SERVICE LIFE OF AIRPLANE STRUCTURES

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Abstract

Test-analytical results of investigating into the fatigue, fail-safety and damage tolerance of Russian aircraft are presented. Stresses in wide-body aircraft structures are given. Fatigue life curves for wing and fuselage structures are generated. Residual strength data are presented for these structures having a skin crack under the broken stiffener. Generalized curves for skin crack duration under the broken stiffener are presented.

1 Introduction

The design and service experience of the transport aircraft for the recent 50 years has shown that to ensure the aircraft reliability and efficiency it is required to provide in their structures the solution of three concepts simultaneously, i.e. safe life, fail-safe and damage tolerance. The comprehensive data about these structural features have been collected by now in the course of testing coupons, panels and full-scale structures. This presentation generalizes the information in order to estimate tolerable stresses, fatigue performances, fail safety, damage tolerance of wide-body aircraft fabricated of modern materials. The generalization covered the results of the joint TsAGI activities with Antonov, Ilyushin, Tupolev, Yakovlev companies.

2 Stress

The stresses have been estimated in the wing and fuselage structures of contemporary wide-body aircraft (Table 1). The tolerable maximum stresses have been found in the wide body aircraft structures as they have the highest stresses. The materials having improved strength, fatigue and crack resistance properties were used in them. The stresses σ^{ult} under the ultimate loads represent the maximum tolerable stresses in static strength criteria. Equivalent skin stresses σ_{equiv} represent the maximum cycle stresses having the stress ratio R=0. The structural damage per cycle, σ_{equiv} , is equal to the structural damage for all the stress cycles during one typical flight. The stresses σ^*_{equiv} are equal to the sum of stresses σ and additional stresses $\Delta\sigma$ under the loading of stiffeners (rivets, bolts). The values of σ_{equiv} and σ^*_{equiv} in fatigue have been calculated by conservative methods of Refs. [1, 2] using the linear damage accumulation hypothesis. The values of σ_{equiv} in crack growth with regard for their retardation have been outlined by the author, based on special test results. The stresses σ_{equiv} have been found for the wide-body aircraft having flight duration of 7-8 hours. The crack growth has been inspected mainly by visual aids at annual intervals (600-700 flights). It is supposed that such inspections detect reliably the skin cracks 50 mm long under the broken stiffener. While evaluating the stresses given in Table 1 the analysis covered the stresses in the aircraft structures both for Russian aircraft and those data published in Refs. [7-12].

3 Design goal

The aircraft structural lifetime is limited by fatigue of longitudinal joints in the wing lower surface panels and those in the fuselage skin longitudinal joints. These joints are affected by multiple site crack initiation at the end of the fatigue life. Hence the lifetime for the longitudinal joints is estimated by safe-life concept. The fatigue life of the longitudinal joints depends on material properties and assembly technology. Figs.1 and 2 present minimum experimental fatigue lives of longitudinal joints in the wing panels and in the fuselage skin laps. These data are taken from lifetime tests of full-scale structures. Different aircraft models have different symbols for the experimental points. The cracks 2-15 mm long have been detected in the longitudinal joints at the mentioned lives. The relations of minimum lifetime values versus equivalent stresses σ^*_{equiv} have been determined from the experimental data (Figs.1 and 2). While estimating the minimum lifetime values the fact was taken into account that each experimental point in Figs.1 and 2 is the result of testing full-scale structures having thousands of similar stress concentrators in the longitudinal joints.

It follows from Figs.1 and 2 that at the equivalent stresses σ^*_{equiv} given in Table 1 the lifetime of 20000 flights is ensured in the structures of contemporary wide-body airplanes.

4 Fail safe

To ensure fail-safe concept it is required that the structure has standardized damages in the form of two-bay skin crack under the broken stiffener to maintain the strength under the limit load of $P^{lim} = 0.67P^{ult}$. To evaluate this criterion, test-analytical results have been generalized in terms of residual strength of the wing and fuselage structures (Figs. 3 to 5). Here the experimental data were used about the residual strength of panels and full-scale structures published in Refs. [3-6]. The residual strength for the structure having such damages has been analyzed using the methods presented in Ref. [4]. The calculations utilized linear fracture mechanics.

