A98-31648

ICAS-98-5,7,4

ENSURING DAMAGE TOLERANCE OF ADVANCED AND AGING AIRCRAFT

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Abstract

Standardizing approaches to establishing service life for Russian transport aircraft structures are outlined. Damage tolerance criteria ensuring advanced aircraft damage tolerance are presented. The basis is given for specifying fracture toughness and crack growth rate of structural materials. Results of investigating residual strength criteria for full-scale structures having widespread fatigue damages are listed. Fatigue strength and crack resistance of new and aging aircraft structures are compared. Principles to determine corrosion damage growth duration for aircraft structures are outlined.

Introduction

The main principle to ensure safe operation of modern passenger aircraft structures is the damage tolerance concept. To maintain damage tolerance of advanced aircraft new damage tolerance criteria are developed, requirements to material crack resistance are formulated (so as to meet damage tolerance criteria), and new structural engineer's approaches for airframe damage tolerance are studied.

One of the most important problems in aviation now is to ensure safe operation of high-time (aging) aircraft. To solve it damage tolerance of structures having widespread fatigue damage (WFD) is studied as well as degradation of structure crack resistance characteristics during long-term operation, and safe operation of structures with corrosion is ensured. Russian experience in dealing with the above-mentioned problems is outlined below (see also [1, 2, 3]).

Aircraft Airworthiness Standards

In the 1950-70s the USSR aircraft airworthiness standards stated the only concept to ensure safety during long-term operation – it was the safe life concept, see[4]. In 1976, equal rights have been given to the principle of "operational survivability" along with the safe life concept. In the USSR and Russian practice the "operational survivability" concept includes both damage tolerance and fail safe concepts. In 1994, Aviation Regulations were introduced for transports (AP 25.571), where the "operational survivability" concept (which is hereinafter called "damage tolerance" for simplicity) is adopted as the main one.

Airworthiness standards for the USSR civil aircraft paid much attention to the results of laboratory test of full scale structures for fatigue strength and damage tolerance. There are no Russian aircraft types whose structures were not subjected to fatigue strength tests at a high safety factor related to the design service life. Several copies of each full-scale aircraft structure type were tested, including those flown for some time.

Damage Tolerance Criteria

The structure tolerant to a damage should satisfy certain requirements. The main criteria of damage tolerance include standardized damages (Figs. 1 and 2), standardized residual strength, and specified fatigue crack growth duration (Fig. 3).

The standardized damages comprise almost all damages ever detected in aircraft structures during fatigue tests and operation. These damages were the result of fatigue, corrosion and accidents. Sizes of standardized damages are defined by generalizing the data about various aircraft structure types. During this analysis the damages were classified by proceeding from structural designs of load-bearing elements. Statistical data were gathered for each damage class including cracks detected during full scale tests and operation; thereafter, integral occurrence frequences of crack sizes were outlined. While establishing sizes of standardized damages, researchers also took into account that stiffening elements (such as stringers and frames) and joints may terminate fracture. The standardized damages should be easily detectable – mainly during external visual inspection of the airframe. A wing structure with standardized damages should sustain no less than 67 percent of the ultimate load P^{ult}. A pressurized fuselage structure with standardized damages should sustain overpressure of no less than 110 percent of operational overpressure ΔP^{op} in the fuselage.

To decrease the probability of failure in a structure with standardized damages that has adequate residual strength some requirements to fatigue crack growth duration have been formulated (Fig. 3). These requirements are based on the analysis of aircraft maintenance intervals. Satisfaction of these requirements provides cost-effective structural inspections during periodical aircraft maintenance.

21st ICAS Congress
13-18 September 1998
Melbourne, Australia

Requirements to Materials

To obey damage tolerance criteria (Figs. 1, 2 and 3) in advanced airplanes the required crack resistance of their structural materials has been established; see [1, 2]. These requirements are the result of "experiment + numerical analysis" activities. Cracked material fracture was investigated using specimens and standard methods. Residual strength and fatigue crack growth rate were calculated by using linear fracture mechanics relations. The effects of element plasticity, rivet/bolt compliance, local bending stresses, thin skin bulging (in pressurized fuselages near cracks) on fracture characteristics were taken into account by using appropriate dimensionless correcting factors. These factors were evaluated by generalizing the test data and introduced into equations to calculate stress intensity factors. Test data on residual strength were generalized upon testing about 200 large-size panels and about 100 full-scale aircraft structures. To obtain fatigue crack growth rates, some 150 crack growth duration curves were generated for large-size stiffened panels, and some 200 fatigue crack growth duration curves, for full-scale aircraft structures.

The required values of skin material fracture toughness were based on residual strength, firstly, of the wing with a two-bay skin crack and a broken center stringer (Fig. 1) and, secondly, of a fuselage with a two-bay skin crack and a broken frame in the crack center (Fig. 2). The following tensile stresses were assumed for the residual strength analysis with such standardized damages: 260 MPa in the lower wing skin and 170 MPa in the upper wing skin for limit loads $P^{lim} = 0.67\ P^{ult}$; 120 MPa in the fuselage skin in the circumferential direction under overpressure of 1.1 P^{op}. These stresses are based on data about stresses in skins of modern wide-body airplanes. Principal results of residual strength study for stiffened structures are presented in Figs. 4, 5, and 6. The required values of apparent material fracture toughness K_{app} were determined for 1200 mm wide plates tested without eliminating bulging near the crack (i.e., without guides). The values of K_{app} for thick plates from wing skin material with and without bulging are almost identical. The values of K_{app} for thin sheets (1.8 mm thick) from fuselage skin material tested with bulging eliminated exceed by approximately 30% the K_{app} values obtained in tests with free bulging. It should be born in mind that under static loading the stiffened structure having an initial crack length $2a_o$ equal to two distances between stiffeners, $2b$, fails in many cases at the relative final crack length $(2a_o + \Delta 2a)/b \approx 2.5$, where $\Delta 2a$ is a crack increment. The residual strength of the structure with such a crack depends on stringer strength (Fig. 5). The results of the above study (see also [1, 2]) have shown that the required fracture toughness values K_{app} are 155 MPa$\sqrt{m}$ for lower wing skin; 77 MPa$\sqrt{m}$ for upper wing skin; 140 MPa$\sqrt{m}$ in fuselage sheets with free bulging, and 182 MPa$\sqrt{m}$ with bulging eliminated. These fracture toughness values are attained in the materials used in the newest aircraft structures, refer to [5, 6].

