frac{\Delta\sigma}{2K}\right)^{1/n} \tag{1}$$

- for ranges $\Delta\varepsilon_i$ of subsequent load spectrum cycles, determining respective number of cycles to destruction $2N_{f,i}$ respective for the Manson-Coffin curve of the tested material in the form of:

$$\frac{\Delta\varepsilon}{2}=\varepsilon_f\left(2N_f\right)^b+\frac{\sigma_f-\sigma_{śr}}{E}\left(2N_f\right)^c \tag{2}$$

- calculating partial damages $D_i=\frac{1}{2N_{f,i}}$ initiated by subsequent load cycles,
- calculating total damage $D = \Sigma D_i$ respective for a given load spectrum,
- calculating durability $N_I = 1/D$ to crack initiation, as reversal of total damage.

With regard to fatigue propagation analysis, i.e. analysis of fatigue crack growth to critical sizes resulting in destruction of structure, the algorithm should comprise the following (KŁYSZ 2001, KŁYSZ 2002):

- determining the geometric configuration of the structural component along with the crack, of a given starting length, most frequently accepted as the minimum detectible length, e.g. with non-destructive methods,
- adopting the appropriate form of stress intensity factor shape function K, respective for the configuration of the element and cracks as well as load type,
- determining material characteristics of tested material, including propagation equation coefficients and perhaps the range of their application, crack growth delay model coefficients, K_{th} threshold value and K_c critical value of the stress intensity factor.

The computational procedure for each load cycle then proceeds as follows:

- the value range of the stress intensity factor ΔK is determined for given crack length value a and the cycle stress range $\Delta\sigma$,
- the crack growth da in a given cycle is calculated from the propagation equation,
- by relating current load value to the history of previous loads, an assessment is made whether conditions arise for crack growth delay as a result of an overload cycle and possibly calculate the retardation coefficient value according to the adopted crack growth retardation model,
- the determined crack growth value or reduced value, with the use of the retardation factor, is added to the current crack size.

The computational procedure is repeated until the critical value of the stress intensity factor K_c is reached for the new crack size value and subsequent load cycle. It is the criteria equivalent to destroying the construction element. The number of cycles or load spectra after which the K_c value was reached is fatigue propagation for the analyzed case of element geometry, applied type of material and given load.

3. Example of fatigue analysis

3.1. Analysis of crack initiation in aircraft turbine engine compressor blades

A turbine engine blade is a very complex structural component with respect to geometric and durability aspects. Taking into account an elliptical crack in the blade requires the inclusion of a submodel in which the top of the crack is surrounded by special elements (singular). A significant limiting factor is also the fact that a new discretization network of the finite element method must be created and long calculations repeated for each propagating crack growth.

Based on the following, calculated with the use of the finite element method (KRZESIŃSKI et al. 1994):

– stress distribution in the compressor blade resulting from centrifugal forces,
– relation of changes in the free vibration frequency of blades to engine compressor revolutions,
– stress intensity factor distributions for shapes and layouts close to actual fatigue cracks,

as well as (WSK report, Rzeszów, 1987):

– blades stresses, registered in lab fatigue tests,

the time to initiation of fatigue crack at the base of potential damage can be determined in selected areas of the blade (BUKOWSKI et al. 1994, KŁYSZ 1999b). Below are presented the results of such an analysis for first-degree compressor blades of an SO-3 engine on a TS-11 aircraft of 18H2N4WA material.

