Implementation of Damage Tolerance in High Cycle Fatigue Systems

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ABSTRACT: *Damage tolerance is positioned to replace safe-life methodologies for designing aerospace structures. The argument for implementing a fracture mechanics based damage tolerance approach is established in the fundamental fact that aerospace structures typically fail from cracks. Therefore, if the scientific means for predicting fatigue crack growth in a structure is feasible, this approach should deliver the most accurate representation of component life. Implementing damage tolerance (DT) into high cycle fatigue (HCF) components will require a shift from traditional DT methods that rely on an initial nondestructive inspection (NDI) flaw. The accumulation of cycles in a HCF component will produce a classic DT design that is either unmanageable because of frequent inspectio, or unrealistic because the design will be too heavy to operate for a reasonable life. Furthermore, once a crack in an HCF component begins propagating, the predicted time to failure is sometimes less than one flight hour, which does not leave ample time for NDI. Therefore, designing an HCF component will require basing the lifing analysis on an initial flaw that is "undetectable" by NDI. In this paper the author will formulate a methodolgy for implementing damage tolerant design in a high cycle fatigue system.*

INTRODUCTION

The fracture mechanics based damage tolerance design philosophy is founded on the ability of a structure to maintain integrity with a crack. The dimension of the crack, that is assumed to be inherent to the structure, is defined by the ability of non-destructive inspection (NDI) methods. The NDI capabilities for a given structure are identified by the accessibility to the component and the NDI method that is being used. For example, an aircraft fuselage outer skin is readily accessible whereas a jet engine turbine blade is not, and dye penetrant is easily applied to nearly any component while ultrasonic inspection requires highly trained mechanics and specialized tools. The United Stated Air Force has researched this subject extensively [1] and defines an initial crack size for fuselage and wing structure to be 1.27 mm, based on component accessibility and NDI capabilities. The international space community has also implemented damage tolerance for the International Space Station [2]. Where the initial crack size is dependant upon the inspection technique used and the component and crack geometries, *i.e.* eddy current of an unobstructed, unpainted surface for a part-through crack yields an initial crack size of 2.50 mm. Utilizing these NDI initial flaw sizes, a fracture mechanics approach is taken to determine the life of the component and inspection intervals can be set based on component accessibility and cost.

In a high cycle fatigue environment, this approach to damage tolerance breaks down [3]. This is primarily because the amount of cycles generated per hour of operation can exceed 60,000 [4, 5]. In this case, an NDI based initial crack size similar to those described above, will predict component failure within hours of installation. This life estimate is unrealistic, as many high cycle fatigue components such as propellers and helicopter rotor hubs are safely in service for thousands of hours before retirement based on safe-life designs. Therefore, to implement damage tolerance into these high cycle fatigue environments, other means of determining initial crack sizes and inspection intervals must be investigated.

In this paper, the author will discuss a process for determining an initial crack size based on manufacturing tolerances using the concept of an equivalent initial flaw size (EIFS) [6]. EIFS will also be discussed as an applicable means to define crack sizes from historical stress life data to define damage from field events, such as corrosion and foreign object damage (FOD). The issue of inspecting for cracks smaller than NDI capabilities will also be examined and finally a methodology for implementing damage tolerance into the design of high cycle fatigue systems will be proposed.

DAMAGE TOLERANCE WITHOUT INSPECTION FOR CRACKS

Implementing classic damage tolerance (DT) into high cycle fatigue (HCF) components will require a shift from traditional DT methods. The accumulation of cycles in a HCF component will produce a DT design, based on an initial NDI flaw size, which is either unmanageable because of frequent inspection, or unrealistic because the design will be too heavy to operate for a reasonable life. Furthermore, once a crack in an HCF component begins propagating, the predicted time to failure is sometimes less than one flight hour, which does not leave ample time for NDI. Therefore, designing an HCF component will require basing the lifing analysis on an initial flaw that is "undetectable" by NDI.

The concept of an Equivalent Initial Flaw Size (EIFS) is based on the assumption that the life of a structure is governed by fatigue crack growth [7]. Assuming this to be true, one can assemble Stress-Life (S-N) data that has been generated for certain materials under specific conditions and back-calculate an initial flaw size that would have had to be inherently evident in the material to fail the specimen. For example, Figure 1 is a plot of S-N data for aircraft aluminium. Each data point represents a specimen failure, except the points denoted as "run-outs" where the test was stopped prior to specimen failure. Using this data and fracture mechanics tools, a distribution of EIFS values can then be determined for each condition, as shown in Figure 2. Where the vertical axis of Figure 2 is a cumulative distribution function defined by dividing the EIFS value by the maximum EIFS and the horizontal axis is the EIFS. Using the as-manufactured condition, one can determine a worst-case EIFS that would be inherent in the part prior to service. This would be the basis for designing a damage tolerant component to meet a life requirement.

