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A Quantitative Assessment of the Variables Involved in Crack Propagation Analysis for In-Service Aircraft

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Recent emphasis on the importance of the damage tolerance characteristics of airframe structures requires that structural designers have adequate analytical tools to properly make decisions which affect structural integrity over the service life of the aircraft. Presented are the significant results of an in-depth assessment to enhance the structural information required for structural management of a transport aircraft. One of the major thrusts of this assessment was to determine the sensitivity of the final analysis prediction to each variable and thereby identify those variables requiring additional accuracy and attention. Both analyses and tests were conducted during the assessment to establish the range of variation expected in the important variables. The results of this assessment are presented and supported by test vs analytical comparisons using flight-by-flight test spectra developed from measured real-time history data obtained from service aircraft. The results of this assessment indicate that there are three areas which should receive special attention in the structural assessment of a transport aircraft: material properties da/dN, an analytical flaw size and shape, and the crack propagation law.

Nomenclature

a = half length or depth of crack, in.
c = half surface length of corner crack, in.
da/dN = constant amplitude crack growth rate per cycle, in./cycle
K_c = plane stress or mixed mode fracture toughness, ksi $\sqrt{\text{in.}}$
K_{Ic} = plane strain fracture toughness, ksi $\sqrt{\text{in.}}$
R = ratio of minimum stress to maximum stress
ΔK = stress intensity factor range, ksi $\sqrt{\text{in.}}$

Introduction

THERE exist many recognized variables involved in the prediction of crack propagation for in-service aircraft.[1] The potential individual contribution of each of these variables and the overall effect on crack propagation life must be known if such analysis is to play a role in aircraft structural integrity. An intensive 2 yr multimillion dollar program has been conducted to quantify these variables individually and to assess potential effects on the overall crack propagation prediction for transport aircraft.

The scope of this program encompassed the full range of variables involved in aircraft crack propagation analysis, such as: flight load variations; initial flaw shape, size, location, and growth mode; material fracture properties variation; effect of environment and temperature on growth rate; maximum load level; analytical spectrum development procedures; and crack propagation law.

Extensive empirical data were developed in this assessment. Crack findings from two full-scale fatigue tests were used. Two hundred sixty-three new coupon tests were conducted (static, constant amplitude, and spectrum) and four box beam spectrum tests were run using real-time history bending and torsional load spectra obtained from aircraft flight measured data. This paper presents the significant data generated during this program and summarizes the findings which occurred as a result of interpretation and evaluation of the data.

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Flight Load Variations

Current fatigue and fracture tracking systems are based on the ability to predict adequately the stress experience of aircraft usage and expected future usage with the results applied to such things as the establishment of inspection intervals. Since the methods of predicting stress experience are based on "average statistics" derived from quantities of recorded response data, it is desirable to evaluate the potential deviation that may exist in isolated smaller intervals. The purpose of this study, therefore, was to evaluate the variation in crack growth rates which occur from the application of "long-term" statistical data to short usage periods.

As illustrated in Fig. 1, real-time strain data recorded from 1400 flights were reduced to sequential cycle-by-cycle form. These data were subjected to a linear crack growth analysis at a constant crack length, i.e., the growth rate was computed independently for each flight. The usage data, in the form of hand-recorded pilot logs (cargo, fuel, altitude, etc.), were then used to predict analytically the stresses that would have occurred based on average long-term statistics. These stresses were subjected to an identical crack growth analysis. An evaluation could then be made of the possible variation within a given block of flights by ratioing the sum of the growth rates computed by the two methods. The effect of block size was included in the evaluation by using a window technique where the window size represents the number of flights within the total block being evaluated. Figure 2 shows an example for a window size of 20 flights.

The window is translated one flight at a time until all combinations of 20 sequential flights have been evaluated. This evaluation is made by comparing the sum of the growth rates of the measured data to the sum of the growth rates of the average data within a window. Using this technique, the most severe group of 20 flights relative to the average statistical data may be isolated. Data were generated at various crack lengths but the resulting severity factors were essentially independent of crack length. Severity factors of the most severe and the least severe blocks for various block size are shown in Fig. 3. As expected, when the block size is reduced, the probability of encountering blocks with large deviations away from "average" data is increased. For block sizes greater than 300 flights (approximately 1500 flight hr), average statistics appear satisfactory. For blocks smaller than 300 flights, variability should be considered. Single flight variation from the average was found to be as much as 15 times as severe as the average. This variation in severity for

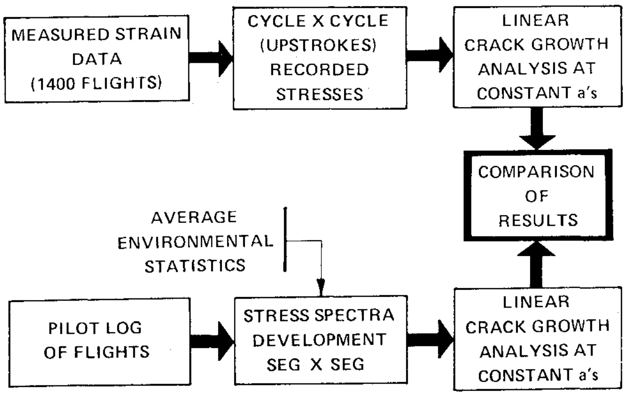

Fig. 1 Flight load variations study—approach.