Blade load is estimated based on extensometer measurements of stresses during engine work (Fig. 1). The blade load spectrum was adopted for calculations, composed of the following:

– static component (average) induced by centrifugal forces from rotating compressor disk. According to the distribution of stress calculated by the finite element method, respective for revolution speed of 15 600 rpm, the value of stress amounts to 270 MPa at the location that experiences the most cracks in the blade ridge. Respectively, the stress experienced by the blade's trailing edge amounts to 50 MPa. Centrifugal stress changes proportionally to the square of compressor revolution,
– dynamic component induced by blade vibrations with free vibration frequency. The first form of free vibrations was adopted for the analysis. Changes in the free vibration frequency of blades, as dependent upon revolutions, was determined based on the calculation results obtained with the finite element method (KRZESIŃSKI et al. 1994), in the form of:

$$f_{\omega} = f_{o} + 1.227 \cdot \omega + 1.244 \cdot \omega^{2} \tag{3}$$

where f_o – is the frequency of free vibrations of a given blade without revolutions, while the revolutions are entered in thousands per minute. The assumption adopted is that this relationship is true for all blades of this type; only typical f_o values for particular blades (relatively easy to determine) were taken into account.

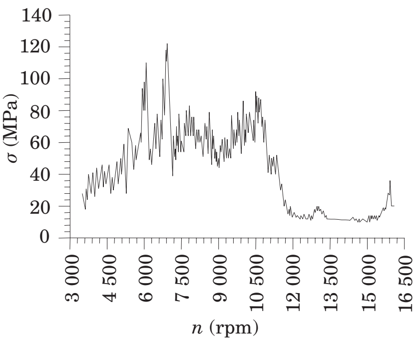


Fig. 1. Change in stress amplitude measured with an extensometer on the blade as a function of engine speed

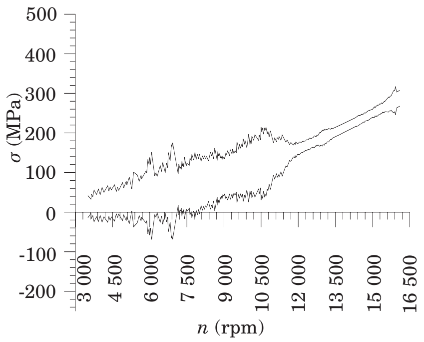


Fig. 2. Blade load spectrum during a test on the engine test bed

To conduct comparative fatigue calculations, the load spectra for these blades were also determined, taking into account changes in the level of amplitude vibrations (during engine work), initiated by:

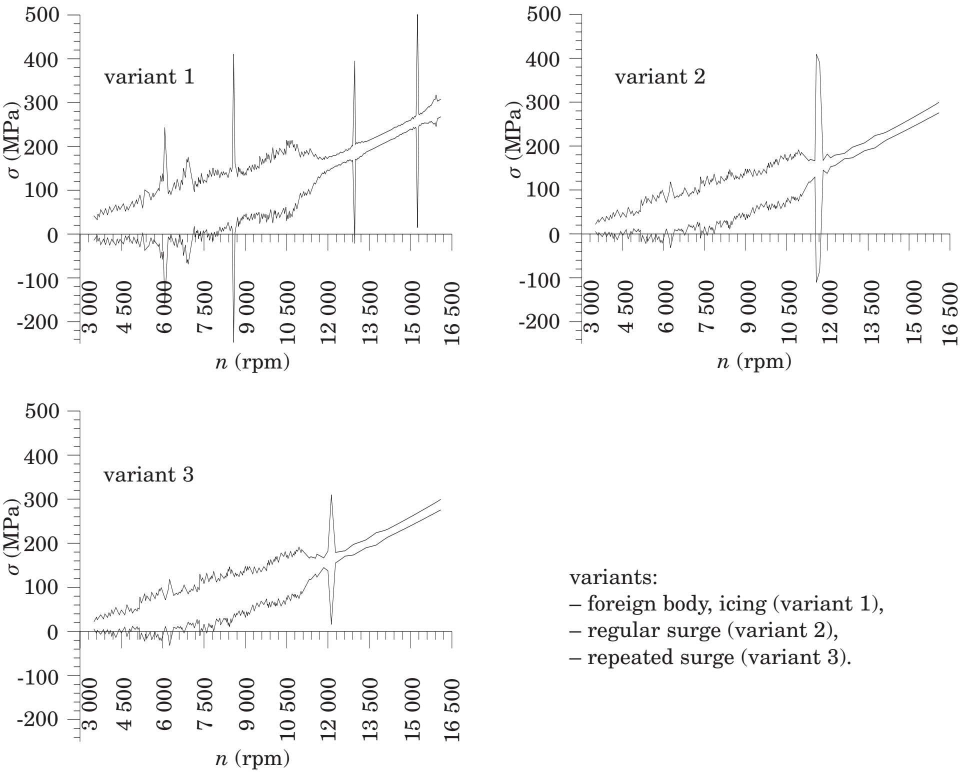
– placing a tarpaulin set with parameters 125×100×45 mm at the entrance of the engine (variant 1),
– regular surge (variant 2),
– repeated surge (variant 3).

