PROBLEMS OF ENSURING SERVICE LIFE THE SECOND GENERATION SUPERSONIC TRANSPORT

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Abstract

On the basis of summarizing the experience of developing supersonic aircraft some aircraft structural locations are found out whose shape is identified by the lifetime concepts. The lifetime properties of SST-2 lower wing panels have been analyzed when the titanium alloy welded wing box was used. The relation has been generated for the advanced Al-alloy structure between the lifetime and the cruise Mach number. Potential application of composite materials in tip of wing has been analysed.

1 Introduction

The development of highly effective and economically reasonable supersonic transport of the second generation (SST-2) envisages the necessity to ensure structural service life for 80000 flight hours including 60000 flight hours of the supersonic flight that 2.5 times exceeds the service life of supersonic transports of the first generation (Tu-144, Concorde). The task to get such long service life is especially difficult because it is requires simultaneously with the service life increase to provide the aircraft structure relative weight decrease no less than 20% as compared to the relative structural weight of the first generation SST. The problem becomes still more difficult as to create highly effective aircraft some layouts are under investigation that differs greatly from those of the first generation aircraft. The comparison of aircraft performances for the first and the second generations presented in Table 1 has shown that the new generation aircraft have larger fuselage and wing aspect ratios and smaller wing thickness-to-chord ratio.

Table 1

Performance	SST-1	SST-2
Design life goal, flight hours	30000	80000
Wing aspect ratio	1,55	2,9
Wing thickness-to-chord ratio, %	3.2	2.5
Fuselage aspect ratio	19,7	29

All this makes the project realisation more difficult from the viewpoint of ensuring the structural service life.

Today different concepts for cruise flight velocity selection are being studied. The European researchers utilise cruise Mach number about M≈2; American researchers consider the candidate aircraft having M≈2,4.

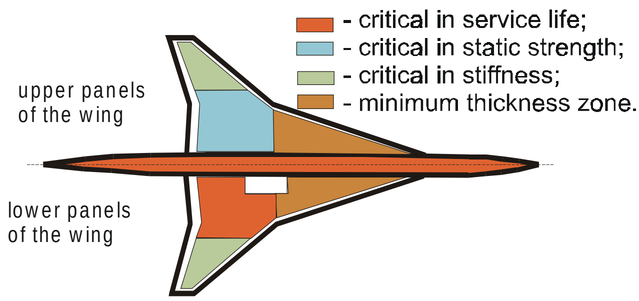


Fig.1 Design criteria for the wing and fuselage panels.

The increase of Mach number until 2,4 requires absolutely new approaches to selecting heat resistant structural materials, to developing quite new structural-technological solutions.

From the view point of strength ensuring the aircraft structure may be arbitrary divided into four zones (fig. 1):

- critical in service life;
- critical in static strength;
- critical in stiffness;
- minimum thickness zone.

It is obvious from the figure that the service life concept is dominating for significant structural zones including lower panels of the wing primary box and pressurised fuselage panel.

This presentation considers the task to ensure the structural service life in these zones for supersonic transport of the second generation designed for cruise Mach number M≈2.

2 Potential material distribution in the structure

During the structural design of the SST-2 the task to select the structural alloys becomes more urgent than for subsonic transports. This is envisages by the need to take into account the action of the elevated temperature in the supersonic flight and extremely strict requirements in weight efficiency. The study results presented in Refs.[1,2] demonstrate that in the SST-2 structure a combination of different materials will be used including Ti and Al-alloys and composite materials.

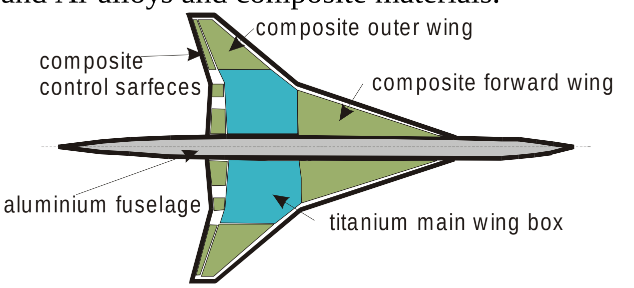


Fig.2 Possible material usage for the wing and fuselage.

