ANALYTICALLY - EXPERIMENTAL STUDY OF DAMAGE TOLERANCE OF AIRCRAFT STRUCTURES

Mr. Boris G. Nesterenko
Moscow Institute of Physics and Technology (State University)
Department of Aerodynamics and Flight Engineering, Russia

Keywords: *damage tolerance, fatigue crack growth, residual strength, degradation*

Abstract

Presented in this paper are the results of the research on some actual problems on ensuring damage tolerance of the airplanes. The first problem considered is the fatigue crack growth under random spectra. The analytically experimental study of fatigue crack growth in specimens was conducted at a regular and random spectrum of loading. Generalized Willenborg model was used for the calculations of crack growth under the spectrum of transport airplane wing loading. The second problem considered is the residual strength of the stiffened structure such as skin with stringers. An effective engineering method using the R- curve of the material was proposed for analysis of the residual strength. In that case the experimental investigation of the R- curve of the material of a passenger airplane fuselage skin was conducted. The results of the analysis were compared to the experimental results, published in the literature. Also the experimental research of the residual strength in case of multiple site fatigue damages was carried out and the criterion of the residual strength in such case was proposed. The third problem considered is the influence of the airplane long operation on the fatigue properties of structure material. In the tests of the specimens that were cut from a wing skin of long operated airplanes, the change of material properties was observed.

1 Introduction

Service life and safe operation of airplanes is ensured in terms of damage tolerance principles Damage tolerance characteristics of airplane structure is the fatigue crack growth under random load spectra and the residual strength of the structure. Currently there are several analytical models of crack growth under random spectra. But there is no universal model of analysis. The state of analysis methods shows also the need to improve methods of residual strength analysis of the stiffened structures. In standard methods of analysis of the residual strength of the structures like skin stiffened by stringers (spars) there is no account of stable crack increase in skin under static loading of that structure. Neglect of such crack increase in the analysis leads to sufficient errors. But the residual strength analysis using R-curves gives the ability to increase the accuracy of analysis. The problem of the residual strength of longitudinal joints of the pressurized fuselage skin with multiple site cracks (MSD) is particularly important for ensuring safe operation of aged aircraft structure. The analysis of longitudinal lap joints fracture is quite complicated task due to bending of the joint while tension. There were proposed some criteria of the residual strength of structure elements with MSD, but they are complicated to apply them for the analysis of full-scale structure. More simple criteria of the residual strength are required. While ensuring safe operation of the aging airplanes arises the question of degradation of structure material fatigue characteristics due to cyclic loading, corrosion and possible “aging” of the material. Test-analytical investigation was made of considered problems on crack growth analysis, material degradation, method of residual strength analysis using R-curves was proposed for the stiffened structure as well as criterion of the residual strength in case of multiple site cracks.

2 Analysis of crack growth under random load spectra

Service life, threshold and interval of airplane structure inspections are defined by the crack growth duration in principle elements. Appearance of cracks during airplane long operation due to fatigue, corrosion and casual damages as well as due to initial flaw requires the advanced methods of crack growth analysis. Load spectrum of airplane structure has variable random loads. Experimental investigations of load interaction show that load spectra type and composition, and order of cyclic load alternation have a sufficient effect on crack growth. Principally this effect is the retardation in crack growth after action of high tensile load. Retardation is caused by the residual compressive stresses in plastic zone near the crack tip. In case of sufficiently high load in comparison with following one crack stop is possible. Another effect that must be taken into account is the influence of compressive loads after high tensile ones. Compression can strongly decrease the retardation, and that leads to acceleration in crack growth. There is a number of load interaction models for crack growth analysis.

To determine duration and rates of crack growth as characteristics of damage tolerance a test–analytic investigation of fatigue crack growth in the skin of lower and upper wing surface of passenger airplane was carried out. Generalized Willenborg model of load interaction [1] was investigated in this work. Tests of specimens on crack growth were carried out in TsAGI, software was developed for the crack growth analysis in case of random load spectra.

2.1 Experiments

The purpose of the specimen tests under random load spectra was to obtain the values of crack growth duration in case of real operational loads on airplane structure and comparison of the test results with analytical ones to improve the method of the analysis.

2.1.1 Specimens and test equipment

Experimental investigations of the fatigue crack growth rates and durations were carried out on the standard plane specimens of 440x100x8mm size with central crack. Initial central through crack was 6.3 mm. Cracks were made with electric spark. Specimens were of aluminum alloy 2324-T39 and 7055–T77 for the investigations of crack growth in skin of lower and upper wing surfaces correspondingly. Tests were carried out on electrohydraulic rigs EGM25 and MTS25, that allow to modulate quasi-random loads. Cracks were visually observed using optical microscope. To simplify observation the scale-rulers with the scale interval of 0.5 mm were attached to each specimen. Microscope scale interval was taken of 0.1 mm.

2.1.2 Regular loading

To define the crack growth rate vs. stress intensity factor 8 specimens of 2324-T39 alloy and 8 specimens of 7055-T77 alloy were fatigue tested under constant loads. Cyclic loading was sinusoidal. Cycle aspect ratio and maximum load level were variable. Loading frequency was 5 Hz. The purpose of these tests was to obtain parameters of Walker equation. This equation was used in the analysis of crack growth under random load spectra. Figure 1 shows the test results of 2324-T39 alloy specimens for the different cycle aspect ratios.

2.1.3 Random loading

Test of 3 standard central cracked specimens of 2324-T39 alloy (skin of lower wing surface) and of 3 specimens of 7055-T77 alloy (skin of upper wing surface) were carried out under TC1 and TC2 load spectra correspondingly. Spectra TC1 and TC2 are the typical load spectra of the lower and upper wing surfaces of the passenger airplane like Boeing-767 [2,3]. Specimens were loaded by blocks, one block was of 5000 flights of five different types (A, B, C, D, E) , that were mixed in random way. Maximum load frequency in quasi-random spectra was limited to 10 Hz.