The residual strength of the lower wing surface structure having two-bay skin crack under the broken stringer is ensured at the limit stresses $\sigma^{lim} = 0.67\ \sigma^{ult}$ (Fig. 3, Table 1). It should be noted that when there are skin cracks under the broken caps of front or rear spar the residual strength of the lower wing surface [5] is not ensured at the stresses taken from Table 1

The residual strength in the structures of the upper wing surface fabricated of high-strength alloys having the standardized damaged [13] and stress value $\sigma^{lim} = 0.67\ \sigma^{ult}_{tens}$, defined from Table 1. It should be noted that in such structures a drastic crack length increase at the stresses of 160-170 MPa is observed.

In the structures of pressurized fuselages having longitudinal two-bay skin cracks under the broken frame the residual strength is ensured at the stresses $\sigma = 1.15\ \dfrac{Pr}{t}$ (Table 1) in the following cases (Fig. 4):

- critical stress intensity factor in skin material is $K_{app} = 135\ \text{MPa}\sqrt{\text{m}}$ and stoppers applied;
- there are no stoppers, but $K_{app} \approx 170\ \text{MPa}\sqrt{\text{m}}$.

n the structures of pressurized fuselages having lateral two-bay skin cracks under the broken stringer the residual strength is ensured at the stresses $\sigma^{lim} = 0.67\ \sigma^{ult}$ (Table 1) in those cases when the stringers are fabricated of high-strength materials of 7000 series [13]. The residual strength of the structure having D16 stringers is approximately 220 MPa (Fig. 5).

5 Damage tolerance

To ensure damage tolerance concept it is required that the crack growth duration from the reliably detectable to the tolerable length is no less than the aircraft structure inspection interval. The safety factor for crack growth duration is assumed equal to 2. Visual inspections are assumed principal. Inspection interval is 700 flights (once a year). The reliably detectable damage is assumed to be the skin crack having the length $2a$ = 50 mm under the broken stiffener (stringer, frame). The generalized average curves of crack growth in the wing and fuselage structures are presented in Figs. 6 to 8. These figures also demonstrate the material properties of the structures under testing.

It follows from these curves that the damage tolerance concept is ensured in the wing and fuselage structures at the specified stresses and visual inspection intervals.

6 Conclusion

The values of tolerable stresses in the wing and fuselage structures of contemporary wide-body aircraft have been determined.

The relations of the wing and fuselage design goals for modern transports versus cyclic stresses are defined.

The residual strengths for the wing and fuselage structures with two-bay skin crack under the broken stiffener have been outlined to estimate fail-safe parameters of these airplane structures.

The generalized average curves of skin crack growth duration from visually detectable till tolerable lengths under broken stiffener are plotted for damage tolerance estimation.

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WIDE BODY AIRCRAFT STRUCTURAL STRESSES

Table 1

Lower wing surface			Upper wing surface				Fuselage		
Ultimate tensile stresses σ^{ult}, MPa	Equivalent pulsating (R=0) stresses		Ultimate compressive stresses, σ^{ult}_{comp} MPa	Ultimate tensile stresses, σ^{ult}_{tens} MPa	Equivalent pulsating (R=0) stresses		Ultimate longitudinal tensile stresses, σ^{ult}, MPa	Equivalent pulsating (R=0) longitudinal tensile stresses, σ_{equiv} MPa	Hoop stresses, **Pr/t**, MPa
	σ_{equiv} MPa	σ^*_{equiv} MPa			σ_{equiv} MPa	σ^*_{equiv} MPa			
380	$0{,}45\sigma^{ult}$ [1)]	$0{,}5\sigma^{ult}$ [1)]	490	245	$0{,}55\sigma^{ult}_{tens}$ [1)]	$0{,}6\sigma^{ult}_{tens}$ [1)]	360	$0{,}35\sigma^{ult}$ [1)]	105
	$0{,}35\sigma^{ult}$ [2)]				$0{,}25\sigma^{ult}_{tens}$ [2)]			$0{,}3\sigma^{ult}$ [2)]	