Fig. 2 presents the potential distribution of structural materials. Ti-alloy is proposed to be applied in the severely loaded wing areas including the primary wing box.

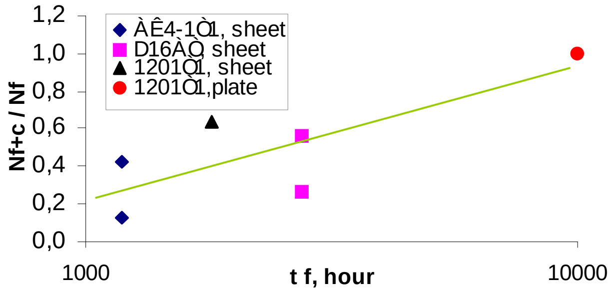
The investigation has shown that the wing primary box fabricated of Ti-alloy is 15-20% lighter than that of Al-alloy. A commonly used Al-alloy is considered for the pressurised fuselage structure in terms of costs and resistance to impact loads. It is supposed to fabricate from the composite materials a slightly loaded leading wing part and the wing trailing part where stiffness due to flutter should be ensured.

3 Approach to Al-alloy selection

Material properties for the SST structure should satisfy both traditional requirements to the materials for long-operated subsonic aircraft and some special requirements due to long action of elevated temperatures. It is known that high creep resistance is one of the main requirements to the supersonic aircraft materials for long temperature service life. It is dictated by the limitation on the value of the whole residual strain of the aircraft structure [3].

It was stated, however, that such simplified approach may result in the fact that enough high fatigue and crack resistances would not be ensured. To find out the connection between the creep properties and fatigue and crack resistances a series of tests was conducted including the study of creep, long-time static strength, sensitivity to the notch and the crack under creep, as well as to fracture resistance under combined effect of

Fig.3 Relation of the ratio of life under fatigue and creep to fatigue life versus fracture time



cyclic loads and creep simulating the loading conditions for the wing lower surface of the SST-2 [4].

Fig. 3 gives the relation of the ratio of life under fatigue and creep to fatigue life N_{f+c}/N_f versus fracture time t_f during the repeated strength tests. These data demonstrate the correlation between N_{f+c}/N_f and t_f. The largest life decrease due to creep (2,3 times) is observed for the alloy AK4-1T1 having the smallest long-term strength of the notched specimens t_f. For the plate of 1201T1 alloy that has very large long-term strength in the notched specimen (t_f >10^4 hours) the damaging effect of the creep is not observed.

Fig.4 Crack resistance under combined loading and heating

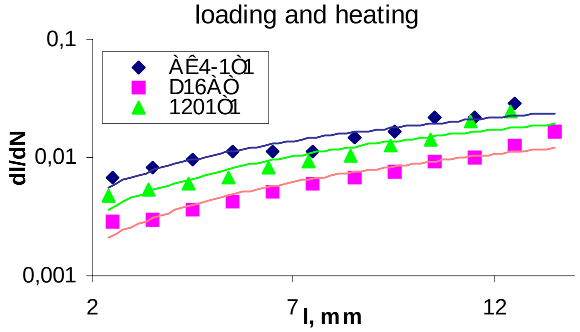


Fig. 4 presents the study results of Al-alloy crack resistance under combined loading. These data also demonstrate that in terms of crack growth resistance the better are the long-term strength properties the smaller is the crack growth rate under joint action of fatigue and creep. Thus the data studied show that the long-term static strength with the notch $\mathbf{t_f}$ may serve as the characteristic of the material behaviour under combined action of mechanical loads and creep both for the stage of crack initiation and for that of crack growth, and so it may be used for comparative estimates of fracture resistance for structural materials.

