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Residual Life Estimation of Cracked Aircraft Structural Components

The subject of this investigation is focused on developing computation procedure for strength analysis of damaged aircraft structural components with respect to fatigue and fracture mechanics. For that purpose, here will be defined computation procedures for residual life estimation of aircraft structural components such as wing skin and attachment lugs under cyclic loads of constant amplitude and load spectrum. A special aspect of this investigation is based on using of the Strain Energy Density (SED)method in residual life estimation of structural elements with initial cracks. To determine analytic formulae for the stress intensity factors here singular finite elements are used. Verification of computation procedures for residual life estimations will be supported with corresponding experimental tests for determination of low cyclic fatigue properties of materials and corresponding parameters of fracture mechanics, including fatigue tests of representative aircraft structural elements.

Keywords: *aircraft constructions, damage tolerance approach, fracture mechanics, crack growth, finite element method*

1. INTRODUCTION

Because of careful detail-design, practically any structure contains stress concentration due to holes. Prior to the 1970s, the prevailing engineering philosophy of aircraft structures was to ensure that airworthiness was main–tained with a single part broken, a redundancy require–ment known as fail-safety. However, advances in fracture mechanics, along with infamous catastrophic fatigue fai–lures such as those in the DeHavilland Comet, prompted a change in requirements for aircraft. Since 1970 USAF has developed the damage tolerance approach [1-3]. For the application of fracture mechanics principles to cracks emanating from holes or between two holes, knowledge of the stress intensity is a prerequisite [4-6]. Fatigue life prediction concepts can be differentiated by the failure criterion applied, i.e. crack initiation, crack propagation, or – encompassing both – total life [7]. As yet no agreement exists on the crack size at which the initiation phase ends; the selection of the initial flaw size for the crack propagation phase may also create difficulties.

It was discovered that a phenomenon known as "multiple-site damage" could cause many small cracks in the structure, which grow slowly by themselves, to join one another over time, creating a much larger crack, and significantly reducing the expected time until failure [8,9]. Initially, the only philosophy for designing against fatigue of aerospace structures was the safe-life appro–ach, which means designing for a finite service life during which significant fatigue damage will not occur. The fail-safe approach requires designing for an adequate service life without significant damage, but also enabling operation beyond the actual life at which such damage occurs.

This philosophy differs from the original fail-safe approach in two major aspects:

(1) The possibility of cracks already in a new structure must be accounted for;
(2) Structures may be inspectable or non-inspectable in service, i.e. there is an option for designing structures that are not intended to be inspected during service life.

Methods for design against fatigue failure are under constant improvement. In order to optimize constructions the designer is often forced to use the properties of the materials as efficiently as possible. One way to improve the fatigue life predictions may be to use relations between crack growth rate and the stress intensity factor range. To determine residual life of damaged structural components here are used two crack growth methods: (1) conventional Forman`s crack growth method [10] and (2) crack growth model based on the strain energy density method [9,11]. The last method uses the low cycle fatigue properties in the crack growth model. Damage Tolerance approach assu–mes the components have a preexisting flaw from which a crack will grow under dynamic loads. This assumption makes it possible to account for in-service or manu–facturing defects in determining the dynamic life. The Damage Tolerance Methodology uses fracture mechanics to predict the fatigue crack growth in a structure. Design based on damage tolerance criteria often deals with notched components giving rise to localized stress concen–trations that, in brittle materials, may generate a crack leading to a catastrophic failure or to a shortening of the assessed structural life. For a successful implementation of the damage tolerance philosophy [1] to the design and in-service operation of structures subjected to fatigue loading, it is crucial to have reliable crack growth prediction tools.

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2. CRACK GROWTH MODELS

Conventional Forman`s crack growth model is defined in the form [10]

$$\frac{da}{dN}=\frac{C\left(\Delta K\right)^{n}}{\left(1-R\right)K_{C}-\Delta K} \tag{1}$$

where K_C is the fracture toughness C, n – are experi–mentally derived material parameters. The strain energy density method can be written as [11,13,14]

$$\frac{da}{dN}=\frac{\left(1-n'\right)\psi}{4EI_{n'}\sigma'_{f}\,\varepsilon'_{f}}\left(\Delta K_{I}-\Delta K_{th}\right)^{2} \tag{2}$$

where: $\sigma_f^{/}$ is cyclic yield strength and $\varepsilon_f^{/}$ - fatigue ductility coefficient, ΔK_I is the range of stress intensity factor, ψ - constant depending on the strain hardening exponent $n^{/}$, $I_n^{/}$ - the non-dimensional parameter depen–ding on $n^{/}$. ΔK_{th} is the range of threshold stress intensity factor and is the function of the stress ratio, i.e.,