The required crack growth rates for skin materials were determined by the detailed visual airframe inspection (Fig. 3) enabling reliable detection of 25-mm long cracks in the wing skin and 75-mm long cracks in the fuselage skin. Inspection interval is assumed to be 2000 flights. The reliability factor for crack growth scatter is assumed to equal 2. The analysis determined such crack growth rates that give average duration of crack growth from a reliably detectable size to the standardized crack size equal to 4000 flights. The crack growth rate was calculated using equivalent stresses σ_{eqv} for the cycle stress ratio R = 0.05. The "damage per one cycle with σ_{eqv}" is equal to structure damage due to all cycles during one typical flight. On the basis of stress spectrum analysis for state-of-the-art wide body aircraft structures the following σ_{eqv} values are assumed: 175 MPa in lower wing skin; 140 MPa in upper wing skin; and 110 MPa in fuselage skin (circumferential stresses). The account in the calculation was taken of crack growth arrest due to interaction of loads with different amplitudes as well as of crack growth acceleration due to external corrosive environment effect. The results of crack growth calculation are presented in Fig. 7 [1, 2]. The required crack growth rate d2a/dN at stress intensity factor range ΔK = 31 MPa$\sqrt{m}$ and the stress ratio R=0.05 are 0.0015 mm/cycle for lower wing skin; 0.0036 mm/cycle for upper wing skin; and 0.0028 mm/cycle for fuselage skin. Crack growth rates d2a/dN for widely used Al/Cu-alloys at ΔK = 31 MPa$\sqrt{m}$ are approximately equal to 0.004 mm/cycle. Adding zirconium to these alloys decreases the crack growth rate by a factor of 1.5 – 2 (see [5, 7]). Al/Zr-alloys are utilized in advanced aircraft [5].

Residual Strength of Structures with WFD

Widespread fatigue damage (WFD) initiates in primary structural sections including a lot of elements having almost identical values of fatigue life. Table 1 gives residual strength data for full-scale structures with WFD.

The apparent fracture stresses in the net section, $\sigma^{app}_{fr.net}$, were defined with due regard to reduction of the section area in primary elements due to the presence of holes and initial cracks. The structural cross section length where the net area and σ^{app} were calculated was stated as follows. The initial length was equal to the length of the zone with WFD. In the calculation this initial length was "extended to the left" by adding two lengths of the leftmost crack, and "to the right", by adding two lengths of the rightmost crack. While calculating critical fracture stresses $\sigma^{c}_{fr.net}$, additional attention was paid to section weakening due to crack length increase during stable crack growth stage (Fig. 8). The values of these stresses

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were compared to the yield strength $\sigma_{0.2}$ values. Stress intensity factors K_{fract} at which the structure fails were calculated according to usual (approved) methods. These factors were compared to fracture toughness for plane strain, K_{1c}, or to the apparent fracture toughness for plane stress state, K_{app}. The apparent fracture toughness K_{app} was determined by using sheets with free bulging near the crack.

Structural residual strength criteria are those fracture parameters whose relative values are 1.0. It follows from Table 1 that for structures made of brittle materials (like V95T1 and V93T1) residual strength criteria are linear fracture mechanics criteria $K_{fract}=K_{app}$ or $K_{fract}=K_{1c}$. For many structures made of plastic materials (like D16T) having WFD with interacting cracks residual strength criterion is the stress $\sigma^{c}_{fr.net}$ equal to the yield strength $\sigma_{0.2}$ calculated with correction for stable crack growth. Some structures failed at $\sigma^{c}_{fr.net} < \sigma_{0.2}$; this can be explained by the effect of local bending stresses difficult to calculate. During residual strength test of two pressurized fuselages with skin notches simulating the through multiple-site cracks no notch growth was observed. It should be noted that during residual strength test of one pressurized fuselage with multiple-site circumferential skin cracks in the upper part a substantial stable crack growth was detected. Test results for this fuselage are not included into Table 1, as the structure did not completely fail. There are also such multiple-site fatigue crack patterns (in structures made of the plastic alloy D16T) for which the residual strength criterion may be formulated as $K_{fract}= K_{app}$.

On Crack and Fatigue Resistance Degradation

In the early 1980s special tests were conducted on wide specimens cut out of aircraft wing and fuselage skins operated for 25 years (from the late 1950s) and on newly fabricated materials of the same types manufactured in the early 1980s. Tested were clad sheets of naturally aged copper-based alloys (D16ATV and D16ATNV) as well as clad sheets of artificially aged zinc-based alloy (V95AT1V); extruded panels of naturally aged copper-based alloy (D16T) and extruded panels of artificially aged zinc-based alloy (V95T1). The specimens were tested with free bulging near cracks. The ultimate strength σ_b, yield strength $\sigma_{0.2}$, and elongation δ were determined using other specimens. The crack growth rate was found by testing the same wide specimens that were thereafter used for determining residual strength. Fatigue crack growth rate (d2a/dN) tests were conducted for stress increment $\Delta\sigma$ =130 MPa, cycle stress ratio R=0.05, and loading frequency f=0.2Hz. Table 2 presents experimental study results on apparent fracture toughness K_{app}, while Table 3 shows the fatigue crack growth rate d2a/dN for the stress intensity factor range ΔK = 31–62 MPa$\sqrt{m}$. The specimen numbers in Tables 2 and 3 are unified. The numbers with index "*o*" are for specimens of old aircraft skins, while the numbers with index "*n*" are for specimens of newly fabricated materials. Some general tendencies are observed with the materials of all aircraft types studied:

– residual strength of materials in airplanes operated for a long time (from the late 1950s) is lower than that of new materials of the same type made in the early 1980s;

– ultimate and yield strength of materials in high-time aircraft is higher than the characteristics of new materials of the same type;

– fatigue crack growth rate for the range ΔK = 31–62 MPa$\sqrt{m}$ in the materials of high-time aircraft is higher than that of new materials of the same type.