However to ensure enough high level of the structural long-term strength $\mathbf{t_f}$ it is required that the material has low creep resistance as in this case it is less sensitive to the notch and the crack. Besides, it is known that the SST-2 structure must satisfy the strict requirement to the tolerable whole residual strain. Thus in accordance with the Russian Strength standards for flying vehicles this strain should not exceed 0.2% by the end of the lifetime. Hence, the requirement of high creep resistance for the SST-2 contradicts the one of low notch sensitivity. So from the view point of ensuring enough long service life the most advanced for this aircraft structure is the material having low creep resistance under high stresses typical for location in the vicinity of the concentrators and on the contrary the high creep resistance in the stress range acting in the regular sections of the airframe structure.

The analysis of known Al-alloys show that the reasonable operational characteristics for the SST-2 structure are available in the new heat resistant Al-alloys AK4-2chT1 and 1215T1.

Fig.5 Ratio of AK4-2chT1 and 1215 alloy properties to those of AK4-1chT1 alloy

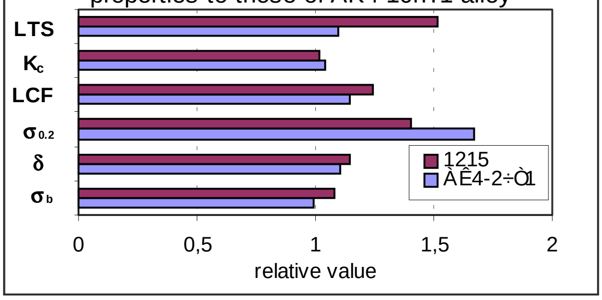


Fig.5 presents the ratio of properties for AK4-2chT1 and 1215T1 clad sheets to the properties for AK4-1chT1 alloy applied in the structure of Tupolev TU-144. It is obvious from the figure that 1215T1 sheets have 10% higher strength properties (σ_b, $\sigma_{0,2}$) as compared to AK4-1chT1 alloy and exceed (1.5 times) it in long-term strength. It enables to consider 1215T1 alloy as the most applicable one for the pressurised fuselage structure at the present stage.

4 Service life estimate of Al-alloy pressurised fuselage panels

The clad pressurised fuselage structure was affected by the cyclic excess pressure and by heating during the supersonic flight. Non-linear relation that took into account the damages caused defined the life by fatigue and creep:

$$\frac{N}{N_f} + \frac{N^* t_{fl}}{t_{cr}} = A(\frac{N^* t_{fl}}{t_{cr}})\text{ ,where}$$

N - structure life, flights;

N_f - specimen life under room temperature, flights;

t_{fl} - cruise flight duration, hours;

t_{cr} - repeated strength, hours;

$A(\frac{N^* t_{fl}}{t_{cr}})$ - parameter of non-linear damage accumulation.

It was supposed during the calculations that the service life of the pressurised fuselage structure is defined by the life of the longitudinal joint having the effective concentration factor assumed equal to 4. The circumferencial stresses were assumed equal to 100 mPa that corresponded to the level of such stresses in the pressurised fuselages of modern passenger aircraft. The cruise flight Mach number was varied and different Al-alloy was studied.

Fig.6 Relation of pressurized fuselage panel service life versus Mach number

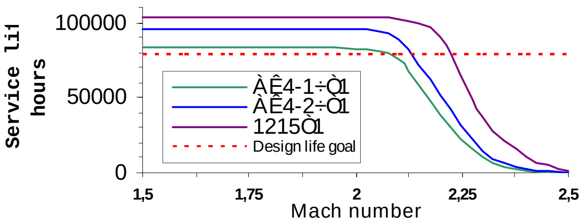


Fig. 6 shows the relation of the service life for regular pressurised fuselage panels versus Mach number. It is seen in the figure that the application of 1215T1 Al-alloy would enable to ensure the design life goal up till Mach numbers $M \leq 2{,}2$. For smaller Mach numbers the opportunity appears to increase the level of curcumferencial stresses, and hence, to decrease the structural weight.

