RESIDUAL STRENGTH ANALYSIS OF STIFFENED PANELS USING R-CURVES OF SKIN MATERIAL

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Abstract

This paper presents a method of the residual strength analysis of stiffened structures with a two-bay crack in the skin under a broken stiffening element. This method takes into account the stable crack growth in the skin by means of R–curve of the skin material. The analyses of the residual strength of riveted and integrally stiffened wing and fuselage panels were carried out using this method developed. Results on the residual strength analyses of these panels were compared with test data. The conclusion was drawn about the accuracy and efficiency of the offered method of analysis.

1 Introduction

Generally applied for residual strength analysis (RSA) of stiffened structures are approximate methods, developed on the basis of linear fracture mechanics. Examination of the contemporary state of these methods showed the need for them to be improved. In most published methods of the RSA of the stiffened structures such as skin stiffened by stringers or frames, there is no account of the stable crack growth in the skin during static loading of the structure. However, the increment of the crack during its stable growth is commensurable with the initial fatigue crack, from which the stable growth starts. For instance, the monotone crack increase in 2024 alloy sheet is about 50–60% of its initial size. Neglect of such crack increase in the analysis leads to the sufficient diminution of the accuracy. In some cases it also leads to the uncertainty in defining a critical element, whether it is a skin or a stiffener. The quantitative description or the crack increase in the skin, the account of the plastic deformations near the crack tip and the definition of the critical element can be made by means of application of R-curve of the skin material.

Application of R-curves in the residual strength analysis was considered in [1]. This paper defined the way to calculate the critical stresses in the skin. Lately, in [2], [3] and [4], the fracture stresses of the stiffened panel skin were calculated using the same approach, regardless the residual strength of stringers. It was assumed that skin was the critical element.

But in [2] the initial skin crack of the full-scale panel was sufficiently smaller than a two-bay distance. In [3] the analyzed panel was with bonded stringers, the dimensions of that panel were 2 times less than that of the full-scale airplane panel, and the initial crack was of one bay length. In [4] the residual strength of the panel was calculated as that of the sheet considering cracks in a stiffener via stress intensity factor increase. The stiffener residual strength was not considered.

The main goal of the investigation presented in a given paper was to develop the effective method of RSA using R-curves. The next step was to calculate the residual strength of the full-scale wing and fuselage riveted and integral panels using the method developed. Attention was paid to the residual strength of panels with regulated damages as the two-bay cracks under the broken stringer (frame). Riveted and integral wing panels of AN-124 airplane (1163T alloy), riveted and integral fuselage panels of IL-86 (D16AT alloy) and riveted fuselage panel of DC-10 were analyzed on the residual strength. The results of the analyses were compared to the appropriate test results to validate the method developed.

2 Background for the residual strength analysis

First, it should be noted that the stable slow crack growth is typical for ductile alloys, such as aluminum alloys of 2000 series.

In case of large two-bay cracks in stiffened skin two criteria of possible failure should be considered simultaneously: the first criterion is the skin failure and another one is failure of the stiffener. It can be noted that in FAA paper [2], [4] and NASA [10] R curves were used only for the residual strength of the skin, while the residual strength of stringers was beyond the scope of the consideration.

For the residual strength analysis of stiffened structure the R-curves of a skin material have to be defined by testing wide specimens of 750 –1200 mm.

Skin stress intensity factors should be defined regarding the influence of a stiffening element (stringer or frame). Calculation of the stresses in the stiffener should be carried out taking into account the skin crack growth.

In the method developed the author used the dimensionless correction factors *?* to calculate the stress intensity factors in the skin as following:

$$\boldsymbol{K} = s\sqrt{p\ a}\cdot \mathrm{C} \qquad (1)$$

and the stiffener (stringer) overload factors *ß* due to the crack influence as

$$s_{\mathrm{str}} = \frac{a\cdot s_{B\ \mathrm{str}}}{ß} \qquad (2)$$

where a is the coefficient of the stiffener strength decrease due to the rivet holes, $s_{?\ \mathrm{str}}$ – is the ultimate strength of the stiffener material.

The corrective factors *C* and *ß* developed in TsAGI and published as "*C,ß vs. 2a/b*" charts [5] (Figure 1) are used in the presented method. In these charts *C* and *ß* factors were calculated using finite element method and justified by data of the residual strength tests of large panels and full-scale structures

For the verification of the developed method, test data on the residual strength of the riveted and integrally stiffened wing and fuselage panels with two-bay skin crack under the broken stringer were used. These data were published in [6], [7] and [8].