In 1998 an additional experiment was carried out to compare fatigue crack growth rate of narrow specimens (160-mm wide) made of D16ATV and D16ATNV. The narrow specimens were cut out of the high-time aircraft wing skin tested earlier. Test methods and results for wide specimens are described above. In addition, narrow specimens of clad sheet from the new material D16AT manufactured in 1993 were tested. Narrow specimens were tested at the same stresses, cycle stress ratio and loading frequency as wide specimens. Results of this experiment may be seen in Fig. 9; these converge with results of testing wide specimens (Table 3). It should be noted that in the experiments described it was difficult to separate the effects of the sheet manufacture process and the service duration on crack resistance. Alloys 1163T and V95pchT2 having less silicon and iron (and with improved process) are utilized instead of alloys D16T and V95T1 (respectively) in structures of Russian advanced aircraft.

Full-scale structures with WFD were investigated in terms of fatigue crack growth during fatigue tests and in operation (Figs. 10 and 11). Relative service times $\overline{T}$ are the ratio of the current service time to the service time at which the structure with WFD has been destroyed. Operational data are presented by the points showing crack size and lifetime of those aircraft where WFD was detected. On the basis of these data and by means of statistical analysis [8] some crack growth curves were generated corresponding to probability p=0.5; 0.05 and 0.001. Experimental data of Figs. 9 and 10 show that multiple-site fatigue cracks initiate much earlier and grow much faster in high-time aircraft structures than in new structure tests.

New aircraft full-scale structures and those taken from the fleet were fatigue tested. Both new aircraft and those with different service times were fatigue tested in the laboratory using similar loading programs. The lives of new and operated aircraft structures of the same type were compared taking into account fatigue damage equivalents

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for tests and operation. Figure 12 presents experimental and analytical data on residual life of four aircraft wings of the same type but having different service times, while Fig. 13 shows experimental and analytical data on residual life of four pressurized fuselages of a different type with various service times. Linear hypothesis of fatigue damage accumulation was used in residual life calculation. It follows from Figs. 12 and 13 that experimental residual life of aircraft structures being operated is much lower than the predicted one.

Corrosion

The problem of ensuring safe operation of structures with corrosion is being solved proceeding from operation experience. It is recommended to ensure standardized residual strength for the structure with standardized damages (Figs. 1 and 2) in the zones prone to corrosion. The corrosion damage growth time is predicted analytically, based on operational sizes of detected corrosion damages and duration of aircraft service with detected damages. The same methods of mathematical statistics as for estimating fatigue crack growth [8] are employed. Examples of estimating corrosion growth in wing and fuselage skins are presented in Figs. 14 and 15.

Conclusion

Structural damage tolerance criteria are formulated for transport aircraft. The size of a standardized damage enabling the aircraft structure to maintain standard residual strength was defined by generalizing the data on aircraft structure damage during tests and in operation. Analysis of maintenance intervals for aircraft resulted in outlining the required duration of crack growth from the reliably detectable size to the standardized size.

The fracture toughness and fatigue growth rates of materials for wings and fuselages (made of Al-alloys) required to ensure damage tolerance criteria in advanced aircraft are presented.

Residual strength criteria are studied for full-scale wing and fuselage structures having widespread fatigue damage. It is stated that fracture of several structures having such damage may be described by linear fracture mechanics criteria – material fracture toughness under plane strain or under plane stress. Many structures manufactured of plastic materials fail under stresses equal to the yield strength for a net section. These stresses should be calculated with consideration of stable crack growth.

The following experimental data were compared: crack resistance and fatigue strength of wing and fuselage skin materials of high-time aircraft and new materials; multiple-site fatigue crack growth duration in new structures during tests and in operation. Theoretical and experimental values of residual and fatigue lives of operated structures were compared. Crack resistance and fatigue strength of high-time aircraft turned out to be decreased in comparison with new structures.

For many types of aging aircraft, corrosion damage growth was estimated on the basis of fleet data. These data are used to specify structural inspection intervals for aging aircraft.

Safe operation of Russian aircraft is ensured by using results of comparing fatigue lives of new and operated full-scale structures; comparing crack resistance of wide specimens cut out of high-time aircraft skins and new skin sheets; generalizing fleet experience (damage data collection and processing using mathematical statistics methods).

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Damaged principal structural element	**Material**	$\frac{\sigma^{app}_{fr.net}}{\sigma_{0.2}}$	$\frac{\sigma^{c}_{fr.net}}{\sigma_{0.2}}$	$\frac{K_{fract}}{K_{app}}$	$\frac{K_{fract}}{K_{1c}}$
Skin and stringers near stringer splice in wing lower surface	D16ATNV D16T	0.8	1.0	0.5	-
Skin, stringers and spar of lower wing surface around stiffening lap edges	D16ATNV D16T	0.9	1.0	0.6	-
Skin and stringers of lower wing surface around stiffening lap edges	D16ATV D16T	0.9	1.0	0.5	-
Skin and stringers of monolithic stiffened panel of wing lower surface near fuel holes in stringer	D16T	0.7	0.83	1.0	-
Spars and shapes of upper wing surface	D16T	0.3	0.47	0.5	-
Splice shapes of upper wing surface	D16T	0.7	1.0	0.75	-
Stringer and lap for circumferential skin splice of pressurized fuselage	D16ATV D16T	0.75	0.88	1.0	-
Pressurized fuselage skin near three-row longitudinal riveted splice	D16ATV	0.57	1.0	0.5	-
Pressurized fuselage skin near two-row longitudinal riveted splice	D16ATV	0.63	1.05	0.9	-
Pressurized fuselage skin near two-row longitudinal riveted splice	D16ATV	0.48	0.85	0.7	-
Pressurized fuselage skin between two frames and between two stringers (19 through-the-thickness notches) (experiment)	D16ATV	0.9	1.0	0.6	-
Pressurized fuselage skin between two frames and between two stringers (19 through-the-thickness notches) (experiment)	D16ATV	0.85	0.85	1.0	-
Strip joining the cylindrical pressurized fuselage with spherical pressure bulkhead	D16ATV	0.16	0.17	0.45	-
Skin and stringer of lower wing surface around stiffening lap edges	V95AT1V V95T1	0.45	0.46	1.0	-
Lap joining the skins of lower wing surface	V95T1V	0.4	.41	0.4	1.0
Wing pivot assembly	V93T1	0.4	0.40	-	1.0

