INFLUENCE OF CORROSION ON THE REQUIRED NUMBER OF AN AIRFRAME INSPECTION

M. Shujauddin Wahab, Yu. M. Paramonov

Riga Technical University, Aviation Institute, Riga, Latvia
Lomonosova 1, Riga, LV-1019, Latvia
Tel.: (+371)-7089950, Fax: (+371)-7089990, E-mail: shujawahab@hotmail.com

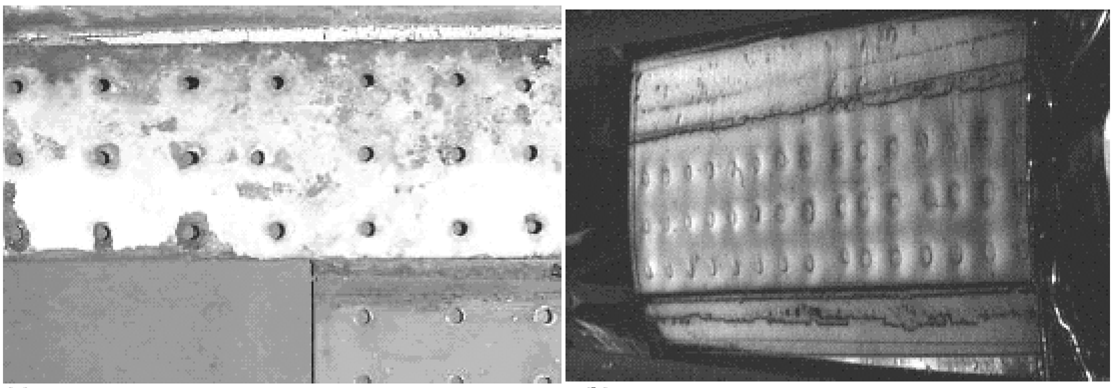
Abstract

The behavior of fuselage splice joints containing multi-site fatigue damage and corrosion is investigated. The objectives of this paper is to develop probabilistic analysis methodology that allows to calculate the probability of fatigue failure as function of specified interval between inspections and to make comparison of the cases when there is and there is not corrosion. As initial information the results of fatigue test of specimens of special type are used [1]. By the use of this information parameters of the stochastic damage growth model were estimated. The probability of failure at the specified number of inspections at the fixed specified life was calculated by the use of Monte Carlo method and related formulas. It was shown that if the specified life is very large then relatively large probability of failure is nearly the same for both cases when there is and there is not corrosion. But if the specified life is relatively small then corrosion increases the probability of failure at the same number of inspection very significantly.

Key words: Fatigue Crack Growth, Corrosion Medium, Failure Probability and Interval Between Inspections

1. Introduction

Fatigue crack growth analysis in the presence of corrosion is an important subject as shown in Figure1 because it can degrade the structural integrity and damage tolerance of fatigue critical structural components in aging aircrafts. Multiple site fatigue damage (MSD) in a longitudinal skin splice has been recognized as a major airworthiness problem. It had a very significant influence in Aloha B-737 incident in 1988.



(a) (b)

Figure 1. *Illustration of a corroded longitudinal fuselage splice from a retired 727: (a) white corrosion product on faying surface, (b) corrosion pillowing detected by D Sight*

For fleet management it is important to know the effects of corrosion in normal service on the durability and damage tolerance (DADT) characteristics of the fuselage. The DADT characteristic of any structure are defined by the crack initiation and growth patterns, the critical crack scenarios that could develop and the number of load cycles it takes for cracks to become detectable and then grow to a critical condition.

2. Test Program

The MSD concept is illustrated by the generic lap splice version of the specimen shown in Figure 2. A finite element model of the loop stress distribution in specimens is also shown. The concept is the use of bonded side straps to simulate the load transfer from cracked areas to surrounding structure that occurs on aircraft. The specimen shown is a 25.4 cm (10 in) wide version designed to be representative of the longitudinal fuselage splices in some narrow body transport aircraft. The splice in the generic specimen comprises two sheets of 1.0 mm (0.04 in) thick 2024-T3 Alclad held by three rows of 4 mm (5/32 in) diameter 20177-T4 rivets (MS20426AD5-5) without adhesive, paint or sealant. The rivet geometry results in a knife-edge countersink.