[1)] - Fatigue, linear hypothesis, conservative analytical method
[2)] - Crack growth with regard for retardation

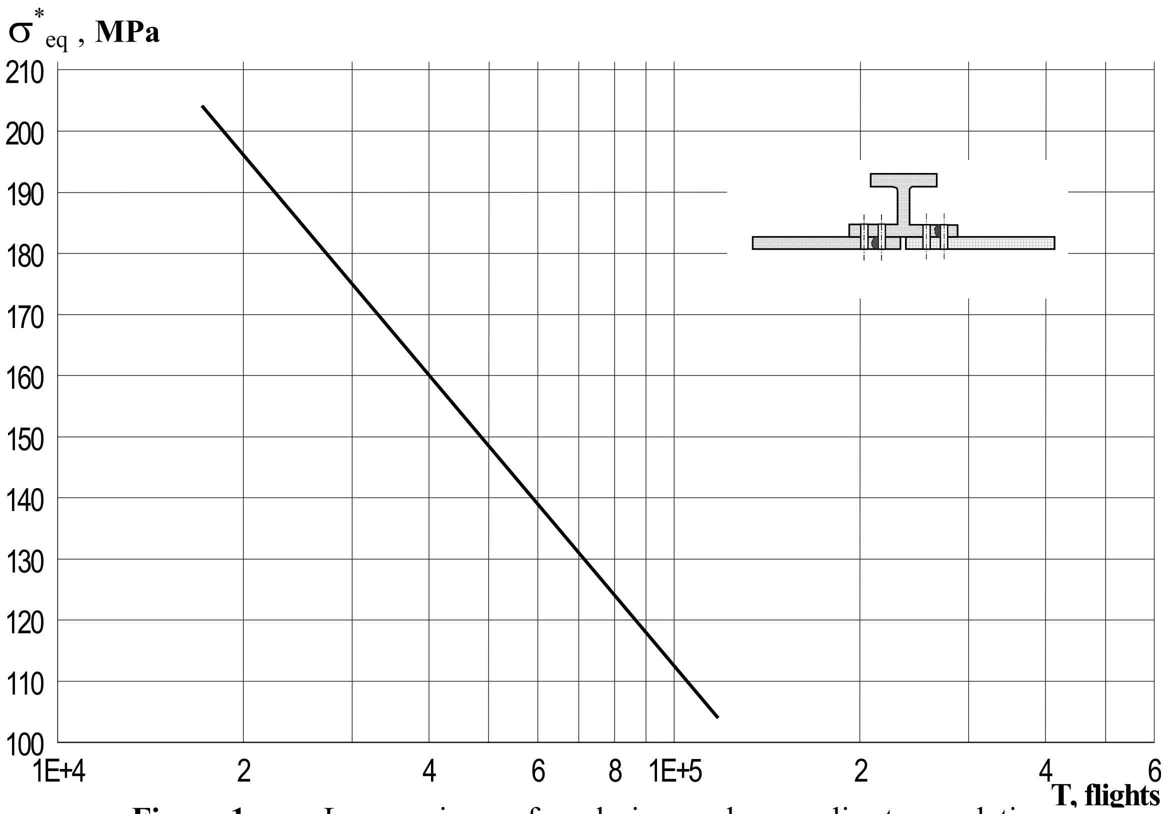

Figure 1. Lower wing surface design goal vs. cyclic stress relation.