Table 1. Residual strength of full-scale aircraft structures with widespread fatigue damages

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Aircraft type	Alloy	Semi-finished item	σ_b, MPa	$\sigma_{0.2}$, MPa	δ, %	Si, %	Fe, %	Specimen number	W, mm	t, mm	2a, mm	K_{app}, MPa $\sqrt{m}$
Wing												
1	D16ATV	sheet	424-443	312-319	21-22	0.28	0.38	1.1*n*	566	4.9	176	98
								1.2*n*	565	4.9	171	100
			472-483	355-367	18-20	0.27	0.4	1.1*o*	568	5	250	91
								1.2*o*	565	5	170	81
2	D16ATHV	sheet	462-490	386-405	8-12	0.28	0.28	2.1*n*	754	3.8	255	102
								2.2*n*	754	3.8	243	103
								2.3*n*	754	3.8	245	103
			476-501	395-425	11-13	0.24	0.3	2.1*o*	710	3.8	225	83
								2.2*o*	747	3.9	220	82
								2.3*o*	713	4.8	223	91
								2.4*o*	692	3.7	152	69
3	D16T	extruded panel	509-547	374-417	12-13	0.27	0.38	3.1*n*	500	6	150	103
								3.1*o*	494	3.2	150	96
								3.2*o*	494	3.2	175	87
								3.3*o*	494	3.2	150	91
4	V95AT1B	sheet	535	440	10			4.1*n*	500	4	160	62
								4.2*n*	500	4	163	60
			545	510	10-11			4.1*o*	580	4	160	53
								4.2*o*	580	4	142	56
								4.3*o*	580	4	140	56
5	B95T1	extruded panel	561-588	506-550	8-9	0.07	0.28	5.1*n*	500	6	160	86
								5.2*n*	500	6	160	87
								5.3*n*	500	6	160	89
			603-634	575-588	7-8	0.22	0.3	5.1*o*	500	6	160	63
								5.2*o*	500	6	160	63
								5.3*o*	500	6	160	54
								5.4*o*	500	6	160	49

Table 2. Material strength in wing and fuselage skins

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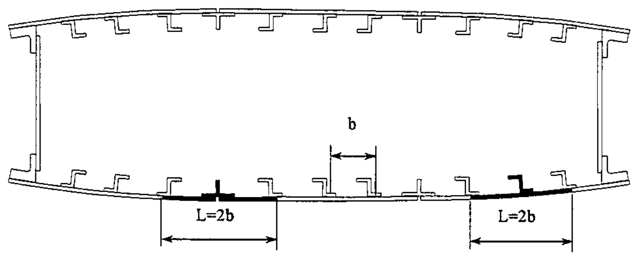
Aircraft type	Alloy	Semi-finished item	σ_b, MPa	$\sigma_{0.2}$, MPa	δ, %	Si, %	Fe, %	Specimen number	W, mm	t, mm	2a, mm	K_{app}, MPa$\sqrt{m}$
Fuselage												
6	D16ATV	sheet						6.1*n*	1200	1.3	599	84
			430-440	320-328	17-19	0.21	0.32	6.1*o*	1195	1.1	597	83
								6.2*o*	1185	1.2	598	87
7	D16ATV	sheet	460-471	356	13-15	0.35	0.27	7.1*n*	1200	1.5	600	96
								7.2*n*	1200	1.5	600	100
			483-488	359-364	13-17	0.35	0.3	7.1*o*	1180	1.5	620	77
								7.2*o*	1246	1.5	600	91

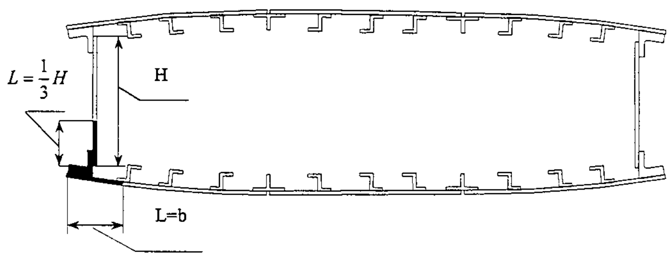
Table 2 (continued). Material strength in wing and fuselage skins

Aircraft type	Alloy	Semifinished item	Specimen number	$(d2a/dN)_{aver.}$, mm/kilocycle at ΔK=31 MPa$\sqrt{m}$	$(d2a/dN)_{aver.}$, mm/kilocycle at ΔK=62 MPa$\sqrt{m}$
Wing					
1	D16ATV	sheet	1.1*n*, 1.2*n*	8.5	131
			1.1*o*, 1.2*o*	36	970
2	D16ATNV	sheet	2.1*n*, 2.2*n*, 2.3*n*	7.5	154
			2.1*o*, 2.2*o*, 2.3*o*, 2.4*o*	17	491
3	D16T	extruded panel	3.1*n*		
			3.1*o*, 3.2*o*, 3.3*o*	12	265
4	V95AT1V	sheet	4.1*n*, 4.2*n*	20	1000
			4.1*o*, 4.2*o*, 4.3*o*	58	5527
5	V95T1	extruded panel	5.1*n*, 5.2*n*, 5.3*n*	13	300
			5.1*o*, 5.2*o*, 5.3*o*, 5.4*o*	20	1200
Fuselage					
6	D16ATV	sheet	6.1*o*, 6.2*o*	5.7	712
7	D16ATV	sheet	7.1*n*, 7.2*n*	3.8	208
			7.1*o*, 7.2*o*	8.5	895

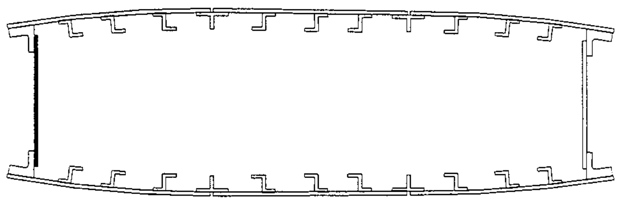
Table 3. Fatigue crack growth rate in wing and fuselage skin materials