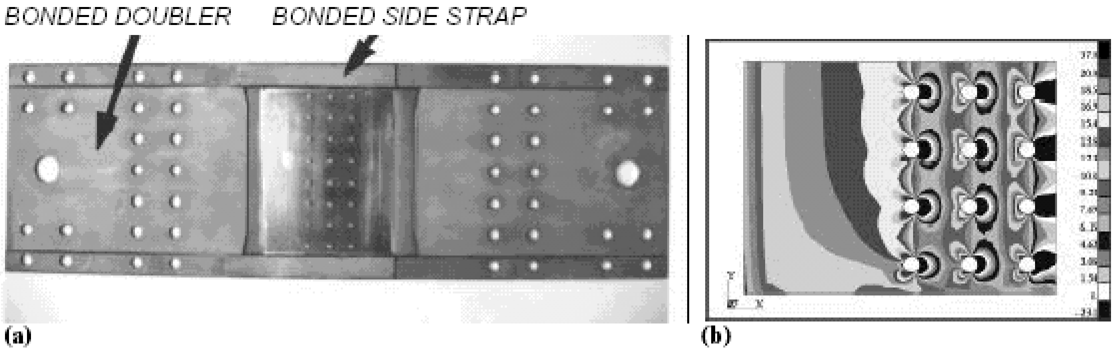


Figure 2. *Illustration of MSD specimen (a) bonded doubler, (b) with a hoop stress distribution at faying surface by finite element prediction*

The average cycle number for the final failure for the corroded specimens is 207640 cycles. As shown in Figure 3, the corrosion damage in this MSD specimen (average thickness loss of between 5% and 6%) was compared with the damage in a section of splice from a Boeing 727 aircraft, shown in Figure 1, which was naturally corroded to a comparable level during 48,665 flights over 24 years. So 1 flight is approximately equivalent to 4.266 cycles.

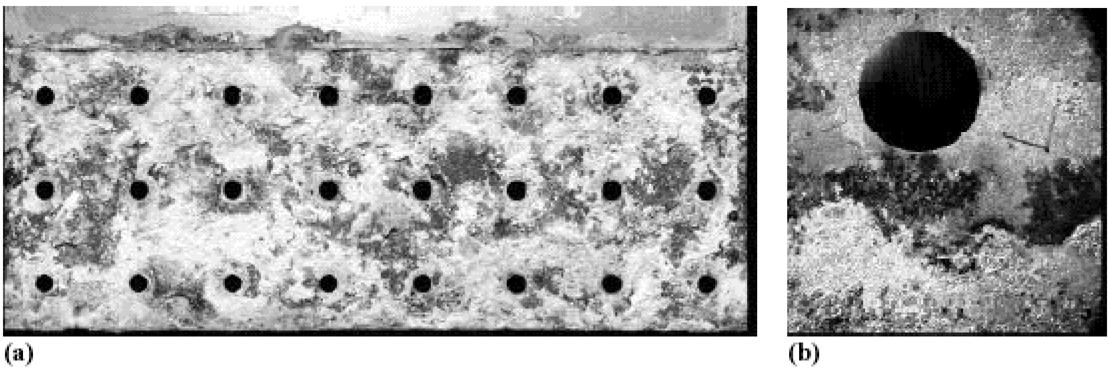


Figure 3. *MSD specimen corroded to 5% to 6% average thickness loss: (a) countersunk sheet with corrosion product still in place; (b) close-up near hole with corrosion product removed*

The combination of corrosion and fatigue assumes that corrosion/fatigue interactions occurs only in the context of pre-existing corrosion and in a dry splice. This is a reasonable approximation for two reasons. First, teardown of aircraft splices and evidence indicates that substantial corrosion often exists without any associated fatigue cracking. Second, the highest in-service loads occur when any moisture in the splice is likely to have frozen.

There are altogether nine MSD specimens out of which five are non-corroded and four are relatively heavily corroded. They all are fatigue tested. These specimens are listed in Table 1 along with their respective fatigue life at visible crack detection, first link up and final failure.

Table 1. Fatigue life of MSD Specimen

Specimen #		Fatigue Life (Cycles)		
		1st observed	1st Linkup	Final failure
Non-corroded	Cgc-f38	387500	491711	501933
	Cgc-f46	314000	398908	403718
	Cgc-f51	304001	381378	392591
	Cgc-f60	290000	368650	378754
	Cgc-f61	368500	473397	481353
			Average Final Failure	431670
Corroded to 5%-6% level	Cgc-cf34	160001		222450
	Cgc-cf43	144000		189074
	Cgc-cf45	104107		177129
	Cgc-cf58	142000		241909
			Average Final Failure	207640

3. Failure Characteristics

A significant difference was noticed in the behavior of the MSD specimen with and without corrosion. The visible crack were observed to start in different scenarios and there were distinct differences in load cycles to first observed cracks, which are shown in Table 1. The five non-corroded specimens showed visible cracks at between 2.9 and 3.88 * 10^5 cycles and failed at between 3.79 and 5.02 * 10^5 cycles. The statistical dispersion of visible crack detection and growth damage accumulation is large, which is a typical phenomenon of MSD specimens. The load cycles to visible crack detection of the non-corroded specimens represented 70% to 80% of their total fatigue life and similar behavior was observed in the corroded specimen. The observed reduction due to corrosion in the mean cycles to visible crack detection was 59% for the specimens corroded to the 5% to 6% level.