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

ΔK_{th0} is the range of threshold stress intensity factor for the stress ratio R = 0 and γ is coefficient (usually, γ = 0.71). Some aspects of practical design with respects residual life estimations are given in references [12, 13, 15, 16]. Finally, the number of cycles until failure can be determined by the integration of the relation for the fatigue crack growth rate:

$$N=B\int_{a_0}^{a_c}\frac{da}{\left(\Delta K_{I}-\Delta K_{th}\right)^{2}},\; B=\frac{4EI_{n'}\sigma'_{f}\varepsilon'_{f}}{\left(1-n'\right)\psi} \tag{4}$$

and

$$\Delta K_{I}=YS\sqrt{\pi a} \tag{5}$$

Equation (4) enables us to determine the crack growth life of different structural components. A very important fact is that equation (4) is easy for application since low cyclic material properties (n', σ'_f, ε'_f) available in the literature are used as parameters. The only important point is the stress intensity factor which, depending on the geo–metry complexity and the type of loading, could be deter–mined by using analytical and/or numerical approaches.

3. NUMERICAL VALIDATION

To validate computation residual fatigue life, estima–tions procedure included here are cracked structural components of light training aircraft, Fig. 1.

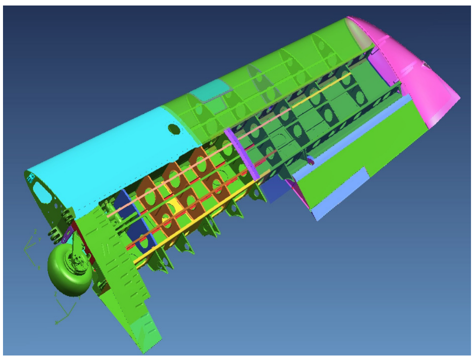
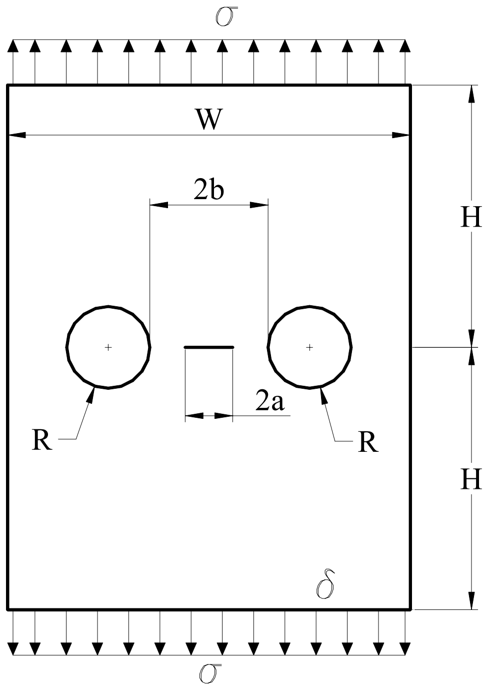


Fig. 1 Wing Structure of Light Training Aircraft

Attention in this investigation is focused on cracked wing skin and an attachment lug structural components.

Residual fatigue life of a skin containing a crack of length 2a symmetrically between two circular holes of radius R subjected, remote from the crack, to a uni–form uniaxial tensile stress σ in a direction perpen–dicular to the crack, Fig. 2, is considered.



R=2 mm
b=6 mm
W=26 mm
H=26 mm
δ=1 mm

Fig. 2 Geometric Properties of a wing skin panel with a Crack between two Circular Holes

In this crack growth analyses of duraluminum wing skin with the next material properties is used:

$E = 7.1\ 10^4$ MPa	-modulus of elasticity
S_y= 334 MPa	-material tensile yield strength
$\sigma_f^{/}$ = 613 MPa	-fatigue strength coefficient
$\varepsilon_f^{/}$ = 0.35	-fatigue ductility coefficient
$n^{/}$ = 0.121	-cyclic strain hardening exponent
$k^{/}$ = 710 Mpa	-cyclic strength coefficient
K_{IC}=120 MPa $m^{1/2}$	-the fracture toughness
$I_n^{/}$ = 3.067	-parameter that is depend on $n^{/}$
ψ = 0.95152	-nondimensional parameter
ΔK_{th0} = 8 MPa $m^{1/2}$	-the range of threshold stress in–tensity factor

Damaged structural component shown in Fig 2 represents part of wing skin in the zone of riveted joint rib/skin.