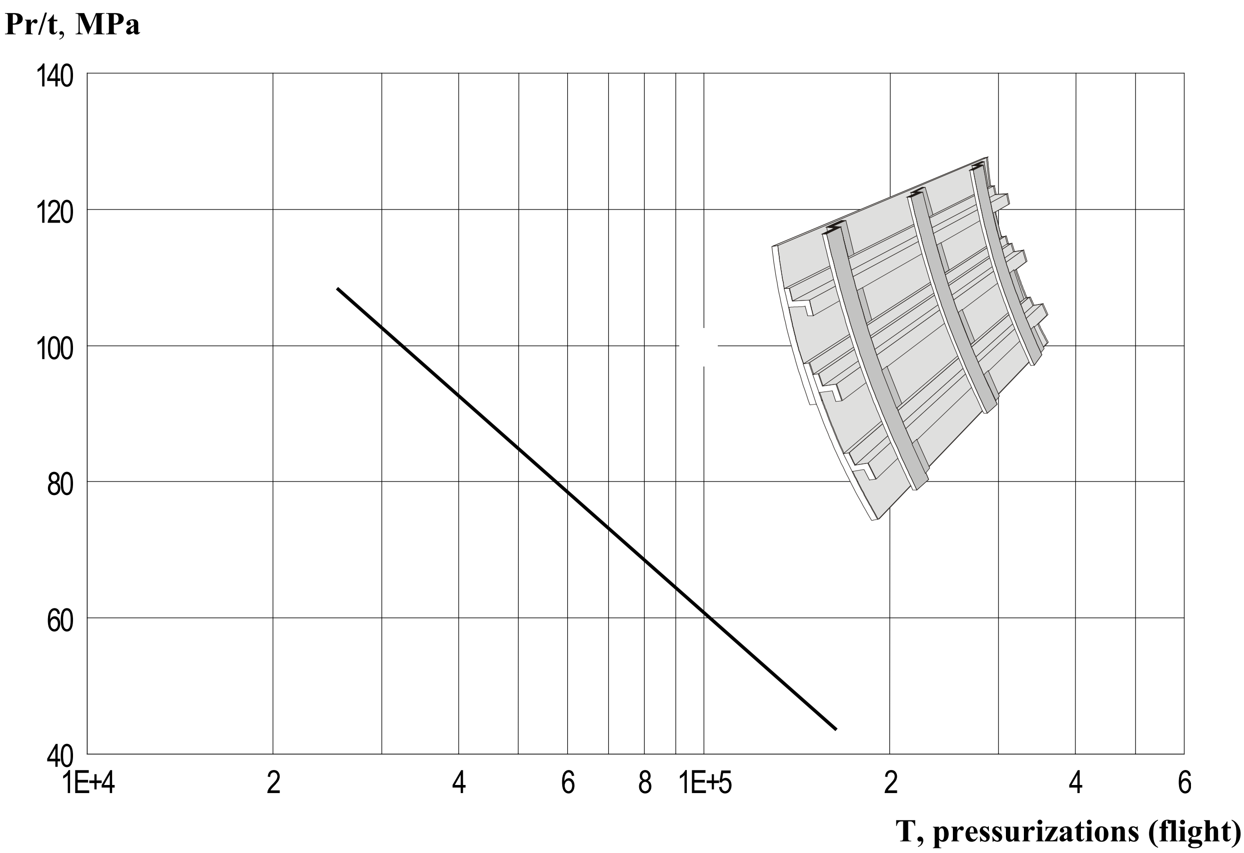

Figure 2. Pressurized fuselage design goal vs hoop stress relation.