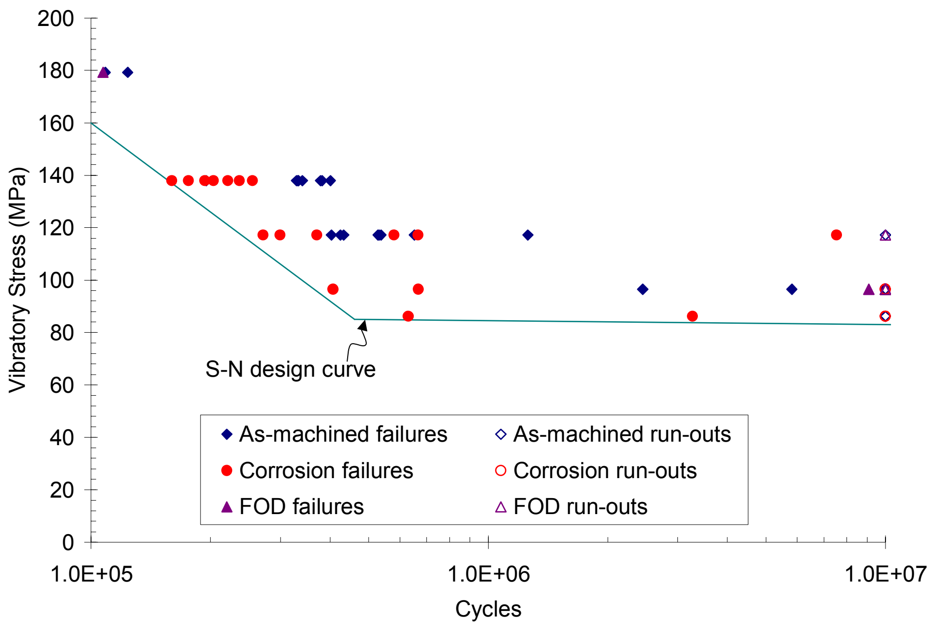

Figure 1: Stress-Life data for aircraft aluminium in the as-manufactured state, with damage and lightly corroded.

The EIFS values for in-service conditions, such as corrosion and mechanical foreign object damage (FOD), can then be used to set inspection intervals. The S-N data for lightly corroded and damaged specimens and the corresponding EIFS estimates are shown in Figures 1 and 2 respectively. It is interesting to note that the estimated FOD EIFS values are smaller than the as-machined specimens. This is attributable to the compressive residual stress that FOD imparts on the structure, thereby retarding crack growth [5]. Since the EIFS values for high cycle fatigue components are typically smaller than an NDI detectable crack size (2.50 mm), inspecting for the EIFS is unrealistic. Based on this formulation, an operator would be required to inspect for the FOD and corrosion damage that leads to fatigue cracking. This is contrary to classic DT where inspection is defined to detect fatigue cracks. It is much more straightforward to inspect for corrosion pitting and foreign object damage than it is to reliably detect a 0.1 mm crack in a structure [8].

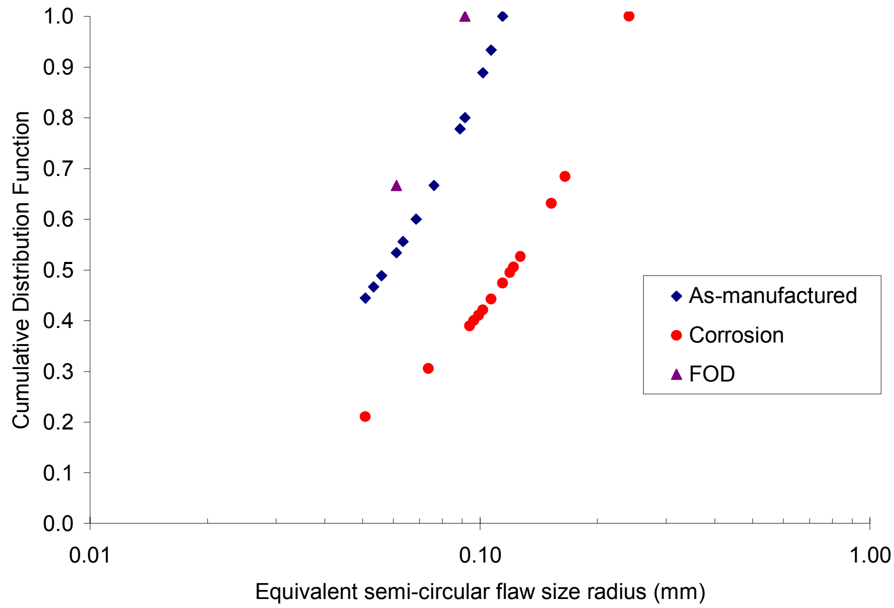

Figure 2: Equivalent Initial Flaw Size (EIFS) data for aircraft aluminium in the as-manufactured state, with damage and lightly corroded.

