FATIGUE AND FRACTURE RESEARCH IN METALS

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INTRODUCTION

Fatigue and fracture of structural materials have always been a concern for design engineers. Notable structural failures have occurred throughout all historical periods. Categories of prominent fatigue and fracture failures have shifted from bridges (1700 to about 1850) to railroads (1850 to early 1900's) to storage tanks (1890's to 1930's) to ships (1940's to 1960) and to aircraft (1940's to present).

Since the 1940's extensive investigations have been undertaken to both explain and solve the fatigue and fracture problems. These studies have revealed that flaws, low metal toughness, and stress concentrations not anticipated in design are responsible for most failures. Conventional design criteria were based on tensile strength, yield stress, and buckling stress. Although these criteria were adequate in many engineering situations they were insufficient under conditions where cracks or crack-like defects were present in the structure.

Fatigue and fracture research at NASA Langley Research Center on monolithic and laminated metals has been concentrated in three areas: (1) stress analyses of two- and three-dimensional cracked bodies, (2) fatigue crack growth, and (3) fracture toughness. Analytical methods have been developed to predict fatigue crack growth and fracture strengths of cracked specimens. Such specimens represent typical aircraft structural details (such as cracks from holes). These specimens were subjected to simple constant-amplitude loading and to more complex flight load histories. Test data from both in-house tests and from the literature are used to substantiate the analytical methods. These analyses have extended the theory of fracture mechanics to deal with fatigue crack growth and fracture of complex crack configurations that are typical of aircraft materials and structural details. Several of these analyses are now used in ASTM standards and ASME codes and by several aircraft companies for damage tolerance studies. This paper will provide a summary of this research area and highlight recent advances in understanding the fatigue and fracture behavior of metals.

MATERIALS TEST AND ANALYSIS METHODOLOGY

Structural components are subjected to cyclic load time histories and operate in various environments. They may develop cracks from material defects, stress concentrations, or inadvertent damage. Such cracks grow at rates that depend upon applied load levels and environmental conditions. Catastrophic failure occurs when the crack reaches a critical length.

Several American Society for Testing and Materials (ASTM) Committees (E9, E24, and G1) are developing or have developed testing standards and test specimens for characterizing fatigue crack growth [1], stress-corrosion cracking, and fracture toughness [2] of engineering materials. The NASA Langley Research Center has contributed to the analyses of several of these standard test specimens. A few of these, shown in figure 1, will be discussed here.

The stress intensity factor (K) (a crack tip characterizing parameter) has been calculated at Langley for several standard laboratory specimens: the compact specimen, the round compact specimen, and the bolt-loaded double-cantilever-beam (DCB) specimen. Compact and round compact specimens, used for fatigue crack growth and fracture toughness tests, were analyzed by Newman [3] using a boundary-collocation analysis. (The round compact specimen has been shown to be about 40 percent cheaper to machine than the rectangular specimen.) The DCB specimen, used in stress-corrosion cracking tests, was analyzed by Fichter [4] using asymptotic and collocation methods.

These analyses provided more accurate K-solutions over a wider range of crack lengths than previous solutions. For example, the failure loads on 2219-T851 aluminum alloy compact specimens are plotted as a function of crack-length-to-width (a/W) ratio in figure 1. The solid and dashed curves show the predicted results using the new and old analyses, respectively. The new analysis [3] was much more accurate than the old analysis at low and high values of a/W. The new K-solutions for the compact specimens are in current ASTM standards [2].