In non-corroded specimens the crack grew with increasing load cycles from the central holes outward forming a pattern of multi-site damage as shown in Figure 4. Changes in gross failure modes were observed in the corroded specimens with the 5% to 6% level. The two dominant failure modes in corroded specimen are: (i) non-uniform MSD – one crack developed from only one site – at the rivet locations in the upper row and (ii) fatigue cracking at one or more sites in the inner (driven) sheet 5.08 to 7.62 mm (0.2 to 0.3 in) below the lower rivet row.

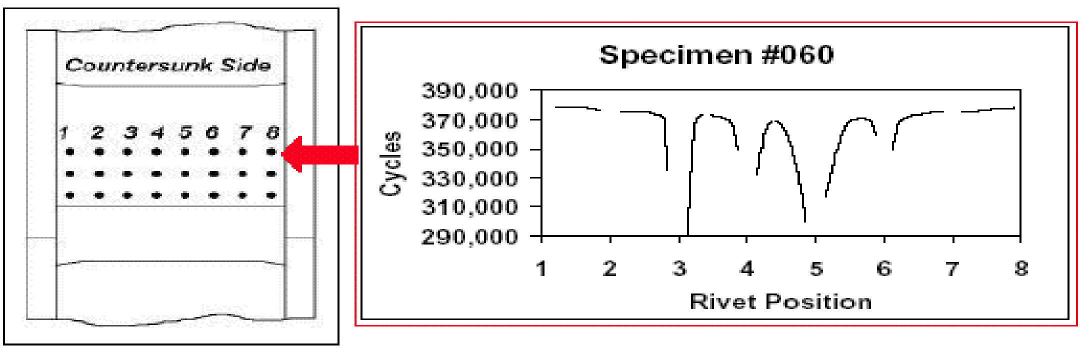


Figure 4. Typical MSD growth pattern in a non-corroded specimen

MSD tends to develop in clusters within the boundaries of a frame-bay. Similarly the linkup of MSD and the formation of a lead crack also tend to occur initially within a frame-bay for this curves that record the sum of all individual crack lengths at any given time. The crack length at a rivet hole is measured from the edge of the drilled hole. For cracks that developed away from the rivet rows, as in some corroded specimens the aggregate crack length is taken as the total tip to tip crack length. Where there were several such cracks in a specimen in an interacting MSD formation, overlapping cracks were regarded as linked cracks. The test data for the crack growth history of the two specimen groups are shown in Figure 5-6. In the non-corroded specimens, first linkup occurred at an aggregate crack length of about 50.8 mm (2 in). Subsequent crack growth was relatively fast and produced a pronounced knee in the growth curve. On the other hand in the corroded specimens the overall crack growth rate was relatively stable during the whole growth period similar to the growth progression of a single crack.

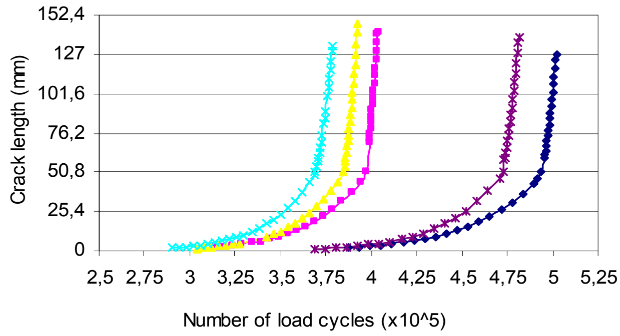


Figure 5. *Crack growth history data of non-corroded specimens*

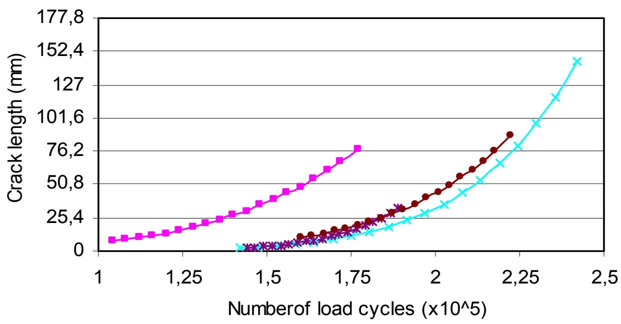


Figure 6. *Crack growth history data of corroded specimens at 5% - 6% level*

With above observation the total service life of a specimen is divided into two or three stages. For non-corroded specimens, the total fatigue life, N_t, is divided into three parts: life to visible cracks or visible damage starting life, N_s, growth life before linkup, N_{g1}, and growth life after linkup, N_{g2}, that follow $N_t = N_s + N_{g1} + N_{g2}$. For the corroded specimens to the 5% to 6% level, a single stage with growth life N_s, is used for the whole growth period because of their relatively stable growth behavior and the total fatigue life is $N_t = N_s + N_g$. The visible damage starting life is the number of load cycles at which the first crack was observed and the total life of a specimen is when the final failure occurred. The growth life is the difference between the total life and the damage starting life $N_g = N_t - N_s$.