5 Potential application of Ti-alloys in wing box.

Ti-alloys have higher specific strength and heat resistance as compared to Al-alloys. Study results show that there are no troubles with the effect of elevated temperatures and creep on Ti-alloy properties in the temperature range typical for the SST-2. Therefore the strength and the service life of details and components fabricated of these alloys is dominated by the properties they have at room temperature.

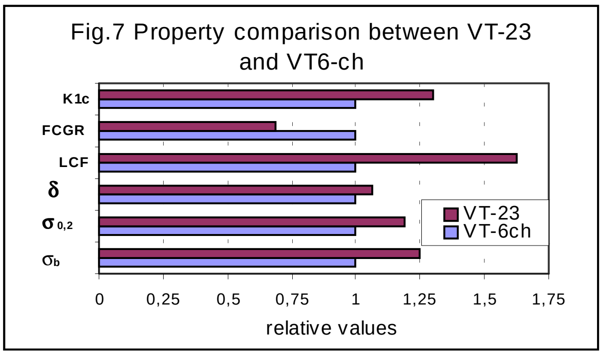


Fig. 7 presents the principal properties of Ti-alloy VT-23 plate as compared to the plate properties for widely used alloy VT-6ch (similar to Ti-6Al-4V). These data demonstrate that VT-23 alloy significantly exceeds VT-6ch alloy in all the aspects considered, and the most advantage (1,5 times) is observed in terms of local cyclic fatigue and fatigue crack growth rates. This enables to consider it as a structural material for SST-2 wing box.

At the same time the application of Ti-alloys gives rise to the task of manufacturing Ti-alloys and their products having the guaranteed properties. The analysis of fatigue strength study results for the specimens and structural elements made of Ti-alloys [5-7] validate the dependence of fatigue properties on the semiproduct structure, surface state after the processing and hardening, higher than for Al-alloys sensitivity to stress concentration, crumple and contact stresses. Strong sensitivity to the welding quality is observed in welded structures.

Fig.8 Relative life of specimens and panels of Vt-6ch

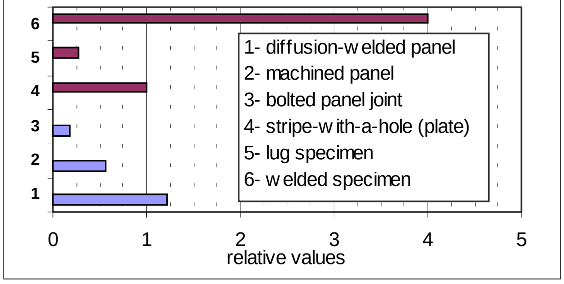


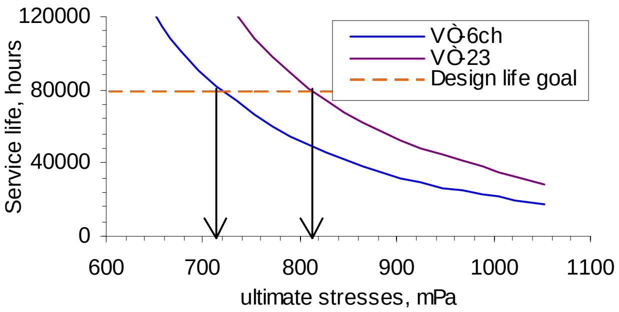
Fig. 8 illustrates the relative lives of specimens and full-scale panels made of VT-6ch Ti-alloy. It can be noted in the figure that essential life decrease is observed both in the lug specimen and in the bolted panel joint as compared to the life of the stripe-with-a-notch specimen. However, for welded panels and specimens some life increase is observed as compared to the stripe-with-a-hole specimen. This leads to the conclusion that the Ti-alloy wing box should be fabricated as a fully welded structure. When the bolted panel joints are required the panel thickness must be increased locally to compensate the life decrease due to crumple stresses.