During lab measurements of stresses for the listed variants, their level would increase at specific impeller speeds (WSK Report, Rzeszów 1987). Modified load spectra were adopted for these variants (Fig. 3) after conducting a similar procedure to determine the fatigue spectrum.

Durability until crack is initiated in the blade was determined based on the algorithm described in the study, assuming that the ridge (or the blade's trailing edge) includes a geometrical notch – simulating material flaw or defects such as inclusions, losses, gaps, corrosion pits, mechanical scratches (Hoeppner 1990) with the stress concentration factor α_k, which is potentially the location of crack initiation. Calculations were performed for a few stress concentration factor α_k values.

The results of crack initiation life on the blade in the location where extensometers were placed, acquired based on the algorithm presented in the study, are presented in Table 1.

With regard to the mentioned computational variants, for which the modified load spectra adopted the shape presented in Fig. 3, the generated



variants:
– foreign body, icing (variant 1),
– regular surge (variant 2),
– repeated surge (variant 3).

Fig. 3. Modifications of blade load spectra during tests on the engine test bed for the variants adopted in the study

Table 1

Durability until blade crack initiation

Specyfication	N_I (number of test bed tests / operating hours)			
Blade no.	α_k = 1	α_k = 2	α_k = 3	α_k = 4
1 (ridge)	$9.77.10^{11}$ $4.07.10^{10}$	$3.09.10^{8}$ $1.29.10^{7}$	$2.37.10^{6}$ $9.89.10^{4}$	51737 2155
2 (ridge)	$6.06.10^{11}$ $2.53.10^{10}$	$1.91.10^{8}$ $7.98.10^{6}$	$1.46.10^{6}$ $6.07.10^{4}$	31041 1293
3 (ridge)	$3.61.10^{11}$ $1.51.10^{10}$	$1.14.10^{8}$ $4.75.10^{6}$	$9.30.10^{5}$ $3.87.10^{4}$	23343 972
4 (edge)	$1.24.10^{12}$ $5.18.10^{10}$	$3.93.10^{8}$ $1.64.10^{7}$	$3.52.10^{6}$ $1.47.10^{5}$	124163 5173

results estimating the period until fatigue crack initiation are presented in Table 2. As shown, the period to fatigue crack initiation drastically falls along with an increasing stress concentration factor.

Taking into account changes in the load spectrum resulting from surging or covering the engine intake with a tarpaulin set (variants 1, 2, 3) significantly affects the fatigue initiation life estimate – a decrease by a few orders of magnitude. In extreme cases the initiation of fatigue cracks in the area of a geometrical notch, defect or damage through operation, occurs over a period smaller than a single trial on the engine test bed.