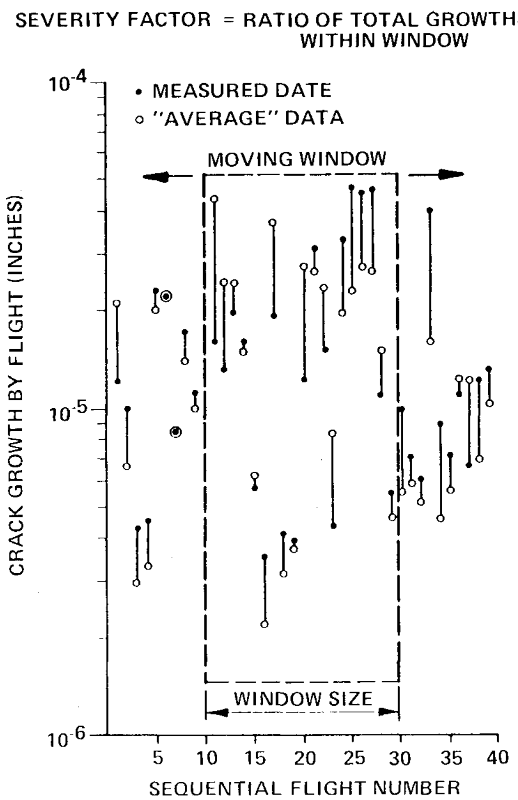

Fig. 2 Severity factor, moving window technique.

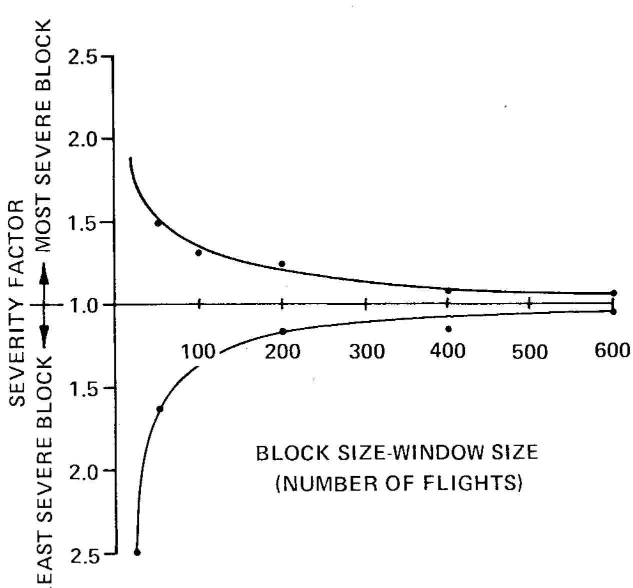

Fig. 3 Severity factor vs number of flights.

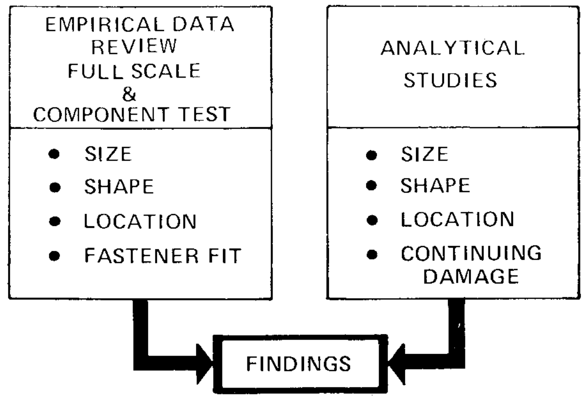

Fig. 4 Flaw model and fastener condition variables.

Table 1 Effect of flaw size, shape, and location results Relative initial flaws to produce equal crack growth interval

Flaw configuration	Equivalent flaw, in.
Single-edged constant ¼ circle	0.05
Single-edged variable shape	0.019
Double-edged variable shape	0.0067
Single-edged thru the thickness	0.0032
Double-edged thru the thickness	0.0016

short usage periods strongly suggests that additional safety factors should be considered when specifying structural inspections with short intervals.

Effect of Flaw Shape and Fastener Condition

The degree to which any crack growth analysis will predict the real-world occurrence is obviously dependent upon being able to simulate adequately the initial and subsequent propagation stages of the crack. Initial shape, size, location, and growth mode of common corner cracks originating at fastener holes are recognized variables which can significantly influence the overall calculation.[2,3] This is particularly true for small ($c<0.10$) corner cracks. This study was conducted using empirical data obtained from two full-scale fatigue test articles which had experienced widespread fatigue cracking, and numerous component tests specifically designed to observe crack propagation. Analytical studies were conducted as a parallel effort to assess the sensitivity of the analysis to the important variables as shown in Fig. 4.

Flaw Shape

Crack data, recorded during nondestructive inspection of two full-scale C-5A fatigue test articles, were reviewed to assess the shapes in which cracking was demonstrated to occur for the overlap spanwise splice joints typical of the C-5A transport. Two important observations resulted from this assessment. First, in a relatively high percentage (18%) of the cases, both the inner and outer panels were cracked. This supports the MIL-A-8344 initial damage assumption specified for dependent cracking when holes are drilled on assembly. Also, the high incidence (71%) of cases shown to have a crack propagating from the opposite side of the hole following element failure supports the continuing damage assumptions of MIL-A-83444.