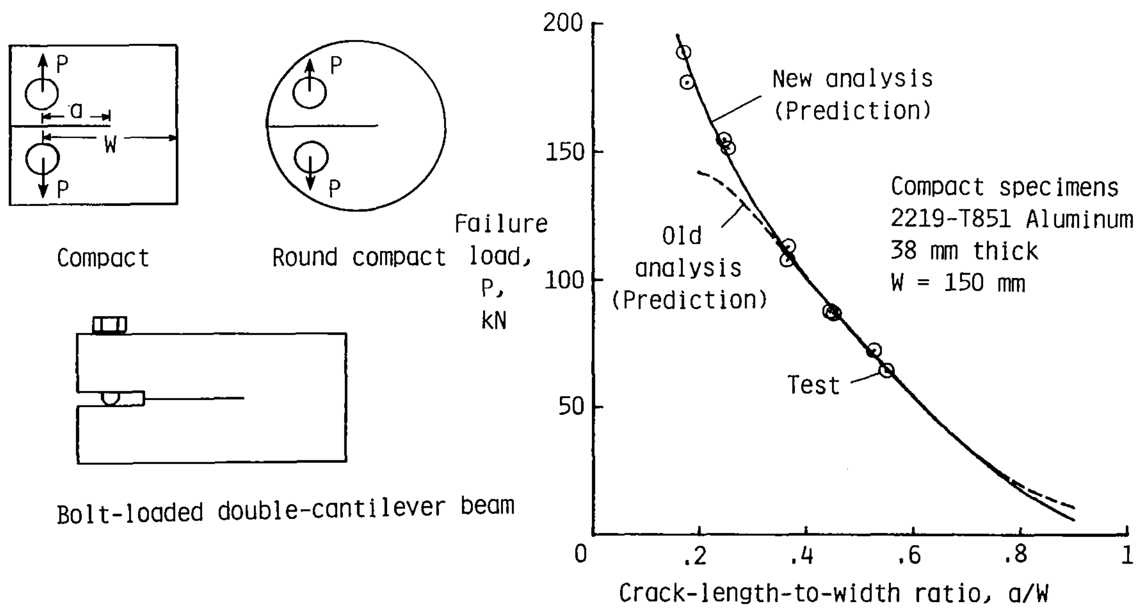


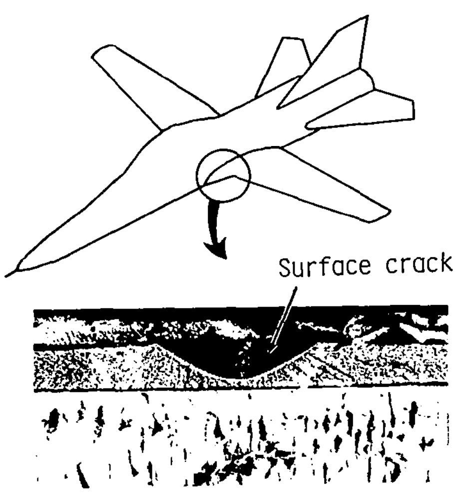
Figure 1.- Laboratory test specimens and experimental verification of analysis.

THE SURACE CRACK

In 1969 the catastrophic failure of the wing on an F-111 aircraft was caused by a surface crack. A photograph of the cracked area is shown in figure 2. This failure, more than any other, initiated present U.S. Air Force airworthiness regulations [5]. These regulations are based on considerations of "damage tolerance." The underlying philosophy for damage tolerance is to acknowledge that accidental or normal service-induced damage is inevitable, and that periodic inspections are needed to detect such damage. Airworthiness is then assured by demonstrating that damage that escapes one inspection will not grow to critical size before the next inspection. Two evaluations must be made: first, the rate of crack growth under expected service loading; and second, the residual static strength with the crack present. Such evaluations employ fracture mechanics analyses, in particular the stress intensity factors for various crack configurations.

Many stress intensity factor (K) solutions have been proposed for the surface crack. However, for large crack-depth-to-plate-thickness ratios (like that in the F-111 failure), the solutions differed considerably (up to 80 percent). To provide an accurate solution, three-dimensional finite-element analyses of semi-elliptical surface cracks subjected to tension and bending loads have been conducted by Raju and Newman [6]. The K-equations for surface cracks in plates under tension and bending loads, as well as in pressurized cylinders, have also been developed [7,8]. The K-equation for the surface crack has been experimentally verified by Newman and Raju [7]. The ratio of predicted to experimental failure loads for brittle epoxy specimens containing various size surface cracks is plotted as a function of crack-depth-to-plate-thickness (a/t) ratio in figure 2. The equation was able to predict failure loads within about ±10 percent of experimental failure loads.

These new K-solutions have been used in an ASTM standard practice on surface crack testing [9], and in the newly revised ASME pressure vessel and piping codes.



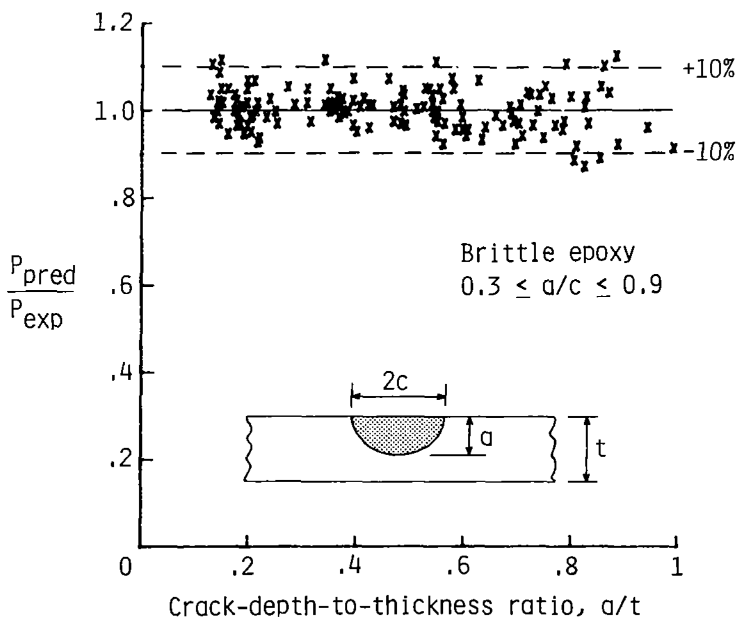


Figure 2.- Aircraft failure due to surface crack and experimental verification of analysis.

