METHODS AND MODELS FOR PREDICTING FATIGUE CRACK GROWTH UNDER RANDOM LOADING

Chang/Hudson, *editors*

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Foreword

The papers in this volume present in detail the methodology and procedure utilized by each individual in the prediction of the fatigue crack growth behaviors and lives for random flight spectrum test cases used in the round-robin analysis conducted by ASTM Task Group E24.06.01 on Application of Fracture Data to Life Prediction. The objective of this round-robin analysis was to assess whether data from constant-amplitude fatigue crack growth tests on center-cracked-tension (CCT) specimens can be used to predict fatigue crack growth lives of CCT specimens subjected to random load sequences.

Related ASTM Publications

Fatigue Crack Growth Measurement and Data Analysis, STP 738 (1981), $39.00, 04-738000-30

Effect of Load Spectrum Variables on Fatigue Crack Initiation and Propagation, STP 714 (1980), $27.00, 04-714000-30

Part-Through Crack Fatigue Life Prediction, STP 687 (1979), $26.25, 04-687000-30

Fatigue Mechanisms, STP 675 (1979), $65.00, 04-675000-30

Service Fatigue Loads Monitoring, Simulation, and Analysis, STP 671 (1979), $29.50, 04-671000-30

Fatigue Testing of Weldments, STP 648 (1978), $28.50, 04-648000-30

Corrosion-Fatigue Technology, STP 642 (1978), $32.00, 04-642000-27

Cyclic Stress-Strain and Plastic Deformation Aspects of Fatigue Crack Growth, STP 637 (1977), $25.00, 04-637000-30

Fatigue Crack Growth Under Spectrum Loads, STP 595 (1976), $34.50, 04-595000-30

Handbook of Fatigue Testing, STP 566 (1974), $17.25, 04-566000-30

Editorial Staff

A Note of Appreciation to Reviewers

This publication is made possible by the authors and, also, the unheralded efforts of the reviewers. This is a body of technical experts whose dedication, sacrifice of time and effort, and collective wisdom in reviewing the papers must be acknowledged. The quality level of ASTM publications is a direct function of their respected opinions. On behalf of ASTM we acknowledge with appreciation their contribution.

ASTM Committee on Publications

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Introduction

One of the principal missions of ASTM Task Group E24.06.01 on Application of Fracture Data to Life Prediction is to provide the members of ASTM Committee E-24 on Fracture Testing with the opportunity to evaluate their in-house capability for predicting fatigue crack growth. As a first step in this evaluation, participating members used constant-amplitude-loading data from compact specimens to predict fatigue crack growth in surface-cracked specimens subjected to constant-amplitude loading. This prediction round-robin analysis culminated in *Part-Through Crack Fatigue Life Prediction, ASTM STP 687,* where the conclusion was reached that, for constant-amplitude loading, data from compact specimens can be used to predict fatigue crack growth in surface-cracked specimens.

Next, participants used constant-amplitude fatigue crack growth data from center-cracked specimens to predict fatigue crack growth in center-cracked specimens subjected to block loadings. Predictions were generally quite good for this round-robin analysis.

Participants subsequently used constant-amplitude fatigue crack growth data from center-cracked specimens to predict fatigue crack growth in center-cracked specimens subjected to pure random loading. Random loading was selected for this round-robin analysis because it is typical of the type of loading which many structures experience in service. Further, random loading produces all of the crack retardation and acceleration effects which cracks experience when propagating in a structure. Thus participants were able to evaluate the ability of their in-house programs to account for these effects.

This volume presents the results of this latest prediction round-robin analysis. It contains six papers. The first paper presents the data given to the participants and a comparison of the predicted and test results. The remaining five papers describe the in-house procedures used to predict the lives of 13 test specimens. Review of these five papers shows that, although six independent procedures were used in making the predictions, the ratios of the predicted lives to the test lives ranged from only 0.58 to 2.52 for all predictions. Further, the vast majority of the ratios were much closer to one. Considering that the scatter in fatigue crack growth rates can range from two to four in constant-amplitude-loading tests, these ratios are quite good.

Special thanks are due G. A. Vroman, chairman of Task Group E24.06.01, for his support and encouragement of this round-robin analysis, and to J. B.

Chang for his leadership in distributing the basic data and compiling the round-robin results.

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J. B. Chang[1]

Round-Robin Crack Growth Predictions on Center-Cracked Tension Specimens under Random Spectrum Loading

REFERENCE: Chang, J. B., **"Round-Robin Crack Growth Predictions on Center-Cracked Tension Specimens under Random Spectrum Loading,"** *Methods and Models for Predicting Fatigue Crack Growth under Random Loading, ASTM STP 748,* J. B. Chang and C. M. Hudson, Eds., American Society for Testing and Materials, 1981, pp. 3–40.