Results of the tests of 2324-T39 alloy and of 7055-T77 alloy specimens are presented on the Figure 2.

2.2 Analysis

Before analyzing the crack growth in case of random loading, load spectra were treated and so called ‘full cycles’ were derived. Method of full cycles generation from the random spectrum was similar to the “Rain flow” method but the consequence of peaks was taken into account. Thus initial spectra were modified and used in computer program of the crack growth analysis. Analyses were carried out for the crack growth in skin of the lower wing surface in case of TC1 spectrum and for the crack growth in skin of the upper wing surface in case of TC2 spectrum. Walker equation of crack growth rate vs. stress intensity factor was used in the following way:

$$\frac{da}{dN} = C\left[Z\,K_{\max}\right]^{p}$$

$$\begin{cases} Z = (1-R)^{\alpha}, & 0 < R < 1.0 \\ Z = (1-\beta R), & -1.0 < R \le 0 \end{cases}$$

Parameters of Walker equation were obtained after treating the results of regular loading tests. Constants C, p, α were defined as test data approximation of the diagram “da/dN–ΔK” and test diagram “2a–N” and then mean values C, p, α of constants in the equation were chosen

The values of Walker equation constants are presented in the Table 1.

Following analysis shows that for more accurate calculation of crack growth under random spectrum approximation of the regular loading test data must be carried out for the specified range of cracks length and delta stress intensity factors. This range is from threshold values up to about 80% of maximum values. It can be explained that while approaching to the critical values of crack lengths or stress intensity factors crack growth rates sufficiently increase. Crack growth duration in random loading case will be mainly defined by the range of crack size about 80% of critical crack size. Thus test data on regular loading should be approximated in the range of crack size up to about 80% of its critical value. Such approach can increase the accuracy of the analysis. It should be noted that it is rather difficult to define the Walker equation constants that will give the required accuracy of crack growth rates in wide range of cycle aspect ratios.

Results of the analysis of crack growth in skin of the lower and upper wing surfaces are presented on the Figure 2.

Accuracy of the analysis for the lower wing surface is within 6% - 19% range of error in comparison with separate test data and is 12% in mean. So, used Willenborg model describes good enough the load interaction in such spectra like TC1 where compressive loads are not sufficient in their amount.

For the upper wing surface sufficient discrepancy of analytical and test results is observed. Analysis accuracy in this case is in range of 12,7% to 52% of error in comparison with separate test data and is 24% in mean. So, used Willenborg model describes could not be applied for the spectrum where compressive loads are dominative in comparison with tensile ones. It follows that there is a need to develop a model that will take into account the effect of compression on crack growth rate. The possible way is to introduce in the Willenborg model some acceleration factor similar to the retardation factor. Operating with plastic zone size, we can consider that high compressive load would “decrease” this zone r_p, appeared from overload, to some zone r_p^c. Correlation between these two sizes r_p and r_p^c should be defined by some factor of proportionality, determined experimentally from the coupon tests with sufficient compression in random load spectra.

3 Method of the residual strength analysis of the stiffened structure using R – curves

This method of analysis was developed to define the residual strength of the stiffened wing and fuselage structures having a two-bay crack in a skin. Currently applied methods of the residual strength analysis of such structures based on linear fracture mechanics do not take into account the stable increase of the crack under static loading of the structure up to fracture. Experience shows the necessity of this stable growth account in the analysis. In the residual strength tests of panels of 2024-T3 alloy the increase of the original crack length was up to 50–60%. The neglect of such increase in the analysis leads to the sufficient errors. Also in some cases this leads to the uncertainty in determination of critical element in terms of residual strength – will it be the skin or the stringer. Proposed method takes into account the crack increase noted using R-curves [4]. R-curve is the relation between stress intensity factor K_R and delta of efficient crack length $\Delta 2\boldsymbol{a}_{\text{eff}}$.

$$\boldsymbol{K_R} = \sigma\sqrt{\pi\, \boldsymbol{a_{eff}}}\ ,$$

$\boldsymbol{a_{eff}} = \boldsymbol{a}_0 + \Delta\boldsymbol{a} + \boldsymbol{r}$ - efficient crack half-length,

$\boldsymbol{a}_0$ – initial crack half-length in skin,

$\Delta\boldsymbol{a}$ – crack half-length increment,

$\boldsymbol{r} = \dfrac{1}{2\pi}\left(\dfrac{\boldsymbol{K}}{\sigma_{0,2}}\right)^2$ – plastic zone on crack the tip.

R-curve characterizes quantitatively the ability of material to withstand stable increase of crack from initial size till fracture under static loading [5].

For the development of this analytical method the nomographic chart was used to define the stresses in stringers and stress intensity factors in the cracked skin of stiffened structure (Figure 3) as well as test data on the residual strength of full-scale stiffened fuselage panels of Russian passenger widebody airplane, referred further as RPWA, and DC-10 airplane.

Test on wide skin sheets were carried out and R-curves were obtained for the skin of aluminum alloys of 2024-T3 type of pressurized fuselages. Tests were carried out in accordance with the Standard ASTM E-561-94 [6] on MTS250 test rig. Specimen width was 760 mm. Using R-curves obtained the analysis of the residual strength of fuselage panels of RPWA [7] and DC-10 [8] was made by the method developed.

The method proposed is the following.

The graph of dependence between crack length in skin and stress level in structure (R_σ-curve) and graph of dependence between fracture stresses of stringer σ_{STR} and crack length should be plotted in the coordinates "$\boldsymbol{\sigma}$ –$\boldsymbol{2a}_{\boldsymbol{eff}}$", where $\boldsymbol{\sigma}$ is the stress in the structure, $\boldsymbol{2a}_{\text{eff}}$ is the effective crack length in skin. Fracture stresses of stringer can be calculated as $\boldsymbol{\sigma}_{\text{str}} = \dfrac{\alpha \cdot \sigma_{\boldsymbol{в}\ \text{str}}}{\boldsymbol{\beta}}$. Intersection point of these two curves on the plot corresponds to the residual strength of the structure.