The results of an analytical study to assess the effects of flaw size, shape, and location are shown in Table 1. The 0.05 in. quarter circular initial flaw specified in MIL-A-83444 for cracks at holes was chosen as the baseline for this study. Crack growth data for each of the variable shapes[2] shown were generated using a typical flight-by-flight stress spectra for an inner wing lower surface location. The thickness of the structural member analyzed was 0.25 in. The equivalent initial

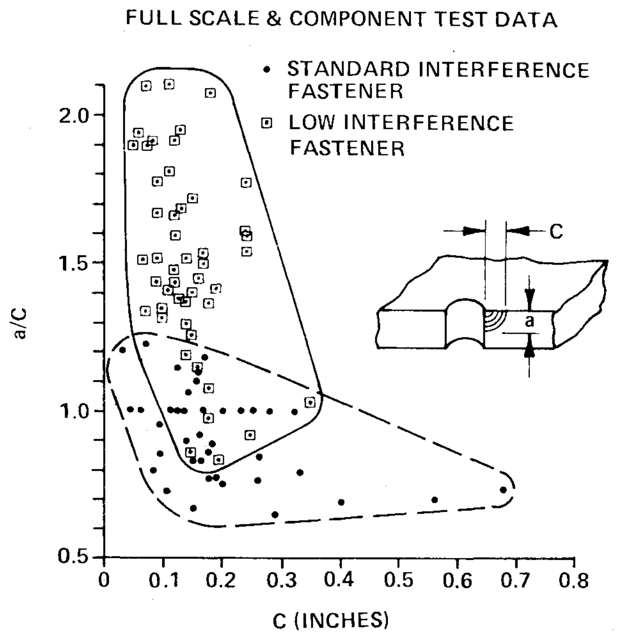

Fig. 5 Empirical review of fastener condition effects on flaw shape.

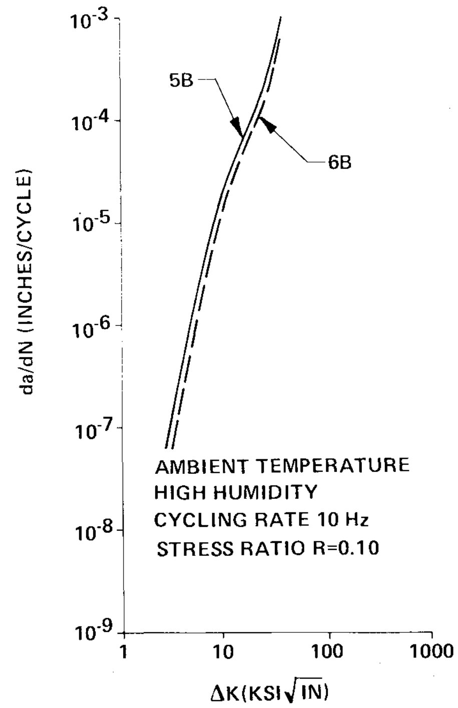

Fig. 6 Variation in constant amplitude da/dN within same extrusion.

flaw was varied to find the size that gave life equal to that of the 0.05 in. quarter circle shape.

It is apparent that the effect of the flaw configuration is significant, and suggests that the 0.05 in. quarter circular crack specified in MIL-A-83444 is not overly conservative for other initial flaw shapes.

Fastener Condition

The effects of fastener interference on the shape of cracks propagating from holes was assessed by reviewing the results of tests conducted with the fasteners installed at known interference levels. Figure 5 illustrates the effect of fastener interference on the shape of cracks propagating from fastener holes. Crack shape data were obtained from photographs taken during fractographic examination of the cracks. The data were obtained from two full-scale fatigue test articles

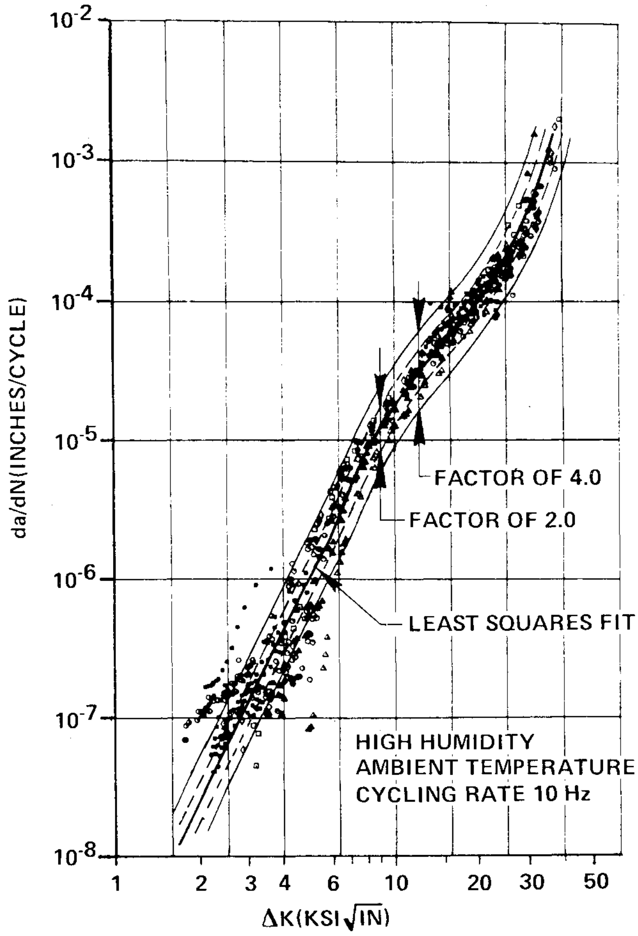

Fig. 7 Variation in growth rate at $R = 0.10$.

where the fasteners were installed with standard interference, and component tests where the fasteners were installed intentionally with near-zero interference. Cracks propagating from holes with low-interference fasteners tend to grow elliptical ($a/c = 1.0$). Cracks propagating from holes with standard interference tend to grow more nearly quarter circular in shape ($a/c = 1.0$). These data lead to the conclusion that effective fastener interference is required to justify a quarter circular crack growth mode assumption.

