11TH INTERNATIONAL SCIENTIFIC CONFERENCE
ON DEFENSIVE TECHNOLOGIES
OTEH 2024
Serbia, Tara, 9-11 October 2024

FAILURE ANALYSIS OF AIRCRAFT STRUCTURAL ELEMENTS

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Abstract: *The attention in the work is focused on the strength analysis of the vital elements of the aircraft structure. Special attention is focused on defining critical locations from the aspect of possible failures during exploitation. Then, initial damage in the form of initial cracking is assumed in the defined critical zone. The assumed crack sizes are in accordance with the recommendation on allowable damages. For the cracks defined in this way and the corresponding load spectrum, the propagation of the crack until the effective fracture is analyzed. Different crack propagation laws can be used for this purpose. In this paper, the various models were used. The finite element method (FEM) was used for the structural analysis, where special 6-node singular finite elements were used around the crack tip. Since there are always peaks within the real load spectrum, they are modeled using in-house software. This paper deals with research in the domain of estimating the failure of aircraft structures in critical locations under the effect of cyclic loads. The main goal is to provide an efficient computation method. The previous method is illustrated on the analysis of the failure of parts of vital connections of aircraft structures.*

Keywords: *aircraft, spectrum loading, overload effects, residual fatigue life estimation*

1. INTRODUCTION

In general, failures occur when a component or structure is no longer able to withstand the stresses imposed on it during operation. Commonly, failures are associated with stress concentrations, which can occur for several reasons including:

- Design errors, e.g. the presence of holes, notches, and tight fillet radii;
- The microstructure of the material may contain voids, inclusions etc.;
- Corrosive attack of the material, e.g. pitting, can also generate a local stress concentration.

From our records and case histories data, an assessment can be made of the frequency of failure modes (Table 1). This reveals that the incidence of fatigue failure dominates the distribution in aircraft. This would suggest, therefore, that fatigue is the predominant failure mode in service. The detection and rectification of corrosion damage on in-service aircraft, however, consumes more effort than the repair of fatigue cracking. The high occurrence of fatigue failure observed probably reflects the destructive nature of this failure mode, while corrosive attack is generally slower than fatigue, and usually more easily spotted and rectified during routine maintenance. Fatigue is a process whereby cracking occurs under the influence of repeated or cyclic stresses, which are normally substantially below the nominal yield strength of the material. Components that fail by fatigue usually undergo three separate stages of crack growth, which are described as follows:

- Initiation of a fatigue crack. This can be influenced by stress concentrations such as material defects or design.
- Propagation of the fatigue crack. This is progressive cyclic growth of the crack.
- Final sudden failure. Eventually, the propagating crack reaches a critical size at which the remaining material cannot support the applied loads and sudden rupture occurs. Fatigue failures generally leave characteristic markings on the fracture surface of cracks from which the failure investigator can deduce a great deal of information. The most obvious are the classic 'beach marks', which are commonly observed macroscopically. Beach marks indicate successive positions of the advancing crack front and are usually the first telltale signs that the mode of crack growth is fatigue. Fatigue fractures tend to be relatively smooth near the origin and show slight roughening of the surface as the

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DOI:10.5937/OTEH24015M

crack progresses. There tends to be little or no macroscopic ductility associated with fatigue cracking.

Detailed examination of the fracture surface in a scanning electron microscope (SEM) usually shows evidence of fatigue estimations (dependant on the material), which represent one cycle of load and crack propagation. If the magnitude of load cycle remains constant, the striations normally appear closer near the origin, gradually increasing in spacing as the crack front progresses due to the increasing stress at the crack tip.

By taking measurements of striation spacing at various distances from the origin to the end of the crack, it is possible to estimate the total number of load cycles to cause failure. If the cause of the loading can be determined, the number of cycles to failure can then be used to estimate the time required for crack growth.

Fatigue cracking is the most common cause of structural failure in aircraft, even though the laboratory fatigue behavior of most metals and alloys is well understood.