MOST COMMON CRACK IN AEROSPACE STRUCTURES

Corner cracks and through cracks at holes are among the most common flaws in aircraft structures. The photograph in figure 3 shows a corner crack at a hole in a plank on a fractured wing structure. Accurate stress analyses of such configurations are needed to reliably predict crack growth rates and fracture strengths, and to establish inspection intervals so that failures like this one can be avoided.

A three-dimensional finite-element analysis was again conducted at Langley by Raju and Newman [10] to determine stress intensity factors for quarter-elliptical corner cracks at the edge of a hole under various loading. These stress intensity factor solutions are used to predict the number of load cycles required to grow a crack from a small defect at the edge of a hole to failure. Figure 3 shows a comparison between experimental and predicted crack length versus number of cycles for a corner crack growing from a circular hole in a 7075-T651 aluminum alloy specimen subjected to constant-amplitude tensile loading. The predicted crack propagation life was in good agreement with the experimental value.

Equations for stress intensity factors for other three-dimensional crack configurations (such as an embedded elliptical crack, a quarter-elliptical corner crack, a semi-elliptical surface crack, and semi-elliptical surface cracks at a hole in finite-thickness plates) have also been developed by Newman and Raju [11]. Several aircraft companies have incorporated these equations into computer programs for the design of damage-tolerant structures.

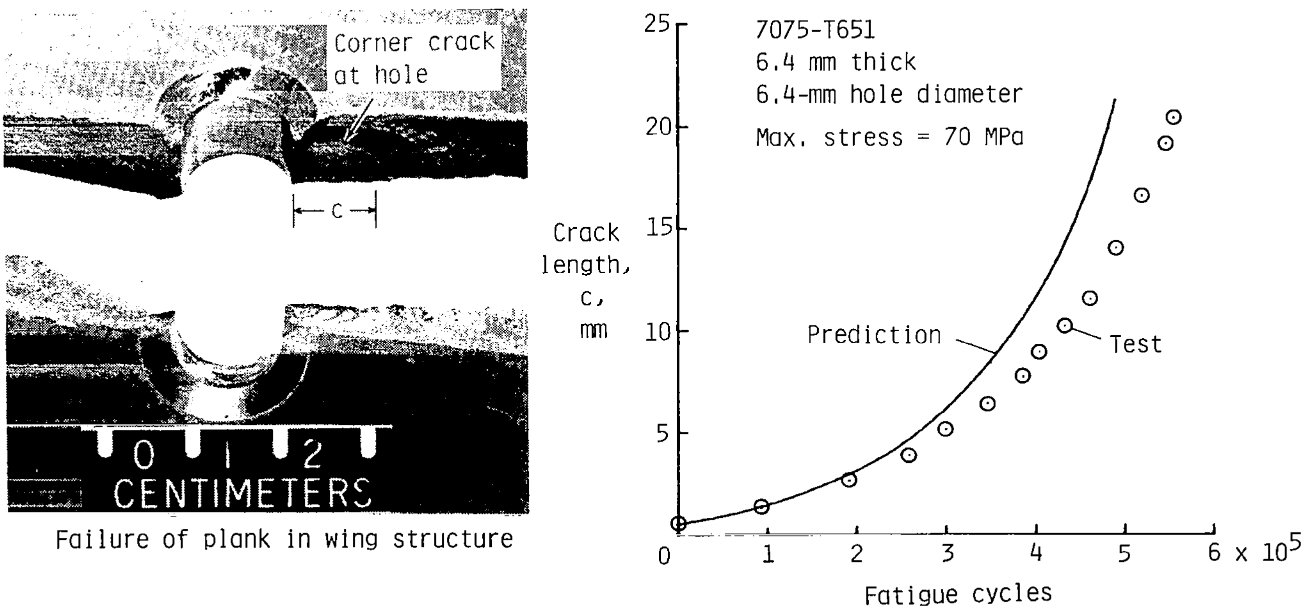


Figure 3.- Wing failure due to corner crack at hole and experimental verification of analysis.

THE CRACK CLOSURE PHENOMENON

Elber [12] was the first to identify and quantify the phenomenon of crack closure during cyclic loading. He showed that the crack surfaces near the crack tip close before a specimen is completely unloaded. Such behavior was not expected in an elastic specimen until an infinitesimal compressive load was applied.