Material Fracture Properties Variation

It has long been recognized that material property variation occurs in the static strength properties of aluminum alloys. A generally accepted procedure to establish static material properties for design purposes with a known confidence and probability is utilized in the U.S. Department of Defense Materials Handbook 5 (MIL-HDBK-5). No such statistical procedure with universal acceptance exists for the application of material fracture properties data. The purpose of this study was, therefore, to quantify the variation that may exist in material fracture properties of 7075-T6511 extrusion and evaluate this potential effect on the structural integrity of the aircraft.

Variation in Test Crack Growth Rate da/dN

It is generally recognized that some of the apparent variation that occurs in material properties crack growth rate data is due to uncontrollable factors in testing techniques and data recording.[4] Four tests were conducted to evaluate the variation in constant amplitude da/dN obtained from nominally identical tests. The results of two of these tests, each conducted using material from the same 7075-T6511 extrusion with every effort made to eliminate testing variation, are shown in Fig. 6. The variation (scatter) was shown to represent a range of approximately 2.0 on da/dN.

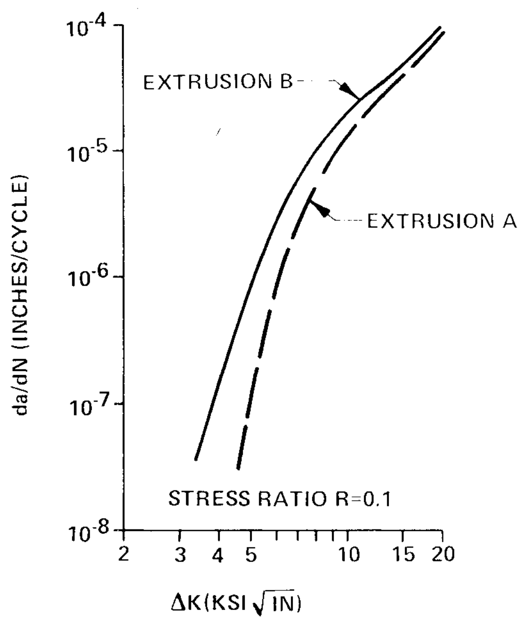

Fig. 8 Variation in growth rate for different extrusions.

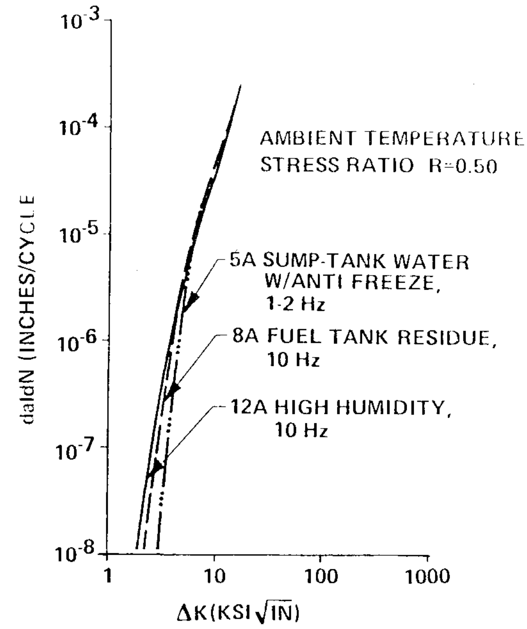

Fig. 10 Effects of environment on growth rate at $R = 0.50$.

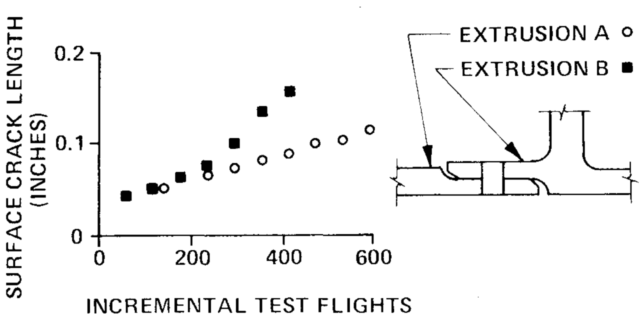

Fig. 9 Variation in spectrum crack growth.

These tests established the baseline variation in da/dN which can be expected in nominally identical tests and provided valuable insight into the interpretation of other comparative tests. The results of these tests were used as an aid in separating the effects of test scatter from apparent variation for other variables evaluated.

Forty-five tests, each from a different material stock, were conducted to evaluate the variation in constant amplitude da/dN of 7075-T6511 extrusions. The tests were conducted in consistent environment at a ratio of minimum-to-maximum stress of $R = 0.1$. The data obtained from the tests conducted at $R = 0.1$ are shown in Fig. 7. A least-squares curve fit is shown representing the mean of the data. Total scatter bands of 2.0 and 4.0 are also shown. A total scatter of 4.0 represents a factor of ± 2.0 from the mean. Nearly all of the data for growth rates above 10^{-6} in./cycle fall within a total scatter of 4.0. There are numerous data points falling outside a total scatter of 4.0 for growth rates less than 10^{-6} in./cycle; however, previous investigations[4] have shown this not to be unusual.