ABSTRACT: A round-robin analysis was conducted by ASTM Task Group E24.06.01 on Application of Fracture Data to Life Prediction to predict the fatigue crack growth in 2219-T851 aluminum center-cracked-tension (CCT) specimens subjected to flight loadings in random cycle-by-cycle format. Baseline data furnished to each participant of the round-robin analysis are described. These data consisted of constant amplitude crack growth data, specimen dimensions, initial crack sizes, and test spectrum tables. Analytical predictions and their correlation to test data are summarized.

KEY WORDS: fatigue crack growth, random flight loadings, 2219-T851 aluminum, center cracked tension specimen, fatigue life predictions

Random flight loadings are of variable amplitude in nature. Various load interactions take place in these loadings that affect the growth behavior of a crack. Significant effects observed by many investigators can be summarized as follows:

1. Tensile overloads cause retardation of the crack growth in general. A sufficiently high overload cycle may stop the growth of a crack completely [*1,2*].[2]
2. Compressive loads in compression-tension load cycles cause the acceleration of the crack growth [*3,4*].

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[2]The italic numbers in brackets refer to the list of references appended to this paper.

3. Compressive loads in tension-compression load cycles reduce the retardation effect caused by the tensile overload [5-7].

For a random flight spectrum as shown in Fig. 1, all of the aforementioned load interaction effects on the crack growth behavior will occur for cracks in the airframe structures. The need to have a crack growth prediction methodology able to account for the load interaction effects is obvious. The implementation of the fracture control plan on airframe structures is a typical example. The airplane structural integrity program specified in Military Standard MIL-STD-1530A [8] requires the ability to accurately predict crack growth under spectrum loadings. To neglect the crack growth retardation caused by the tensile overload can lead to unnecessary weight increases and cost penalties. On the other hand, an unsafe design will result if the acceleration effect and the reduction of the retardation effect caused by the compressive loads are not accounted for in the analysis.

A round-robin analysis to predict fatigue crack growth under random spectrum loadings was recently performed by members of ASTM Task Group E24.06.01 on Application of Fracture Data to Life Prediction. The objective was to determine whether data from constant-amplitude fatigue crack growth tests of center-cracked-tension (CCT) specimens can be used to predict fatigue crack growth lives of CCT specimens subjected to random load sequences. Each member of the Task Group was provided by the author with the material baseline (constant-amplitude) crack growth rate data, specimen dimensions, the initial crack sizes of each test, and the random spectrum

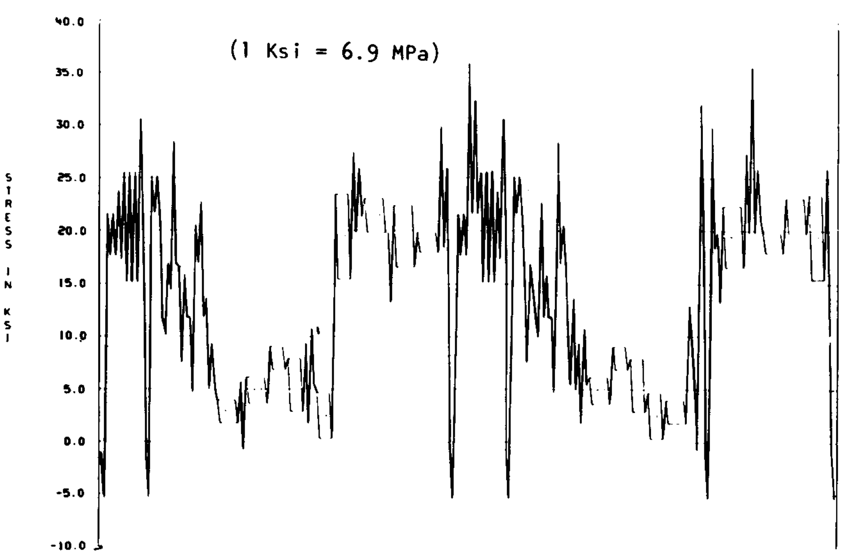

FIG. 1—*Typical aircraft wing spectrum.*

tables. The analytical effort was required for participants to predict the number of loading cycles needed for a crack growing from an initial size to a final size, using their own prediction methodology and computer codes. This paper describes the data furnished to members of the Task Group and summarizes the results of the round-robin predictions.

Experimental Data Base

The experimental data base used in the round-robin analysis was obtained from a research program currently being conducted at Rockwell International, North American Aircraft Division, for the U.S. Air Force [*9*]. The objective of this research effort is to upgrade the crack growth analysis technology required for the implementation of the damage tolerance control procedures for any aircraft system. One of the primary tasks performed in this program was to develop a streamlined fatigue crack growth life prediction methodology used for assessing the damage tolerant ability of a structural component in the detailed design stage.

