SERVICE LIFE OF A TITANIUM MAIN WING BOX FOR SECOND GENERATION SUPERSONIC COMMERCIAL TRANSPORT

Dementyev A.D.,Rodchenko T.S., Shcherban K.S

TsAGI, Zhukovsky, 140180, Moscow region, Russia

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Abstract

On the basis of summarizing the experience of developing supersonic aircraft the lifetime properties of SST-2 lower wing panels have been analyzed when the titanium alloy welded wing box was used. The fatigue curve of welded joints of panels from a titanium alloy Vt6ch is obtained, it was used for the estimation of durability. The analysis of the critical sizes of cracks, crack growth duration from the visually detected size up to the critical size is carried out. For an estimation of a beginning of inspections in operation, the analysis of crack growth duration from a probable technological imperfection in a welded joint up to the critical size is carried out.

1 Introduction

The specific feature of the second generation supersonic commercial transport (SCT-2) is the need to ensure the structural service life of 25000 flights (80000 flight hours) that is 2,5 times larger than the first generation SST structures (Tupolev Tu-144, Concorde). This task becomes still more difficult to solve as alongside with the enhancement of the service life the structural relative weight should be decreased at least for 20 per cent. The contemporary research of potential structural layouts has shown that it may be reasonable to make a wing box as the integrally welded Ti-alloy structure. In this case the primary wing box designed for static strength is about 20 per cent lighter than that of Al alloy.

However these potentials are not always realized due to the structural drawbacks typical for Ti alloy welded structures. They are as follows:

- fatigue strength sensitivity to the type of the material applied;
- initiation of residual stresses of the significant value (till the yield strength level) and strains causing the shrinkage and buckling of the welded structures;
- sensitivity of mechanic and fatigue properties in welded joints to the parameters of the technological welding process;
- presence of flaws in the welded joints significantly decreasing the fatigue strength.

Hence while designing the integrally welded Ti alloy structure it is very important to select the required level of design stresses in the lower panels with regard for the factors affecting fatigue and damage tolerance parameters. If the design stresses are not selected thoroughly enough it results either in the structural overweight or in the lifetime decrease.

2 Fatigue life analysis

Integrally welded wing box structure fabricated of Ti alloy Vt-23 has been under consideration. The life of such a structure is dominated by the longitudinal and lateral joints connecting panel skin with stringers and ribs, as well as by welded joints between panels, and also by the welded joints between spar caps and panels. Therefore the allowable design stress level was found based on life analysis in the welded joints.

The welded joint life has been outlined by the degree relation:

$$N = \frac{10^c}{\sigma_{\text{equiv}}{}^m n} \text{ [flights], where}$$

c and m – parameters of design fatigue curve;

σ_{equiv} – equivalent stress;

n – factor accounting for effect of technological defects.

The equivalent stress σ_{equiv} has been outlined by idealizing the flight stress sequence by the complete cycle emphasizing method, by recalculating the emphasized cycles into the pulsating cycles and by the linear accumulation of the fatigue damage.

The asymmetric cycles «σ_{max}-σ_{min}» have been recalculated into the pulsating ones σ_{pulse} using empirical formulas:

$$\sigma_{puls} = \sqrt{\sigma_{\max}(\sigma_{\max} - \sigma_{\min})} \text{, while}$$

$$\frac{\sigma_{\max} + \sigma_{\min}}{2} \geq 0;$$

$$\sigma_{puls} = \frac{1}{\sqrt{2}}(1{,}2\sigma_{\max} - 0{,}8\sigma_{\min}) \text{, while}$$

$$\frac{\sigma_{\max} + \sigma_{\min}}{2} \leq 0,\ \sigma_{\max} > 0;$$

$\sigma_{puls} = 0$, while $\sigma_{max} < 0$.

It was assumed that the stresses on the wing box lower surface are proportional to the ultimate stresses at all the stages of the typical flight. The curve of relative stresses σ/σ_{ult} is given in Fig.1. The flight stress curve was the result of multiplying the relative stress curve by the ultimate stresses.