Table 2

Durability until blade cra*ck* initiation for the variants adopted in the study

Specyfication	N_I (number of test bed tests / operating hours)			
Blade no.	α_k = 1	α_k = 2	α_k = 3	α_k = 4
1 (ridge) variant 2	$4.22.10^7$ $1.76.10^6$	7067 294	40 1.7	1 0
2 (ridge) variant 3	$8.47.10^{10}$ $3.53.10^9$	$1.31.10^7$ $5.47.10^5$	$5.79.10^4$ 2412	1129 47
3 (ridge) variant 2	$2.94.10^{10}$ $1.22.10^9$	$5.50.10^6$ $2.29.10^5$	$2.92.10^4$ 1218	628 26
4 (edge) variant 3	$3.28.10^{11}$ $1.37.10^{10}$	$5.55.10^7$ $2.31.10^6$	$2.22.10^5$ 9252	2624 109
5 (edge) variant 3	$3.32.10^7$ $1.39.10^6$	6921 288	41 1.7	-

3.2. Analysis of fatigue crack growth in aircraft wing beam

A crack in the main wing beam near the bolts connecting it to the aircraft's fuselage always results in serious consequences. Cracks in this area develop quickly on the lower lug, i.e. in the positive stress zone, while slowly on the upper lug, i.e. in the compressed zone. The analysis presented below is an attempt to determine the durability of the beam taking into account the load spectra registered during operation.

In order to determine the load spectrum of the wing and fuselage connection, two sets of information were used (SOBCZYKIEWICZ et al. 1983, BUKOWSKI et al. 1994b), coming from:

– operation, based on 4443 flights, lasting 117 646 minutes, carried out in various units on training/fighter jets. The set of information relates to training and combat flight structures with duration provided for their particular types (tasks, number and ceiling);
– measurements of fuselage and wing loads for typical flight stages, conducted during 6 flights. Each of the flights was performed by a different pilot, according to the same program and attempting to acquire maximum loads.

The load spectra respective for the 6 pilots (presented in Fig. 4) enabled the analysis of the impact of differences between piloting methods on aviation structure durability (KŁYSZ 1999, KŁYSZ 2002, BUKOWSKI et al. 1994b, BUKOWSKI et al. 1994c). As shown, despite the pilots receiving the same task to execute, the maximum loads that were registered during the flights varied by up to around 50% (e.g. between pilots 2 and 4 and pilots 3 and 6). Cycle sequences, or the durations of particular flight phases in particular spectra, were also different. Results of durability assessment N_I until the initiation of fatigue crack in the lug of the aircraft's wing beam (in the form of the number of flights carried out by particular pilots), respective for each of the 6 flights, determined based on the above described procedure, is presented in Table 3. A difference of a few orders of magnitude between durability respectively for pilots no. 2 and 4, compared with the remaining pilots, is typical.

Assuming, as a starting point to assess fatigue propagation life of the aircraft wing beam lug damage in the form of a quarter elliptic crack of a minimum size of 0.2 mm, detectable with flaw detection methods, the periods (number of flights N_P) required for the crack to reach critical size were assessed – respective for each of the 6 flights (KŁYSZ 1999, KŁYSZ 2002). The analysis was conducted based on the Paris model, applying stress intensity factor in the form of (SOBCZYKIEWICZ et al. 1983, LIU 1974):

$$\Delta K = 1.26 \cdot \beta \cdot \Delta\sigma \cdot \sqrt{\bar{a}} \cdot f\left(\frac{\bar{a}}{r}\right),$$

where:

$$\bar{a} = \frac{a}{\sqrt{2}},$$

$f(a/r)$ – the so-called Bowie function for a single cross slot propagating from a hole with the radius r (SOBCZYKIEWICZ et al. 1983);

β – coefficient taking into account the width (W) and thickness (h) of the lug;

$$\beta = \sqrt{\frac{1}{b}\,\mathrm{tg}(b)}\,\frac{0.752 + 4.04 \cdot b + 0.37\left(1 - \sin(b)\right)^3}{\cos(b)},$$

$$b = \frac{\pi \cdot a^2}{4h(W - 2r)},$$

W, r, h – geometric sizes of the wing beam lug.

The number of flights N_P, acquired by this method, until aircraft wing beam lug ruptures (as a result of exceeding the critical value of the stress intensity factor for the beam's material) due to the operation of the aircraft according to particular load spectra, is presented in Table 3 (in column N_P).