The investigations of the recent few years have shown that the most advancing welding method is the diffusion one. The technological process of the diffusion welding provides the fabrication of joints without microflaws having the properties of plasticity, strength, low and high cyclic fatigue and fracture toughness on the level of the basic metal. Ref.[6] outlines the study results of strength and fatigue characteristics of diffusion-welded panel and structurally similar extruded panels. This study has shown that the life of diffusion-welded panels is no less than two times higher than that of the extruded ones.

To estimate the service life of the wing box some calculations have been done if the wing box is a fully welded structure. Ti-alloys VT-6ch and VT-23 have been analysed. Different levels of design stresses in the range from 600 mPa to 1050 mPa.

Fig.9 presents the relation of regular panel service life on the lower surface of the wing box versus the value of the design stresses. It is obvious from the figure that the design life goal was ensured when the design stress level was equal to 700 mPa for VT-6ch material. When Ti-alloy VT-23 was applied the level of design stresses can be increased till about 800 mPa, that in turn would result in the weight decrease of the wing box.

Fig.9 Relation of service life in the wing box panels versus ultimate stress



To estimate the effect of design stress level and structural parameters of Ti-alloy VT-23 lower wing panel on its damage tolerance characteristics a crack was studied in the panel skin under the broken stringer. The damage tolerance of the wing panel for the SST-2 was carried out based on the linear elastic fracture mechanics using the automatic system ALTAY [8].

Fig.10 Relative stringer stiffness effect on the critical crack length

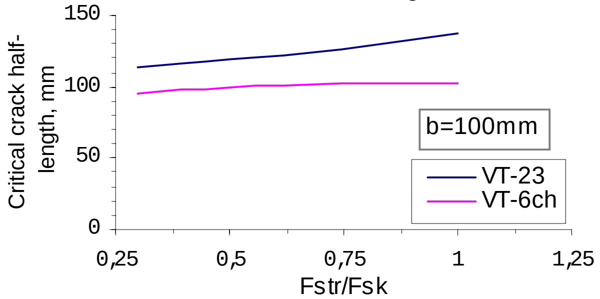
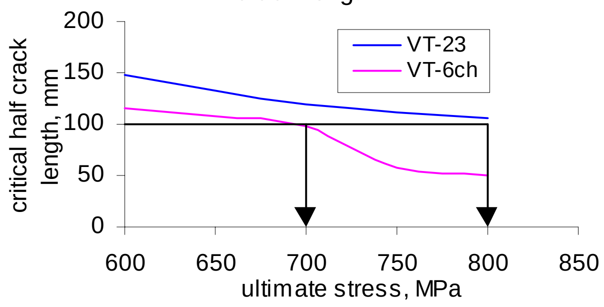


Fig. 10 illustrates the relation of the critical crack half-length l_c versus the relative stringer stiffness F_{str}/F_{sk} (F_{str} - stringer area, F_{sk} - skin area) under the ultimate stress σ_{ult} = 700 mPa.

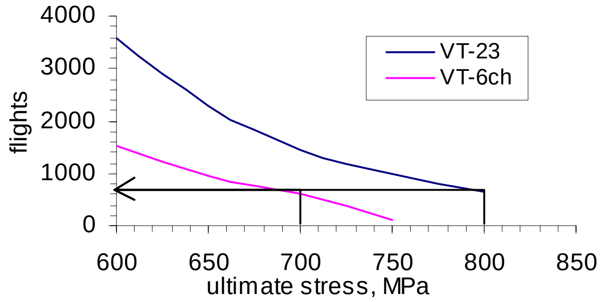
Fig.11 Ultimate stress effect on the critical crack length



The relation between the critical crack half-length l_c and the design stresses at F_{str}/F_{sk} = 0,5 is shown in Fig. 11.