R_σ-curve shows the relation between crack length $\boldsymbol{2a}$ in the stiffened skin (i.e. taking into account the influence of the stringer) and the level of applied stresses $\boldsymbol{\sigma}$. R_σ-curve should be obtained from the re-account of non-stiffened skin fracture by corrective factors. More simply it can be presented graphically. The graph of fracture of non-stiffened skin with the initial crack of $2a_0$ must be plotted in coordinates K – $2a_{\text{eff}}$, where $\boldsymbol{K}$ is stress intensity factor in skin, $\boldsymbol{2a}_{\text{eff}}$ is effective crack length in skin. This graph is R-curve of skin material, moved to point $\boldsymbol{2a}_0$ of initial crack. Then on the same plot the set of curves of stress intensity factors in stiffened skin $\boldsymbol{K} = \sigma_{\boldsymbol{i}}\sqrt{\pi\, \boldsymbol{a_{eff}}} \cdot \boldsymbol{C}$ vs. crack length for given constant stresses $\boldsymbol{\sigma}_{\mathbf{i}}$ = *const* in structure should be drawn. The influence of stringer is accounted by the factor $\boldsymbol{C}$, that depend on current crack length. Factor $\boldsymbol{C}$ can be defined analytically or by finite element method and must be specified experimentally. In the analysis the values of $\boldsymbol{C}$ factors were taken from the nomographic charts of TsAGI [9]. Points of intersections of these curves with displaced R_K-curve will define R_σ-curve as correspondence of $\boldsymbol{2a}_{\text{i}}$ to stresses $\boldsymbol{\sigma}_{\text{i}}$=*const*. In other words the R_σ –curve defined in this way is the solution of equation systems:

$$\begin{cases} \text{K} = \mathbf{K_R}\ (2\text{a} - 2\text{a}_0) & (1) \\ \text{K} = \sigma_{\boldsymbol{i}}\sqrt{\pi\, \text{a}_{\text{eff}}} \cdot \boldsymbol{C}(\text{a}_{\text{eff}}) & (2) \end{cases}$$

(1) corresponds the R-curve, displaced to $2a_0$;
(2) is the stress intensity factors in the stiffened skin for given constant stresses $\boldsymbol{\sigma}_{\text{i}}$ = const ;

The results of the analyses were compared with test data on the residual strength of full-scale fuselage panels of RPWA and

DC-10 airplane. The difference in residual strength was in scope of 3-4 %.

While analysis of the fracture stresses of the stringer the values α $\sigma_{\text{B str}}$ for stringers of D16AT and 7055-T6 alloys were equal to 400 MPa and 500 MPa correspondingly. Factor α gives the decrease of stringer strength due to the rivet holes, $\sigma_{\text{B str}}$ is the ultimate strength of stringer material. The value $\alpha \cdot \sigma_{\text{B str}}$ were defined taking into account combined effect of bending and tensile stresses in stringer. Corrective factor **β** of stringer overload depends on current crack length. Factor **β** can be defined analytically or by finite element method. In the analysis the values of **β** factors were taken from the charts of TsAGI [9].

4 Criterion of the residual strength in case of multiple site cracks

The problem of the residual strength of the longitudinal joints of the pressurized fuselage skin with MSD is particularly important for ensuring safe operation of aged aircraft structure. Such multiple site cracks were the reason of accident with Boeing 737 in 1988. The analysis of longitudinal lap joints fracture is quite complicated task due to bending of the joint while tension. After this accident a lot of attention was paid to the residual strength of the fuselage skin longitudinal joints. The analysis of longitudinal lap joints is rather complex problem due to the bending effect while joint tension.

Currently there are some criteria of the residual strength of structure elements with MSD. These criteria operate with plastic zone size near crack tips [10] and with the crack tip open angle [11], but they are complicated to apply them for the analysis of full-scale structure. More simple criteria of the residual strength are required.. For estimation of the residual strength of the structure with MSD a simple fracture criterion was proposed :

$$\sigma_{fracture}^{net} = \alpha \cdot \sigma_{0,2}$$

where $\sigma_{0,2}$ is the yield strength of the material, α is the dimensionless empiric factor. Net fracture stress $\sigma_{fracture}^{net}$ is calculated regarding the effect of cross section reduction of load-bearing elements by the rivet holes and multiple site cracks. To define empiric factor α special tests investigations were conducted.

Specimens of two rivet row longitudinal lap joints of fuselage skin were tested. The material was 2024-T3. Specimen width was 462 mm. Each fastener row had 21 rivets. Rivet holes were not countersunk. Tests of the specimens till fracture were conducted under cyclic load with $\Delta\sigma$ = 110 MPa and aspect ratio R = 0,1 on MTS250 hydraulic machine. Stress level of 110 MPa corresponds to the circumferential stresses caused by pressurizing of airbus fuselage. In the fracture sections of the specimens multiple cracks were observed near most of the holes. Sizes and locations of the cracks observed in the tests correspond to those in real full-scale structure. Measurements of crack sizes were made by optic microscope. Squares of multiple fatigue cracks in the cross section of the joints were calculated and the values of net stresses σ^{net} were defined. These values were compared to the yield strength of the material and proposed criterion was verified. The values obtained of the factor α are in range of 0.48 to 0.57. That is in agreement with data on full-scale structure tests.

5 Degradation

To answer the question of degradation tests of two specimen types were conducted. Fist specimens were cut out of wing skin of aged Russian turbo-prop airplane (D16AT alloy) and fuselage skin of long operated turbojet Russian airplane (D16AT alloy). Another similar specimens were made of new sheets of corresponding alloys. Initial through crack was 30mm. Specimen size was 450x160 mm. Specimen thickness was 1,5mm for the fuselage and 5mm for the wing skin. Tests were carried out on MTS100 test rig under sinusoidal loading with σ_{max}=130 MPa and σ_{min}=3 MPa and frequency of 0,17 Hz. Crack size was measured visually by microscope and by foil gauges.