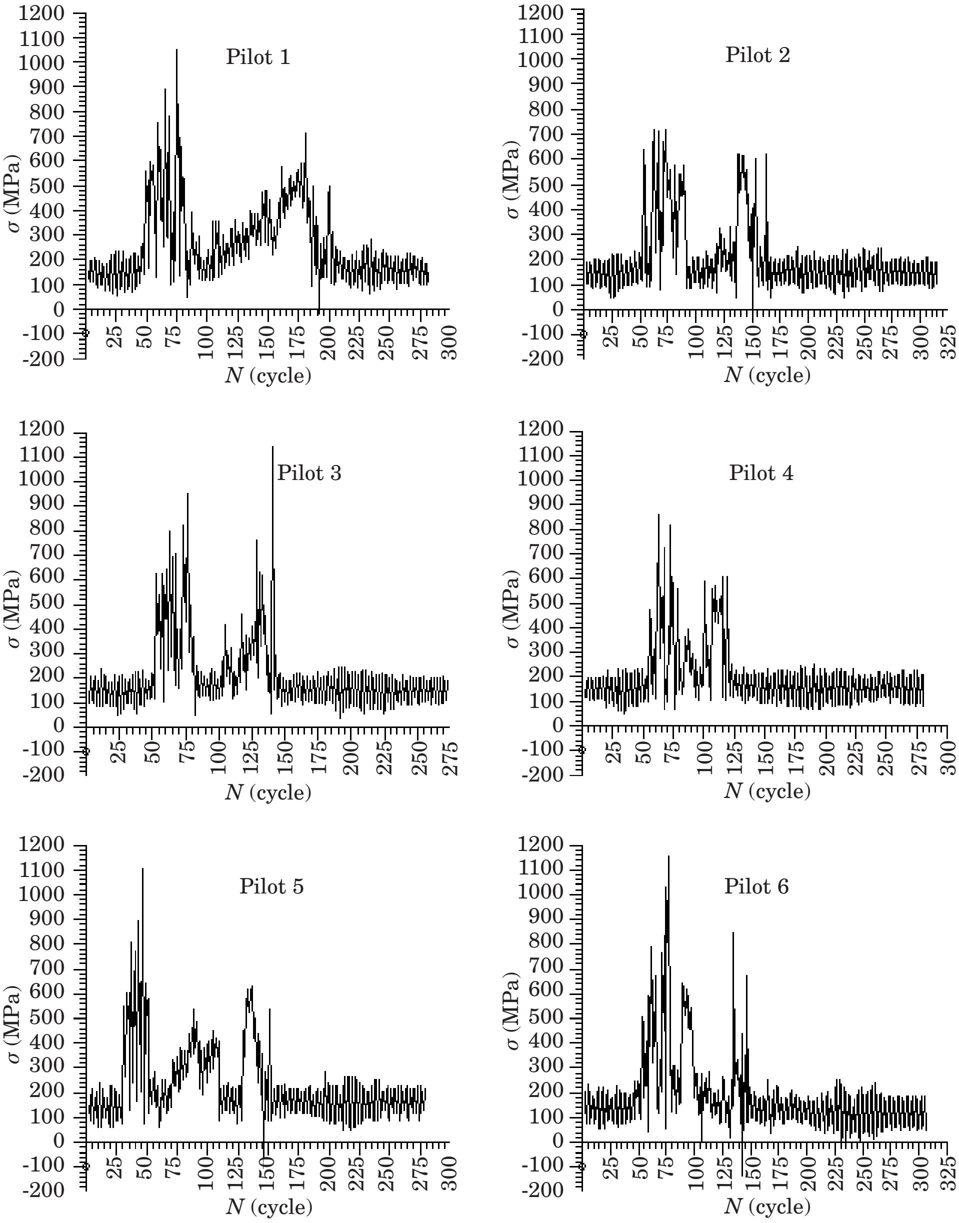


Fig. 4. Load spectra in the wing beam connection acquired by 6 pilots carrying out the same flight task