Materials and their design can be taken into consideration so that the probability of fatigue cracks occurring can be reduced, but it is often the case that the possibility cannot be removed completely. Therefore many aircraft structural components are designed with a safe or inspection-free life, below which fatigue cracking should not be a cause for concern. The fact that fatigue failures still occur, however, indicates the complex nature of this problem. There are many variables that influence fatigue, some of which are the mean stress, peak stress, frequency of loading, temperature, environment, material microstructure, surface finish, and residual stresses. Many of these factors are taken into account when determining the safe life of a component and, therefore, the majority of fatigue failures in aircraft causing catastrophic failure tend to be those that initiate as the result of unforeseen circumstances.

Material surface defects such as forging laps or surface cracking can increase the local stress, producing a concentration at these points that could initiate fatigue much quicker than would be expected. However, many aircraft components are thoroughly inspected by non-destructive techniques after manufacturing and these types of defects are usually detected and rectified. Stress concentrations caused by surface defects such as scratches and wear tend to be more common as these may not be present at build, but can be introduced during service. Another common cause of stress concentration is corrosion, which can lead to fatigue crack initiation.

Table 2 shows a summary of the common fatigue crack initiation sites observed in aircraft [1,2] that have led to accidents.

Ductile or overload failure occurs when a material has been exposed to an applied load at a relatively slow rate to the breaking point of the material. This results in a ductile fracture of the material, with the fracture surface exhibiting tearing of the metal and plastic deformation.

On rapid application of a load, fast fracture or brittle failure can occur. Microscopic examination of brittle fractures reveals intergranular or transgranular facets on the fracture surface.

Corrosion is the chemical degradation of metals as a result of a reaction with the environment. It usually results in failure of components when the metal wastes to such an extent that the remaining material cannot support the applied loads or the corrosion renders the component susceptible to failure by some other mode (e.g. fatigue).

Extensive work has been carried out on the rates and types of corrosion observed in different materials so that selecting a suitable material in terms of corrosion resistance for a known environment is relatively straightforward. In aircraft structures, however, the strength to weight ratio can be a more desirable property than corrosion resistance and in these circumstances the most suitable material cannot always be used. In cases like this, measures must be taken to limit corrosion, which most commonly involve the use of a coating, such as a paint system, to act as a barrier to the environment. There are various forms of corrosion that exist, each of which poses different problems to aircraft structures.

The most common types of corrosion observed are discussed below:

- Uniform corrosion, as its name suggests, is corrosion that occurs without appreciable localized attack, resulting in uniform thinning.
- Pitting corrosion is a localized form of attack, in which pits develop in a material causing localized perforation of the material. Pitting corrosion occurs when one area of a metal surface becomes anodic with respect to the rest of the surface of the material. The pits formed by this type of attack are generally very small and, therefore, difficult to detect during routine inspection. Pitting attack cancause failure by perforation with very little weight loss to the material.
- Crevice corrosion occurs when localized changes in the corrosive environment exist and lead to accelerated localized attack. These changes in the localized corrosive environment are generated by the existence of narrow crevices that contain a stagnant environment, which results in a difference in concentration of the cathode reactant between the crevice region and the external surface of the material. Crevices can be formed at joints between two materials, e.g. riveted, threaded, or welded structures, contact of a metal with a nonmetallic material, or a deposit of debris on the metal surface.
- Galvanic corrosion occurs when dissimilar metals are in direct electrical contact in a corrosive environment. This results in enhanced and aggressive corrosion of the less noble metal and protection of the more noble metal of the bimetallic couple. This type of corrosion can be recognized by severe corrosion near to the junction of the two dissimilar metals, while the remaining surfaces are relatively corrosion-product free. Galvanic corrosion is generally a result of poor design and materials selection.
- Stress corrosion cracking is a mechanical-environmental failure process in which tensile stress and environmental attack combine to initiate and propagate a fracture. Failure by stress corrosion cracking is frequently caused by simultaneous exposure to an apparently mild chemical environment and to a tensile stress well below the yield strength of the material. The stress required for failure can originate from in-service conditions or from residual stress during component manufacturing.

Hydrogen embrittlement is a failure process that results from the retention or absorption of hydrogen in metals, usually in combination with applied tensile or residual stresses. It most frequently occurs in high-strength steels (>1100 MPa). For aircraft components, the common source of hydrogen embrittlement is hydrogen absorption during manufacturing processes such as pickling and electroplating.

