Damage Tolerance Design

Damage Tolerance Design for Wing Components – Procedure Standardization

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ABSTRACT: Fatigue analysis of mechanical components, in many cases, leads to underestimated lives and thus greater costs, so a different philosophy was developed, Damage Tolerance Design. Using this theory, the designer no longer assumes a perfect component but rather the existence of an initial damage that is allowed to propagate. However, that damage is detected and repaired within the safety limits placed. The purpose of this paper will be the definition and standardization of procedures to be followed by an aircraft contractor, OGMA, in order to implement a Damage Tolerant Design for the components. Once the procedure is approved by the Aeronautical Authorities it will have significant impact on costs reduction, without compromising the safety of the repairs. The paper is mostly focused towards wing structures, particularly riveted joints. To do so, stress concentration factors, residual strength determination, crack growth analysis and inspection intervals must be correctly defined, determined and analyzed. Furthermore, the initial damage on the structure must also be correctly assumed, in terms of shape, size, direction and quantity. Computational methods for determining some of these parameters are mandatory.

Keywords: Damage Tolerance, Crack Propagation, Riveted Joint, Inspection Chart.

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1. Theoretical Background

Since the end of the Second World War, with the increasing operational life of the aircrafts, fatigue related failures started to occur. Therefore, methods to prevent them where developed. The first one was Safe Life that only takes into account fatigue life issues, thus it has severe economic implications [1].

In order to obtain a safe, but economically viable mechanical component a different approach was needed, and thus in the 70's, Damage Tolerance (DT) was created. It is usually described as "*the ability of aircraft structure to sustain anticipated loads in the presence of fatigue, corrosion or accidental damage until such damage is detected through inspections or malfunctions and repaired*" [2].

As so, using DT, the structural engineer no longer assumes a perfect structural part, like for a safe life component, but rather assumes that the new part already has a defect that will eventually evolve leading to the catastrophic failure of the component [3].

This theory might seem too conservative, but analysis of in service fractures and cracking instances have indicated that manufacturing errors are a major source of initial damage. As so, the consideration of initial damage in the form of cracks or equivalent damage is absolutely necessary to ensure safety [3].

In order to fully understand damage tolerance principles, it is important the comprehension of some subjects, mostly related to Fatigue and Fracture Mechanics.

In the field of the Linear Elastic Fracture Mechanics, it is important to define the stress intensity factor for stress mode I, K_I, the most critical. It is an essentially elastic concept that gives an indication of the stresses intensity and severity near the crack's tip. This factor and its variation are used to describe fatigue crack growth resistance of materials [4].

$$K_I = \beta\sigma\sqrt{\pi a} \quad (1)$$

Where a defines the crack length, σ the stress, and β the geometrical shape factor.

Also the limiting value, K_{IC} is a defining property of the material, occurring when the crack reaches its critical size, known as fracture toughness [4].

Fatigue is the primary collapse reason for structural components, as so it's study and comprehension is of the most importance [1].

Fatigue is the consequence of cyclic loads, and has two important and distinct phases, crack initiation and crack growth [5].

In order to predict the fatigue life of a material, a diagram must be determined, called S-N curve or Wöhler curve. The line drawn in it indicates the critical stress level and number of cycles for a material to fail due to fatigue effects.

The Miner's Rule is the most common method to determine the damage that a component can withstand, due to fatigue, because of its simplicity and good results. It is defined knowing the fatigue limits of a material stipulated by the S-N curve. It states that a component's capability to absorb damage is cumulative, as so expression 2 defines the rule, where n is the number of cycles carried out by the component and N the maximum number of cycles the component can withstand:

$$\sum \frac{n_i}{N_i} = 1 \quad (2)$$

For design purposes this sum is often chose as 1, but its value can vary from 0.61 to 1.45 [5].

In the study of the fatigue, it is also very important to clearly define and determine the crack growth rate (*da/dN*). Paris, [5], noticed a relationship between the crack growth rate and the stress intensity factor. Thus, several crack growth laws where developed.