3 R-curves of skin materials

To obtain the R-curves of the skin 2024-T3 alloy of pressurized fuselages, tests of wide panels were carried out in TsAGI. Tests were fulfilled in accordance with the ASTM E-561-94 standard.

All the specimens were tested on the electro-hydraulic test rig MTS250. To obtain R-curve test data were processed also in accordance with the ASTM E-561-94 standard.

The test R-curve of the aluminum fuselage skin sheet of 1.8 mm is shown in Figure 2. Test R-curves of aluminum wing panel of 1163T alloy were taken from [9].

4 Principles of the residual strength analysis using R-curve

The principle of K_R-curve application for the residual strength analysis of stiffened skin with a crack is shown in Figure 3. Calculated and drawn is the set of the curves of stress intensity factors K_R in the skin vs. effective crack length 2*a* for constant stresses *s*. Plotted on the same graph is the K_R-curve shifted to the point of the initial skin crack length $2a_0$. Critical level of the stresses in the skin for the considered initial crack is defined from the corresponding "$K – 2a_{eff}$" curve that is tangent to K_R-curve of the skin material. For the higher stresses the rapid failure will take place. For lower stresses crack would stop to grow when reaching the value defined by the intersection point of "$K – 2a_{eff}$"- curve with the K_R-curve.

In case of the stiffened structure "skin with crack + stringer" the residual strength is defined as following: for the less durable skin, i.e. critical in comparison with the stringer (Figure 4, case a)), the analysis should be carried out as mentioned above. For the relation of the skin and stringer's strength similar to that shown in Figure 4, case 2, the analysis should be carried out for the stringer.

For the case 2 the shifted K_R-curve should be transformed into stresses vs. effective crack length dependence for the given structure with the initial crack. More simply and obviously this transformation can be presented graphically. Plotted in the "s – $2a_{eff}$" coordinates are the values of σ, that correspond to the intersection points of the set of "$K = s_i \sqrt{p\, a_{eff}} \cdot C$" -curves, where σ_i=const, with the K_R-curve of skin material, shifted to the point of the initial crack $2a_0$. In such a way the "σ_R- curve" for the skin can be obtained. In other words, the σ_R -curve is the solution of system of equations:

$$\begin{cases} K = \boldsymbol{K}_R\left(2a - 2a_0\right) & (3) \\ K = \boldsymbol{s}_i \sqrt{p\, a_{eff}} \cdot \boldsymbol{C}\left(a_{eff}\right) & (4) \end{cases}$$

Equation (3) corresponds to the K_R-curve, shifted to initial crack $2a_0$; (4) is the stress intensity factors in the stiffened skin for given constant stresses σ_i =const;

This σ_R–curve of the skin defines the crack increase for the given stress level applied to the structure. On the same graph the fracture stresses in stringer σ_{str} should be plotted vs. $2a_{eff}$ taking into account the influence of the skin and the crack increase using equation (2). The intersection point of these two curves will define the stresses in the structure that cause the failure of the stringer and of all the structure – Figure 5.

5 Validity and accuracy of the method

To verify the method and to determine the accuracy, the analyses of residual strength of the fuselage panels of IL-86, DC-10 airplanes and the wing panel of Antonov airplane were carried out and compared with the corresponding test data.

In the residual strength analyses of the panels the following assumption were made:

- alloys D16ATB and 2024-T3 are equivalent
- R-curves for all the thickness within the range of 1.8 – 2.2 mm range are the same
- R-curves slightly depend on the direction of the roll, i.e. R-curves for "LT" and "TL" direction are similar.

In the curved fuselage panels of IL-86 airplane, the skin and stringer were of aluminum alloy D16AT. The panels were 1870 mm wide with 11 stringers and 6 frames. Fuselage panels were tested on the electrohydraulic rig by the tensile load. In the panel with one cut stringer the initial fatigue crack in the skin was grown up to the length of $2a$ =300 mm and then the residual strength of the panel was determined.

In the residual strength analysis of IL-86 panel by the method developed the value $a \cdot \sigma_{B\ str}$ for the stringer of D16AT alloy was taken equal to 400 MPa when calculating the fracture stresses of the stringer vs. crack length $2a$. This value $\alpha \cdot \sigma_{B\ str}$= 400 MPa was obtained in TsAGI [5] regarding the combined action of bending and tensile stresses in the stringer. The residual strength analysis of the curved fuselage panels of IL-86 airplane gives the fracture stresses σ = 218 MPa as shown in Figure 5.

In accordance with the test data two fuselage panels of IL-86 with the same damages failed under the gross stresses ~ 220 MPa [7]. Thus the discrepancy between analytical and test data is about 2%.