The parameters of the design fatigue curve c and m were found based on fatigue test results for Ti-alloy VT6ch specimens having welded joint made by argon-arc welding (ArAEW). These curves have been compared in Fig.2 with those for specimens with free holes. The experimental data have been approximated by the root-mean square method by the degree function using average life values for each stress level and so the average fatigue curve has been outlined.

The comparison of average fatigue curves shows that the life of the welded joint equal to that of the free hole specimen may be the result of higher stress levels. For instance, the specimens with welded joints have the life of 10^5 cycles at the stress of 524 MPa, while the free hole specimens have it at 230 MPa. It demonstrates the significant advantage of welded joints. However the presence of separate gas pours or unweldings larger than 0,2-0,3 mm close to the surface of the welded joints decreases the life more than twice [1].To take into account the potential life decrease due to technological flaws it was assumed that n=2.

The estimate of root-mean square deviation σ_{lgN} for each stress level demonstrates that for the welded joints it is in the range of 0,133 to 0,156. To take into account the life scatter the design curve was assumed the one corresponding to the lower tolerant life boundary having probability level p=0,999 and confidence level γ =0,9 (Fig.2).

Fig.3 presents the results of analytical life estimate for integrally welded wing box made either of Ti alloy VT6ch or the advanced Ti alloy Vt23 depending on the level of the ultimate stresses. It is clear that the life of Vt6ch wing box is 35700 flights at the design stress level of 800 MPa that is 1,4 times larger than the required life of 25000 flights. For Vt23 wing box life increases till 52000 flights.

3 Analysis of crack critical length and its growth duration

A crack in the panel skin under the broken stringer has been considered to evaluate the crack critical length and its growth duration under different levels of ultimate stresses in the lower panel of the wing box made of VT6ch and Vt23 alloys. It was assumed that the ratio of stringer cross section area to the skin area was $F_{str}/F_{skin} = 0{,}5$, and stringer space was b = 130 mm. The analysis was based on linear fracture mechanics using automatic ALTAI system [2].

The relations between the critical half-lengths and ultimate stresses are presented in Fig.4. The relations demonstrate that the application of Ti alloy Vt23 enables to ensure the critical crack length equal to two bays between stringers $2l_c=2b=260$ mm under the ultimate stresses in the wing box lower panels equal to $\sigma_{ult}=760$ MPa. For Ti alloy Vt6ch these stresses do not exceed 610 MPa.

Crack growth time from minimum detectable for visual inspections crack length (100 mm) till the critical one related to the ultimate stresses is presented in Fig.5. The relations show that the crack in the lower panels of Vt23 wing box is growing for 1000 flights at ultimate stresses of 740 MPa. The crack in the lower panels of Vt6ch wing box is growing for 1000 flights at ultimate stresse of 615 MPa.

4 Life evaluation before inspections

To evaluate the life of the wing box structure before the inspection start the growth time of the fatigue crack due to the manufacturing flaw from the length of $2l_0$ till the critical length $2l_c$ has been analyzed. Crack growth in the wing panel made of Vt23 Ti-alloy has been investigated. The section of the rear spar has been studied having the reduced panel thickness of 8,5 mm. The calculations were performed by cycle-by-cycle method using the linear damage accumulation model. The following parameters from Foreman equation were used in the calculations: $C=7{,}51\cdot10^{-6}$, $n=2{,}27$, $K_c=890$ kg/mm$^{3/2}$. As the initial flaw some surface semi-round crack was assumed having the length $2l_0=0{,}3$ mm and depth $a=0{,}15$ mm. Such a flaw has simulated the surface pours or unweldings that may happen in the welded joints and that may result in the fatigue crack [1,3].

Fatigue crack growth curves from the initial length of $2l_0=0{,}3$ mm till the critical one for different levels of ultimate stresses are given in Fig.6. These data resulted in the crack growth duration from the flaw of $2l_0$ till the critical length $2l_c$ depending on the level of the ultimate stress σ_{ult} (Fig.7). It is obvious from the curves that the crack growth time till the visually detectable length is more than 95 per cent of its growth till the critical length. At the design stresses of 750 MPa the crack growth time till the critical length is 50000 flights and it is twice as large as the required value of 25000 flights.