Test data on crack growth rates in tested specimens show that crack resistance properties of materials after their operation are worse than of non-operated ones. It can be seen distinctly in case of the wing skin on Figure 5. After heat treatment of the wing skin specimens the decrease in crack growth was observed. Nevertheless degradation in case of thin skin can not be surely stated. Test results point on possible degradation of aluminum alloys. For the quantitative estimation of degradation more experiments should be carried out to accumulate statistic data.

6 Conclusion

Test–analytical investigation of crack growth in the specimens of 2324–T39 and 7055–T77 alloys was carried out. Using test data constants in Walker equation were defined. Accuracy of analysis using Generalized Willenborg model was investigated by comparison with test data. Efficiency of Willenborg model was shown for the analysis of the lower wing skin and sufficient difference of test and analytic results in case of crack growth in the upper wing skin.

Residual strength analysis method using R-curves was developed for the analysis of the stiffened structure. R-curves of the fuselage skin material aluminum alloy 2024-T3 were obtained while method development. Efficiency of the developed method was shown by the comparison with the test results published of the residual strength of the fuselage panels with tow–bay crack of Russian passenger wide body airplane and DC-10.

The tests of the fuselage longitudinal lap joints with two rivet rows were conducted. Multiple site cracks formed in joints while tests. On the basis of the tests conducted the criterion of the residual strength $\sigma^{net}_{fracture} = \alpha \cdot \sigma_{0,2}$ was proposed and α factors were obtained.

Test investigation of material properties degradation was accomplished. Sufficient degradation of crack resistance was observed in the material of the passenger airplane wing skin of 5mm. Degradation of crack resistance in 1,5mm fuselage skin can not be stated.

7 Acknowledgements

Author would like to thank the head of R&D Department 18 of Airframe Integrity of Central Aerohydrodynamic Institute for the ability to accomplish experimental investigations and working team for collaboration in tests.

References

[1] J.P. Gallagher, P.C. Miedlar, C.M. Cross, M. L. Papps *Crack 93 System,* User manual,University of Dayton Research Institute, UDR-TR-93-103

[2] Ulf G. Goranson "*Damage tolerance. Facts and Fiction* " 17 Symposium of the International Committee on Aeronautical Fatigue, Stockholm Sweden, June 9, 1993

[3] Fowler R.F and Watanabe R.T. "*Development of jet transport airframe test spectra"* Issue of Boeing commercial airplanes – Seattle, Washington, USA, May 1989,

[4] T. Swift *"Damage tolerance capability"*, Journal Fatigue, 1994, Vol. 16 Number 1, pp. 75-94,

[5] D.Broek *"Fracture mechanics"* –High School , Moscow 1980

[6] *Standard Practice for R-curve Determination*, ASTM E 56.

[7] Nesterenko G.I., Selikhov A.F. Application of damage tolerance concept in developing wide-body aircraft.- In "*Strength of aircraft structures*", Moscow, Mashinostroyeniye, 1982, p.151-189 (in Russian)

[8] Swift T. The application of fracture mechanics in the development of the DC-10 fuselage. – *AGARD-AG-176,* v.*5*, 1974, p.227-287.

[9] Nesterenko G.I. Basis of lifetime machine design. – In "*Mashinostroyeniye. Encyclopedia. Vol.IV-3. Machine reliability*", - Moscow, Mashinostroyeniye, 1998, p.408-439 (in Russian)

[10] Swift T. Design of redundant structures. – *AGARD-LS-97*, 1978, p.9.1 – 9.23.

[11] B.R. Seshadri, J.C.Newman Jr., D.S. Dawicke and R.D. Young *"Fracture analysis of the FAA/ NASA Wide stiffened panels"* , Proceedings of the Second Joint NASA/FAA/DoD conference on aging Aircraft 1998, Williamsburg, Virginia, USA