Variation in Spectrum Crack Growth Rate

The variation in constant amplitude da/dN for different material stock raises the question of whether or not this same characteristic would hold true under spectrum loading. Two different basic extrusions which demonstrated significantly different constant amplitude da/dN were selected for this evaluation. A comparison of the da/dN test results is shown in Fig. 8. A test specimen using these two different material stocks was fabricated such that each specimen would receive identical spectrum loading. This was accomplished by using these two pieces of material to form the corner splice in a box-beam test specimen. The splice was representative of the actual aircraft spanwise splice geometry and fastener system. The box beam was loaded with axial and torsional loads to simulate a flight-by-flight stress history based on measured real-time history data obtained from a service aircraft. Crack growth measurements from preflawed holes were recorded during the test and presented in Fig. 9. Examination of the test results presented in Figs. 8 and 9 shows that the same extrusion that demonstrated the slowest constant amplitude crack growth rate also demonstrated the slowest spectrum crack growth. Spectrum crack growth differed by a factor of approximately 2.0. The results of these tests provide positive evidence that the same trends exhibited in constant amplitude da/dN testing are also present under spectrum loading.

Variation in Fracture Toughness (K_{Ic} and K_c)

A total of 121 tests were conducted to assess the effects of temperature and basic material variation on fracture toughness. All tests conducted to assess the effects of temperature utilized test specimens fabricated from the same basic 7075-T6511 extrusion. The variation in fracture toughness for the temperatures tested was well within normal test scatter and was not considered significant. Ninety-one valid tests, each from a different material stock, were obtained to evaluate the variation in fracture toughness (K_{Ic} and K_c) of 7075-T6511 extrusions. Valid K_{Ic} values obtained from compact tension specimens ranged 19.3-31.6 ksi $\sqrt{\text{in.}}$ K_c tests were conducted for two different thicknesses using center-cracked tension specimens 24 in. long by 8 in. wide. The variation in fracture toughness was shown to be quite large but not unusual for this material.

Effects of Environment and Temperature on Growth Rate (da/dN)

It is usually not practical to simulate exactly the range of possible influencing parameters to which the front of a crack in an aircraft structure will be exposed to over a prolonged period of time. It is desirable, therefore, to evaluate the potential effects that variations in these parameters may have on crack growth rate, and then select for analysis purposes the data, or combinations of data, most applicable to the specific aircraft's operation. The variation which can be expected to occur in nominally identical tests was found to be approximately 2.0 as shown in Fig. 7. It therefore becomes important to temper any judgment rendered with the realization that the apparent variations which do occur may be due all or in part to test scatter.

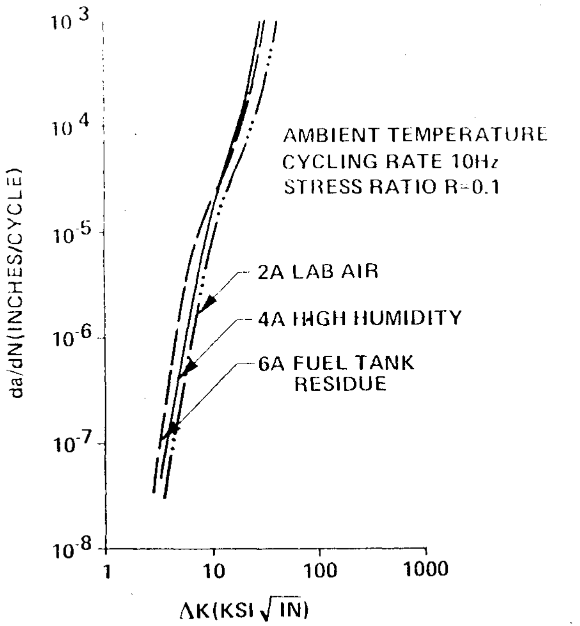

Fig. 11 Effects of environment on growth rate at $R=0.10$.

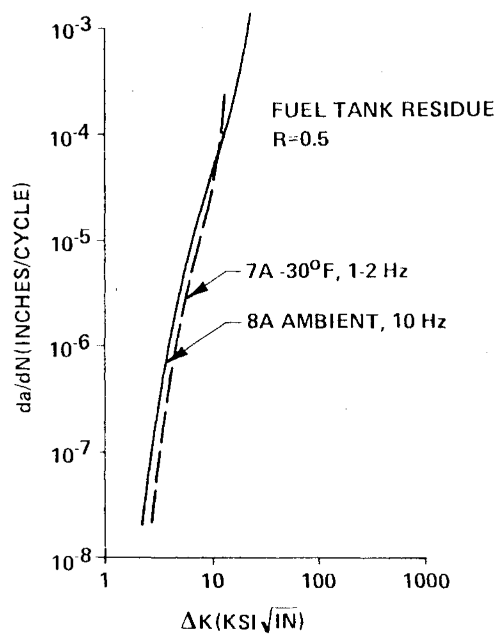

Fig. 12 Effects of temperature on growth rate at low temperature.

Effect of Environment

The environments selected for these comparative tests were laboratory air, high humidity (80-100%), actual fuel tank residue, and Air Force specification sump tank water.[5] Fuel tank residue was obtained from a force aircraft and analyzed to determine the chemical composition. Chemical analysis showed the fuel tank residue to be primarily a solution of water and 14% anti-icing agent with an odor similar to fuel. The solution had a freezing point of 19°F and a pH of 6.0. The results of the tests are presented in Figs. 10 and 11. The most significant finding from these tests was that the high-humidity testing gives results very similar to the actual fuel tank residue environment. Tests conducted in laboratory air demonstrated the slowest growth rate, as expected. Tests conducted in the USAF sump tank water surprisingly demonstrated growth rates less than either the high-humidity or fuel tank residue environments over most of the growth range tested.