Crack growth duration from the minimum detectable by visual control crack half-length l_0=50 mm till the critical length as a function of ultimate stresses is given in Fig. 12.

Fig.12 Ultirmate stress effect on crack growth duration



Using the above curves for the given inspection interval some optimum ratios of stringer-to-skin stiffness, as well as the level of ultimate stresses. The calculations done demonstrate that for two-bay fracture of the stringer and the panel can be ensured at the stresses σ_{ult} = 800 mPa and F_{str}/F_{sk} = 0,35. In this case the inspection interval is about 750 flights.

6 Potential application of composite materials in tip of wing

The operational experience of composite material structures witnesses that technological damages (flaws like pores and delaminations) and impact damages (dents, cracks and holes) arising both during the manufacture and the operation result in the significant decrease of the structural strength and fatigue. Therefore, some requirements have been formulated to the structures and the methods to find out their applicability in terms of the required strength and service life properties. They are based on the damage tolerance and fail safe concepts. There are some tool methods to inspect pores, delaminations and impact damages that enables to reject the structures having the damage sizes exceeding the threshold values. The structure with technological damages whose sizes do not exceed threshold values should bear the ultimate load. The damage growth duration starting from the initial size till the critical one should not be less than design life goal. The structure having operational damages of visually detectable sizes should satisfy less strict requirements. The structure with a through thickness damage being detected during the pre-flight inspection should bear the operational load and have the life corresponding to the inspection interval between the major repair.

There is a structural technological approach to increasing the static strength of composite structures having stress concentrators by introducing into the basic material the stripes having the longitudinal layers of the basic material completely replaced by the layers of some other composite but with the similar orientation angle [10-13]. These high-modulus or low-modulus stripes play the role of crack stoppers. The static failure of the structure having the through-thickness crack (notch) was described by two-parametric model of fracture mechanics. The failure criterion was assumed the critical value of stress intensity factor K_c, being determined with regard for the correction to the cracking zone in the damage tip [9]. The effect of the stopper width b and its space w on the tolerable damages has been analytically investigated by the damage tolerance concept for carbon-plastic skin with a through-thickness crack having a starting crack length of $2l_0$.

One location in the wing box panel lower (upper) skin has been studied under compression (tension) and shear. The crack stoppers were the carbon-glass plastic stripes being introduced into the carbon plastic skin in such a way that their longitudinal layers oriented along the action of the axis force have been replaced by the glass-plastic layers with orientation angle $\alpha=\pm45°$.

The ultimate stresses σ_x and τ_{xy} were the stresses of the static skin failure with a crack having $2l_0$=50 mm and symmetrical relative to the longitudinal axis of the carbon-plastic stripe. Here the fatigue life N of the skin with this damage must be equal to the design life goal ~25,000 flights. It was assumed that the cracked skin ($2l_0$=100 mm) symmetric relative to the stopper longitudinal axis should not be fractured at the limit stresses being 67% of the ultimate stresses. The damage growth life from $2l_0$ till the limit size was assumed equal to the cycle number between the major repairs (~5000 flights).

To ensure the weight efficiency of the composite skin as compared to the Al-alloy one the values of the ultimate stresses were assumed the same as for the aluminium skin, i.e. σ_x=337mPa.

For the skin materials the high-modulus carbon plastic and glass plastic were assumed that had the basic design properties of the mono-layer as follows:

#	Property	Carbon plastic	Glass plastic
1	Young modulus, mPa	23000	55000
2	Ultimate strength (mPa) at: -tension -compression;	 1400 1100	 1100 700

It was assumed in the calculations that the ratio of shear and normal stresses was τ/σ=0.196 and the ratio b/w=0.2. First, a damage having the initial length $2l_0$=50mm was considered located symmetrically relative to the axis of the carbon-plastic stripe.

Fig.13 Relation of ultimate stress versus stopper space.