To develop analytic expressions for stress intensity factors (SIF`s), by using finite elements, it is necessary to determine (SIF`s) for various crack lengths. In Fig. 3 the FE model for initial crack length a_0 =2 mm is shown. In Table 1 are given values of the SIF determined by FEM for various values of the crack lengths.

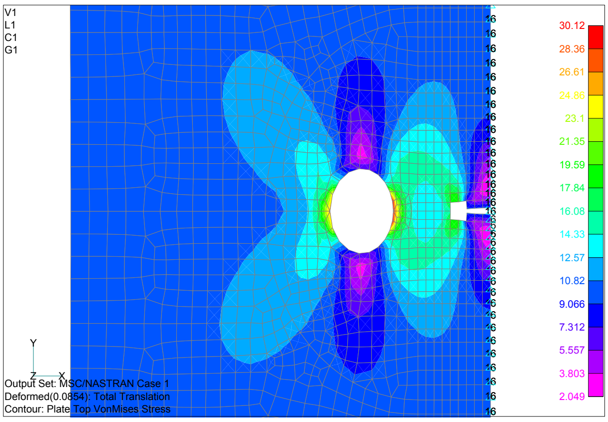


Fig. 3 Finite Element Model of Plate with Crack Between two Circular Holes

Table 1 Comparison of SIF by FEM with Analytic Solutions for Different Crack lengths

a (mm)	2	2.5	3	3.5	4
K_I (daN/mm$^{3/2}$)	26.077	29.426	32.289	36.225	40.196
a/b	0.333	0.417	0.500	0.583	0.667
$K_O = \sigma\sqrt{\pi a}$	25.060	28.018	30.692	33.151	35.440
$Y=K_I/K_{O\ (FEM)}$	1.041	1.047	1.059	1.090	1.134
$Y=K_I/K_{O\ (ANAL)}$	1.09	1.1	1.12	1.145	1.19
Difference between ANALYTIC and FEM (%)	4.5	4.8	5.5	4.8	4.7

To determine corrective function Y we started with analytic expression of stress intensity factor in the next form:

$$K_I = Y\,S\sqrt{\pi\,a} \tag{4}$$

where **Y**- corrective function, **S**-nominal stress and **a** – crack length. Here, corrective function **Y** is unknown and will be determined by finite elements. For that purpose special 6-node singular finite elements around crack tip will be used. Using FEM here are determined the stress intensity factors for various crack length, Table 1. Using these SIF`s for various crack lengths by applying FEM here are a corrective function is defined in polynomial forms:

$$Y_4^{MKE} = 0.152 + 1.2883\,a - 0.68483\,a^2 + 0.15667\,a^3 - 0.01267\,a^4 \tag{5}$$

and

$$Y_5^{MKE} = 0.90288 + 0.09774\,a - 0.0058\,a^2 - 0.0053\,a^3 - 0.0007\,a^4 + 0.00047\,a^5 \tag{6}$$

Analytic form of corrective function is the form as follows:

$$Y^{ANAL}{}_I = \left(\begin{array}{l} 1.08899 + 0.04369\left(\dfrac{a}{b}\right) - 1.77302\left(\dfrac{a}{b}\right)^2 + \\ 9.21212\left(\dfrac{a}{b}\right)^5 - 15.8683\left(\dfrac{a}{b}\right)^4 + \\ 9.48718\left(\dfrac{a}{b}\right)^5 \end{array} \right) \tag{7}$$

The difference between present Finite Element results and analytic results for the corrective function is within 5%, Table 1.

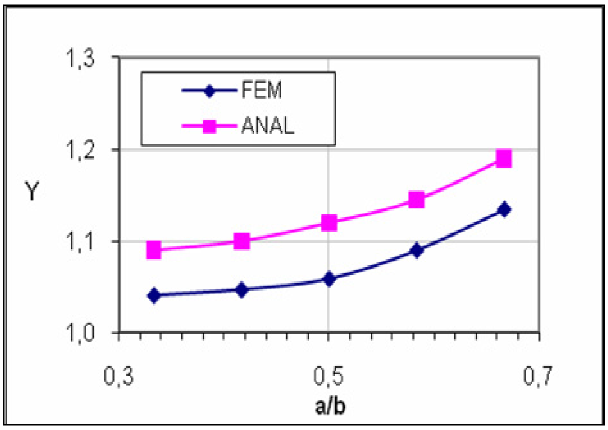


Fig. 4 Comparissons of Corrective Function Determined by FEM and Analytic Solution

Using expressions for stress intensity factors obta–ined by finite elements, for skin with a crack between two circular holes, according to expressions (6) and (7) and relation for crack growth is defined based on Strain Energy Density (2) relation **a-N,** Fig. 5.