2. ESTIMATION OF TOTAL FATIGUE LIFE

During exploitation, aircraft are exposed to cyclic loads of constant amplitude and spectrum. Basically, for life estimates during exploitation, life estimates are made until the appearance of initial damage from one side, as well as the remaining life estimate, i.e. In the presence of damage

2.1 Assessment of the age until the appearance of initial damage.

The Smith-Watson-Topper relation SWT relation was used to estimate the age until the appearance of initial damage. The Smith-Watson-Topper (SWT) relation for describing the low-cycle fatigue curve has the form:

$$P_{SWT} = \sqrt{\sigma_{max} \frac{\Delta\varepsilon}{2} E} = \sqrt{\left(\sigma_f^{'}\right)^2 \left(N_f\right)^{2b} + E\sigma_f^{'}\varepsilon_f^{'}\left(N_f\right)^{b+c}} \qquad (4)$$

whereby the influence of medium stress is included via dependence

$$\sigma_{max} = \sigma_m + \frac{\Delta\sigma}{2} \qquad (5)$$

The notation P_{SWT} in (4) refers to the Smith-Watson-Topper parameter. The relation SWT (4) defines that there is no fatigue damage in situations where the value of the maximum stress, σ_{max}, is zero or has a negative value, which is not entirely true.

2.2 Estimation of remaining life using the strain energy density method

In this work fatigue crack growth method based on energy concept is considered and then it is necessary to determine the energy absorbed untill failure [3-6]. This energy can be calculated by using cyclic stress-strain curve. As a result the energy absorbed untill failure become [6]:

$$W_c = \frac{4}{1+n^{/}} \sigma^{/}_f \varepsilon^{/}_f \qquad (1)$$

where $\sigma_f^{/}$ is cyclic yield strength and $\varepsilon_f^{/}$ - fatigue ductility coefficient. Fatigue crack growth rate can be obtained in the next form [7]:

$$\frac{da}{dN} = \frac{\left(1-n^{/}\right)\psi}{4E\, I_{n^{/}}\, \sigma^{/}_f\, \varepsilon^{/}_f} \left(\Delta K_I - \Delta K_{th}\right)^2, \qquad (2)$$

where ΔK_I is the range of stress intensity factor, ψ - constant depending on the strain hardening exponent $n^{/}$, $I_n^{/}$ - the non-dimensional parameter depending on ΔK_{th} is the range of threshold stress intensity factor and is function of stress ratio i.e.

$$\Delta K_{th} = \Delta K_{th0}(1-R)^{\gamma}, \qquad (3)$$

ΔK_{th0} is the range of threshold stress intensity factor for the stress ratio R = 0 and γ is coefficient (usually, $\gamma = 0.71$). Equation (2) enables us to determine crack growth life of different structural component. Very important fact is that equation (2) is easy for application since low cyclic material properties ($n^{/}$, $\sigma_f^{/}$, ε_f are used as parameters.

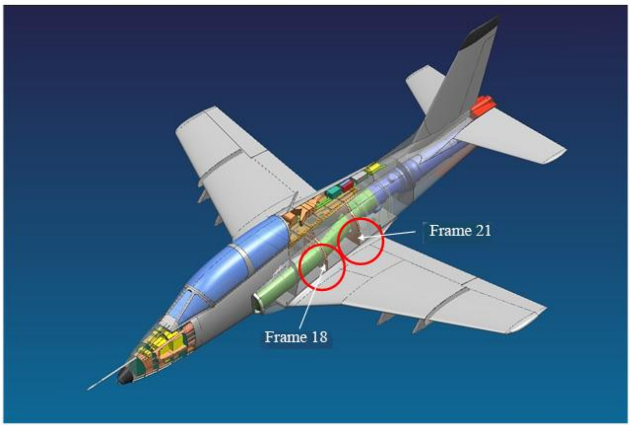


Fig.1 Aircraft structure

Figures 2 and 3 show the results of the structural analysis, in this case the stress states, which were determined by the precise structural analysis based on FEM. These are basically the critical zones on the aircraft, the failure of which could lead to the effective breakage and even the loss of the aircraft itself.