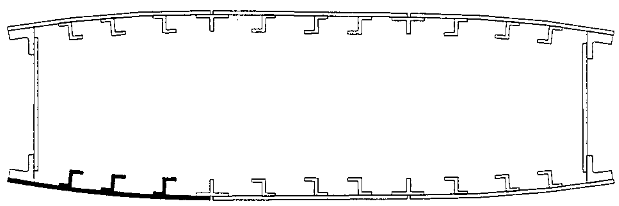
Simultaneous initiation of one crack per panel; crack length underneath broken stringer is equal to two bay lengths



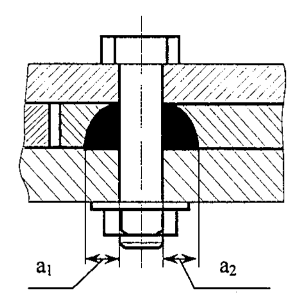
Spar cap broken; spar web crack length = 1/3 of web height; skin crack is equal to one bay length



Spar web broken



One panel broken



$a_1 = a_2 = 25$ mm;
corner and surface cracks in thick elements

FIGURE 1 - Standardized Wing Damages

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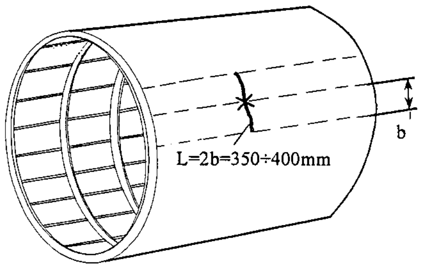
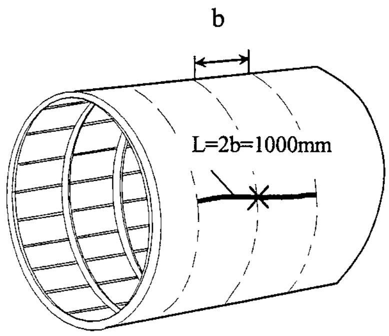
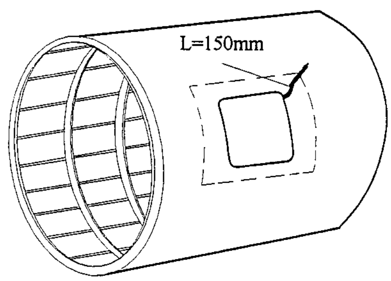


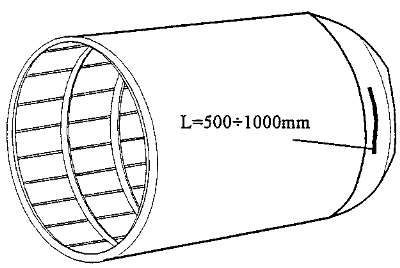
Transverse skin crack length = two bay lengths; stringer broken



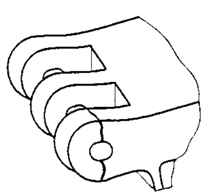
Longitudinal skin crack length = two bay lengths; frame broken



Cutout-initiated crack 150 mm long; skin and edge shape damaged



Skin crack in pressure bulkhead



One element broken in the joint between fuselage and wing,
fuselage and empennage etc.