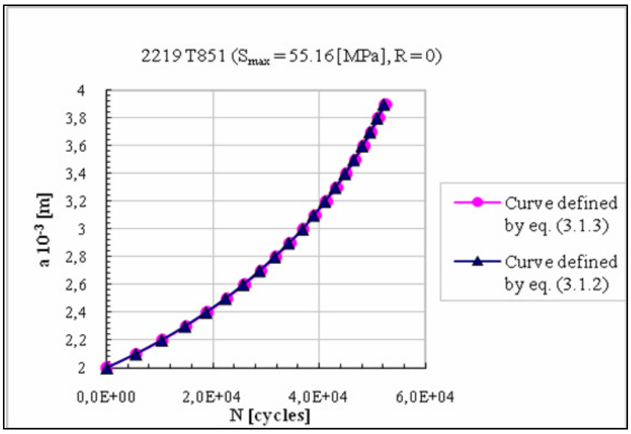


Fig. 5 Crack Growth Analysis of Plate with Crack Between two Circular Holes using SED and derived analytic expressions for SIF using singular finite elements

This means that the relation **a-N,** Fig. 5**,** or residual life estimation of a skin/plate with a crack between two circular holes for cyclic loads S_{max}=55.16 MPa (R=0) determined by SED and analytic expressions of stress intensity factors is defined by FEM. Presented compu–tation procedure, which combines finite element method to establish analytic expressions for SIF`s and SED in which are used cyclic material properties, represent a general approach for residual life estimation of aircraft structural components.

3.1 Residual Life Estimation of Cracked Lugs

Here are considered cracked aircraft attachment lugs, Fig 6. Once a finite element solution has been obtained, Fig. 7, the values of the stress intensity factor can be extracted from it. To determine Stress Intensity Factors of cracked aircraft attachment lugs, here the method based on extrapolation of displacements around tip of crack is used.