The Paris Law is given by [5]:

$$\frac{da}{dN} = C\,\Delta K^m \quad (3)$$

Forman proposed a different solution that accounts for the stress ratio, R, and fracture toughness [5]:

$$\frac{da}{dN} = \frac{C_F\,\Delta K^{m_F}}{(1-R)K_{IC} - \Delta K} \quad (4)$$

C and m are material constants, which are different for each growth law, even if they cover the same data set for the material [4].

More complex laws exist, but are mostly used in computational methods for the determination of this parameter [6].

It is also important to notice that all of these formulas are built in order to adapt to trends, and have no physical value, being experimental testing the only way to accurately determine stress intensity factor versus crack growth rate [5].

2. Stress Concentration

Mechanical components should maintain constant section, or its alteration should be very smooth, otherwise stress concentration will occur. Stress concentration greatly promotes fatigue crack growth [5].

The stress concentration factor is an essentially elastic and dimensionless parameter that relates the nominal tension applied and the local tension:

$$K_t = \frac{\sigma_{local}}{\sigma_{nominal}} \tag{5}$$

Fatigue in fastener holes is responsible for the greater part of aircrafts' components failures. Therefore, stress concentration determination is very important in joints [7].

There are two important types of components that would need to be riveted to an aircraft skin panel, splices and doublers (Figure 1). Splices allow load transfer between components, as doublers pick up load, in order to relieve stresses on other components [8].

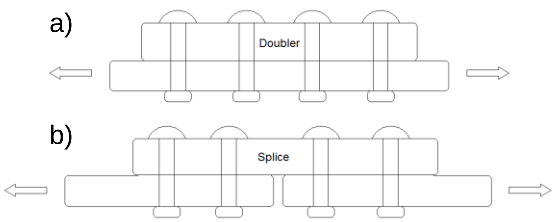

Figure 1 – a) Doublers; b) Splices

Thus, for these structures two types of loads exist. Transfer loads, which are loads transmitted through the fastener, and bypass loads that are, on the other hand, the loads still carried by the skin after a row of fasteners.

In order to determine the stress concentration factor for riveted joints there are two important methods: an adaptation of the FEM and a correlation developed by Tom Swift [1] [9].

The correlation method relates test data and equations and is proven to have good results. It models both fasteners and panels with springs, and through a displacement analysis the bypass and transfer loads are determined [1].

Swift proposed equation 6 to describe the fastener spring constant:

$$C = \frac{E_{plate}D}{\left[A + B\left(\frac{D}{t_{skin}} + \frac{D}{t_{splice/doubler}}\right)\right]} \tag{6}$$

$$for \begin{cases} steel\ fastener \quad A = 1{,}666;\ \ B = 0{,}86 \\ aluminum\ fastener \quad A = 5;\ \ B = 0{,}8 \end{cases}$$

The formulas are applied to a splice or doubler of thickness *t*, width *w*, with fastener holes of diameter *D* and Young Modulus *E*.

For the panels, using the normal force equation, where *A* is the cross sectional area, the spring constant is determined through and *L* the length of the plate:

$$K = \frac{AE_{panel}}{L_{plate}} \tag{7}$$

Knowing the fastener constant, *C*, and using for the plate springs, *K*, it is possible to build a matrix that relates fastener and panel displacement, where the only unknowns are the transfer loads in each fastener. [1]

Knowing the transfer loads (*ΔP*), the bypass load (*P*) is easy to determine, and thus the maximum stress can be determined using [8]:

$$\sigma_{max} = \frac{K_{tb}\theta\Delta P}{Dt} + \frac{K_{tg}P}{wt} \qquad (8)$$

K_{tb} represents the stress concentration factor for bearing stress, K_{tg} the stress concentration factor for bypass gross area stress and θ the bearing distribution factor, and they can be obtained by graphics or formulas, through the geometrical data of the plates and hole

The determination of the stress concentration factor, using a FEM approach, is made by modelling the fastener as a circular beam, cantilevered in both extremities, with a diameter given by [1]:

$$D_{model} = \sqrt[4]{\frac{64L_{fast}^3}{12E_{fast}C\pi}} \qquad (9)$$

The fastener constant, C, is determined using Swift's equation. The construction of the finite element model is simple (Figure 2). It is common to assume that the length of the fastener (L_{fast}) is unitary to decrease the calculation time.