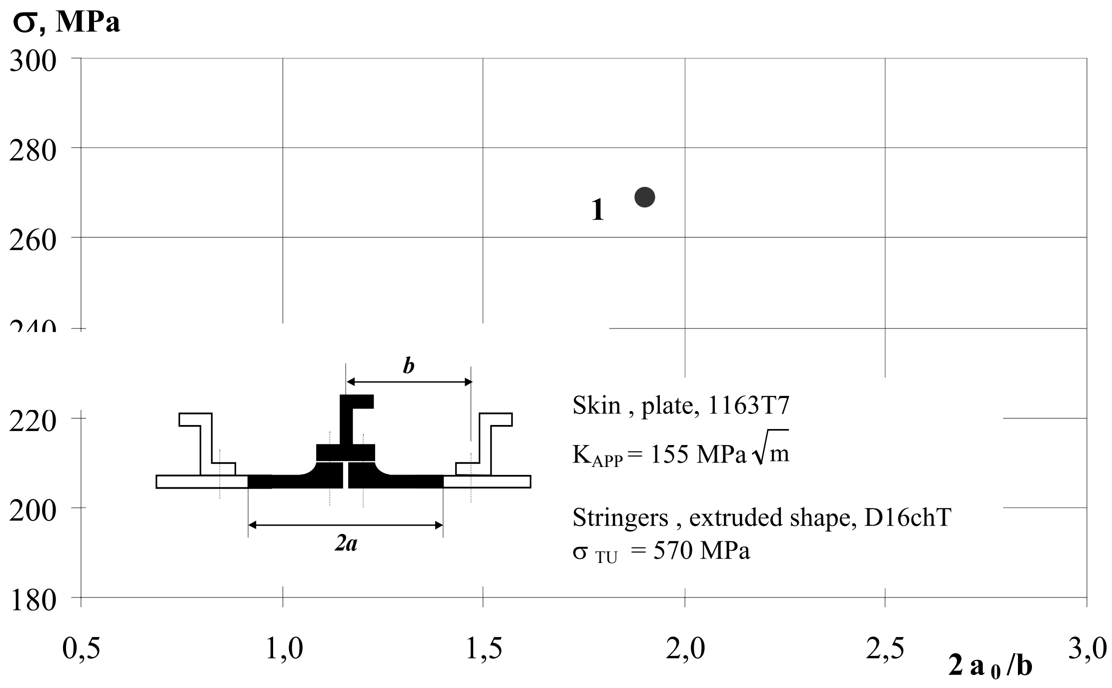

Figure 3. Residual strength of the lower wing surface having two-bay skin crack under the broken stringer

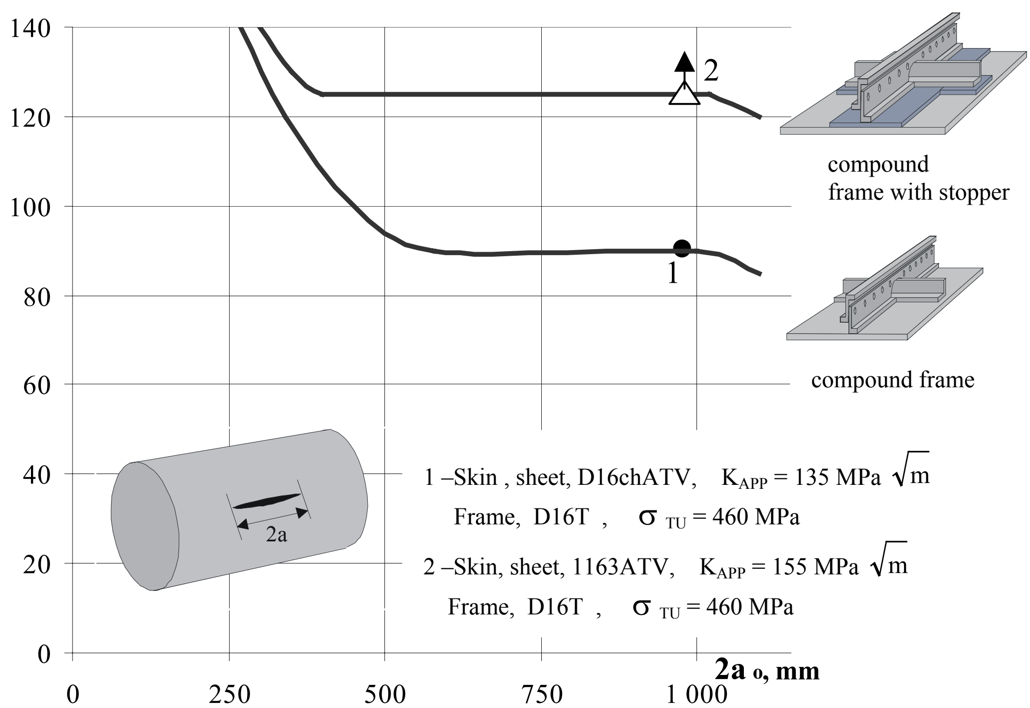

Figure 4 Residual strength of the pressurized fuselage having longitudinal two-bay crack in skin under the broken frame