EXAMPLE PROBLEM

The feasibility of implementing damage tolerant design in a simple lug under high cycle fatigue conditions is examined. A schematic of the lug is shown in Figure 3 defining the length, width, hole radius and loading direction. The thickness and consequently the weight will be determined by the analyses. The lug will be manufactured from aircraft aluminium, will undergo a series of variable amplitude load excursions, and will be exposed to a corrosive environment and foreign object damage. The design will be undertaken using a Stress-Life analysis [9], damage tolerance based on NDI detectable flaws, and the above described HCF DT method. The flight life requirement for the component is 10,000 hours of service. The loading spectrum is graphically depicted in Figure 4 for one flight hour. It is also assumed that this part is readily inspectable using eddy current to find a 2.50 mm crack reliably [2]. Furthermore, the lug will be inspected 3 times prior to retirement, or reissue to service based on comprehensive inspection and refurbishment.

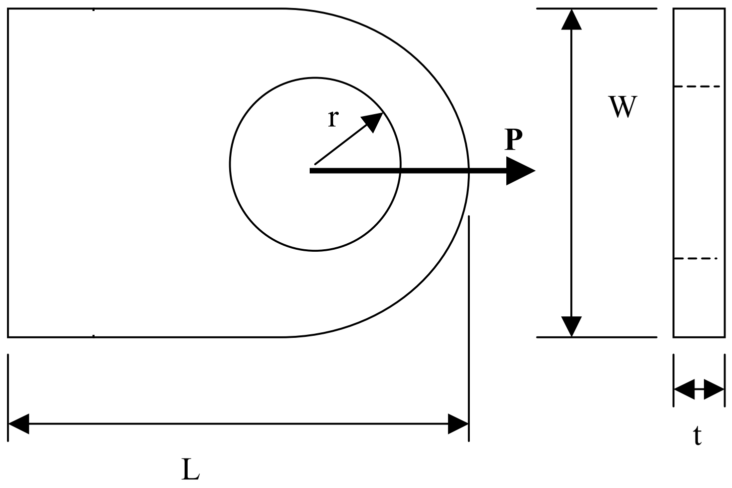

Figure 3: Definition of lug problem (L = 0.5m, W = 0.25 m and r = 0.1 m).

The required S-N material data is presented in Figure 1 for the Stress-Life analysis and the loading spectrum is defined in Figure 4. The damage tolerance analysis further requires da/dN vs. ΔK data [10], which is depicted for a stress ratio of 0.1 in Figure 5. The density of the aluminium alloy is assumed to be 2.81 g/cm^3 for purposes of calculating structural weight.

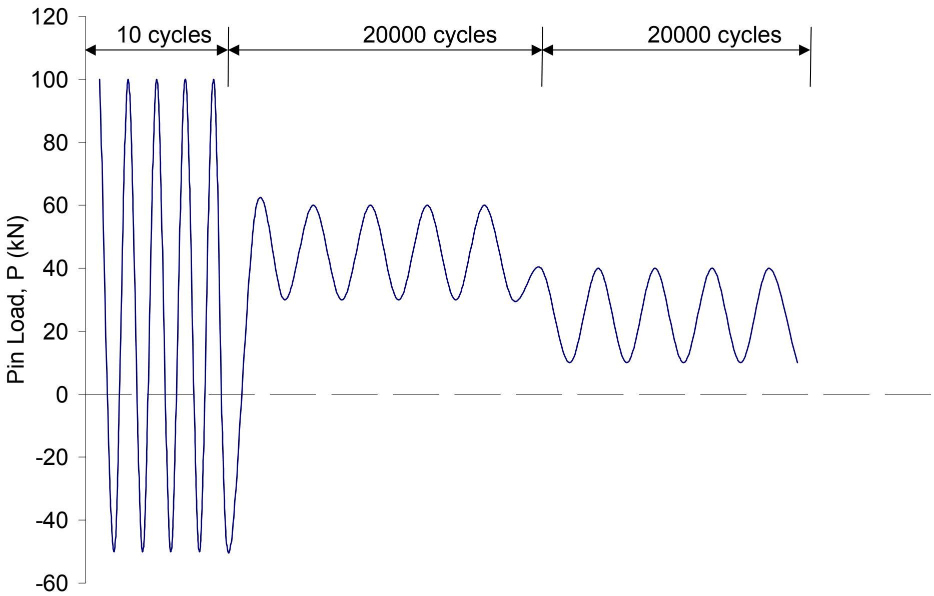

Figure 4: Schematic of the component loading for one flight hour.