5 Conclusions

For the structure of integrally welded wing box for SCT-2 made of Ti alloy Vt23 there are some potentials to ensure the required service life of 25000 flights at the level of ultimate stresses about 750 MPa. Thus:

- the critical length of the crack in the lower panel under the broken stringer is no less than two-bay inter-stringer distance;
- fatigue crack growth time from the visually detectable length (100 mm) till the critical one is about 1000 flights;
- fatigue crack growth time from the manufacturing flaw till the critical length is about 50000 flights.

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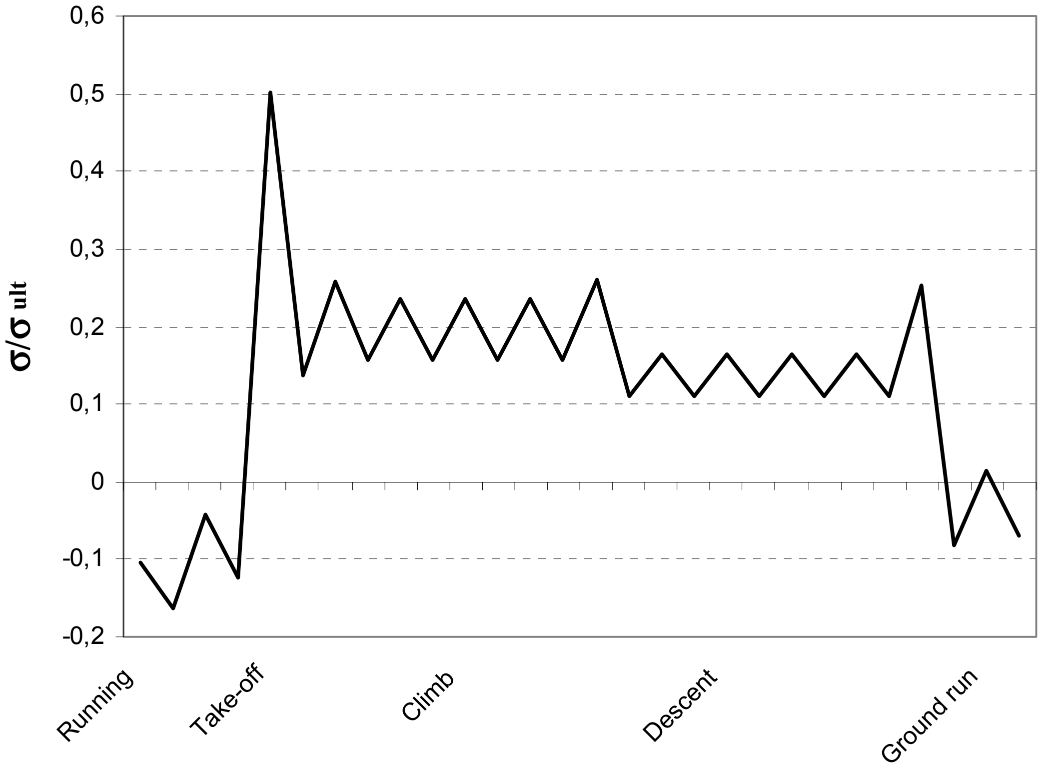