In a modern transport aircraft, the critical length of a single longitudinal crack in a fuselage skin is typically in excess of the frame bays about 10.16 cm (40 in). Crack growth rates are high when a lead crack reaches a length of several inches. The presence of MSD in adjacent frame-bays could reduce the critical length of the lead crack. Therefore first the splice is considered to have failed when the first linkup occurs at which the length of the aggregate lead crack $\mathbf{a}$, reaches a specific value $\mathbf{a_{lk}}$. Second the splice is considered to have failed when the aggregate crack length reaches a critical value $\mathbf{a_{cr}}$. The specific crack length for linkup and the critical crack length for final failure are taken from the mean values of the crack length obtained fro the test data corresponding to the linkup and failure life, respectively. For corroded specimen, only the final failure is considered.

4. Damage Starting Life and Stochastic Growth Model

4.1. Curve-Fitting of Test Data

The test data are fitted by the help of growth function, expressed as:

$$a = C_1 * N^{C_2} \tag{1}$$

Where **a** is the aggregate crack length, **N** is the number of load cycles, $\mathbf{C_1}$ and $\mathbf{C_2}$ are two constants taken from the Table 2.

Table 2. Constants in fitted growth curves

Specimen #		Growth stage 1		Growth stage 2	
		C_1	C_2	C_1	C_2
Non-corroded	Cgc-f38	1.39e-9	13.18	1.90e-40	57.69
	Cgc-f46	2.16e-8	13.29	1.21e-35	58.84
	Cgc-f51	5.59e-9	14.62	1.97e-30	51.30
	Cgc-f60	2.48e-8	13.93	6.60e-22	37.86
	Cgc-f61	1.70e-10	14.92	4.52e-35	51.40
Corroded to 5% - 6% level	Cgc-cf34	0.0194	6.48		
	Cgc-cf43	0.0028	9.65		
	Cgc-cf45	0.2412	4.44		
	Cgc-cf58	0.0057	7.82		

4.2. Determination of Fatigue Crack Growth Function Parameters

It is assumed that fatigue crack rate of some items of airframe is defined by Paris's formula [2]:

$$\frac{da}{dt} = C\left(\Delta K\right)^m = Q * a^{m/2} \tag{2}$$

a – size of a crack

t – service time

Where

$$Q = C\left(\lambda\left(\sigma_{\max} - \sigma_{\min}\right)\sqrt{\pi}\right)^m \tag{3}$$

C,m – crack growth function parameters

λ - takes into account the width of the panel, influence of stringers

σ_{max} , σ_{min} – maximum and minimum stress in a flight

The solution of this differential equation for m≠2 is:

$$a(t) = a(o) \Big/ \left(1 - \mu\left(a(o)\right)^{\mu} Qt\right)^{1/\mu} \tag{4}$$

Where, $\mu = m/2 - 1$ (5)

μ – depends on the material characteristics

$$a(o) = \left(1 \Big/ \left(\mu Q t_n + a_n^{-\mu}\right)\right)^{1/\mu} \tag{6}$$

a(0) – equivalent beginning size of a crack

Residual strength

$$\sigma_p(t) = K_c \big/ \lambda\sqrt{\pi a(t)} \tag{7}$$

K_c – critical value of stress intensity factor

Lets investigate the results of data processing for crack growth during fatigue experiments. Transferring to logarithm scale and putting new designation, we will accept following form of description of crack growth speed.

$$y = \ln(da/dt) = \ln Q + m/2 * \ln a = b_0 + b_1 * x \tag{8}$$

Where, $b_0 = \ln Q$, $b_1 = m/2$ (9)

Then sequence $\{(a_i, t_i), I=1,\ldots, n\}$ transmits to sequence (y_i , x_i), $I = \{1,\ldots,n-1\}$, where:

$$x_i = \ln((a_{i+1} + a_i)/2); \tag{10}$$

$$y_i = \ln(\Delta a/\Delta t)_I = \ln((a_{i+1} - a_i)/(t_{i+1} - t_i)), \tag{11}$$