Flat fuselage panels of DC-10 airplane were stiffened by stringers and frames fabricated of high-strength alloy 7075-T6. The skin was of 2024-T3 alloy. The width of the panel was 1520 mm. The stringer in the panel was cut, the crack in the skin grown up by the cyclic loads [6].

In the residual strength analysis of IL-86 panel by the method developed the value $a \cdot \sigma_{B\ str}$ for the stringer of 7075-T6 alloy was taken equal to 500 MPa when calculating the fracture stresses of the stringer vs. crack length $2a$. This value was also obtained in TsAGI [5].

The residual strength analysis of the curved fuselage panels of DC-10 airplane gives the fracture stresses σ = 278 MPa as shown in Figure 6. Test data of the residual strength of DC-10 fuselage panel with the broken stringer and initial crack $2a$ =315 mm are σ=286 MPa [6]. The discrepancy between analytical and test data is 3%.

The difference in the fracture stresses of IL-86 and DC -10 is due to the fact that stringers in DC-10 fuselage panels were made of high-strength alloy 7075-T6, while stringers in the fuselage panels of IL-86 airplane were fabricated of less durable alloy D16AT.

The verification of the developed method was made also by comparing analytic and test values of the residual strength of the integrally stiffened fuselage panels of IL-86 and integrally stiffened wing panel of Antonov airplane.

In the residual strength analyses of the riveted and integrally stiffened wing panels, R-curves of 1163T alloy plates were taken from the publication [8]. The analyses of the riveted and integrally stiffened panels were carried out using the same procedure, described above.

The geometry of the curved integral fuselage panels of IL-86 was the same as for the riveted panels of this airplane. Two types of these panels were tested by the same program. The integral panel was fabricated from the plate of D16chT alloy by milling.

The residual strength analyses of the integral stiffened fuselage panel of IL-86 airplane with the cut stringer and the initial fatigue crack of 300 mm gives the fracture stresses σ =213 MPa. Results of the analysis are shown in Figure 7.

In accordance with the test data the failure of IL-86 integral stiffened fuselage panel with the same damages was under the gross stresses σ = 214 MPa . Thus the discrepancy between analytical and test data is less than 0.5%.

Riveted and integral stiffened wing panels of Antonov airplane were fabricated of 1163T alloy. The skin of the riveted panels was fabricated from the plates; integral panels were made by mean of extrusion. The panel width was 750 mm. Each panel was of 5 stringers. One stringer was cut in these panels, and the fatigue cracks were grown up in the skins.

The residual strength analyses of the riveted wing panel of Antonov airplane with the broken stringer and the initial fatigue crack of 270 mm gives the value of the fracture stresses σ =229 MPa. Results of the analysis are shown on the Figure 8. Test value of the residual strength of the panel with the broken stringer and initial fatigue crack σ =230 MPa [8]. The discrepancy between analytical and test data is about 0.5%.

The residual strength analyses of the integral wing panel of Antonov airplane with the broken stringer and the initial fatigue crack of 250 mm gives the fracture stresses σ=232 MPa. Results of the analysis are shown on the Figure 9. By test data the failure of the integral panel with the same damages happened at 242 MPa. The discrepancy is about 4%.

6 Conclusion

The developed method of the residual strength analysis of the stiffened structure does not require any complicated calculations.

This method allows to analyze the residual strength of a stiffened structure with two-bay crack in the skin under a broken stringer taking into account the initial size of the skin crack, further crack increase while static loading of the structure and the size of the plastic zone near the crack tip.

Validity and the accuracy of the developed method of the residual strength analysis were evaluated by comparing analytical values of the residual strength with experimental ones of riveted and integral stiffened fuselage panels of IL-86 airplane, riveted fuselage panels of DC-10 airplane, integral and riveted wing panels of Antonov airplane. The discrepancy between analytical and test data of the wing and fuselage residual strength is in the range of 1-5%.

The method developed of the residual strength analysis can be useful for the investigation of damage tolerance of new advanced and in-service airplane.

Using the same method, the analyses of the residual strength of integral and riveted panels with two-bay crack in the skin under the broken stringer indicate that in the structures fabricated of the similar plastic aluminum alloys, the side stringers with rivet holes in riveted panels have the same strength criterion $\alpha \cdot \sigma_B$ as the side stringers with small cracks in the integral panels.