The reason for crack closure is depicted in figure 4. The presence of a plastic zone ahead of a crack tip (shown with double crosshatch) is accepted as a feature of crack tip behavior. But the feature that had been commonly ignored was the plastically deformed material left in the wake of a crack. As the load on a specimen is reduced, this deformed material comes together. It then transfers compressive forces across the crack surfaces before the load is reduced to zero. Many investigators have conducted detailed experiments to verify crack closure. The intuitive consequence of this behavior is that those portions of a load cycle during which the crack tip is closed will not contribute to crack growth.

The closure behavior has been analyzed by Newman using an elastic-plastic finite-element analysis [13]. This analysis showed that cracks open and close at predictable stress levels. Figure 4 shows a comparison between experimental and predicted crack-opening stresses normalized by the maximum applied stress (S_o/S_{max}) plotted against stress ratio (ratio of minimum to maximum applied stress). The predicted values agreed well with the experimental values. The finite-element analysis was, however, very complicated and required a large computer. More recently, Newman [14] developed a simple strip-yield model of closure that needs only a small computer but can accommodate many hundreds of thousands of variable load cycles. The use of this model to predict crack growth under aircraft spectrum loading will be discussed later.

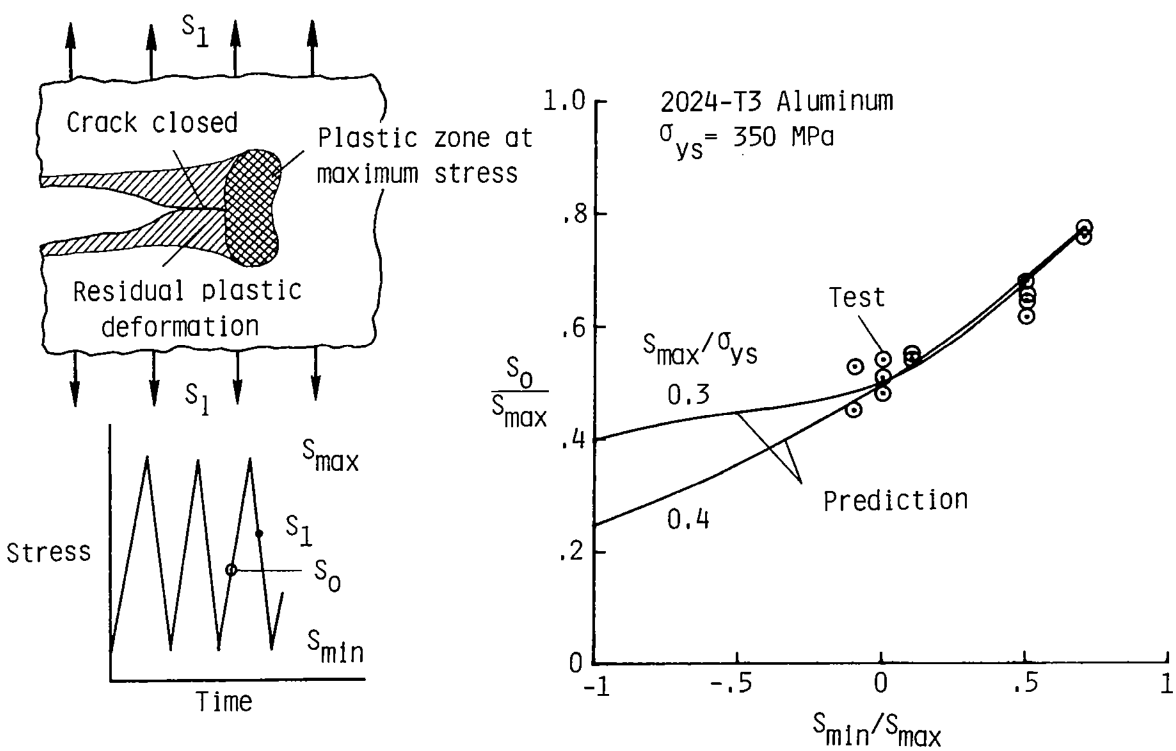


Figure 4.- Fatigue crack closure concept: test and analysis.

Most studies on fatigue crack growth have been conducted on "large" cracks with lengths in excess of 2 mm. However, in many engineering structures crack growth from "small" preexisting flaws is a major portion of the component's fatigue life. But the growth of small cracks (10^{-2} to 1 mm) in plates and at notches differs from that of large cracks. Such behavior is illustrated in figure 5, in which the crack growth rate is plotted against the stress intensity factor range, ΔK, for a constant ratio of minimum to maximum load, R. The solid curve shows a typical result usually obtained from tests with large cracks. At low growth rates the threshold stress intensity factor range, ΔK_{th}, is usually obtained from load reduction tests. Some typical results for small cracks are shown by the dashed curves. These results show that small cracks grow faster than large cracks at the same ΔK level and that they also grow at ΔK levels below threshold.