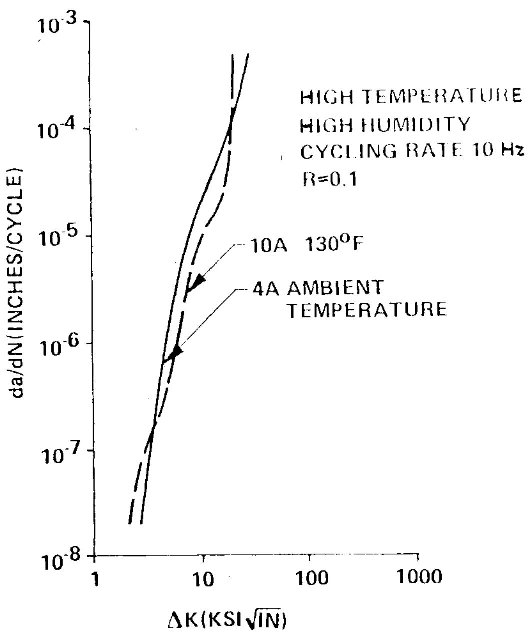

Fig. 13 Effects of temperature on growth rate at high temperature.

Effect of Temperature

Constant amplitude testing was conducted to evaluate the effects of high and low temperature on growth rate da/dN. The results of the high- and low-temperature tests are shown compared to ambient temperature tests in Figs. 12 and 13, respectively. The low temperature of $-30°F$ is considered to be the lower limit of the temperature that the fuel of a large transport aircraft will attain on a cold day. This temperature was selected based on flight tests conducted by Pan American Airways using a transport aircraft during the winter of 1960 and 1961.[6] The $-30°F$ tank temperature was exceeded on only 10 of 2811 flights during the PAA tests and the coldest tank temperature recorded was $-38°F$. The cold-temperature tests produced slower crack growth rates than the ambient temperature tests in the low ΔK range. It was concluded that while cold temperature has a moderate effect on constant amplitude growth rate it is not a significant factor, since only a relatively small portion of the total overall spectrum crack growth could be attributed to the high-altitude cruise segments where the tank is cooled. The results of the high-temperature tests are questionable since the data show an inconsistent trend at different values of ΔK, but little overall effect is observed if the crack growth rates are combined for all values of ΔK. Here again, the aircraft exposure expected at 130°F is minimal.

Effect of Maximum Load Level and Fracture Toughness on Crack Propagation Analysis

Analytical sensitivity studies were conducted to evaluate the effect of variations in maximum load level and fracture toughness on critical crack length and the associated crack growth interval. The effect of variations in maximum load level and fracture toughness on critical crack length was found to be significant. The analytical critical crack length was found to be 0.80 in. at the limit stress and 2.9 in. at the maximum stress occurring once in the missions. However, the total variation in the crack propagation interval was less than 10% when calculated from an initial flaw of 0.05 in. It was concluded that variations in fracture toughness and load level have a significant effect on critical crack lengths, but crack propagation life is relatively insensitive to these variables even though the range of the variables may be relatively large. Crack propagation life was found to vary approximately 2% for each 10% variance in either fracture toughness or maximum loads.

Table 2 Spectrum development sensitivity studies

Analytical parameters	Baseline	Variations
Cycle sequence	Low-high	High-low, low-high-low, random
Cycle definition	Mean-min-max-mean	Mean-max-min-mean, max-min-max, min-max-min
Variable-stress increment size (bandwidth), psi	1000	500, 1500
Low-stress truncation, psi	500	250, 1000
High-stress truncation	1/flt/500 h	1/flt; 1/flt/1000 h

Table 3 Results of sensitivity studies for 0.05 in. crack length

	Cycle sequence			Cycle definition			Incremental bandwidth		Truncation stress			
Baseline	Low-high						1000 psi		Low 500 psi		High 1/500 h	
Variation	High-low	Low high-low	Random				500	1500	250	1000	1/flt	1/1000 h
Inner wing	1.08	1.01	1.02	0.84	1.23	1.06	0.95	1.06	0.97	0.98	0.74	1.04
Outer wing	1.22	1.05	1.03	0.92	1.08	1.00	1.01	1.06	0.99	1.01	0.70	1.03

Ratio >1.0 indicates faster growth rate than baseline.

This conclusion on insensitivity to critical length may be highly dependent on local detail design factors however. Also, long critical crack lengths are naturally very desirable from an inspection viewpoint.

Analytical Spectrum Development Procedures

A study was conducted to evaluate the assumptions and procedures inherent to the development of analytical stress spectra. This study consisted of sensitivity analyses and comparative tests.

Analytical Sensitivity Studies

Sensitivity analyses were conducted to provide a quantitative evaluation of the effects of each individual parameter of the spectrum development procedure on the crack growth prediction. Crack growth rates for the selected baseline procedure were first generated. Each parameter of the baseline spectrum was then systematically varied and the crack growth rate computed for the revised spectrum. A list of the parameters studied and a description of the baseline procedure is presented in Table 2. Also presented in Table 2 is a description of the different variations of each parameter studied. The studies were conducted for both an inner and outer wing location using crack growth rates computed at five crack lengths ranging 0.005-0.30 in. The results of the analytical studies for the inner and outer wing locations at one crack length (0.05 in.) are summarized in Table 3. The sensitivity index is defined as the ratio of the growth rate of the studied variable to the growth rate of the baseline spectrum. An index value greater than 1.0 means that the studied variation is more severe than the baseline.