FIGURE 2 - Standardized Fuselage Damages

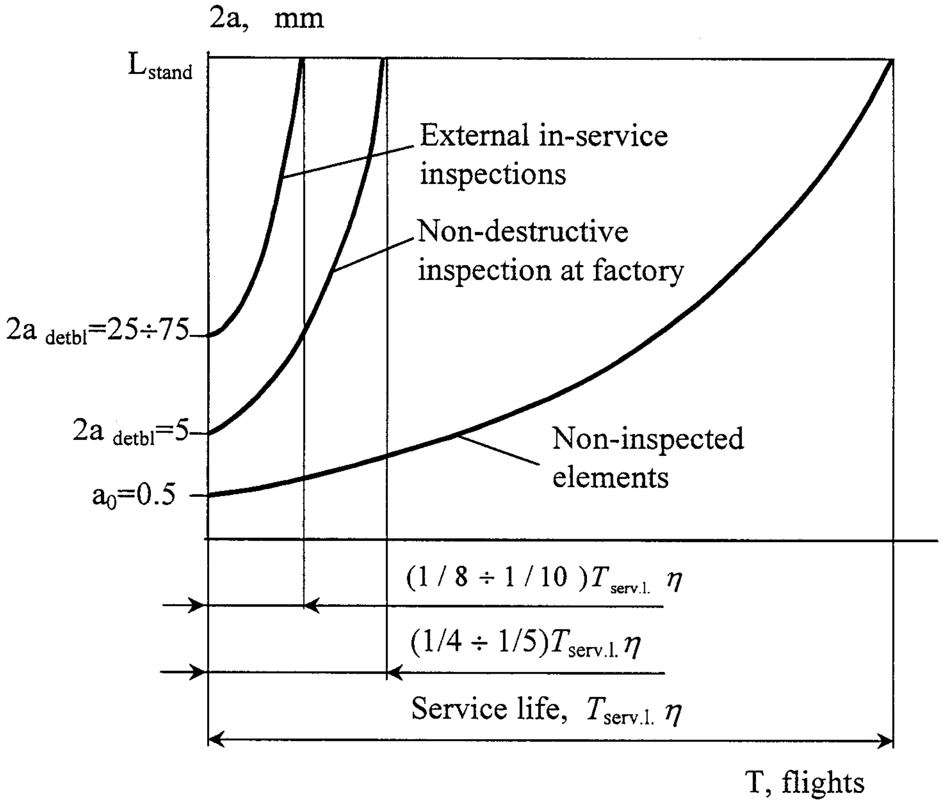


FIGURE 3 - Requirements to Crack Growth Time

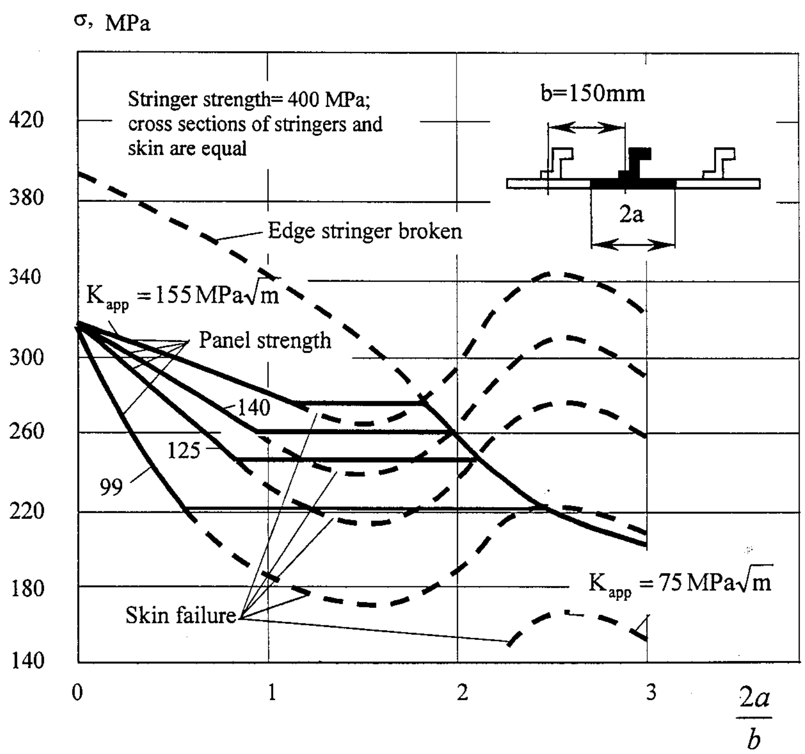


FIGURE 4 - Skin Fracture Toughness Effect on the Stiffened Panel Residual Strength

Nesterenko

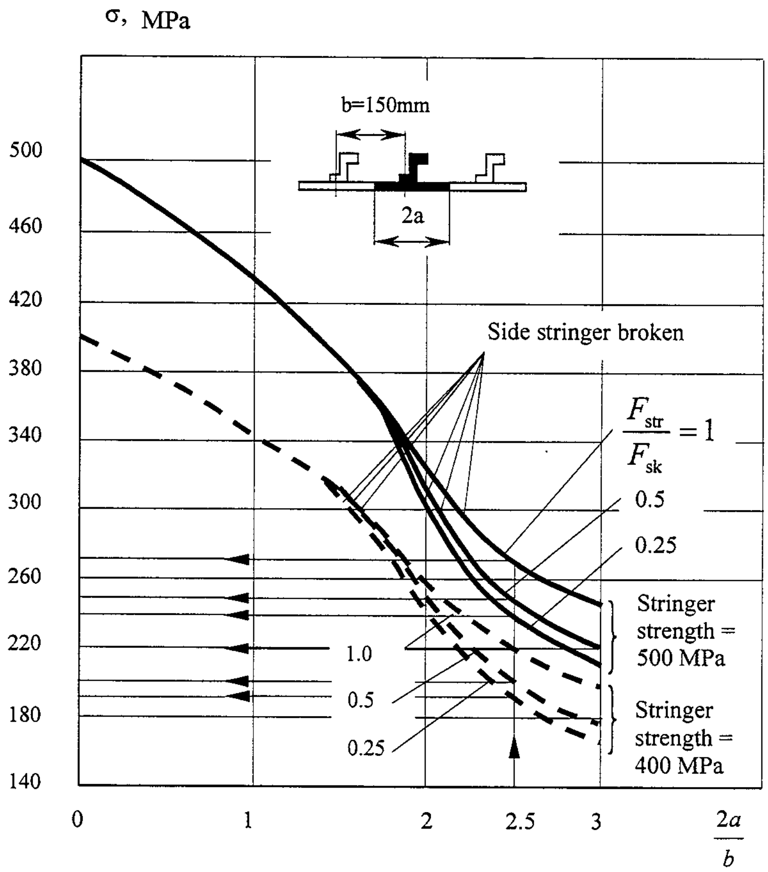


FIGURE 5 - Panel Residual Strength: Dependence on Stringer Strength and Stringer F_{str} to Skin F_{sk} Area Ratio