Table 1: Wing-fuselage test results

Spec. No.	**Experimental results**		**Numerical estimation of total life**
1	788	$\overline{N}_{bl} = 806.85$ (Middle value)	**$N_{bl,i}$=505** **$N_{bl,p}$=52** **$N_{bl,t}$=557**
2	988		
3	946		
4	892		
5	686		
6	612		
7	736		

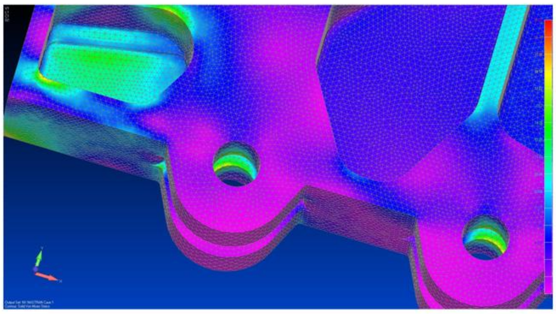


Fig. 2 The stress state in the forward connection of the

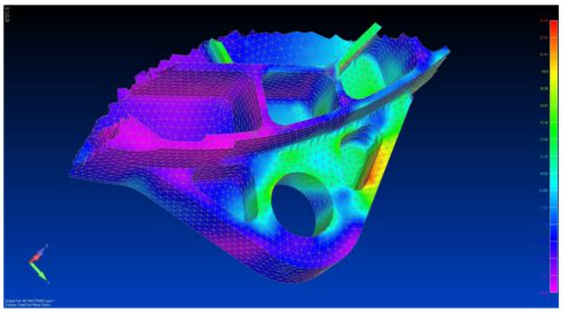


Fig. 3 The stress state in the rear connection of the wing fuselage using FEM

3.VALIDATION OF CALCULATION RESULTS

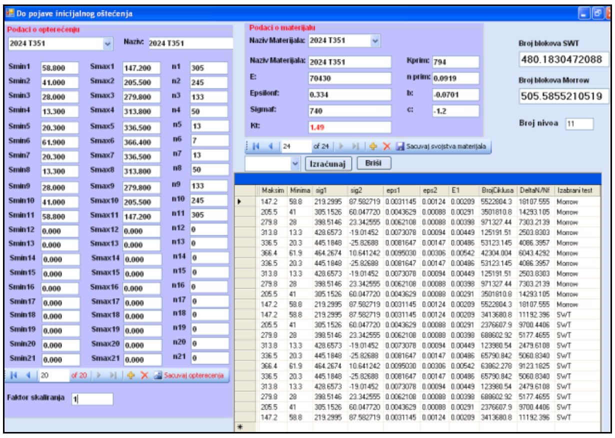
In order to validate the results of the assessment of the total life under the influence of the load spectrum, a part of the connection of the wing fuselage of the ORAO aircraft is shown. On Fig. 4 shows part of the wing-fuselage connection of the aircraft. This part of the wing-fuselage connection was tested under static and dynamic loads.

Fig. 4 Part of the wing fuselage connection

Fig. 5 The fracture appearance of the wing-fuselage connection part after the fatigue test

Table 1 shows the complete results of the examination of the part of the wing-fuselage connection of the aircraft, as well as comparisons with the results of the calculation assessment. In Table 2, for the experimental result for the number of blocks, the mean value of the number of blocks is taken, where one block corresponds to 50 hours of aircraft flight. In that case, the difference between the calculation estimate of the total lifetime and the experiment is about 18.6%, which can be considered that the calculation estimate of the lifetime is slightly conservative.