7 Reference

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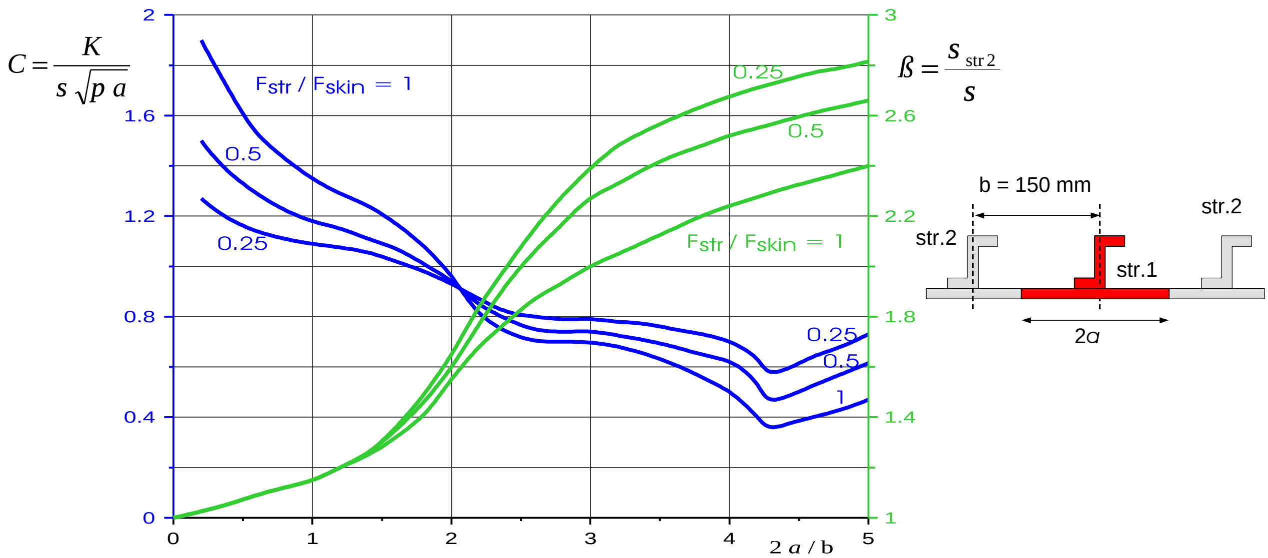

Figure 1 Correction factors ***C*** and *b* for different relations between stringer/skin cross section area