Fig.1 The curve of relative stresses σ/σ_{ult} on the wing box lower surface

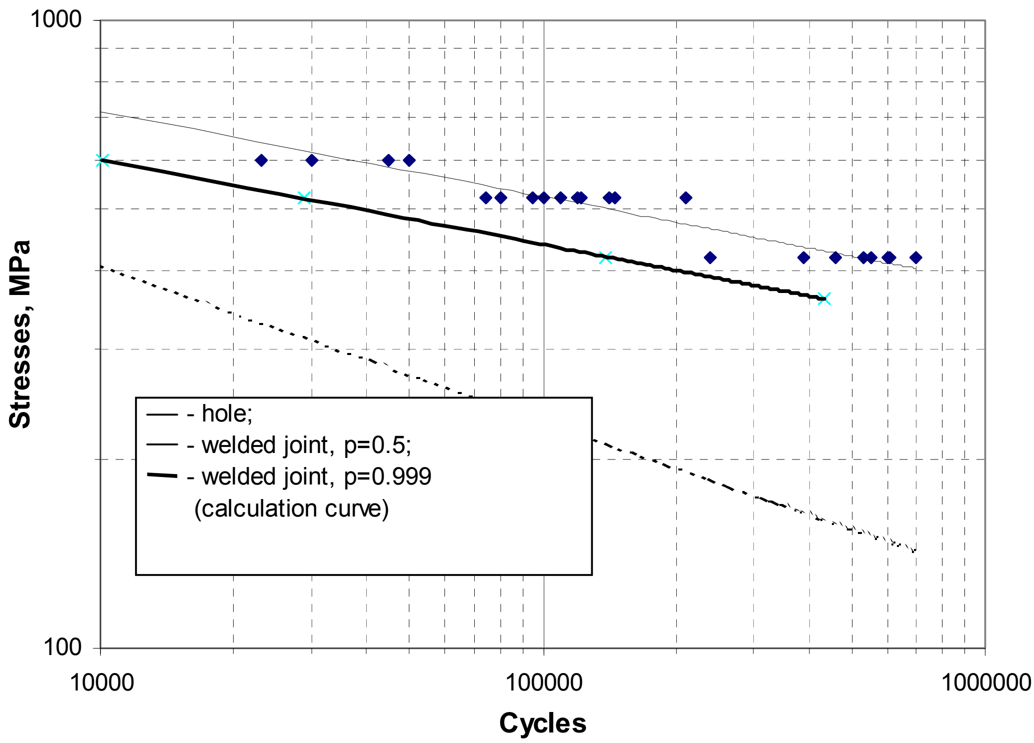

Fig. 2 Fatigue curves of welded joint of titanium alloy Vt6ch.