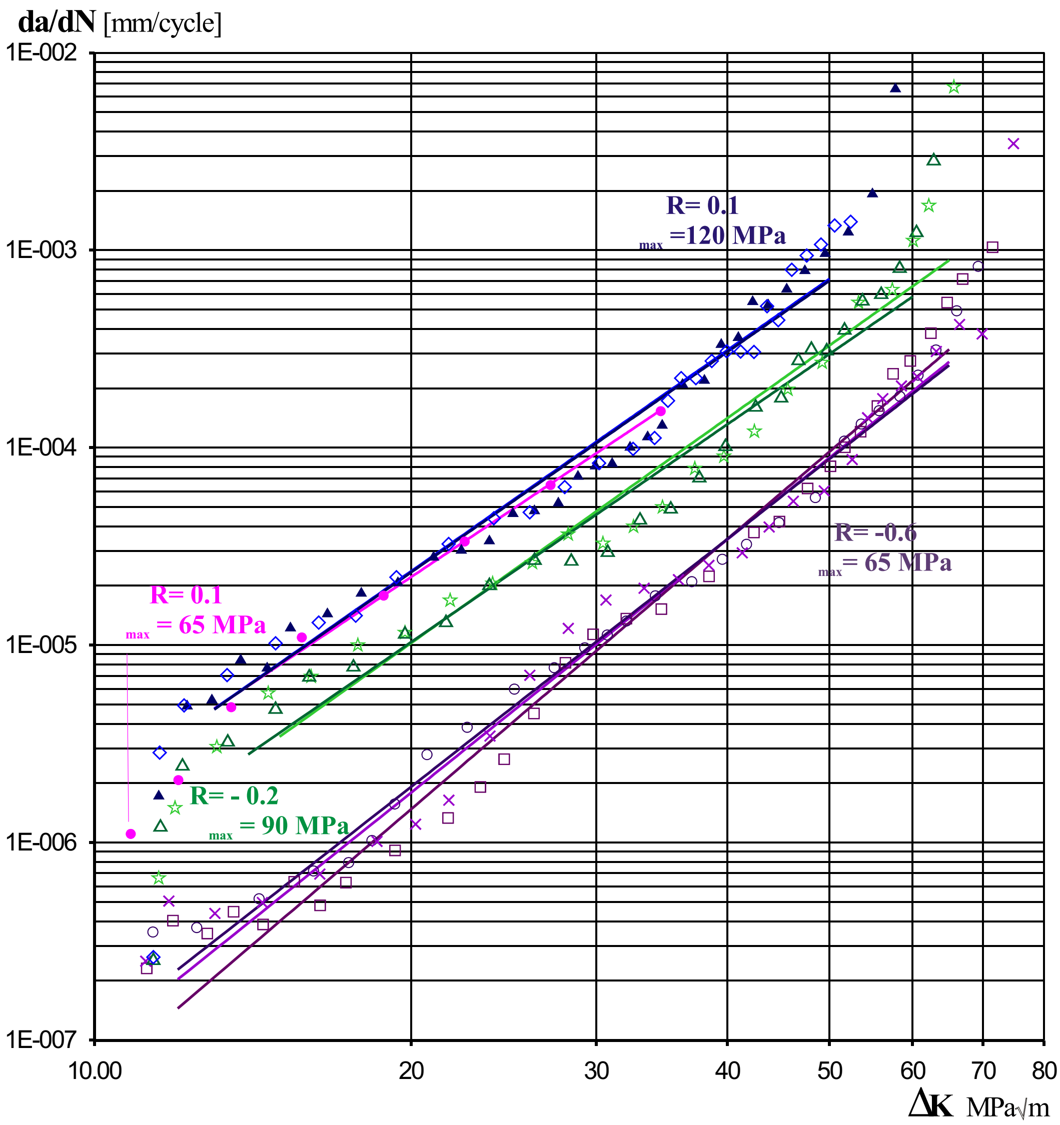
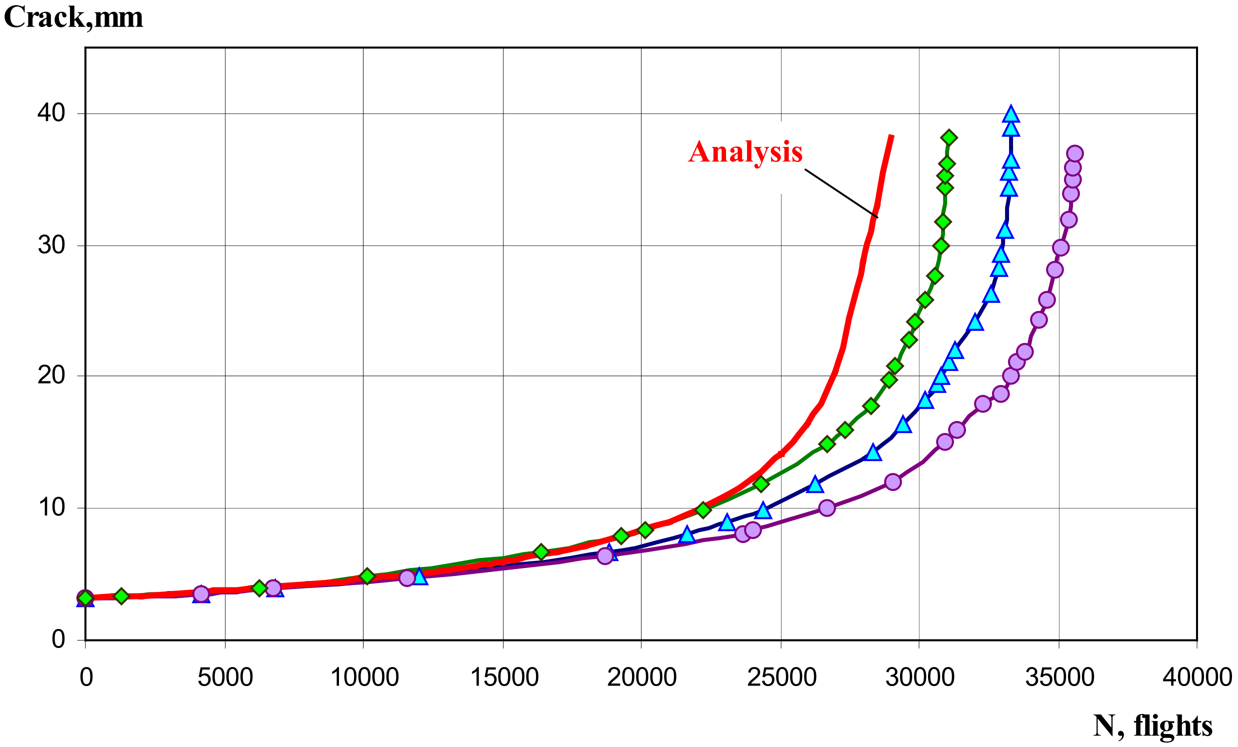