Table 2: Estimation of life to occurrence of wing-fuselage damage according to (SWT) and Morrow


Do pojave inicijalnog oštećenja

Podaci o opterećenju

2024 T351 Naziv: 2024 T351

Smin1	58.800	Smax1	147.200	n1	305
Smin2	41.000	Smax2	205.500	n2	245
Smin3	28.000	Smax3	279.800	n3	133
Smin4	13.300	Smax4	313.800	n4	50
Smin5	20.300	Smax5	336.500	n5	13
Smin6	61.900	Smax6	366.400	n6	7
Smin7	20.300	Smax7	336.500	n7	13
Smin8	13.300	Smax8	313.800	n8	50
Smin9	28.000	Smax9	279.800	n9	133
Smin10	41.000	Smax10	205.500	n10	245
Smin11	58.800	Smax11	147.200	n11	305
Smin12	0.000	Smax12	0.000	n12	0
Smin13	0.000	Smax13	0.000	n13	0
Smin14	0.000	Smax14	0.000	n14	0
Smin15	0.000	Smax15	0.000	n15	0
Smin16	0.000	Smax16	0.000	n16	0
Smin17	0.000	Smax17	0.000	n17	0
Smin18	0.000	Smax18	0.000	n18	0
Smin19	0.000	Smax19	0.000	n19	0
Smin20	0.000	Smax20	0.000	n20	0
Smin21	0.000	Smax21	0.000	n21	0

20 of 20 Sacuvaj opterecenja

Faktor skaliranja: 1

Podaci o materijalu

Naziv Materijala: 2024 T351

Naziv Materijala: 2024 T351

E: 70430

Epsilonf: 0.334

Sigmaf: 740

Kt: 1.49

Kprim: 794

n prim: 0.0919

b: -0.0701

c: -1.2

Broj blokova SWT: 480.1830472088

Broj blokova Morrow: 505.5855210519

Broj nivoa: 11

24 of 24 Sacuvaj svojstva materijala

Maksim	Minima	sig1	sig2	eps1	eps2	E1	BrojCiklusa	DeltaN/Nf	Izabrani test
147.2	58.8	219.2995	87.582719	0.0031145	0.00124	0.00209	5522804.3	18107.555	Morrow
205.5	41	305.1526	60.047720	0.0043629	0.00088	0.00291	3501810.8	14293.105	Morrow
279.8	28	398.5146	23.342555	0.0062108	0.00088	0.00398	971327.44	7303.2139	Morrow
313.8	13.3	428.6573	-19.01452	0.0073078	0.00094	0.00449	125191.51	2503.8303	Morrow
336.5	20.3	445.1848	-25.82688	0.0081647	0.00147	0.00486	53123.145	4086.3957	Morrow
366.4	61.9	464.2674	10.641242	0.0095030	0.00306	0.00542	42304.004	6043.4292	Morrow
336.5	20.3	445.1848	-25.82688	0.0081647	0.00147	0.00486	53123.145	4086.3957	Morrow
313.8	13.3	428.6573	-19.01452	0.0073078	0.00094	0.00449	125191.51	2503.8303	Morrow
279.8	28	398.5146	23.342555	0.0062108	0.00088	0.00398	971327.44	7303.2139	Morrow
205.5	41	305.1526	60.047720	0.0043629	0.00088	0.00291	3501810.8	14293.105	Morrow
147.2	58.8	219.2995	87.582719	0.0031145	0.00124	0.00209	5522804.3	18107.555	Morrow
147.2	58.8	219.2995	87.582719	0.0031145	0.00124	0.00209	3413680.8	11192.396	SWT
205.5	41	305.1526	60.047720	0.0043629	0.00088	0.00291	2376607.9	9700.4406	SWT
279.8	28	398.5146	23.342555	0.0062108	0.00088	0.00398	688602.92	5177.4655	SWT
313.8	13.3	428.6573	-19.01452	0.0073078	0.00094	0.00449	123980.54	2479.6108	SWT
336.5	20.3	445.1848	-25.82688	0.0081647	0.00147	0.00486	65790.842	5060.8340	SWT
366.4	61.9	464.2674	10.641242	0.0095030	0.00306	0.00542	63862.278	9123.1825	SWT
336.5	20.3	445.1848	-25.82688	0.0081647	0.00147	0.00486	65790.842	5060.8340	SWT
313.8	13.3	428.6573	-19.01452	0.0073078	0.00094	0.00449	123980.54	2479.6108	SWT
279.8	28	398.5146	23.342555	0.0062108	0.00088	0.00398	688602.92	5177.4655	SWT
205.5	41	305.1526	60.047720	0.0043629	0.00088	0.00291	2376607.9	9700.4406	SWT
147.2	58.8	219.2995	87.582719	0.0031145	0.00124	0.00209	3413680.8	11192.396	SWT

N$_{bl,\ i}$= 480 (SWT) (Calculation estimate of the life until the appearance of initial damage.)