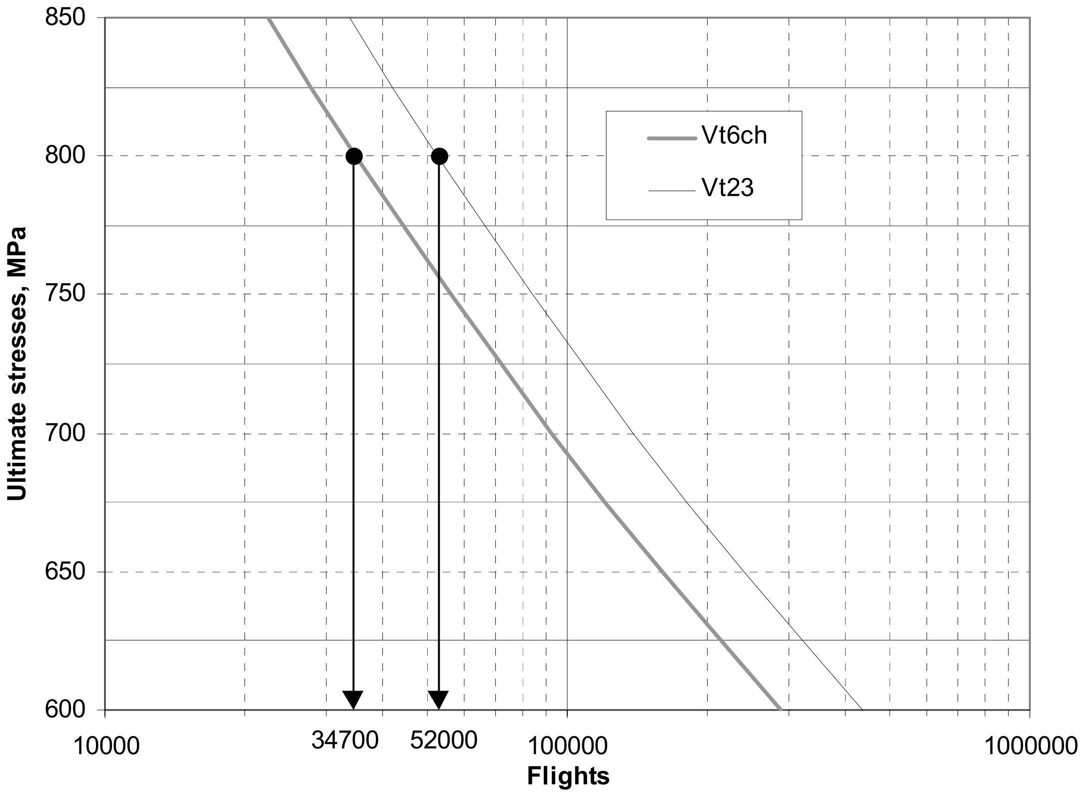

Fig.3 Service life depending on the ultimate stresses for integrally welded wing box made either of Ti alloy VT6ch or the advanced Ti alloy Vt23.