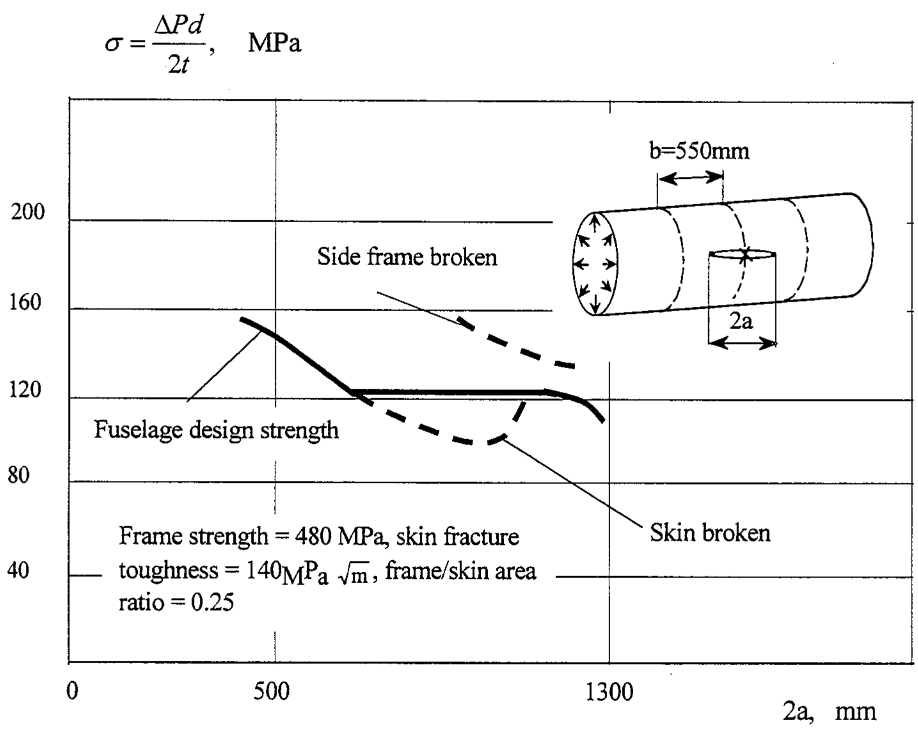


FIGURE 6 - Residual Strength of Pressurized Fuselage

Nesterenko

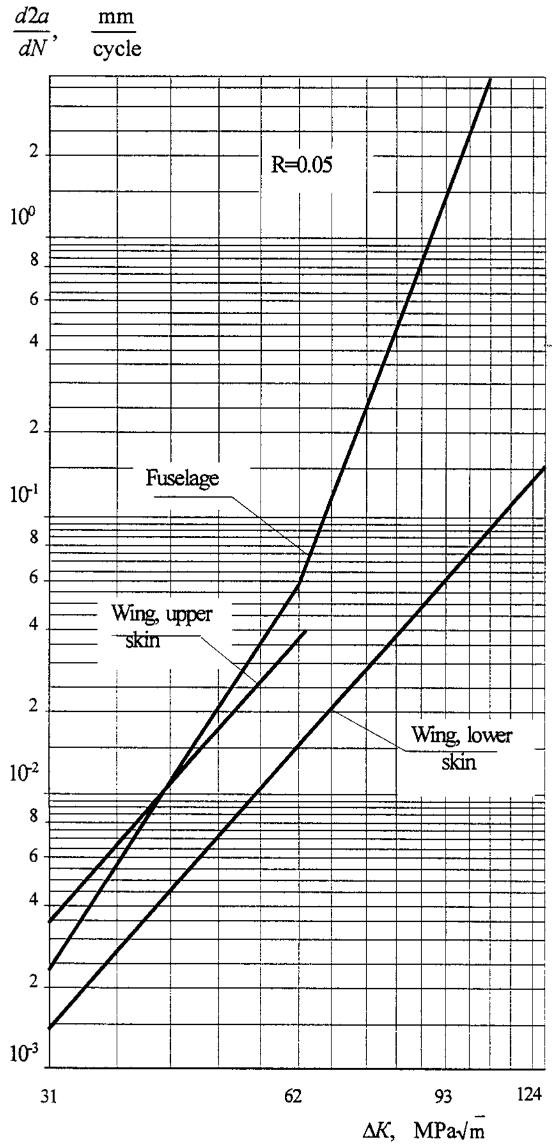


FIGURE 7 - Required Crack Growth Rates in Skin Materials

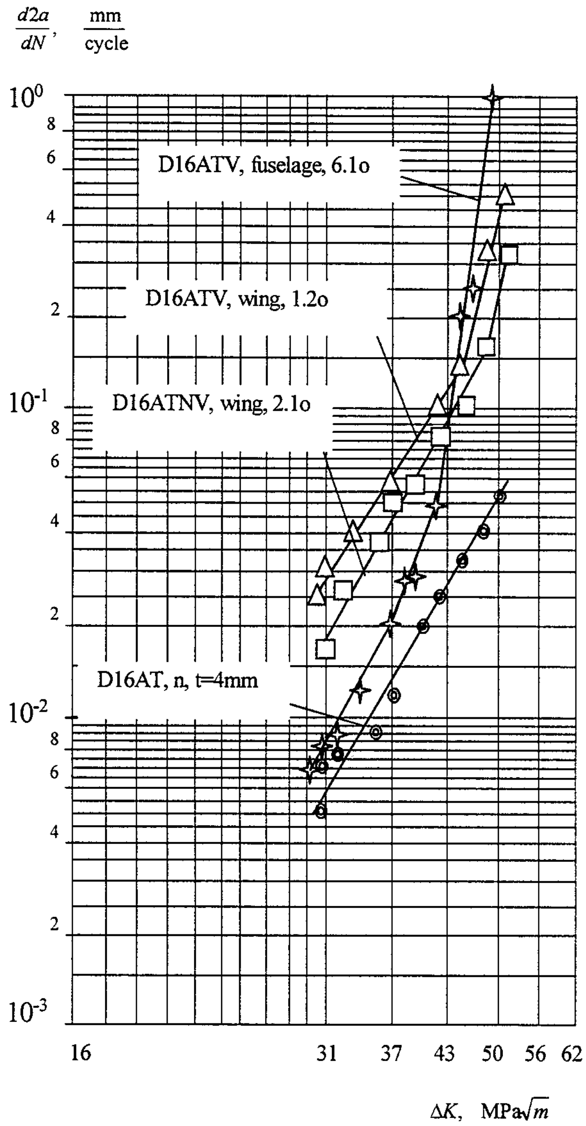


FIGURE 9 - Fatique Crack Growth Rate

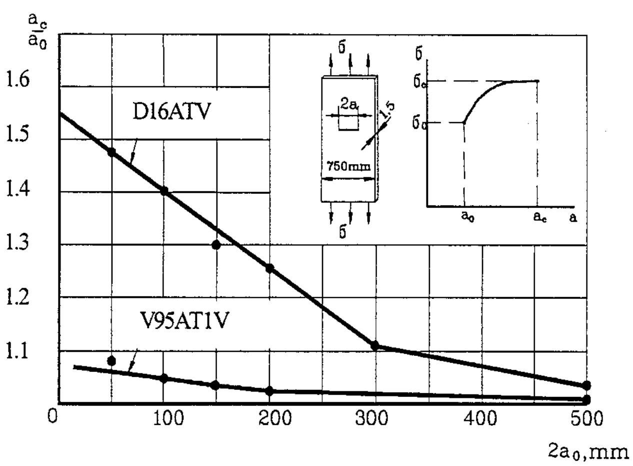


FIGURE 8 - Stable Crack Growth from Sheet Cutouts

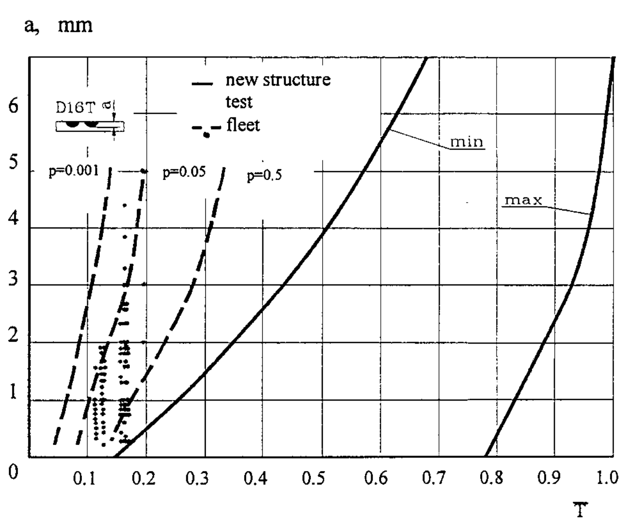


FIGURE 10 - Relative MSD Growth Time: Upper Wing Panel Attachments