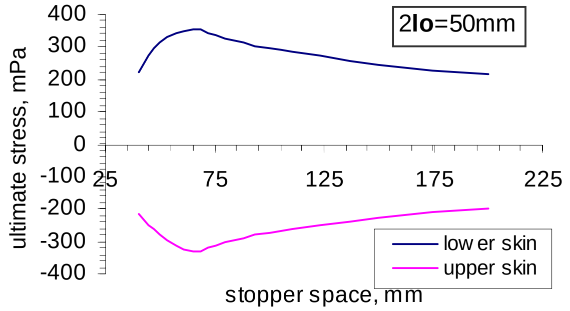


Fig.13 presents the relation between the ultimate stress and the stopper space.

Fig.14 Relation of skin service life versus stopper space

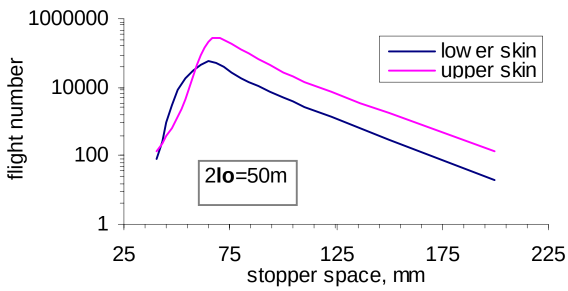


Fig.14 gives the damage growth duration from $2\mathbf{l_0}$ till the limit size depending on the stopper space.

Then the damage having the initial length $2l_0$=100mm was studied that was also symmetrical relative to the axis of the stopper. The stopper space was assumed 65mm. The stopper width was varied.

Fig.15 Relation of limit stress versus stopper

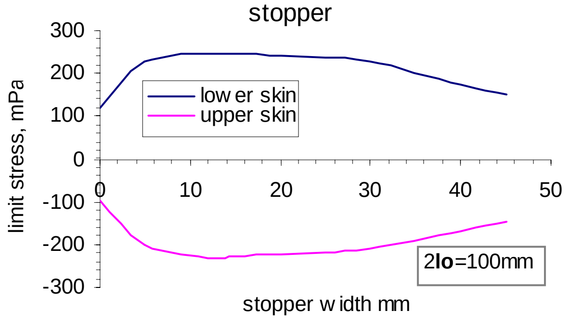


Fig.15 demonstrates the relation of limit stress versus stopper width.

Fig.16 Relation of flight number between inspections versus stopper width

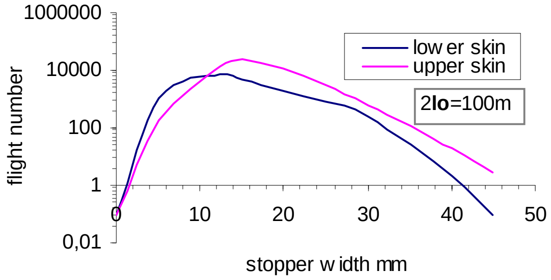


Fig.16 presents the damage growth duration from $2l_0$ till the limit size depending on the stopper width.

It is obvious from the diagrams that to ensure the design life goal of the composite structure there exists some optimum value for the stopper space and its width.

7 Conclusions

The ensuring of structural service life for the supersonic transport of the second generation envisages the need to solve many scientific problems connected with application of new metallic and composite materials, new structural layouts.

The application of modern high-strength Al-alloys (1215T1, AK4-2T1) in the pressurised fuselage structure can ensure the design life goal up till Mach numbers $M \leq 2{,}1$.

For fully welded primary wing box fabricated of one of high-strength Ti-alloys (VT-6ch, VT-23), the required service life and damage tolerance properties may be received at the design stress level up until 700-800 mPa.

Safe operation of structural elements fabricated of composite materials can be provided only when the stopper stripes are used. To ensure the design life goal of the composite structure there exists an optimum value both for the stopper space and for its width.

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