Parameters b_0, b_1 can be found using least – squares method by formulas:

$$b_1 = \frac{\left(\overline{xy} - \bar{x}\cdot\bar{y}\right)}{\left(\overline{x}^2 - \left(\overline{x}\right)^2\right)} \tag{12}$$

$$b_1 = y - b_1 * x \tag{13}$$

Where: (x_i,y_i), $I=\{1,..,n\}$

Using the evaluation of parameters b_0,b_1 can be found estimations:

$$Q = \exp(b_0),\ m = 2b_1,\ C = Q/(\lambda(\sigma_{max} - \sigma_{min}) * \sqrt{\pi})m \tag{14}$$

After finding the estimation parameters Q, m, a(0) for both the non-corroded and corroded specimens, we get their statistical analysis with the help of Excel.

Examples of Relevant curves a(t) and $\sigma_p(t)$ are shown in Figures 7-8.

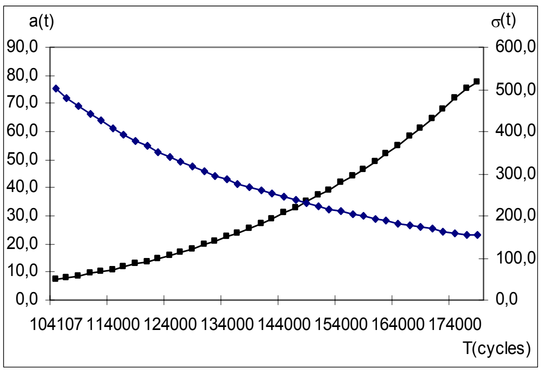


Figure 7. *Fatigue crack growth and loss of residual strength (FCG&RS) for corroded specimen.*

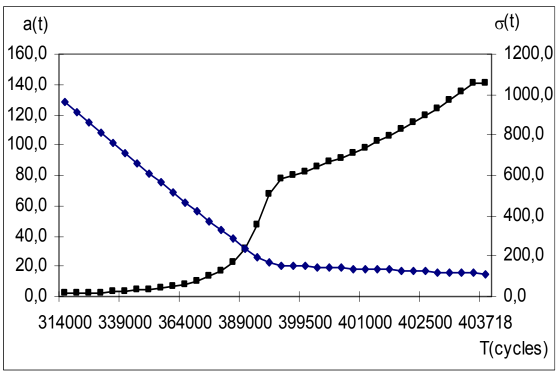


Figure 8. *Fatigue crack growth and loss of of residual strength (FCG&RS) for non-corroded specimen.*

Results of processing four fatigue crack growth data for corroded specimens at 5%-6% level are given in Table 3.

Table 3. Fatigue crack growth parameters for corroded specimens at 5%-6% level

Serial No.	Specimen #	μ	b_o = ln Q	a(0)
1.	Cgc-cf34	-0.15424	-9.75393	2.37E-19
2.	Cgc-cf43	-0.10353	-9.52048	1.48E-27
3.	Cgc-cf45	-0.22511	-9.61464	4.51E-13
4.	Cgc-cf58	-0.12758	-9.70537	1.63E-18
Average		-0.15262	-9.6486	1.13E-13
Standard Deviation		0.052581	0.103096	2.26E-13

Results of processing five fatigue crack growth data for non-corroded specimens are given in Table 4.

Table 4. Fatigue crack growth parameters for non-corroded specimens

Serial No.	Specimen #	μ	b_o = ln Q	a(0)
1.	Cgc-f38	0.432273	-11.2323	0.079004
2.	Cgc-f46	0.44934	-11.0738	0.091935
3.	Cgc-f51	0.334914	-10.6355	0.02707
4.	Cgc-f60	0.249426	-10.5026	0.014186
5.	Cgc-f61	0.322694	-10.7252	0.018577
Average		0.35773	-10.8339	0.046155
Standard Deviation		0.082805	0.307091	0.036475

4.3. Simulation of a Process of Fatigue Crack Inspection

It is assumed that some inspection technology is characterized by two values: a_d and w_i; a_d - the minimum size of a detectable crack and w – is interpreted as probability that the earlier scheduled inspection will be made with required accuracy. Service time when crack becomes detectable t_d and service time to fatigue failure t_f are defined below:

$$t_d = \frac{C_d}{Q} \qquad t_f = \frac{C_f}{Q} \tag{15}$$

we consider, that t_d and t_f are functions of random variable Q.