To explain these differences, the crack closure model mentioned previously was used to study crack growth and closure behavior of small cracks in plates and at notches [15]. At equivalent ΔK levels the model predicts that small cracks should grow faster than large cracks because the applied stress level needed to open a small crack is less than that needed to open a large crack. Results from the model also imply that many of the ΔK_{th} values that have been obtained in tests with large cracks and with load reduction schemes do not apply to the growth of small cracks; that is, the load reduction scheme is causing the threshold. Consequently, the large crack data at low applied stress levels may possibly follow the dash-dot curve.

These results indicate the importance of considering crack closure when using large-crack data to predict the growth of small cracks from preexisting defects.

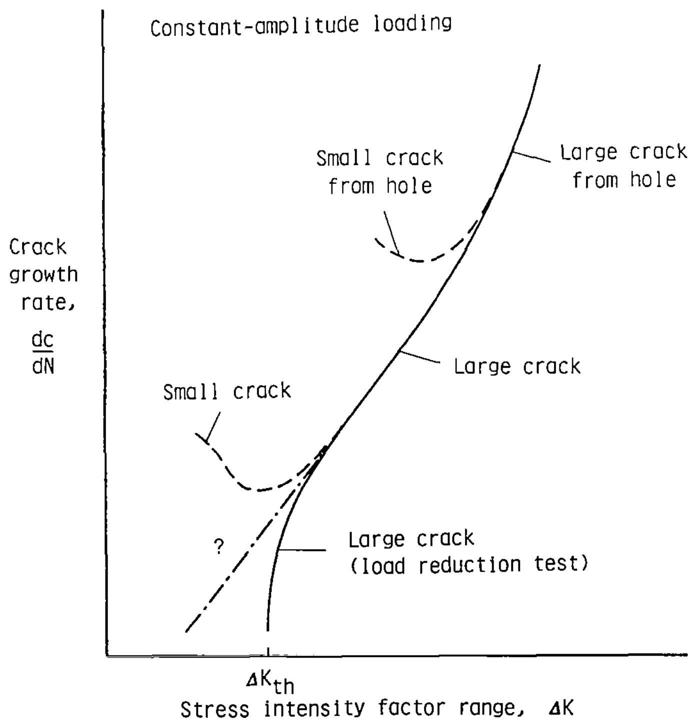


Figure 5.- Typical fatigue crack growth rate data for small and large cracks.

PREDICTION OF CRACK GROWTH UNDER AIRCRAFT SPECTRUM LOADING

Predictions of crack growth rates are used in aircraft design to determine inspection intervals and damage tolerance. Under aircraft spectrum load conditions, load interaction effects sometimes retard and sometimes accelerate crack growth.

Various models have been developed to account for load interaction. Two of the most widely accepted concepts are being studied at Langley. One, the CGR-LaRC computer program developed by Johnson [16], uses the multiparameter yield zone (MPYZ) model. Like the well-known Willenborg model [17], the MPYZ model is based on residual stresses in the crack tip region. The MPYZ model uses crack growth rate data from constant-amplitude loading tests. Four parameters must also be obtained from variable-amplitude load tests.

The other concept is the FAST computer program, which is based on the crack closure model developed by Newman [14]. The closure model requires only crack growth rate data under constant-amplitude loading to predict crack growth under variable-amplitude loading.

ASTM Subcommittee E24.06 on Applications recently completed a round robin study of crack growth predictions under aircraft spectrum loads. The accuracy of the predictions made from the CGR-LaRC and FAST programs on 2219-T851 aluminum alloy specimens for the round robin is shown in figure 6. Both methods gave good results on 13 different aircraft spectrum load tests. The mean value and standard deviation of the ratio of predicted cycles (N_{PRED}) to test cycles (N_{TEST}) for each method are shown in the figure.