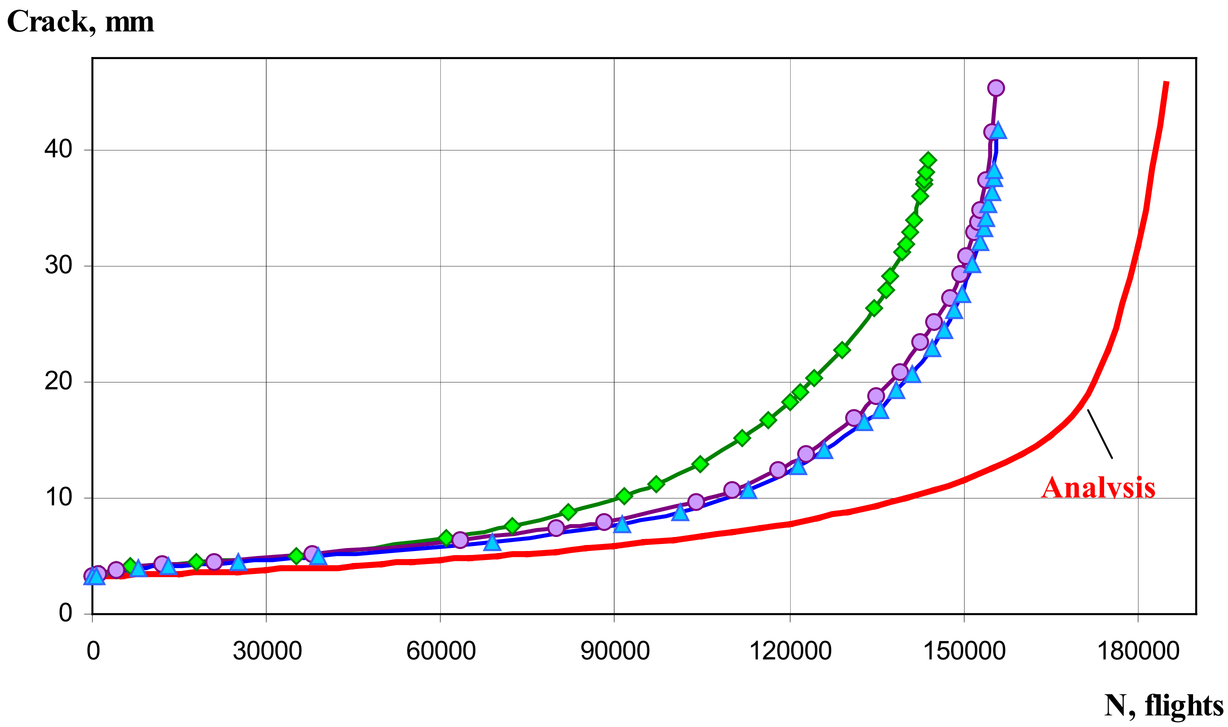
Figure 1 Crack growth rate vs. stress intensity factor in 2324-T39 alloy under constant load spectra of different amplitudes and aspect ratios

Table 1 Parameters in Walker equation

	From database		Corrected from regular loading tests	
Material	**p***	**C***	**p**	**C**
2324-T39	3.9	2.86 E-9	3.7	3.5 E-10
7055-T77	3.5	8.83 x 10	2.9	7.8 E-10



Crack growth under typical load spectra of airplane wing lower surface.
Specimen test data and analysis for 2324-T39 alloy.



Crack growth under typical load spectra of airplane wing upper surface.
Specimen test data and analysis for 7055-T77 alloy.