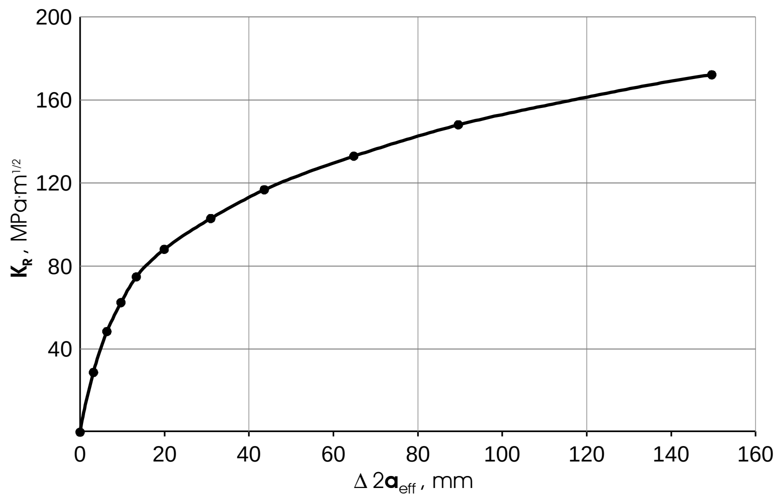

Figure 2 Test R-curve of the 2024-T3 alloy fuselage skin sheet of 1.8 mm

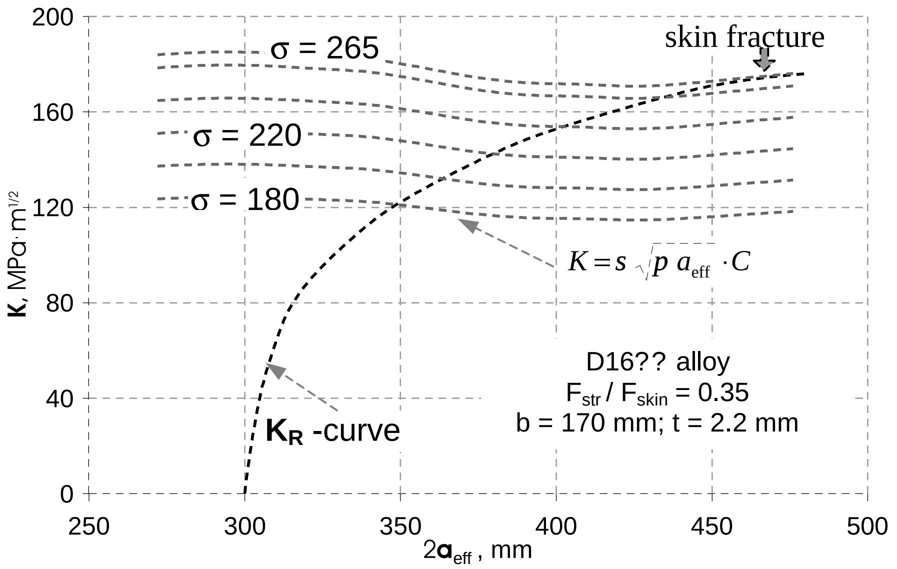

Figure 3 Principle of K_R-curve application for the residual strength analysis of the-stiffened skin with the crack. Riveted fuselage panel of Il-86 airplane

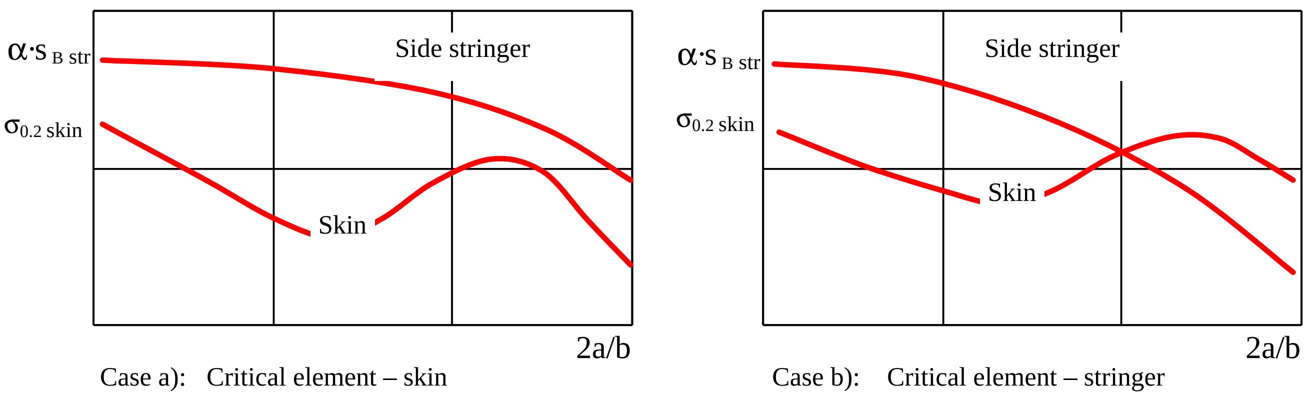

Figure 4 Different design cases of the stiffened structure

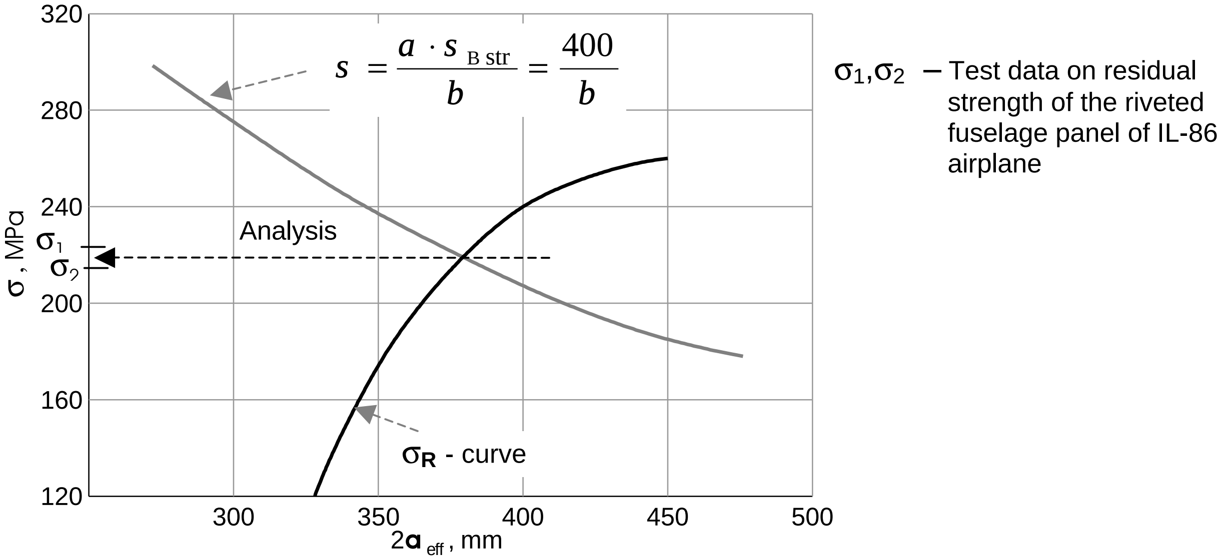

Figure 5 Residual strength analysis of the curved riveted fuselage panel of Il-86 airplane