Table 3

Fatigue analysis results for the aircraft wing beam

Pilot no.	N_I (flights)	N_P (flights)
1	14 145	20
2	2 398 739	1 653
3	2 472	3
4	760 528	2 608
5	5 791	11
6	6 072	5

Similarly as with fatigue initiation life, there is also a difference of a few orders of magnitude between durability respective for pilots no. 2 and 4 compared to the remaining pilots. Differences in methods of aircraft operation by various pilots (flight dynamics), during the execution of the same task, mainly characterized by a difference in load levels at the most intensive stages of flight (reaching 70% – the maximum stress level achieved by pilot no. 2 amounted to 700 MPa, while pilot no. 6 achieved over 1200 MPa) may result in very large differences in aircraft structure durability, reaching a few thousand flights.

4. Summary

The study presents an analysis of problems related to assessing the fatigue life of elements and structures, describing currently developed fatigue test methods and algorithms applied in the above-mentioned analysis. It was pointed out that these are interdisciplinary issues requiring cooperation among specialists from many fields of science.

There is usually no unequivocal, commonly applied solution relating to, for example, the stress intensity factor function shape for the analysis of aviation construction elements of complex form and load conditions. With respect to such elements, the most reliable results are obviously achieved with 1:1 scale tests, on actual objects, and best if jointly performed with tests of the entire aircraft (in flight or at testing workstations). However, this requires large testing and measuring facilities as well as good financial support. Based on the results of durability analysis, the impact of various factors on the course of using elements or structures can be assessed. As an example, the study presents the results of applying algorithms to determine the initiation and propagation periods for fatigue cracks. An assessment was also performed with regard to how the difference in operational load level, respective for different cases of use or assigned to different pilots, impacts the durability of structural components. They indicate the particular meaning of in-flight measurements and flight-by-flight analysis of durability degradation of the structure. Knowledge of actual operational loads of structural components is then necessary to properly assess aviation construction safety and its durability reserve.

The analysis presented in the study is an example of using the results of measurements on actual construction elements, calculations applying the finite element method respective for a given element and load conditions as well as mathematical models for estimating fatigue life, leading to acquire quantitative assessments (initial) and qualitative assessments (more reliable) of the durability of construction elements used in aviation. Each of the mentioned elements of this analysis contributes a significant and mutually complementing input into acquiring a reliable assessment of fatigue life. Only such a combination of many scientific fields creates large possibilities with respect to a practical analysis of structural component damage mechanisms. By possessing these types of effective analytical tools, experimental tests on expensive aeronautical equipment in real conditions can be rationally minimized, and laboratory tests may be appropriately planned. They should mainly pertain to material characteristics and assessments of load spectra of particular construction elements as well as the simulation of appropriate environmental conditions during operation. Meanwhile, analytical methods of assessing durability should become increasingly more accurate, and the models should reflect more reliably the actual relationships and properties of tested objects.

References

AGARD Conference Proceedings, 186, Impact damage Tolerance of structure. 41st Meeting of the Structures and Materials Panel, Ankara 1975.

ATLURI S.N. 1986. *Computational methods in the mechanics of fracture*. Computational methods in mechanics, 2, North Holland.

BUKOWSKI L., KŁYSZ S., KRZESIŃSKI G. 1992. *Obliczanie czasu inicjacji pęknięcia zmęczeniowego*. AERO Technika Lotnicza, 5, I-II.

BUKOWSKI L., KŁYSZ S. 1994a. *Pękanie zmęczeniowe łopatek lotniczych silników turbinowych*. Sprawozdanie ITWL, 11018/I, Warszawa.

BUKOWSKI L., KŁYSZ S. 1994b. *Przykłady awarii konstrukcji lotniczych – belka główna skrzydeł samolotu. Tajniki wytrzymałości (2)*. Aero Business, 2: 21.