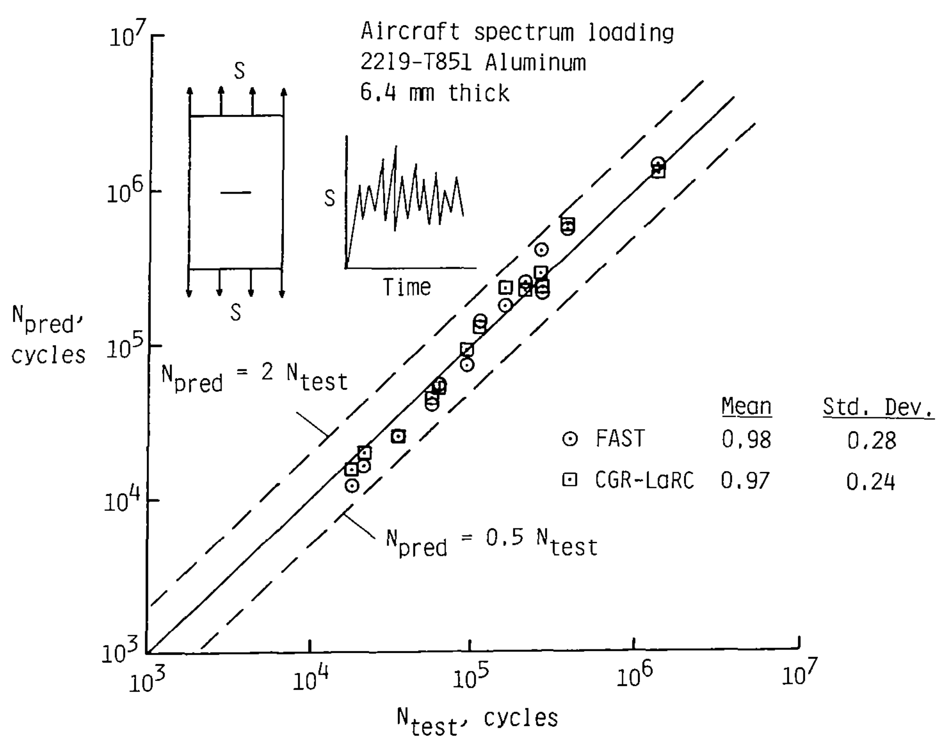


Figure 6.- Comparison of predicted and experimental cycles to failure using two crack growth models.

Under an increasing load, a crack growing in metal leaves plastically stretched material behind. The stretching is very dramatic in ductile materials, as is shown in the photograph of a copper specimen failing (fig. 7). An arrow points out the plastic wake of material formed during the final load cycle of a fatigue test. (The photograph shows the lower half of the crack surface; the upper half was outside the field of view.) Although the plastic wakes are much smaller in aerospace structural materials, their formation still controls stable crack growth and instability. If the plastic wakes were not there, any crack extension (Δa) would precipitate unstable growth and catastrophic failure.

Elastic-plastic finite-element analyses of the crack growth process conducted by Newman [18] have confirmed the importance of the plastic-wake concept. The analysis was used with a critical crack tip opening displacement (CTOD) criterion to study crack initiation, stable crack growth, and instability under monotonic loading to failure for several materials (7075-T651 and 2024-T351 aluminum alloys and 304 stainless steel). Comparisons between calculated and experimentally measured CTOD values (δ_c) at initiation agreed well for compact specimens made of the two aluminum alloys. These critical CTOD values, determined from compact specimens, were used to predict failure loads on laboratory specimens (center-crack tension) and on a structurally configured specimen (typical of aircraft structural details) made of the three materials. Figure 7 shows a comparison between experimental (symbols) and predicted (solid curves) failure loads on center-crack tension specimens made of the three materials. Predicted failure loads were within ±10 percent of the experimental failure loads for these specimens and were generally within ±15 percent of the experimental failure loads on the structurally configured specimens for all three materials.

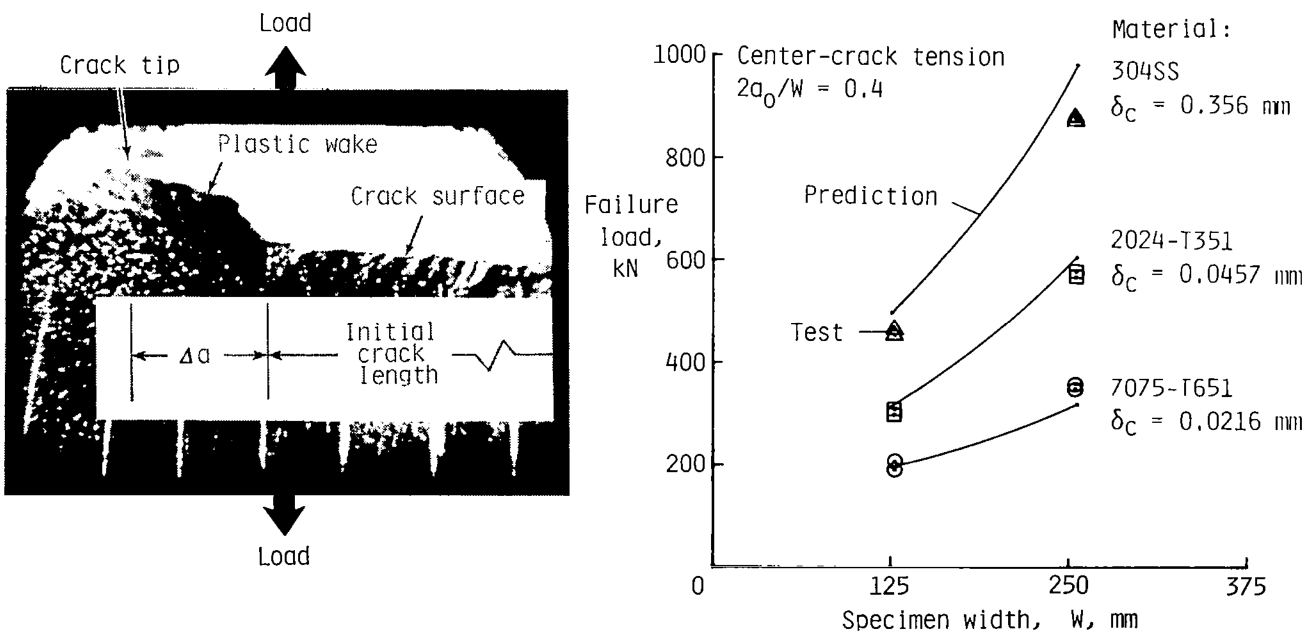


Figure 7.- Plastic-wake concept: test and finite-element predictions.