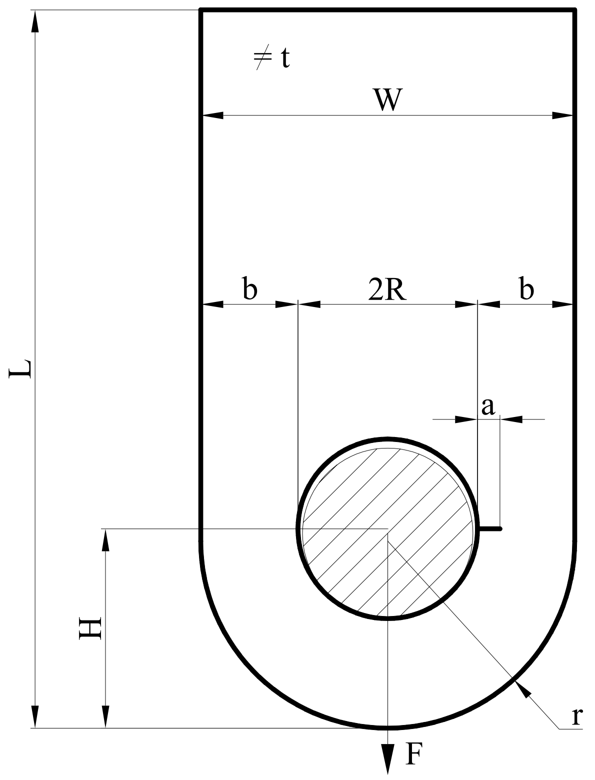


Figure 6 Geometry of cracked lug 2

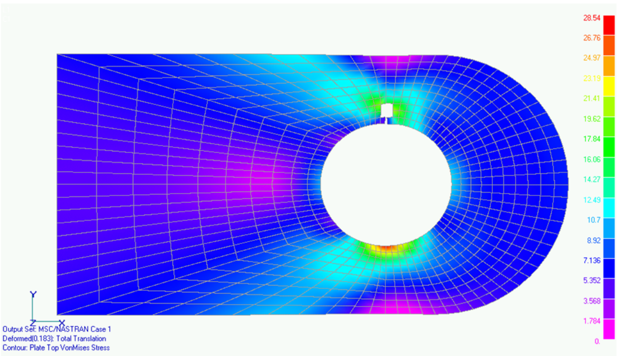


Figure. 7 Finite Element Model of cracked lug with stress distribution

The subject of this analysis is cracked aircraft lugs under cyclic load of constant amplitude and load spectra. For that purpose, conventional Forman crack growth model and crack growth model based on strain energy density method are used. The material of lugs is Aluminum alloy 7075 T7351 with the next material properties: σ_m=432 N/mm^2 ⇔ Tensile strength of material, σ_{02}=334 N/mm^2, K_{IC}=2225 [N/mm$^{3/2}$], Dynamic material properties (Forman`s constants): C=3* $10^{-7}$, n=2.39, Low-Cyclic fatigue material properties: $\sigma_f'$=613 MPa, $\varepsilon_f'$=0.35, $n'$=0.121. The stress intensity factors (SIF`s) of cracked lugs are determined for nominal stress levels: σ_g = σ_{max}=98.1 N/mm^2 and σ_{min}=9.81 N/mm^2. These stresses are determined in net cross-section of the lug. The corresponding forces of lugs are defined as F_{max}= σ_g (w-2R) t = 63716 N and F_{min}= 6371.16 N, that are loaded of lugs. For stress analysis contact pin/lug finite element model is used. For cracked lug, Fig. 6, with initial crack a_0, SIF is determined using finite elements, Fig. 7. For cracked lug No.2, Fig. 6, with crack through the thickness the crack growth behavior under two-level load spectra is considered. The first level of load spectra is defined as: σ_{max}=142,8 N/mm^2, σ_{min}=14,28 N/mm^2 for the first 1000 cycles. The second level of load spectra is defined as: σ_{max}=38,1 N/mm^2 and σ_{min}=14,28 N/mm^2. Crack growth numerical simulation of cracked lug carried out using SED method and conventional Forman`s method, Figure 8.

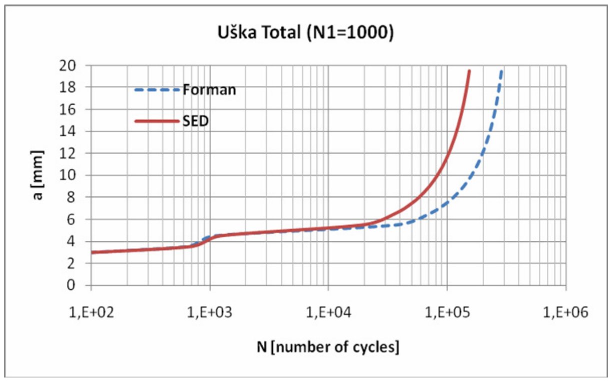


Figure. 8 Comparisons crack growth behavior using SED and Forman`s methods

In Figure 8 are shown the results of crack growth for cracked lug using two methods: (1) conventional Forman`s method [10] and (2) strain energy density method [9,11] (SED).

4. CONCLUSION

This investigation is focused on developing efficient and reliable computation methods for residual fatigue life estimation of damaged structural components. Special attention has been focused on determination of fracture mechanics parameters of structural components such as stress intensity factors of aircraft cracked skins and lugs. Predictions and experimental investigations for fatigue life estimation of an attachment lug and skin under load spectrum were performed. From this investigation following is concluded: A model for the fatigue crack growth is included which incorporates the low cycle fatigue properties of the material. Comparisons of the predicted crack growth rate using strain energy density method with experimental data and conventional Forman`s model points out that this model could be effective used for residual life estimations.

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ПРОЦЕНА ПРЕОСТАЛОГ ВЕКА ЕЛЕМЕНАТА АВИОНСКИХ КОНСТРУКЦИЈА СА ИНИЦИЈАЛНИМ ПРСКОТИНАМА

М. Максимовић, И. Васовић, К. Максимовић, Н. Тришовић, С. Максимовић

Предмет овог истраживања је усмерен на успостављање прорачунске процедуре за анализу чврстоће елемената авионских конструкција са аспекта замора и механике лома. За ту сврху овде ће бити успостављена прорачунска процедура за процену преосталог века елемената авионских конструкција типа дела оплате крила и ушки под дејством цикличних оптерећења константне ампли– туде и спектра оптерећења. Посебан аспект истраживања се односи на примену густине енергије деформације (ГЕД) за процену преосталог века елемената конструкција са иницијалним оштеће– њима типа прскотина.

За одређивање аналитичких израза за факторе интензитета напона овде су коришћени специјални сингуларни коначни елементи. Верификација прорачунских процедура за процене преосталог века је подржана са са аналитичким и експерименталним резултатима укључивши и тестове на замор посебно са аспекта експерименталног одређивања малоцик– лусних заморних карактеристика материјала.