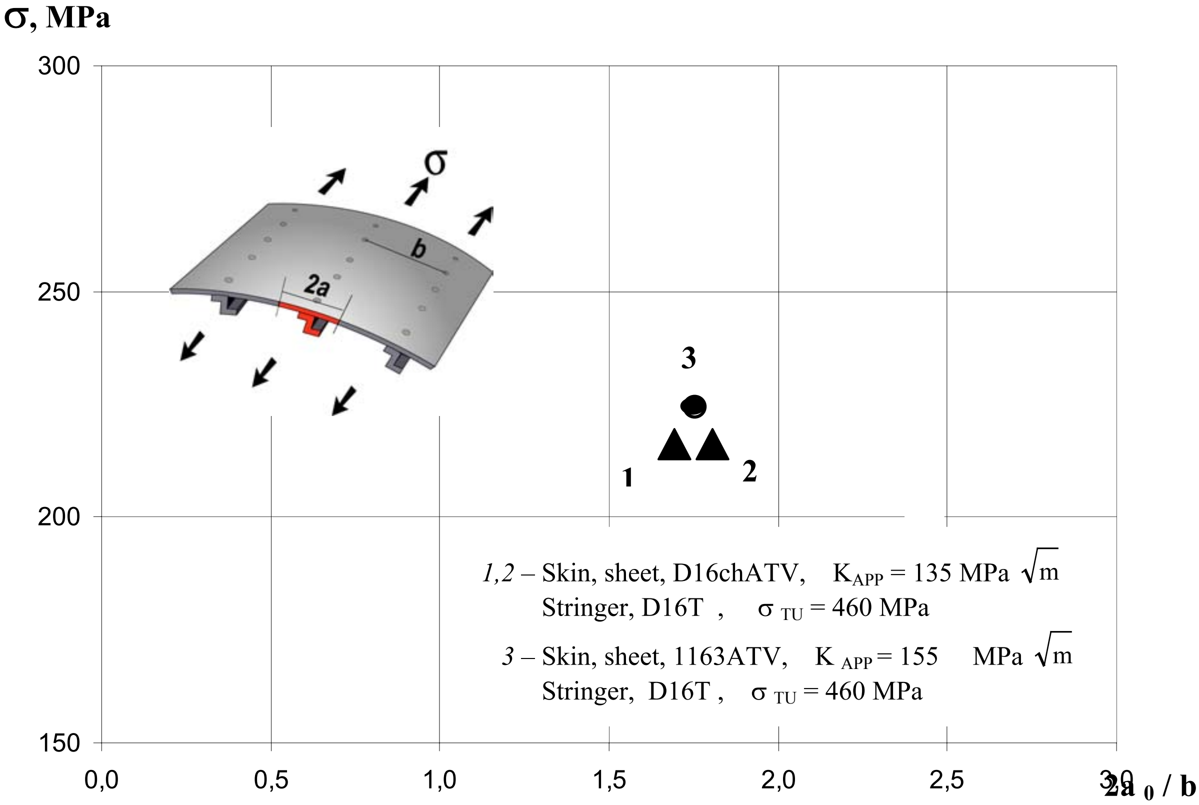

Figure 5 Residual strength of the fuselage having transversal two-bay skin crack under the broken stringer