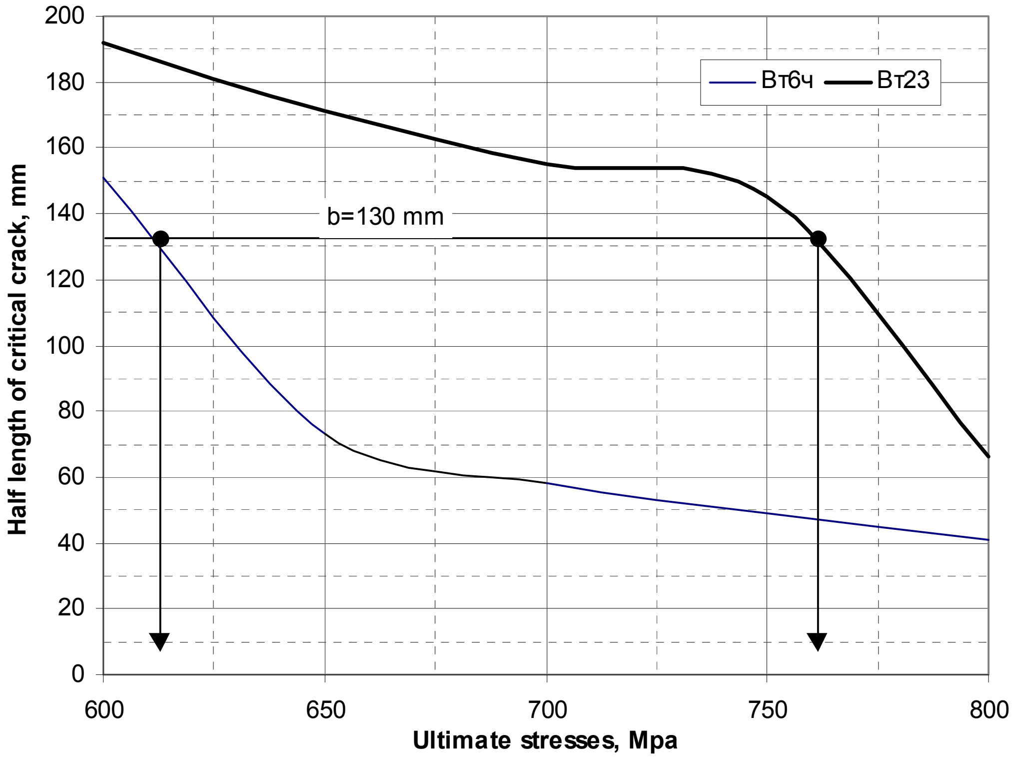

Fig.4 The relations between the critical half-lengths and ultimate stresses

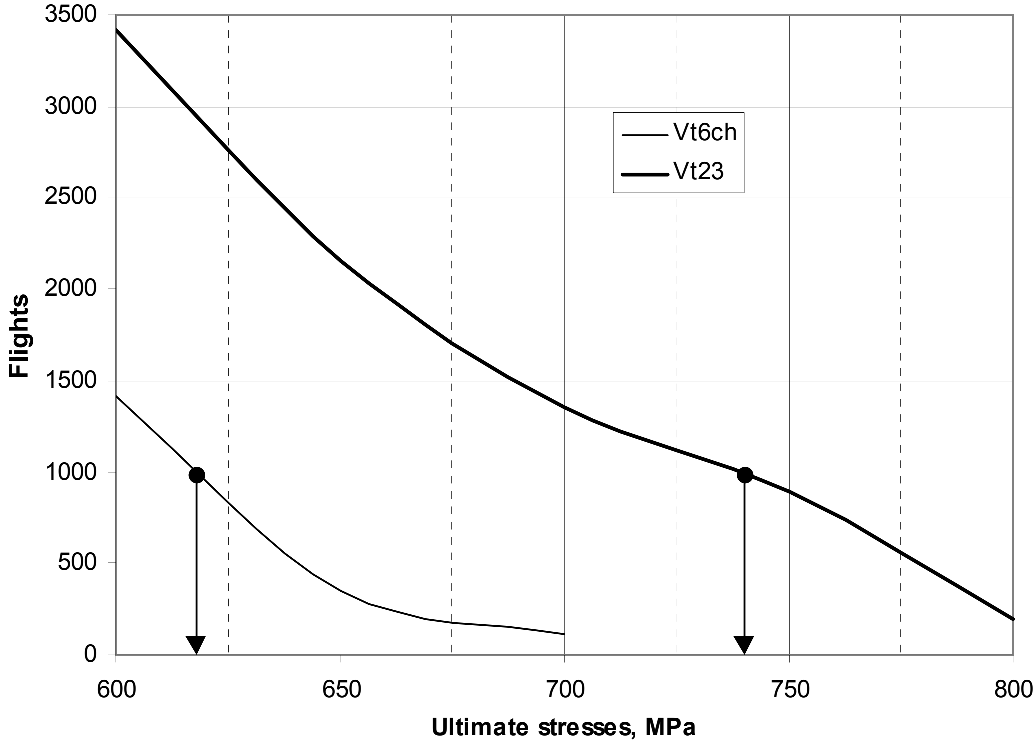

Fig.5 Crack growth time from minimum detectable for visual inspections crack length (100 mm) till the critical one related to the ultimate stresses.