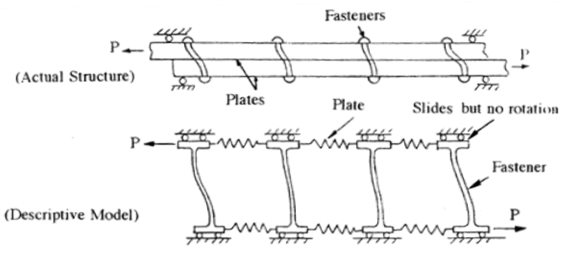

Figure 2 – Finite Element Model for Splices [9]

The FEM software returns the transfer loads, and so, a similar procedure to determine the maximum stress is used.

Once the maximum stress is determined, the stress concentration factor is calculated through its usual formula, a ratio between the maximum local stress and the nominal.

3. Initial Damage Characterization

Damage Tolerance states that a structure enters service with an initial damage. These are assumed to be located in the most critical area, in the most critical quantity, and in the most critical direction (according to the load applied).

The length of such damage is assumed according to its location on the structure. The values can be consulted in tables [3].

Figures 3 and 4 illustrate the shape and length of damages to be assumed near holes and free surfaces, for panel thicknesses superior to 0.05 in and 0.125 in, respectively. If the panel thickness is inferior to this value, a through crack should be assumed, with the same length.

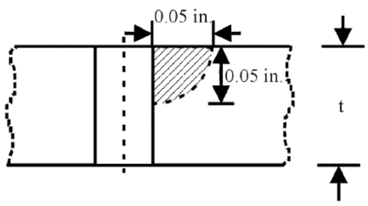

Figure 3 – Initial Damaged near Holes

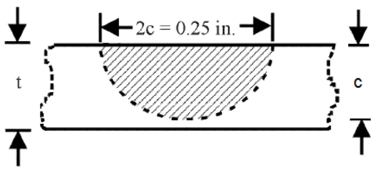

Figure 4 – Initial Damaged in Free Surfaces

After manufacturing Non-Destructive Inspection (NDI) procedures, like visual inspection, liquid penetrant, and Eddy current, among others are conducted. Still the initial damage is not likely to be found, once it is too small.

It's important to notice that for holes the cracks usually begin in the faying surfaces, which means that this surface condition, along with rivet hole quality are very important [10].

Initial damage is always assumed to grow in the worst possible direction, usually perpendicular to the load applied, for mode I, which is the most critical.

Furthermore, studies to determine the worst disposition of the damage around a hole must be made. Still, the most common situation is two cracks growing on opposite sides of a hole.

The quantity of damage that a structure is assumed to have is also an important issue. Multiple Site Damage (MSD) leads to a significant reduction of the fatigue life of a component.

For multiple holes, if the holes are 'new', one crack is introduced for every 10 holes drilled, which is a conservative approach, based on experience. It accounts for accidental manufacture errors and malpractice. If the holes are 'old', cracks should be assumed in every hole, with the length prescribed in the tables for initial damage.

4. Load Spectrum

The loads applied to an in-service component usually have a significant variation through time, as so they are referred to as spectrum.

Spectrums for wings have a rather chaotic shape, due to the effects of turbulence and gusts.

There are two types of load spectrums: either from data collected from an actual aircraft or through computer algorithms. For this document purposes, the spectrums incorporated in the AFGROW software were considered – TWIST and FALSTAFF.

TWIST simulates the loads to which the lower panel of the wing of transport aircraft is subjected. FALSTAFF simulates the loads in the same location, but for a fighter [11] [12].

As mentioned, the spectrums are chaotic, thus to use and compare them a normalization is required. The normalized spectrum will have a sinusoidal shape of constant amplitude.

Using the initial spectrums a medium of the peaks and of the valleys is determined. These two mediums will set the maximum and minimum stresses to be used, in order to create the normalized spectrum. Through them, the medium (σ_m) and alternating stresses (σ_a), for the normalized spectrum, can be determined.

$$\sigma_m = \frac{\sigma_{max}+\sigma_{min}}{2} \qquad \sigma_a = \frac{\sigma_{max}-\sigma_{min}}{2} \qquad (10)\ (11)$$

5. Residual Strength

The existence of cracks has a great influence on the ability of a structure to transfer loads, and as long as the crack grows this capacity starts to decrease. Thus, the concept of residual strength was defined as the limiting load transfer capability of the structure.