Random vs Ordered Flight-by-Flight Testing

The results of the analytical studies indicated that there was no appreciable difference in growth rate between a low-high ordered flight-by-flight spectrum and a random sequence. Testing was conducted to verify this finding. The test specimen was a center-cracked plate 0.25 in. thick, 6 in. wide, and 24 in. long. The test material was a 7075-T6511 extrusion.

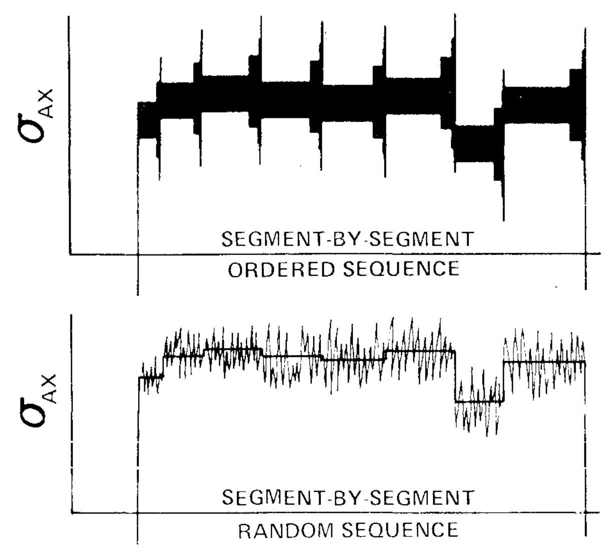

Fig. 14 Random vs ordered flight-by-flight test spectra.

The random and ordered spectra are described pictorially in Fig. 14. The test was conducted by first applying the low-high ordered spectrum, changing to the random spectrum, and then returning to the low-high ordered spectrum. Use of the same specimen in this manner eliminated possible test-to-test scatter and gave a direct comparison. The results of these tests indicated there was no appreciable difference in the growth rate of the two spectra. These results are consistent with the results of previous investigation.[7]

Effects of Compression Cycles and Compression Hold Time

This study was conducted to ensure that the beneficial retarding effects of the overload cycles in a typical lower wing surface stress spectrum were not analytically overestimated. Previous investigations[8,9] have shown that compression-compression cycles and compression hold time may reduce the

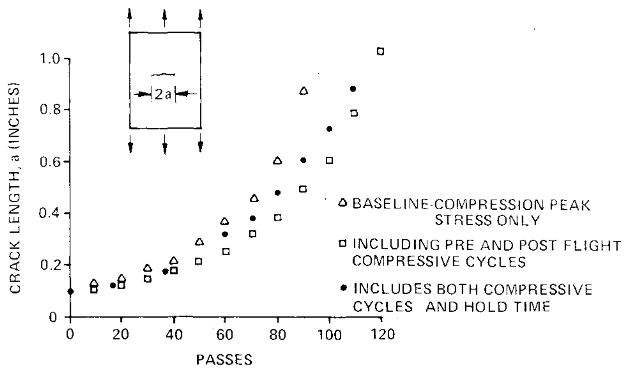

Fig. 15 Compression cycle and cycle hold time test results.

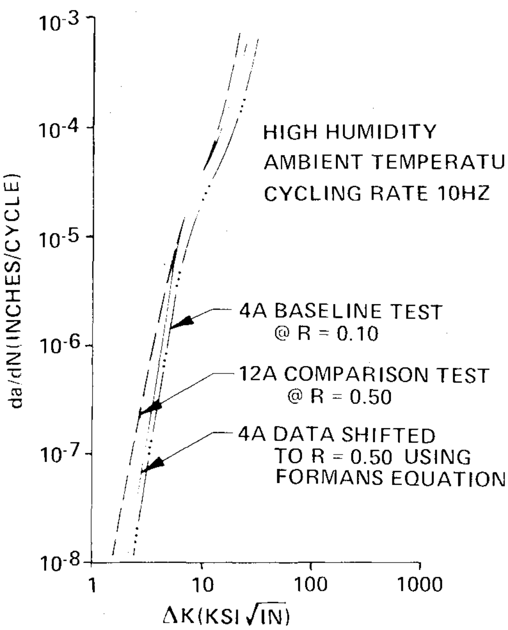

Fig. 16 Effects of *R* shift.

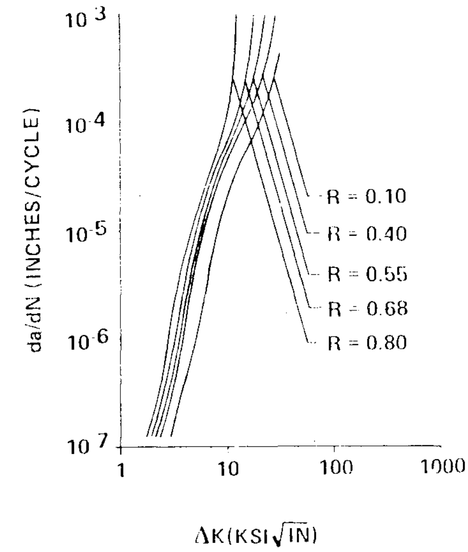

Fig. 17 Parametric d*a*/d*N* vs Δ*K* test results.