C_d – constant for both non-corroded and corroded specimens.

$$C_d = \frac{1-\left(\frac{a(o)}{a_d}\right)^{\mu}}{\mu\left(a(o)\right)^{\mu}} \tag{16}$$

where a_d is equal to 20 mm.
C_f – constant for non-corroded specimens

$$C_f = \frac{1-\left(\frac{a(o)}{\frac{K_c{}^2}{\sigma_{\max}{}^2\pi}}\right)^{\mu}}{\mu a(o)^{\mu}} \qquad \mu \text{ - positive} \tag{17}$$

C_f – constant for corroded specimens

$$C_f = \left[\frac{\left(\left(K_c/\sigma_{\max}\right)^2 * 1/\pi\right)^{\gamma} - \left(a(o)\right)^{\gamma}}{\gamma}\right] \qquad \mu \text{ - negative.} \tag{18}$$

Bar chart of crack undetectable and crack detectable time periods (CUCDTP) in both cases are shown in Figures 9-10.

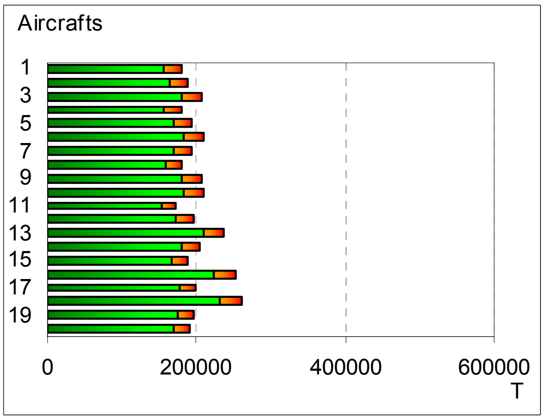


Figure 9. *Bar chart of CUCDTP for the corroded specimens at 5% and 6% level.*

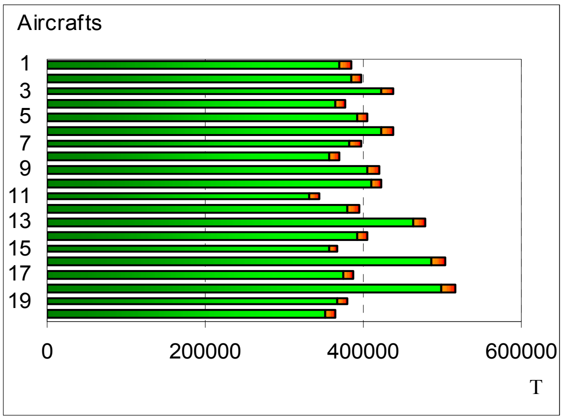


Figure 10. *Bar chart of CUCDTP for the non-corroded specimens.*

4.4. Interval between Inspection and Estimation Fatigue Failure Probability

$$\Delta = \frac{t_{SL}}{(n+1)} \tag{19}$$

where t_{SL} – specified life of an aircraft
n – number of inspections.

If we use Monte Carlo method then the failure probability in the interval $(t_n, t_p)_j$ with r_j inspections on the j-th airplane is defined by formula:

$$\hat{p}_{f1j} = (1-w)^{r_j} \tag{20}$$

w – is a probability that planned inspection will be made with required accuracy

Estimation of mean P_f for N airplanes:

$$\hat{p}_f = \frac{1}{N}\sum_{j=1}^{N} P_{f1j} \tag{21}$$

Relevant curves $P_f = P_f(\Delta)$ are shown in Figures 11-14

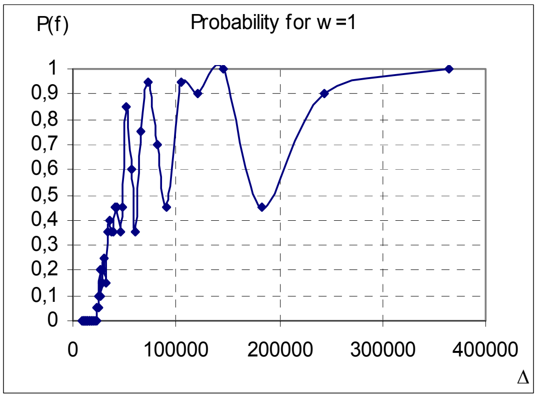


Figure 11. *Failure probability for the corroded speciemn with specified life = 730000 cycles*

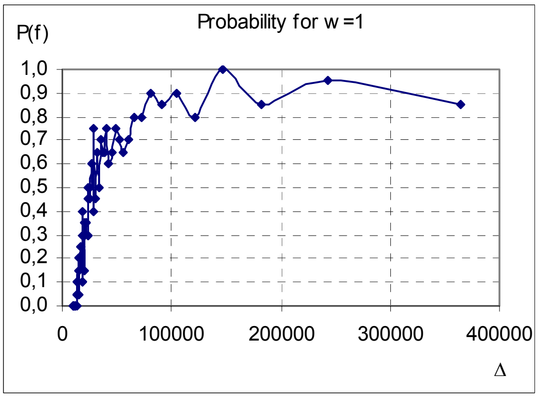


Figure 12. *Failure probability for the non-corroded speciemn with specified life = 730000 cycles*