BUKOWSKI L., KŁYSZ S., SOBCZYKIEWICZ W. 1994c. *Pomiary w locie a szacowanie trwałości zmęczeniowej elementów konstrukcji lotniczych*. I Konferencja *Metody i technika badań statków powietrznych w locie*. Mrągowo, 37-48.

GOSS CZ., KŁYSZ S., WOJNOWSKI W. 2004. *Problemy niskocyklowej trwałości zmęczeniowej wybranych stali i połączeń spawanych*. Wyd. ITWL, Warszawa.

HOEPPNER D.W. 1990. F*uture technology requirements (related to defects and quantative material bahaviour to aid in implementation of damage tolerance for design of engine structures)*. AGARD Report, 769.

ICAF Symposium. *Durability and damage tolerance in aircraft design*. The Proceedings of the 13th Symposium of the International Committee on Aeronautical Fatigue, Pisa 1985.

KŁYSZ S. 1991. *Modelowanie rozwoju pęknięć zmęczeniowych w strefie koncentracji naprężeń*. WAT, Warszawa (rozprawa doktorska).

KŁYSZ S. 1999. *Szacowanie trwałości wybranych materiałów i elementów konstrukcji lotniczych w zakresie rozwoju pęknięć zmęczeniowych*. Prace Naukowe ITWL, 5: 1-180.

KŁYSZ S. 1999. *Analiza inicjacji pęknięć zmęczeniowych w łopatce sprężarki lotniczego silnika turbinowego*. Problemy badań i eksploatacji techniki lotniczej. Wydawnictwo ITWL, 4: 197-207.

KŁYSZ S. 2001. *Rozwój pęknięć zmęczeniowych w materiałach lotniczych i stali konstrukcyjnej z uwzględnieniem przeciążeń*. Prace Naukowe ITWL, 12: 1-151.

KŁYSZ S. 2002. *Wpływ przeciążeń na rozwój pęknięć zmęczeniowych w wybranych materiałach lotniczych*. Prace Naukowe ITWL, 15: 1-147.

KRZESIŃSKI G. i in. 1994. S*tatyczna i dynamiczna analiza wytrzymałościowa łopatek I stopnia silnika SO3 z uwzględnieniem pęknięć zmęczeniowych*. Etap II. Sprawozdanie ITWL, 2478/III.

LIU A.F. 1974. *Stress intensity factor for a corner flow*. Engng Fracture Mech., 4.

Mishnaevsky L.L. jr. 1997. *Methods of the theory of complex systems in modeling of fracture: a brief review*. Engng Fracture Mech., 56 (1): 47-56.

O'DONOGHUE P.E., ATLURI S.N., PIPKINS D.S. 1995. *Computational strategies for fatigue crack growth in three dimensions with application to aircraft components*. Engng Fract. Mech., 52 (1): 51-64.

Opracowanie WSK Rzeszów. *Sprawozdanie Nr RKB-2/17,51/42/87 z pomiaru naprężeń dynamicznych w łopatkach i tarczy wirnika sprężarki I stopnia silnika SO-3,* 37174120, 1987.

PARK J.H., SINGH R., PYO C.R., ATLURI S.N. 1995. *Integrity of aircraft structural elements with multi-site fatigue damage*. Engng Fracture Mech., 51 (3): 361-380.

SOBCZYKIEWICZ W. i in. 1983. *Metoda szacowania okresów przeglądowych silnie obciążonych węzłów konstrukcji stalowej na przykładzie połączenia skrzydło-kadłub wybranego samolotu*. Etap II i Etap III. Praca naukowo-badawcza Politechniki Warszawskiej, 121/501/ /019/4.

TRYON R.G., CRUSE T.A., MAHADEVAN S. 1996. *Development of a reliability-based fatigue life model for gas turbine engine structures*. Engng Fracture Mech., 53 (5): 807-828.

YANG J.N. 1993. *Application of reliability methods to fatigue, quality ssurance and maintenance*. Proceedings 6th Int.Conf. on Structural Safety and Reliability, Innsbruck, 1:1-18.

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