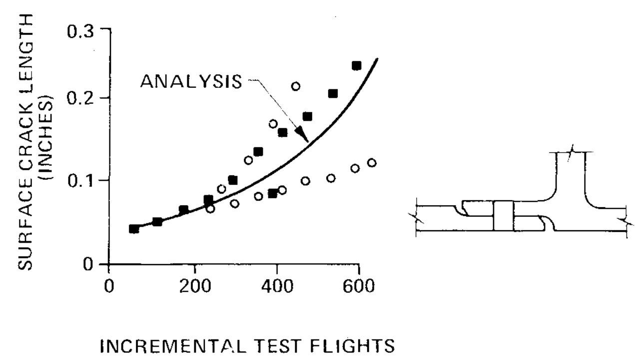

Fig. 18 Comparison of analytical and measured crack propagation.

retarding effects of the tensile overloads. These possible effects are not accounted for using current retardation models. The effects of compression-compression cycles and compression hold time on spectrum crack growth were evaluated using three center-cracked test specimens fabricated from a common 7075-T6511 extrusion. The results of these tests are presented in Fig. 15. All test specimens were 0.25 in. thick, 6.0 in. wide, and 24 in. long. The baseline test had only the maximum preflight compression stress applied between flights. The second test had compression-compression cycles applied during all preflight and postflight segments. The third test had the preflight and postflight compression plus a 15 min hold time in compression to simulate the aircraft wing lower surface condition between flights. These tests indicate that there are no adverse effects from the compression cycles not accounted for in the analysis. Although these data are considered to be part of the normal test-to-test scatter, the effect of the compression cycle, if any, was to decrease the growth rate.

Crack Propagation Law (Equation)

A desirable goal in the development of any analytical procedure is to have the procedure applicable to the solution of as many general problems as possible. It is generally accepted that the rate of crack propagation is a function of the stress intensity factor range ΔK for a given material. The range of ΔK is a function of applied stress, crack length, and local geometric features in the vicinity of the crack. Crack propagation is normally computed by integrating da/dN throughout the spectrum using mean constant amplitude da/dN vs ΔK data. This procedure will yield acceptable results as long as the crack propagation equation employed adequately accounts for the effects of changes in the minimum-to-maximum stress ratio R. Several procedures have been developed and used to account for the shift in stress ratio. Two such procedures commonly used are Forman[10] and Walker.[11] This study evaluated the effects of R shift on constant amplitude growth rate da/dN, and investigated possible methods which could be generally employed to account for the R shift present in typical stress spectra.

Effects of R Shift on da/dN

Constant amplitude da/dN growth rate data obtained from the results of several different 7075-T6511 extrusion tests were used to evaluate the effects of R shift. The results of two typical tests conducted using material from the same basic extrusion at $R=0.1$ and 0.5 are shown in Fig. 16. Also shown in the figure is the da/dN for test 4A ($R=0.1$) shifted to $R=0.5$ using the Forman[10] equation. The shifted data do not match the $R=0.5$ test data at the lower values of ΔK. Although some of this mismatch could be due to test scatter, indications are that potentially unconservative results can be obtained with this procedure.

Development of Parametric da/dN Data

Constant amplitude da/dN vs ΔK was generated to establish crack growth rate data at five different R values. The

Table 4 Analysis sensitivity summary

Variable	Sensitivity of overall analysis
Flight load variations	High[a]
Flaw size and shape	High
Fastener condition	Moderate
Crack growth da/dN	High
Fracture toughness	Low
Environment	Moderate
Temperature	Low
Maximum load level	Low
Spectrum development	Moderate
Spectrum crack propagation law	High

[a]Short term only.

data were used to establish parametric values of da/dN which can be used with an interpolative routine and thus eliminate the need to rely on a generalized equation which extrapolates from a single R value. The tests were conducted from several different material sources to account for the variations which may occur in material properties. The parametric da/dN vs ΔK data are shown in Fig. 17 and represents the mean of the data obtained at each R value. The results of 54 separate tests were used to establish these curves.

Analytical vs Test Crack Propagation Comparisons

The overall ability of the analysis procedure to predict crack propagation rates in complex joints representative of service aircraft was evaluated. This was accomplished by using the crack growth data obtained fractographically from a box beam test program. The tests duplicated all of the features of in-service aircraft crack propagation considered practical to represent in the laboratory. Complex bending and torsion loading interactions measured during actual flight were precisely applied to the test box beams. Fourteen different cracks were used for analytical comparison. Comparison of the analytical to measured crack propagation data for four of the cracks is shown in Fig. 18. It is particularly interesting to note that the measured crack propagation data appear to fall in two distinct groups. One group, which is comprised of the two slower growing cracks has a growth rate approximately one-half the other group. This trend is consistent with the constant amplitude growth rate data obtained for the two extrusions involved, as discussed earlier. The analytical prediction was based on mean parametric da/dN data and is shown to represent the approximate mean of the measured spectrum crack growth data. Retardation effects were accounted for by use of the Hsu model.[12] This comparison clearly illustrates the need for reliable material properties da/dN data.

Summary

This program has vividly illustrated the need to have reliable data on certain variables to which the service aircraft crack propagation predictions are sensitive. Material properties data were shown to exhibit significant variation and are thus considered to be the primary area of uncertainty. The sensitivity of the overall analysis to variations in the variables which have the largest range of variation is summarized in Table 4.

The effect of variation in the remaining variables requires less consideration and in most cases can be adequately accounted for by the application of sound engineering judgment. It was shown that if reliable data and good analytical procedures are available, crack propagation analysis can adequately predict complex problems representative of in-service aircraft.

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