Residual strength requirements for a structure are defined in the Airworthiness Requirements, such as FAR-25 and CS-25. CS-25 states much tighter standards for gust pairs and turbulence, and thus should be used in order to be conservative.

Still, the requirements don't provide formulas for residual strength determination, but rather a group of parameters that allow its calculation, like load factors or gust velocities.

As so, to determine the residual strength for wings some simplifications were made:

- Wing section modelled as a box section idealized with 10 booms, with area *B* (still beam taper effects were considered);
- Wing modelled as a cantilevered beam with a load applied at half the semi span.

If the wing's mass is known, it is possible to transform the load factor into a force, *S*, through Newton's Second Law, which will be applied in the beam cross section.

Using the CS-25 requirements, the distance penetrated in a gust must be known. Through this and the cruise speed, the time duration of the gust may be determined and then the acceleration induced by said gust. As so, a similar reasoning, using Newton's Law can be made for this case.

Knowing the forces, one can determine the momentum applied on the wing, *M*. With these two parameters, one can determine the normal stress and the shear flow distribution [13]:

$$\sigma_z = \frac{M_y I_{xx} - M_x I_{xy}}{I_{xx} I_{yy} - I_{xy}^2} x + \frac{M_x I_{yy} - M_y I_{xy}}{I_{xx} I_{yy} - I_{xy}^2} y \quad (12)$$

$$q_s = -\frac{S_x I_{xx} - S_y I_{xy}}{I_{xx} I_{yy} - I_{xy}^2} \left[\int txds + \sum_{r=1}^{n} x_r B_r \right] - \frac{S_y I_{yy} - S_x I_{xy}}{I_{xx} I_{yy} - I_{xy}^2} \left[\int tyds + \sum_{r=1}^{n} y_r B_r \right] + q_{s,0} \quad (13)$$

The von Mises criterion is then used to combine the effects of shear and normal stresses [13]:

$$\sigma_{res} = \sqrt{\sigma_{max}^2 + 3\tau_{max}^2} \quad (14)$$

The maximum residual strength obtained in the wing for every flight condition, using the requirements, will define the residual strength limit. It is expected that gusts or strong turbulence impose the most restrictive limits.

6. Crack Growth Analysis

Important characteristics, already discussed in this document, will now be gathered. The highest stress concentration area will be chosen and the initial damage will be applied there, for the residual strength value already stipulated and with these parameters two plots will be obtained: number of cycles *vs.* crack growth and stress intensity factor *vs.* growth rate (Figure 5 and Figure 6).

Crack retardation may be also considered. This phenomenon occurs due to overloads, which will slow down the progression of the crack [14].

The stress intensity factor, *ΔK*, is calculated recurring to the loading spectrum chosen, and it uses the difference between the maximum and minimum stresses, *Δσ*.

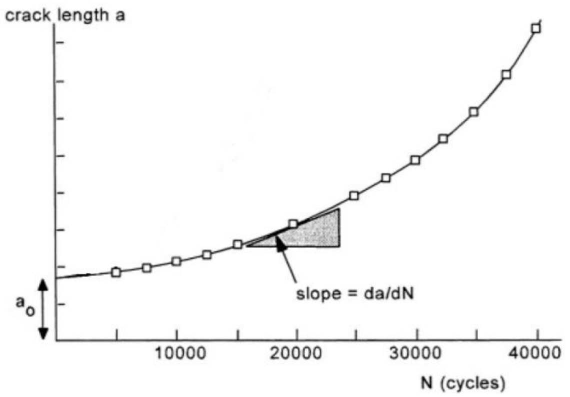

Figure 5 – Crack Length *vs.* Number of Cycles [5]