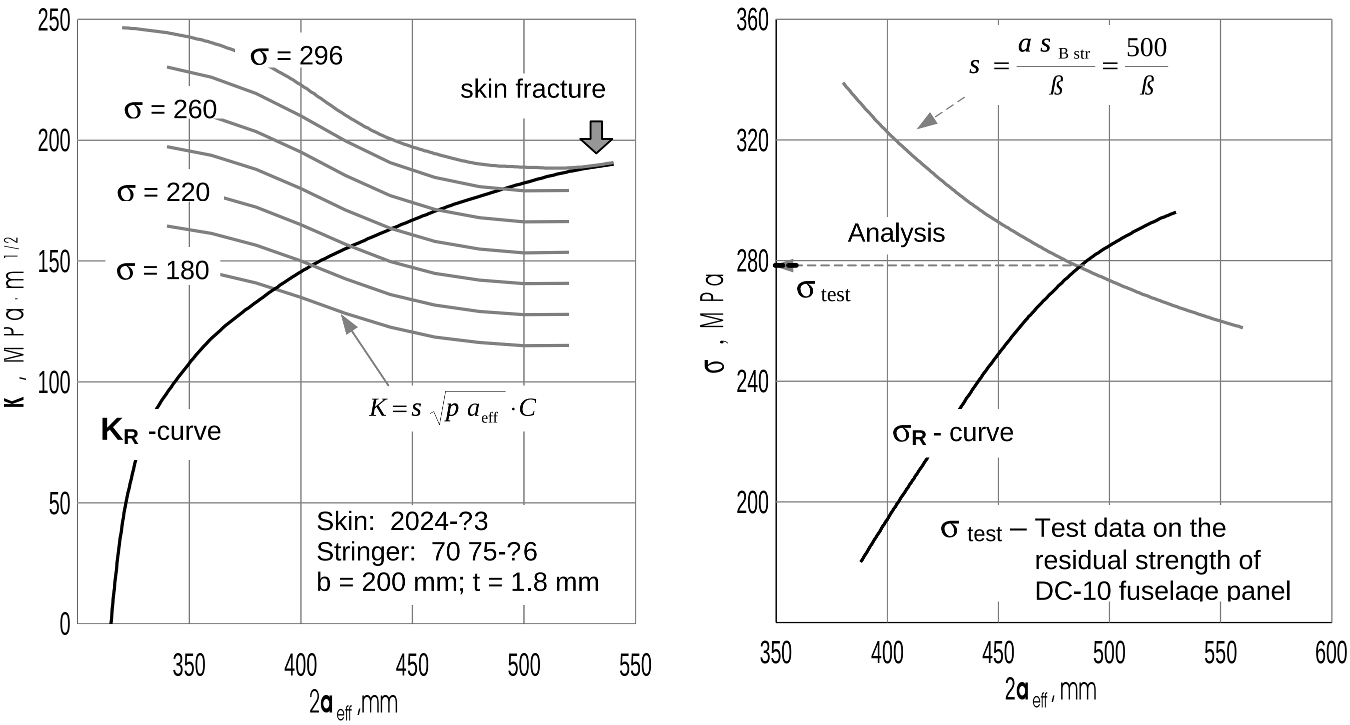

Figure 6 Residual strength analysis of the flat riveted fuselage panel of DC-10 airplane.