Figure 2 Comparison of analyzed crack growth and test results

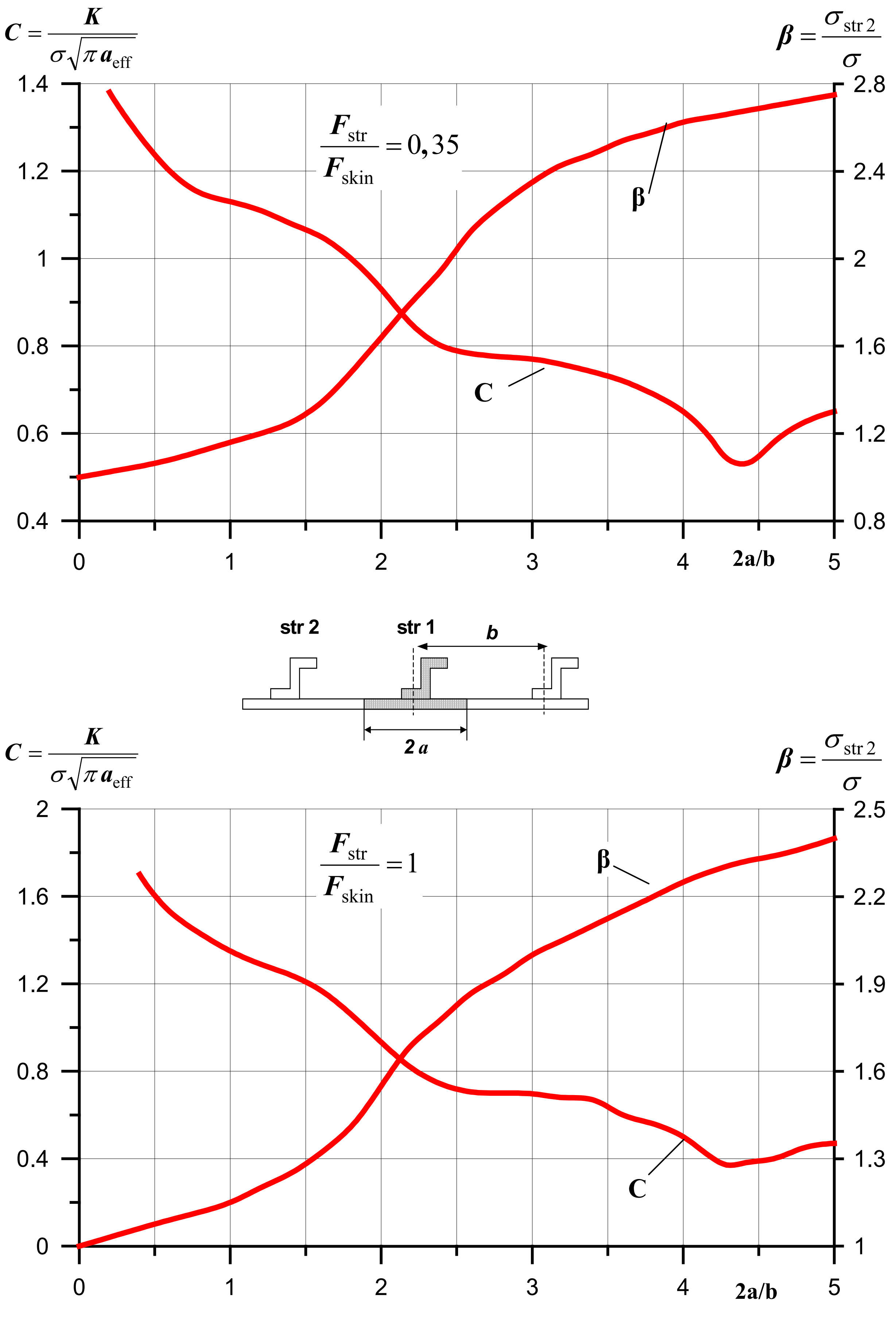
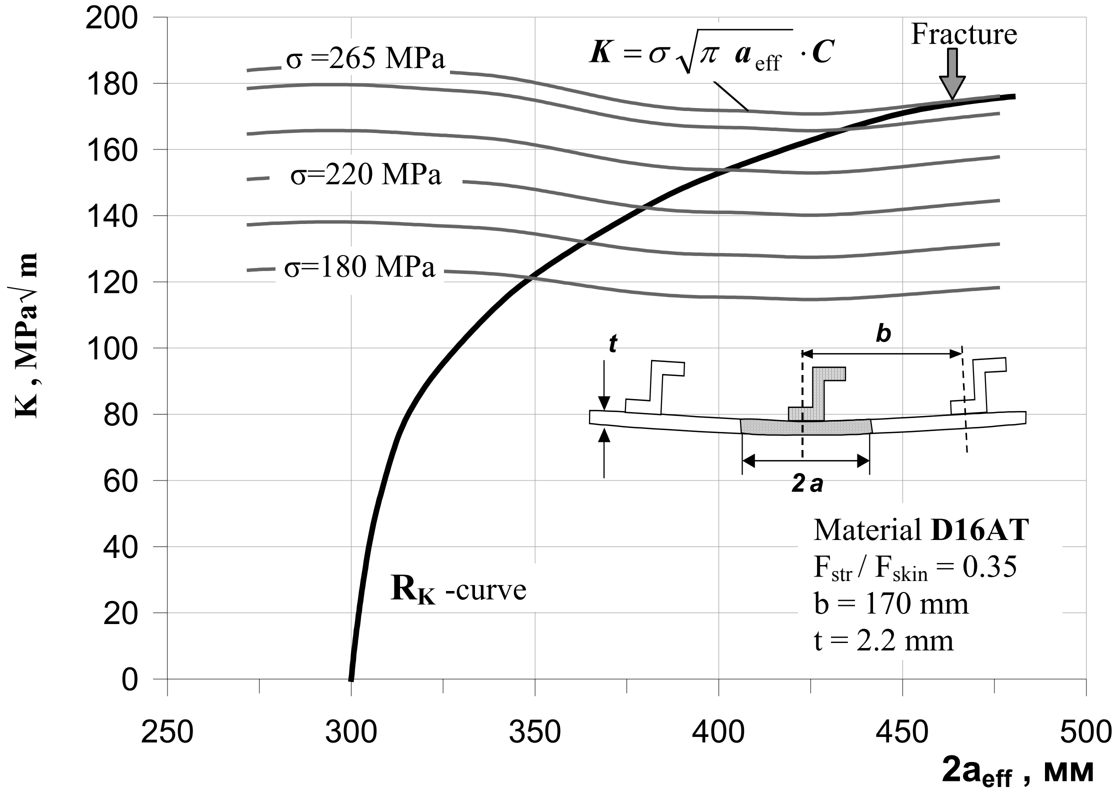


Figure 3 Correction factors **C** и β for the different ratios between stringer and skin squares