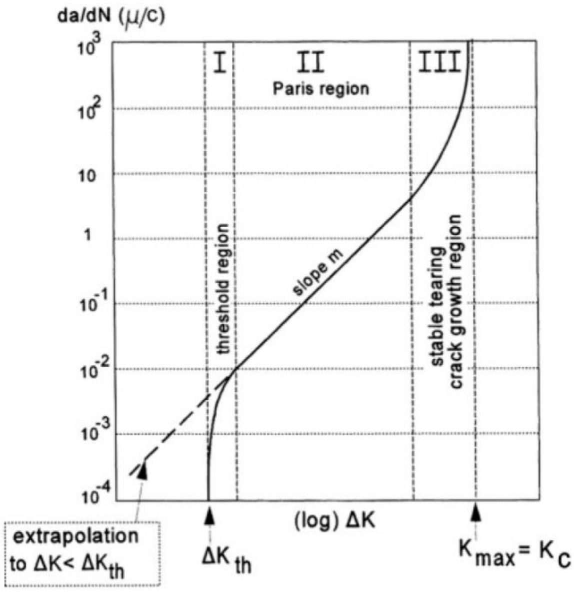

Figure 6 – Stress Intensity Factor vs. Crack Growth Rate [5]

The determination of the two plots presented in Figure 5 and Figure 6 can be made in two different ways: either using an analytical procedure or recurring to computer software like AFGROW.

The software is programmed to do the same procedure of the analytical method, but using more iterations and much more complex growth laws. It also introduces the possibility of using crack retardation.

The analytical procedure follows these steps [15]:

1. Initial damage, a_0
2. Determination of ΔK using the chosen spectrum's stress range ($\Delta\sigma$):

$$\Delta K = \beta \Delta\sigma \sqrt{\pi a} \qquad (15)$$

3. Determination of da/dN applying the Growth Law in use (Paris, Forman, *etc*.);
4. Obtaining a medium (arithmetic or geometrical) of two consecutive values of *da/dN* – $\frac{\overline{da}}{dN}$;
5. Determination of number of cycles for the current increase in crack length:

$$\Delta N = \Delta a / \left[\frac{\overline{da}}{dN} \right] \qquad (16)$$

6. The number of cycles, N, is obtained adding the value for ΔN determined.
7. Adding an increment to the crack size – Δa;
8. The process is then repeated from point 2 until a failure criterion is reached, whether based on Residual Strength, a_{crit}, or based on the fracture toughness of the material, K_{IC}.

It is important to notice that the geometry is very important for the definition of the shape factor β, which must be determined recurring to tables, graphs and formulas [16].

If it is chosen to use the AFGROW software, the process to determine the crack growth rate should follow these stages:

1. Introduction of the Geometry (model), whether user input or software incorporated (if needed the shape factor table may be also inserted);
2. Crack Growth Law selection, usually NASGRO Equation. The component's material will be also selected in this step;
3. Introduction of a Retardation Model (by default no retardation is chosen leading to a more conservative structure and its use is recurrent);
4. Spectrum insertion, with the definition of the Residual Strength value (notice that the spectrum is in percentage, as so a stress multiplication factor is introduced here);
5. Run the software to determine the two plots (ΔK *vs.* da/dN and N *vs.* a).

AFGROW automatically determines the shape factor. Still, user imposed factors can be introduced.

It is important to emphasize that only real fatigue life testing can give the designer a solid design point. Yet, computational and analytical methods, such as these ones, are a good starting point to a first estimation and sizing for the fatigue life of the component under study.

7. Inspection Requirements

The goal of an inspection is the detection of a growing crack, therefore the time definition of the first and subsequent inspections is very important.

The inspections must be made within a

period where the crack is detectable but still not dangerous for the structure. Figure 7 illustrates this interval.

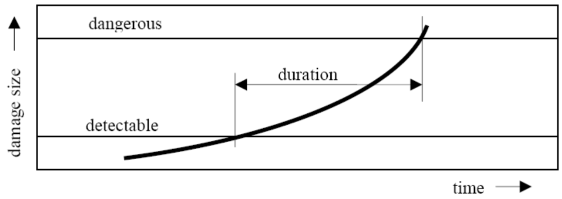

Figure 7 – Crack Detection Interval [17]

Once the crack is detected a repair procedure is adopted.

Only major inspections allow component disassembly, which leads towards a greater importance for visual inspections.