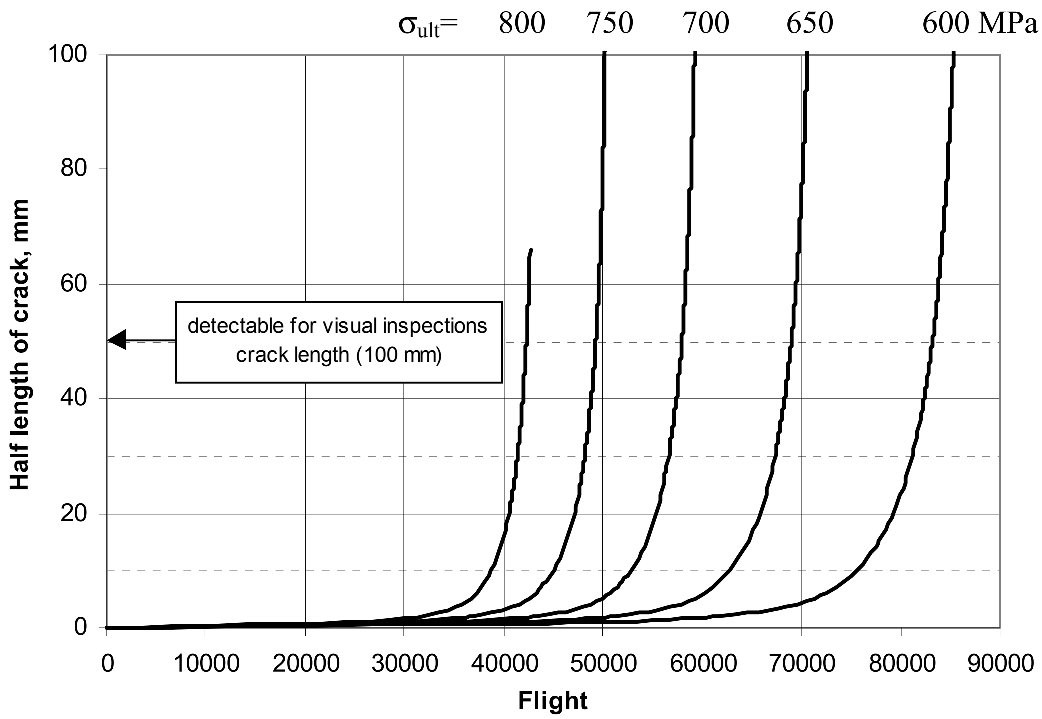

Fig.6 Crack growth from initial defect ($2l_o$=0,3 mm) till critical size.

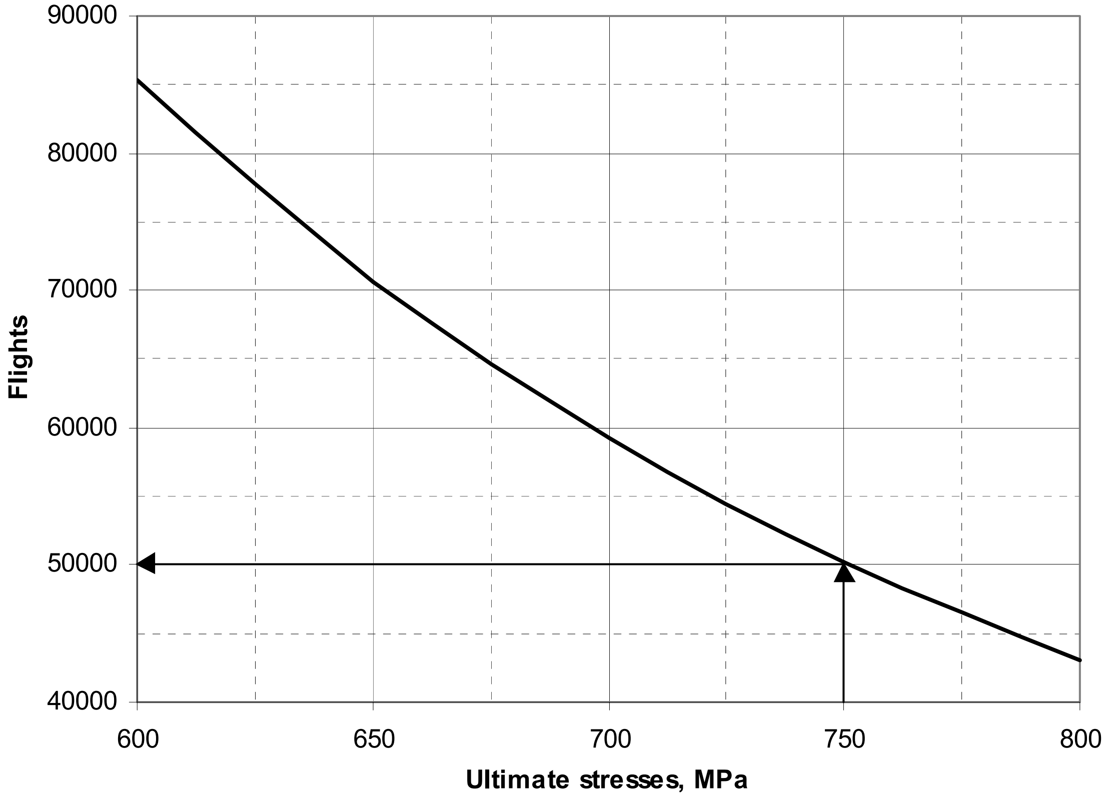

Fig.7 Crack growth time from initial defect length (2*l*o=0,3 mm) till the critical one related to the ultimate stresses.