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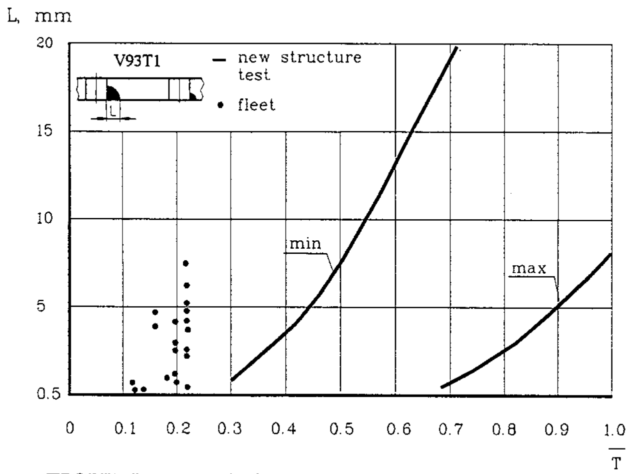

FIGURE 11 - Relative MSD Growth Time: Lower Wing Panel Attachments

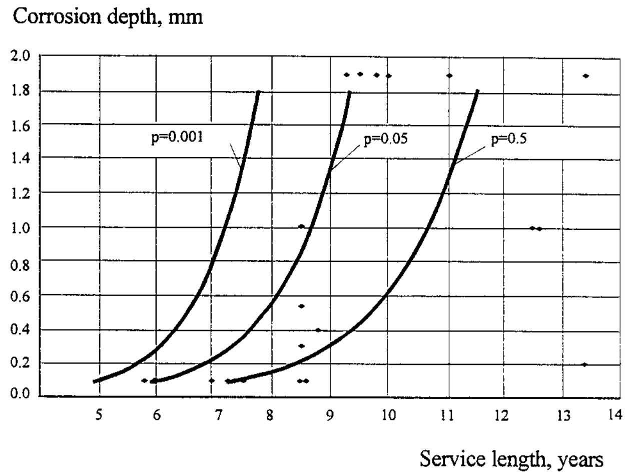

FIGURE 14 - Corrosion Growth in Fuselage Skin

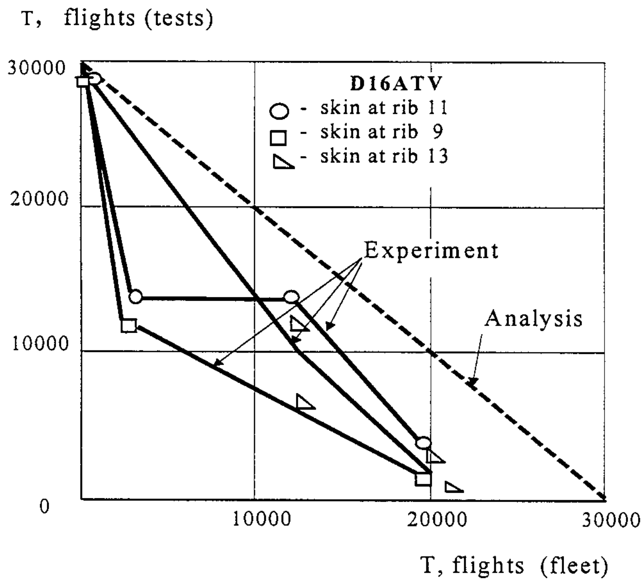

FIGURE 12 - Residual Life of Turboprop Aircraft Wing

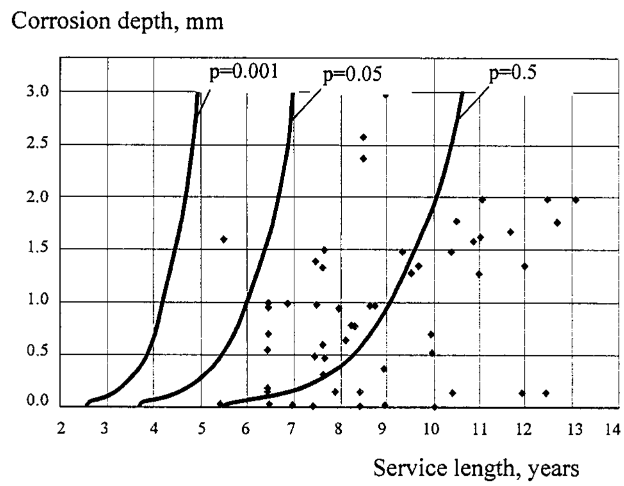

FIGURE 15 - Corrosion Growth in Wing Skin

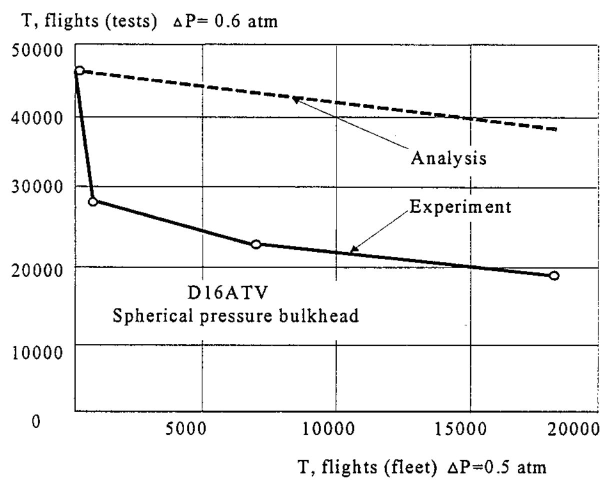

FIGURE 13 - Residual Life of Turbojet Aircraft Fuselage

Nesterenko