Visual inspections should be made by experienced technicians aided with lights, mirrors and magnifying glasses. Still, they can be divided into two major groups: detailed visual inspections and general visual inspections. Compared to the other NDI methods, visual inspections have a very low cost.

It is expected that trained inspectors are able to detect cracks larger than 2 inches in general visual inspections and larger than 0.25 inches in a detailed one [3] [18].

Next, is important to define inspection scatter factors. Their main purpose is to concede several crack detection opportunities, thus preventing the failure of the structure.

There are two types of scatter factors: one concerning the entire life of the component (j_1), and other to define the number of recurrent inspections to be made after the first one (j_2).

Usually a value of 2 is chosen, but values between 2 and 4 may be also assumed for j_1. The second scatter factor is often chosen between 3 and 8.

These factors can be obtained considering a multiplication of several life reduction factors, presented in Table 1 [19].

K_1, K_2	Defined as 2, to imposing inspections at least on half the component's life
K_3	1 for low humidity environment 1.5 for medium humidity environment 2 for high humidity environment
K_4	Special factor to account for unknowns (agreed with Aeronautical Authorities to ensure safety

Table 1 – Life Reduction Factor [19]

$$j_1 = K_1 \times K_3 \times K_4 \quad j_2 = K_2 \times K_3 \times K_4 \quad (17)\ (18)$$

Structures must be designed to withstand a period of two times the component's design life, in order to cover different uncertainties related with crack growth, such as variability of the material's properties, manufacturing quality or inspection reliability [3].

The initial inspection, also known as inspection threshold, is determined using the total life the component is expected to have (determined with crack growth analysis) divided by the total life scatter factor.

$$I_{th} = \frac{N_{crit}}{j_1} \tag{19}$$

Recurrent inspections are made since the first inspection until the crack is detected and repaired. Its interval definition is made considering the number of cycles between the length in which the crack is detectable and the expected moment of failure, available inspection time, divided by an appropriate factor (j_2) [2].

$$I_r = \frac{N_{crit} - N_{detec}}{j_2} \tag{20}$$

Figure 8 is a good example of this interval definition.

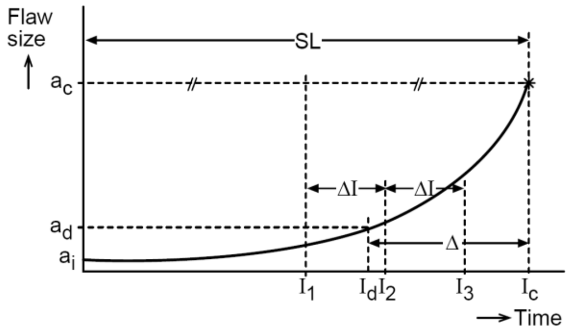

Figure 8 – Inspection Interval [2]

Furthermore, the first inspection interval, apart from the scatter factor chosen, cannot exceed half of the component's expected life [3] [19].

A compromise for the first inspection and inspections interval must be made. Different components will have growing cracks that might have inspection moments spaced in time. In order to reduce the costs, the fewest inspections should be made, and during one as many parts as possible should be inspected. This can be made through component design and proper scatter factor choice.

8. Concluding Remarks

This document contains a standardized procedure to make estimations on damage tolerance of a structure, and thus promoting more accurate inspection timing, which will ultimately lead to smaller costs for the company.

Damage tolerance is estimated over several steps:

1. Determination of the most critical point in terms of stress concentration (crack initiation and propagation is easier at these locations);
2. Definition of the initial damage (length, direction, shape and quantity);
3. Definition of the spectrum to use, whether from a real aircraft or algorithm based, usually selected concerning the type of aircraft that is being analysed;
4. Residual strength limits, calculated from the airworthiness requirements, defining the maximum allowed crack length on the structure;
5. Crack growth determination will enable to compute the number of cycles for the structure to have grown a crack with a prescribed dimension;
6. Inspection procedure definition, emphasizing the definition of the inspection threshold and inspection interval. The construction of the inspection charts is made through these parameters.

When the steps are completed, and a Damage Tolerance Analysis has been conducted, the following must be provided:

- The residual strength as a function of the crack size;
- The crack growth, emphasizing the number of cycles till failure and critical crack length;
- The initial damage assumed;
- The inspection interval.

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