STATE OF THE ART IN FRACTURE ANALYSES FROM ASTM ROUND ROBIN

The results of an experimental and predictive round robin conducted by Langley and ASTM Task Group E24.06.02 on Application of Fracture Analysis Methods are shown in figure 8. The objective of the round robin was to verify whether the fracture analysis methods currently used can or cannot predict, from the fracture results of a "standard" compact specimen, failure loads of complex structural components containing cracks.

Results of fracture tests conducted on various-sized compact specimens of 7075-T651 aluminum alloy, 2024-T351 aluminum alloy, and 304 stainless steel were supplied as baseline data to 17 participants. These participants (from industry, university, and government) used several different fracture analysis methods to predict failure loads on a structurally configured specimen that typified aircraft structural details. The specimen contained three circular holes with a crack emanating from one of the holes (see insert in figure). The specimen was loaded in tension. The failure loads on the structurally configured specimens were unknown to the participants.

The accuracy of the prediction methods was judged by the variation in the ratio of predicted to experimental failure loads, as shown in figure 8 for the 2024-T351 aluminum alloy specimens. The various methods used and the range and mean of the predictions are indicated on the figure. A few methods were able to predict failure loads within about ±10 percent of the experimental loads. Several methods were totally inadequate. On the basis of the predictions made on all three materials, some crack growth resistance methods (with proper limit load calculations), the two-parameter fracture criteria, and the finite-element analyses appear to be the best methods for predicting failure loads on cracked components.

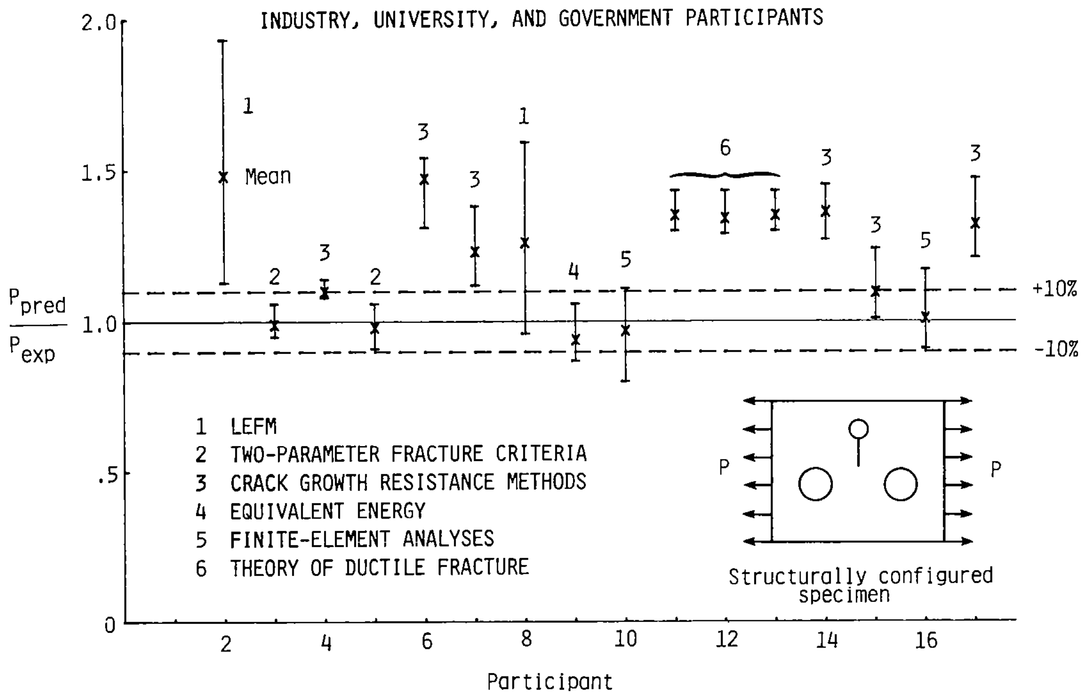


Figure 8.- Summary of predicted to experimental failure loads on 2024-T351 aluminum alloy cracked structural components.