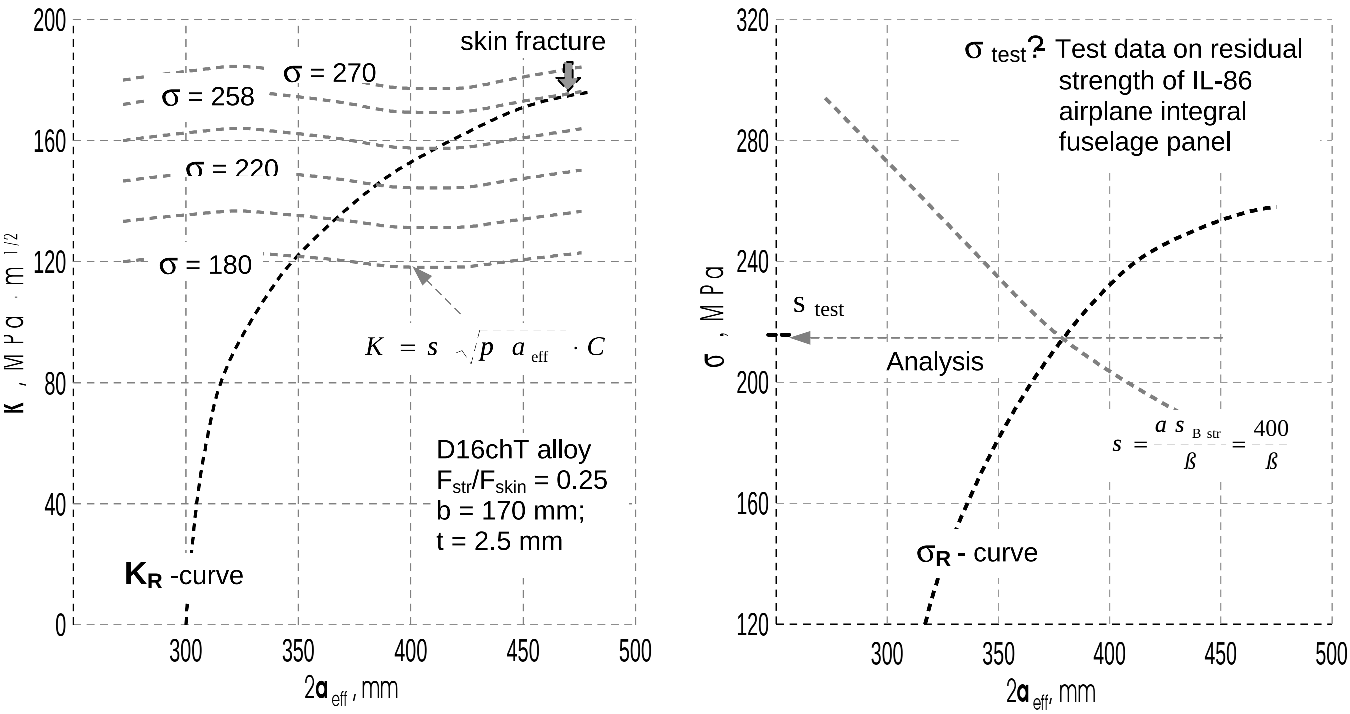

Figure 7 Residual strength analysis of the curved integral fuselage panel of IL-86 airplane