The computed S-N fatigue life, using Miner's analysis [9], of the component is 10,000 hours with a calculated thickness of 18.75 millimetres, imparting a component mass of approximately 1.32 kg. This information is summarized in Table 1 along with the inspection interval, which is set to one quarter of the overall life in the case of S-N. The damage tolerance analysis was performed using an initial crack size of 2.50 mm, specified by the NDI method, and provided a component life of 10,000 hours with a thickness and mass of 97.91 mm and 6.88 kg respectively. This data is also summarized in Table 1 with the prescribed inspection interval, which is one quarter of the overall life. Finally, the HCF damage tolerance approach was performed using an initial flaw size of 0.10 mm, the maximum EIFS based on the as-manufactured data (Fig. 2). The computed fatigue life, thickness and mass of the component are 10,000 hours, 26.59 mm and 1.87 kg respectively. These are summarized along with the computed inspection interval, based on the corrosion EIFS (0.30 mm), in Table 1.

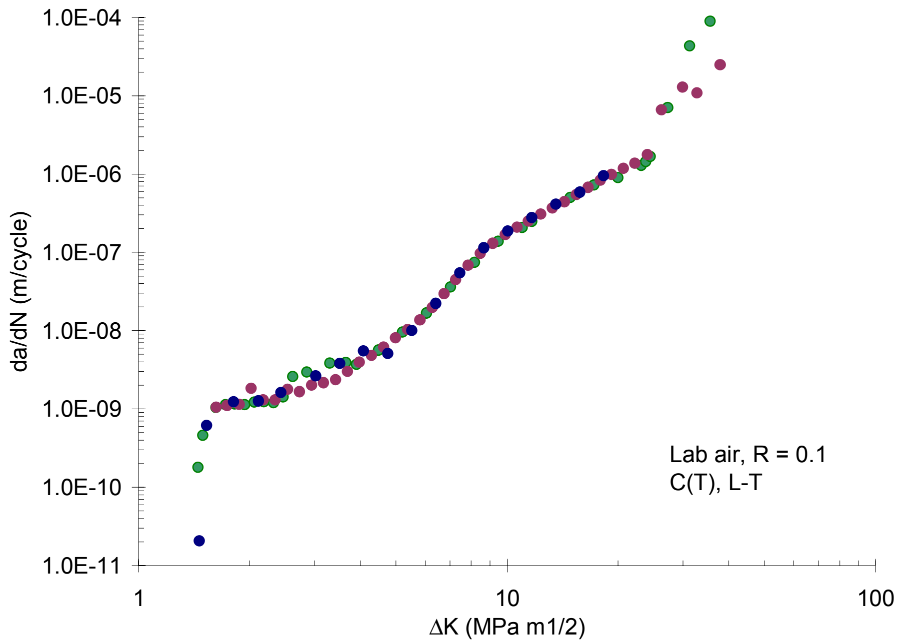

Figure 5: da/dN vs. ΔK data for aircraft aluminium at R = 0.1.

TABLE 1: Comparison of design methods.

Method	Fatigue Life (hours)	Thickness (mm)	Mass (kilograms)	Inspection (hours)
S-N	10,000	18.75	1.32	2,500
DT	10,000	97.91	6.88	2,500
HCF DT	10,000	26.59	1.87	2,250

DISCUSSION

Based on the analysis of the lug using stress-life and two forms of damage tolerance it is apparent, for this example problem, that the stress-life solution gives the lightest component. In comparison, the traditional damage tolerance procedure yielded a part that was 420 percent heavier than the safe-life component and the HCF DT approach yielded a part that was 42 percent heavier. A contributing factor to the difference in the solutions arises from the 40,000 flight cycles that are considered non-damaging in an S-N approach, whereas these cycles cause damage in a DT approach. However, if no crack is detected during NDI, a damage tolerance managed component may continue use. This can lead to component operating times that far exceed the S-N retirement life, possibly offsetting the weight penalty imposed in the above example.

Inspecting for cracks on the order of 0.10 to 0.30 mm is currently not practical. Nor is designing a part four times heavier to provide adequate time to identify detectable cracks. Therefore, for high cycle fatigue components to be designed damage tolerant there must be an improvement in NDI or an alternative to inspecting for cracks. Surface damage, such as corrosion and FOD, can be readily detected, and in most cases repaired, through visual inspection. Understanding the impact this damage has on component life, one can safely maintain structural integrity through damage tolerance without directly inspecting for cracks. Contrary to classic damage tolerance, an operator would be required to inspect for the FOD and corrosion damage that leads to fatigue cracking instead of the cracks themselves. It behoves industry to address high cycle fatigue issues that obstruct the adoption of damage tolerance. The companies that choose to accomplish this task will realize significant improvements in safety and operating cost. Replacing parts without cause is no longer a viable means to fleet management.

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