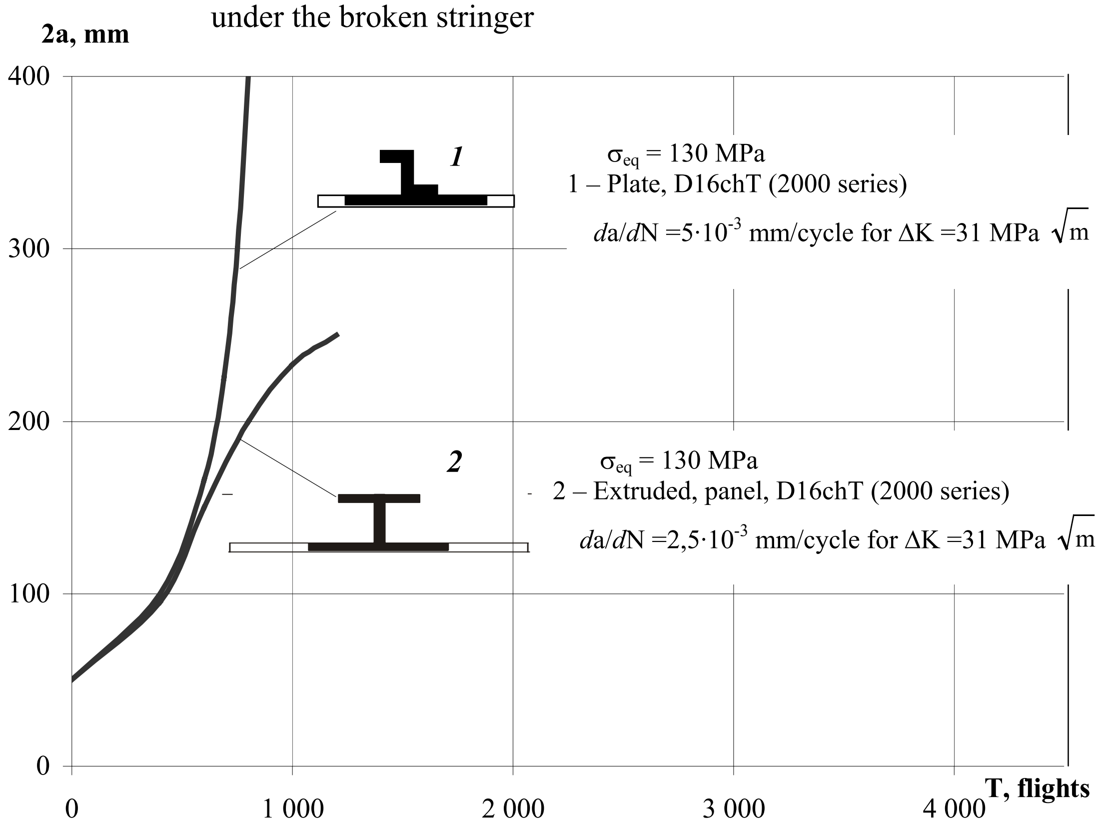

Figure 6 Crack growth duration in the skin under the broken stringer in the lower wing surface.