N$_{bl,\ i}$= 505 (Morrov) (Calculation estimate of the life until the appearance of initial damage.)

3.1 Estimate of the remaining life

In addition to the estimate of the life until the appearance of the initial damage, the results of which are for the part of the wing, under the influence of the load spectrum determined in the previous point and shown in Table 2, it is necessary to make a calculation assessment of the remaining life, that is, during the crack propagation. To estimate the remaining life, it is necessary to assume initial damage in the form of surface cracking. Based on the maximum stress value, determined by the elastic-plastic analysis of FEM, the critical position was defined during exploitation.

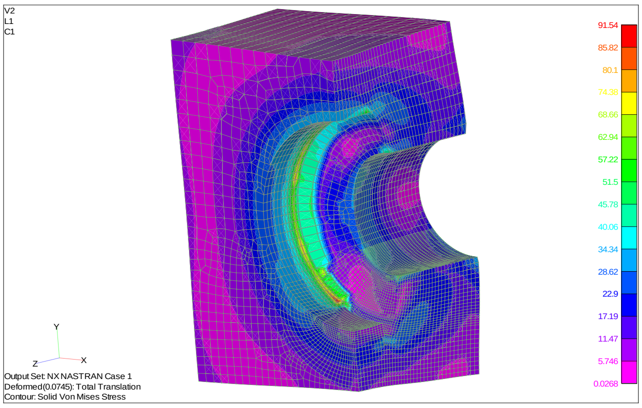


Fig. 6 Distribution of the stress state [MPa] for a crack depth of 3.5 mm

Breakages of vital aircraft links during flight often lead to the loss of the aircraft itself. In order to avoid catastrophic breaks, it is necessary to pay special attention to the critical areas of the aircraft during the design, as well as the experimental verification of the strength of the aircraft structure itself. As a rule, these are vital connections such as the wing-fuselage connection, fuselage-fuselage connection, tail-fuselage connection, and a number of others whose failures lead to catastrophic fractures. One of the important factors for prevention is a precise and detailed analysis of stress conditions in the aircraft structure. For this purpose, as illustrated in the paper, the finite element method (FEM) is used with one and precisely determined mechanical characteristics of the material. Due to efficiency and economy, the low-cycle fatigue characteristics of the material are used in the work, both for the assessment of the life until the appearance of initial damage and after the assessment of the remaining life.

4. CONCLUSION

The attention in the work is focused on the analysis of fractures in aircraft structures or, more precisely, how to prevent fractures during exploitation. Breakages of vital aircraft links during flight often lead to the loss of the aircraft itself. In order to avoid catastrophic fractures, it is necessary to pay special attention to the critical areas of the aircraft during the design, as well as the experimental verification of the strength of the aircraft structure itself. As a rule, these are vital connections such as the wing-fuselage connection, fuselage-fuselage connection, tail-fuselage connection, and a number of others whose failures lead to catastrophic fractures. One of the important factors for prevention is a precise and detailed analysis of stress conditions in the aircraft structure. For this purpose, as illustrated in the paper, the finite element method (FEM) is used with one and precisely determined mechanical characteristics of the material. Due to efficiency and economy, the low-cycle fatigue characteristics of the material are used in the work both for the assessment of the life until the appearance of initial damage and for the assessment of the remaining life. Figures 2 and 3 show the results of the structural analysis, in this case the stress states, which were determined by the precise structural analysis based on FEM. These are basically the critical zones on the aircraft, the failure of which could lead to the effective breakage and even the loss of the aircraft itself.

Acknowledgement:

This research has been supported by the research grant No. 451-03-66/2024-03/ 200066 of the Ministry of Science, Technological Development and Innovation of the Republic of Serbia.

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[6] MSC/NASTRAN software code