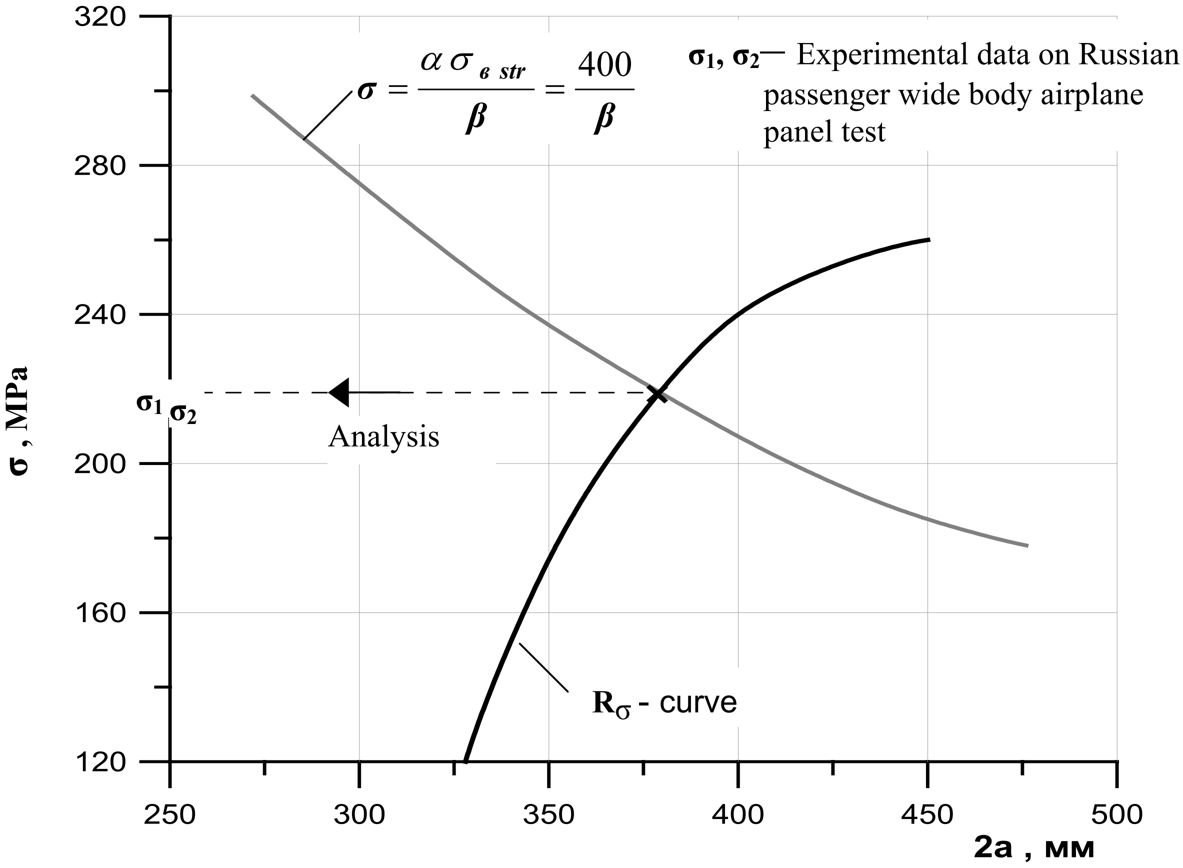
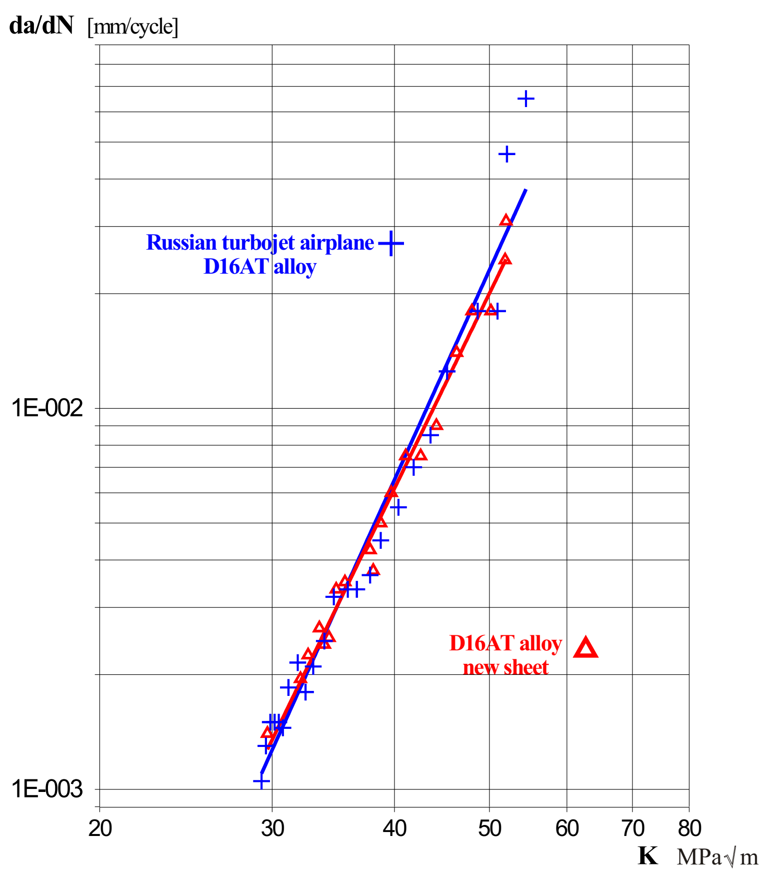
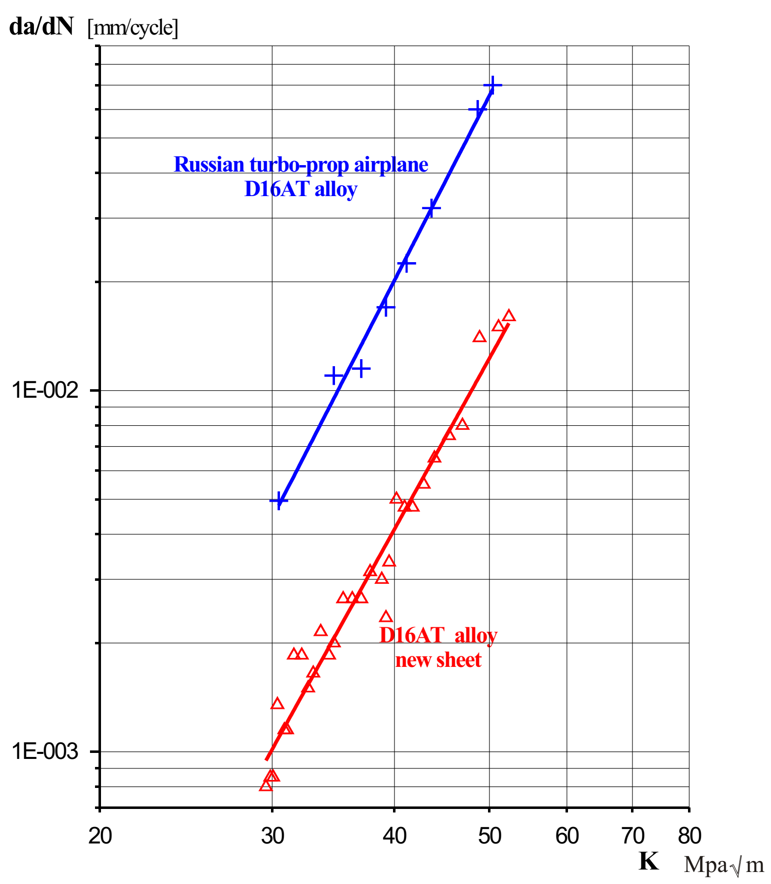


Figure 4 Analysis of fracture stresses for the fuselage panels of the Russian passenger wide body airplane (RPWA)



Crack growth rates in the fuselage skin specimens of 1,5 mm width



Crack growth rates in the wing skin specimens of 5 mm width

Figure 5. Test investigations of material properties degradation