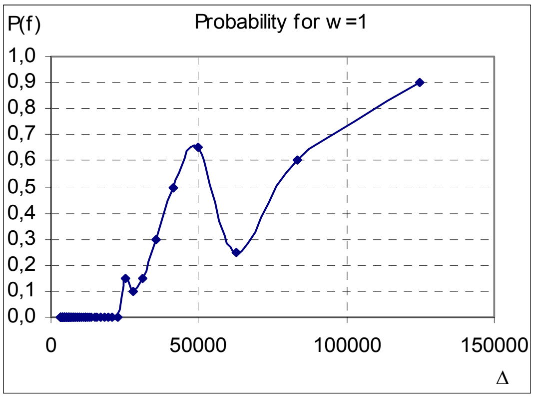


Figure 13. *Failure probability for the corroded speciemn with specified life = 250000 cycles*

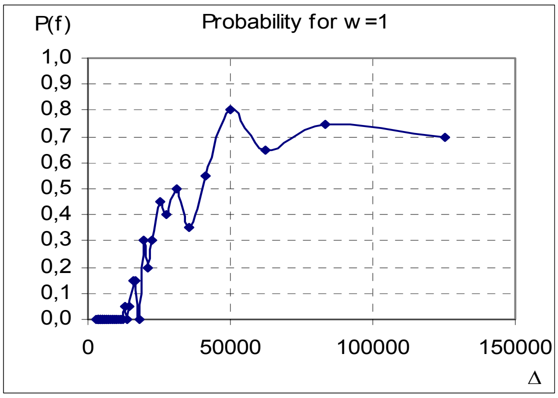


Figure 14. *Failure probability for the non-corroded speciemn with specified life = 250000 cycles*

Failure can happen in any interval $(0,t_1)$, (t_1, t_2), …, (t_{n-1}, t_n), (t_n, t_{n+1}). The probability that the visual fatigue crack appears in the interval and conditional probability that simultaneously the critical time T_f will be less than t_i [3]:

$$p_i^* = P\left(\{t_{i-1} < T_d < t_i\} \cap \{t_{i-1} < T_f < t_i\}\right) = \tag{22}$$

$$p_i^* = P\left(\left\{t_{i-1} < \frac{C_d}{Q} < t_i\right\} \cap \left\{t_{i-1} < \frac{C_f}{Q} < t_i\right\}\right) =$$

$$p_i^* = P\left(\max\left(\frac{C_f}{t_i}, \frac{C_d}{t_i}\right) < Q < \min\left(\frac{C_d}{t_{i-1}}, \frac{C_f}{t_{i-1}}\right)\right) = \tag{23}$$

$$p_i^* = P\left(\frac{C_f}{t_i} < Q < \frac{C_d}{t_{i-1}}\right), \text{ if } \frac{C_d}{t_{i-1}} \le \frac{C_f}{t_i}.$$

So in this case

$$p_i^* = \begin{cases} 0, if \dfrac{C_d}{t_{i-1}} \le \dfrac{C_f}{t_i} \\ \pi_i, if \dfrac{C_d}{t_{i-1}} > \dfrac{C_f}{t_i} \end{cases} \tag{24}$$

$$\pi_i = F_Q\left(\frac{C_d}{t_{i-1}}\right) - F_Q\frac{C_f}{t_i} = \tag{25}$$

$$\pi_i = \phi\left(\frac{\ln(C_d/t_{i-1}) - \theta_0}{\theta_1}\right) - \phi\left(\frac{\ln(C_f/t_i) - \theta_0}{\theta_1}\right)$$

In which θ_0, θ_1 are mean and standard deviation of lnQ. We suppose the normal distribution of lnQ. If we assume that the α is random variable also and use the denotation P_i^* as a function of α then in order to get the mean value of P_i^* we have to do integration.

$$p_i^* = \int_0^\infty p_i^*(X) dF_\alpha(X) \tag{26}$$

And finally the failure probability is defined in the following way:

$$p^* = \sum_{i=1}^{n} p_i^* \tag{27}$$

After we have got function $P_f = P_f(n)$, we can choose n (number of inspections) for allowable failure probability as shown in Figure 15-18.