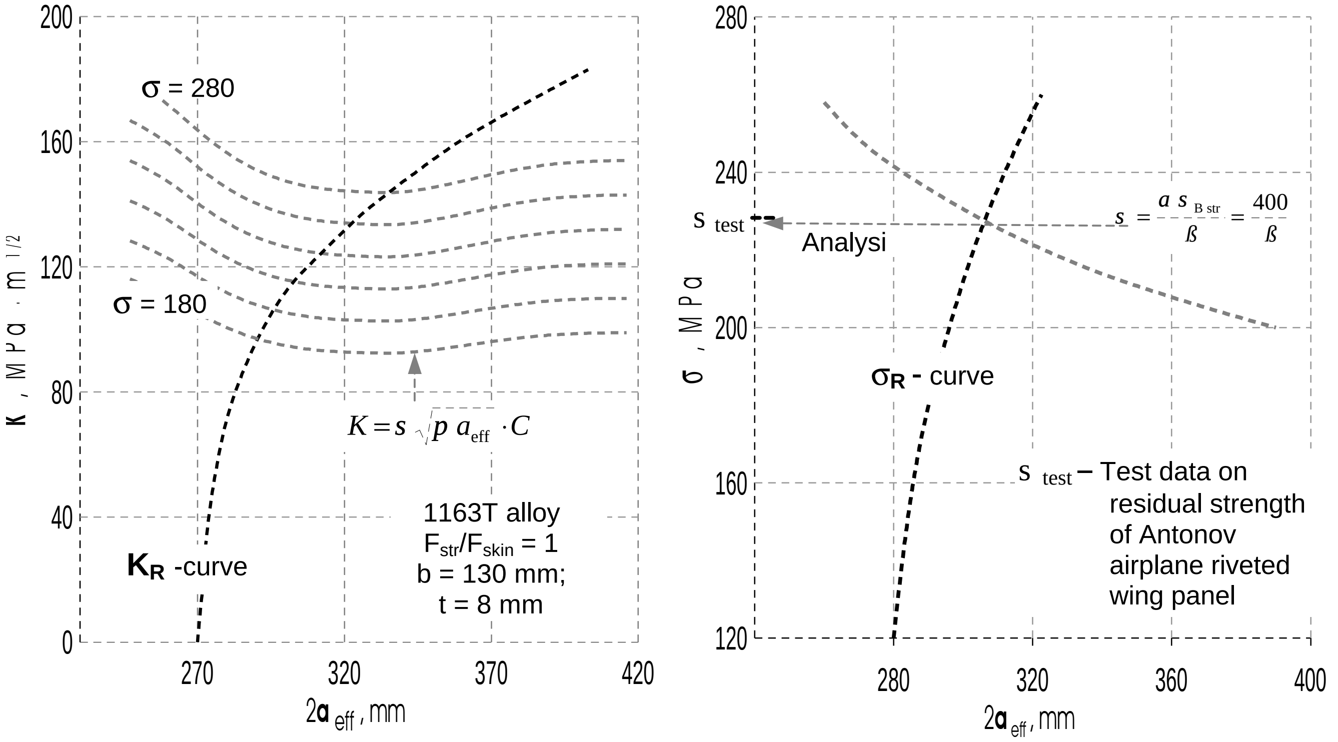

Figure 8 Residual strength analysis of the flat riveted wing panel of Antonov airplane

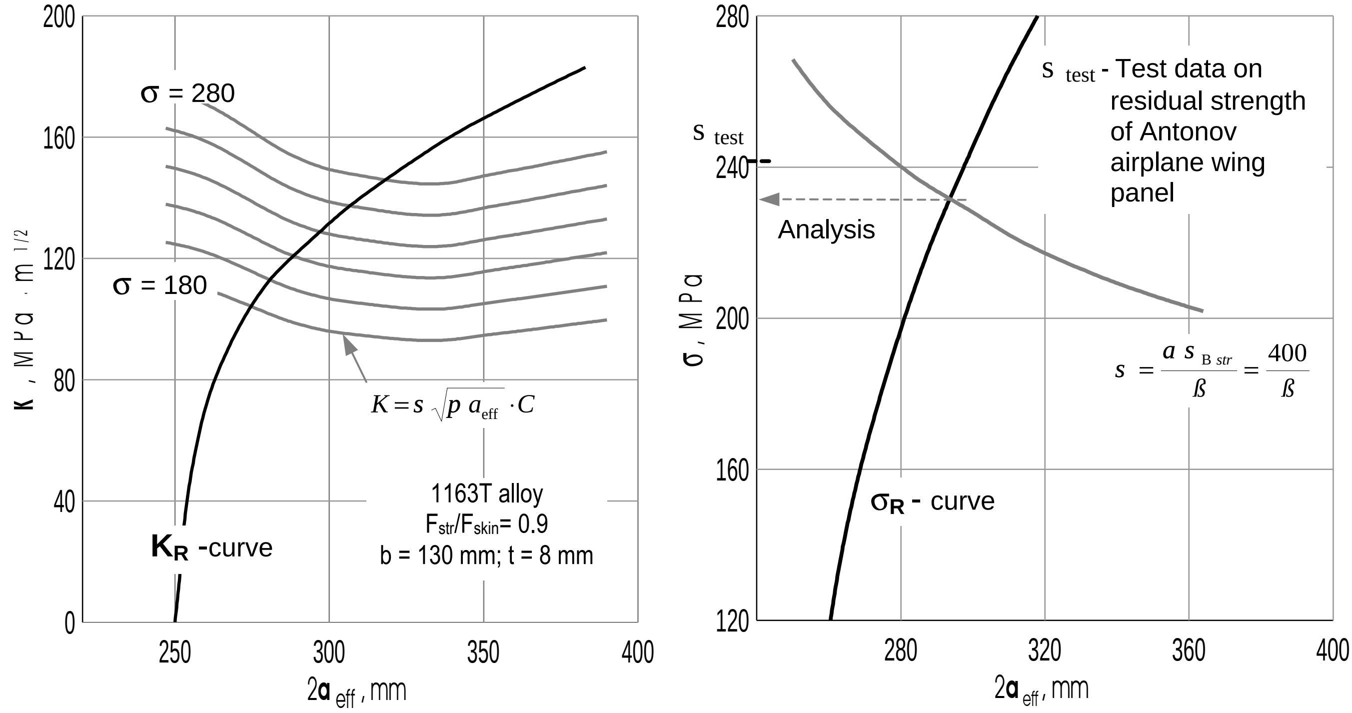

Figure 9 Residual strength analysis of the flat integral wing panel of Antonov airplane