FATIGUE LIVES AND FRACTURE TOUGHNESS OF LAMINATED METALS

Titanium's high strength-to-weight ratio and high temperature stability are desirable structural properties. However, under cyclic loads the crack growth rates are high compared to aluminum alloys at comparable applied-stress-to-density ratios, and its toughness-to-density ratio is low (between brittle D6AC steel and ductile 2024 aluminum). The high crack growth rates and low toughness lead to short damage-tolerant life. To improve the damage-tolerant capability of titanium structures the concept of adhesively bonded titanium sheets was assessed by Johnson [19].

In this investigation six thin sheets of Ti-6Al-4V titanium were bonded together with AF-147 adhesive. They formed a laminated plate 9 mm thick from which specimens were cut. Specimens similar in planform and about the same thickness as the laminate were cut from a monolithic plate. Electric-discharge-machined (EDM) surface flaws were cut into each type of specimen, as shown in figure 9. All specimens were subjected to the same fighter load spectrum. The ratio of maximum spectrum load to specimen density was the same for each specimen. The figure shows that the laminated specimen survived 15 times longer than the monolithic one did. This increased life happened in part because when one ply failed, the others supported the load. Because the adhesive was relatively weak, flaws were inhibited from crossing from one lamina to another.

Toughness tests were also conducted on compact specimens made of both the six-ply titanium laminate and the monolithic plate. Figure 9 shows that the laminated plate has a fracture toughness K_c about 40 percent higher than that of the monolithic plate. The higher toughness implies that a crack in a laminated structure can grow to about twice the length of a crack in a similar monolithic structure before failure. These results show that the adhesive lamination process is a viable way to improve the fatigue life and toughness of titanium for structural applications.

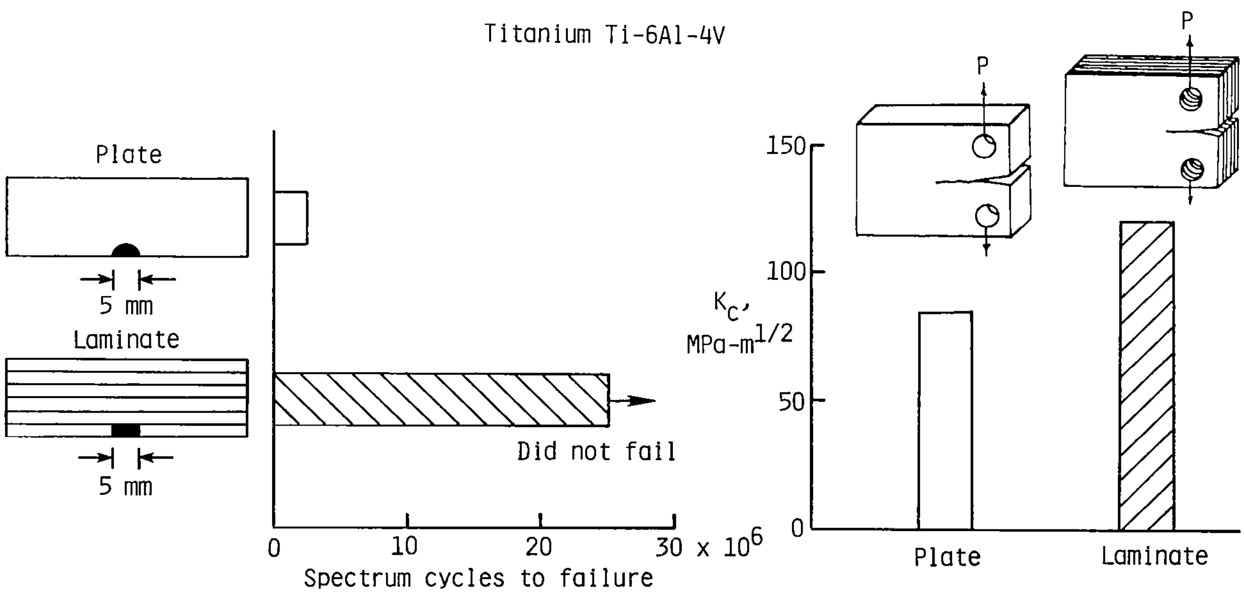


Figure 9.- Fatigue lives and fracture toughness of monolithic and laminated metal.

CURRENT RESEARCH FOR FATIGUE AND FRACTURE OF METALS

- Three-dimensional elastic stress analyses of other crack configurations
- Three-dimensional elastic-plastic analyses of fatigue crack growth and fracture
- Development of a simple fracture model based on the plastic-wake concept
- Scanning electron microscope studies of crack growth under variable-amplitude loading
- Experimental studies on crack closure in thick plates
- Elastic-plastic fracture of surface-cracked plates
- Study effects of holes and bolt loading on the damage tolerance of laminated metal
- Fatigue crack growth measurements in various thickness materials
- Develop and experimentally verify new threshold testing procedures

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