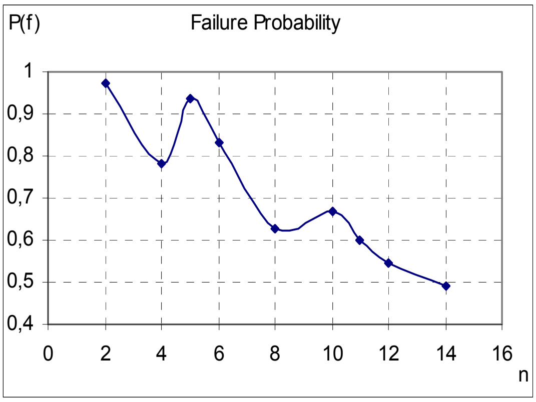


Figure 15. *Failure probability for the corroded speciemn with specified life = 730000 cycles*

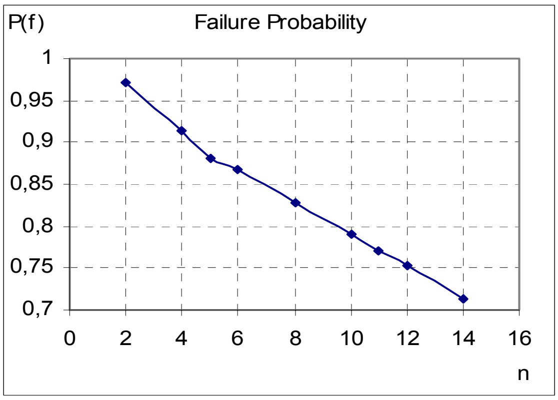


Figure 16*. Failure probability for the non-corroded speciemn with specified life = 730000 cycles*

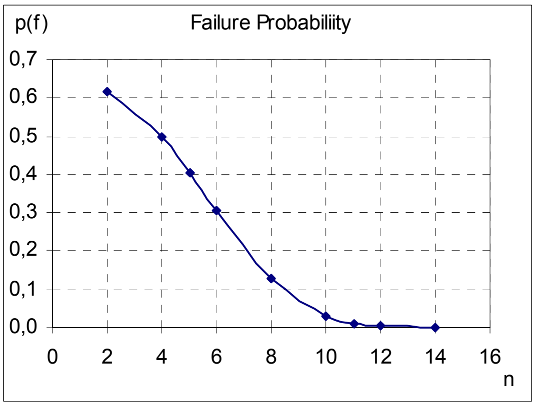


Figure 17. *Failure probability for the corroded speciemn with specified life = 250000 cycles*

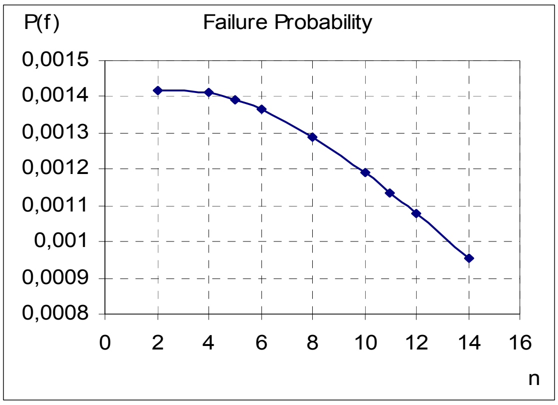


Figure 18*. Failure probability for the non-corroded speciemn with specified life = 250000 cycles*

In the following Table 5., we observe the probability of failure (Pf) for different specified lives (SL). We see that for very large SL (171120 flights) 14 inspections (n=14) give comparable value of Pf and in this case there are not significant difference between corroded and non-corroded specimens. But the difference is very significant for relatively small SL (58603 flights): for 8 inspections Pf is equal to 0.127 for corroded and Pf is equal to 0.00129 for non-corroded specimens.

Table 5. Influence of corrosion on the required number of inspections

Cycles	Flights	Corroded Specimens		Non – Corroded Specimens	
		P_f	n	P_f	n
730000	171120	0.49	14	0.71	14
250000	58603	0.127	8	0.00129	8

5. Conclusions

1. If the specified life is 730000 cycles (171120 flights) then probability of failure is unacceptably large for both cases when there is and there is not corrosion even for inspection number n=14.

2. If the specified life is relatively small that is 250000 cycles (58603 flights) than corrosion increases the probability of failure at the same number of inspections very significantly.

References

1. Dr. Y. Xiong, Mr. G. Eastaugh and Dr. G. Shi, Probabilistic failure analysis of fuselage splice joints with multiple site fatigue damage and corrosion, www.nrc.ca/iar/pdfs/aiaaj99_paper.pdf.
2. Ю. М. Парамонов и В. П. Кузнецов, Живучесть авиационных конструкций. Оценка эффективности эксплуатации по состоянию, МУ по выполнению самостоятельной работы, Рига, 1990, 24с.
3. Yu. M. Paramonov and R. Bharati, Mathematical Models of the Process of Airframe Fatigue Problem Solution for the purpose of Automated System Development, Transactions of RAU, Mechanical Development, 1997, 123-126.