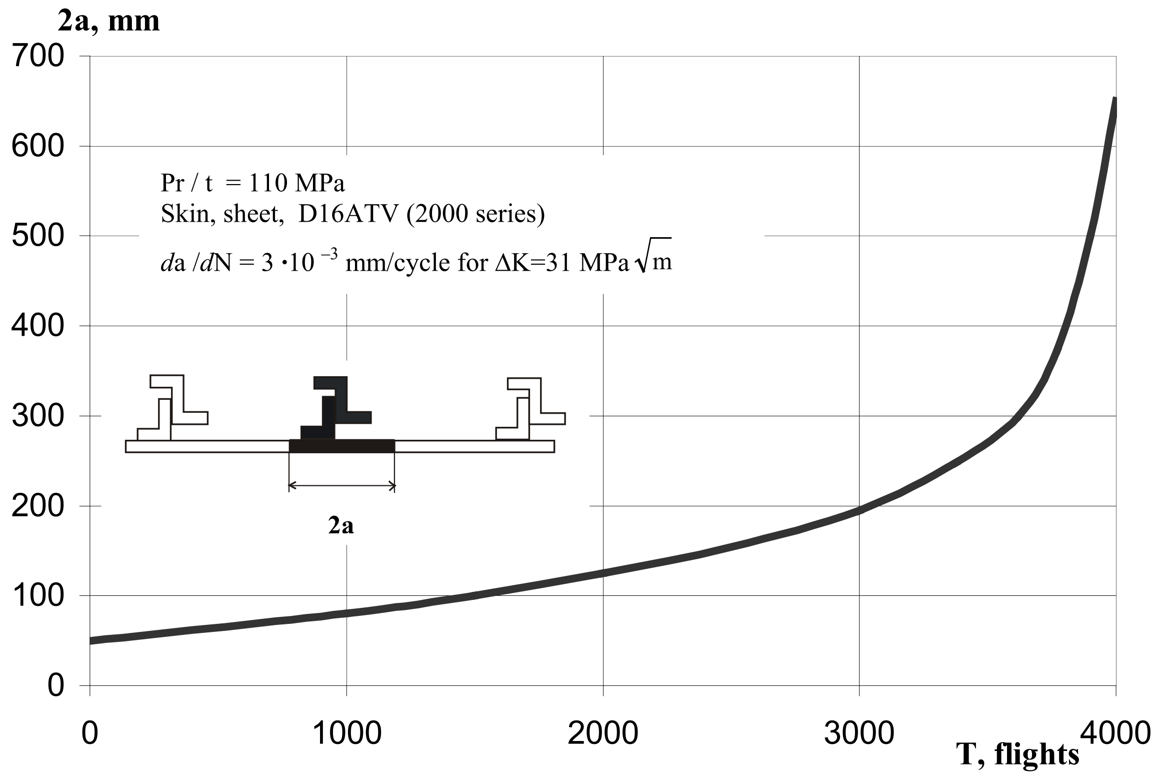

Figure 7 Longitudinal crack growth duration in the fuselage skin

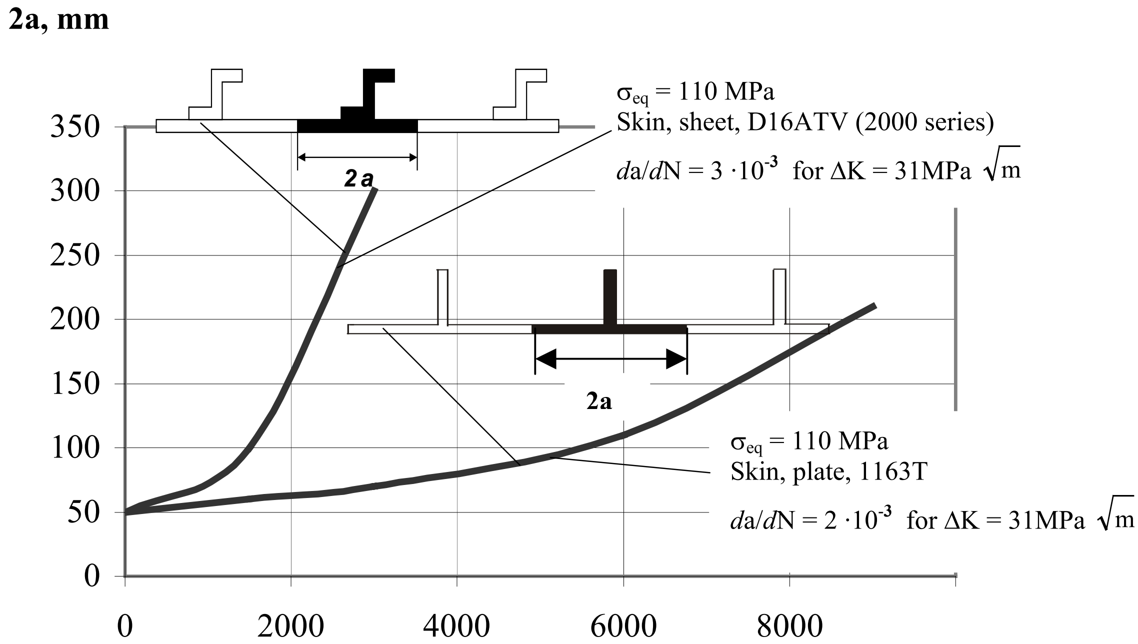

Figure 8 Transversal crack growth duration in the skin under the broken stringer in the fuselage