To aid the formulation of the fatigue crack life prediction methodology, an experimental program was conducted in Phase I of this research program. It consisted of (*1*) baseline (constant-amplitude) crack growth rates and fracture toughness data generation tests, (*2*) constant-amplitude tests with various stress levels and stress ratios, (*3*) single overload/underload or periodic overload/underload tests, (*4*) block loading tests, and (*5*) random cycle-by-cycle spectrum loading tests. All test specimens used in this experimental program were ASTM Tentative for Constant-Load-Amplitude Fatigue Crack Growth Rates above 10^{-8}m/Cycle (E 647-78 T) standard CCT specimens fabricated from 6.35-mm (0.25-in.)-thick 2219-T851 aluminum plates. All plates were from the same lot of material. Figure 2 shows the test specimen configuration. The center notch was fabricated by employing the electrical discharge machining (EDM) process, with the maximum width of the notch less than 0.245 mm (0.01 in.).

The entire test program was conducted on MTS Fatigue Testing Systems. The EDM slot in each specimen was precracked to produce a crack approximately 7.62 mm (0.3 in.) in length including the EDM slot. Precracking was performed under constant-amplitude loading cycled at 55.16 MPa (8 ksi) maximum stress, with stress ratio $R = 0$. All tests were run at a cyclic rate of 6 Hz in ambient laboratory air at room temperature. Cyclic crack growth measurements were obtained using visual optics. The resolution of the crack length measurement was approximately 0.13 mm (0.005 in.).

Baseline Crack Growth Rates and Fracture Toughness Data

Baseline fatigue crack growth rate data presented in the fatigue crack growth rate versus the stress-intensity-factor range (da/dN versus ΔK) plot

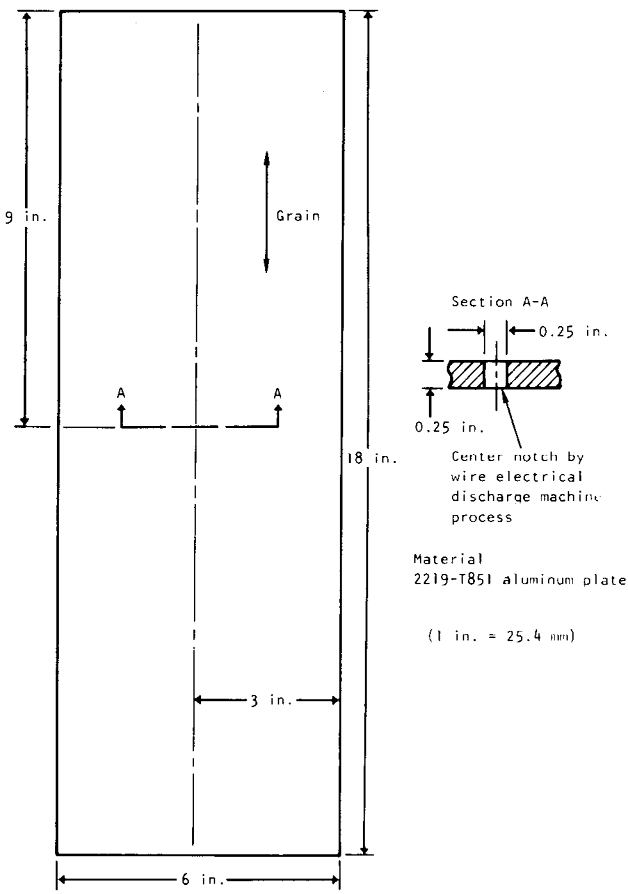

FIG. 2—*Test specimen configuration.*

together with the fracture toughness (K_c) value for 2219-T851 aluminum were furnished to members of the Task Group. Figure 3 shows the baseline crack growth rate data. They were obtained from constant-amplitude tests with the maximum stress kept at 138.1 MPa (20 ksi) level. The stress ratios tested were $R = 0.01$, 0.2, 0.3, and 0.7. Figure 3 was plotted by an interactive graphics computer program, PLOTRATE [*10*]. It employs the seven-point polynomial method as recommended by ASTM E 647 to determine da/dN from the crack size versus elapsed cycles (a versus N) data. Values of ΔK were calculated using the ASTM standard formula for CCT specimens.

The fracture toughness value for the 6.35-mm (0.25-in.)-thick 2219-T851 aluminum plate used in the test program was determined from the static fracture tests employing the same CCT specimens. An average value, K_c = 70.85 MPa $\sqrt{\text{m}}$ (65 ksi $\sqrt{\text{in.}}$), was obtained.

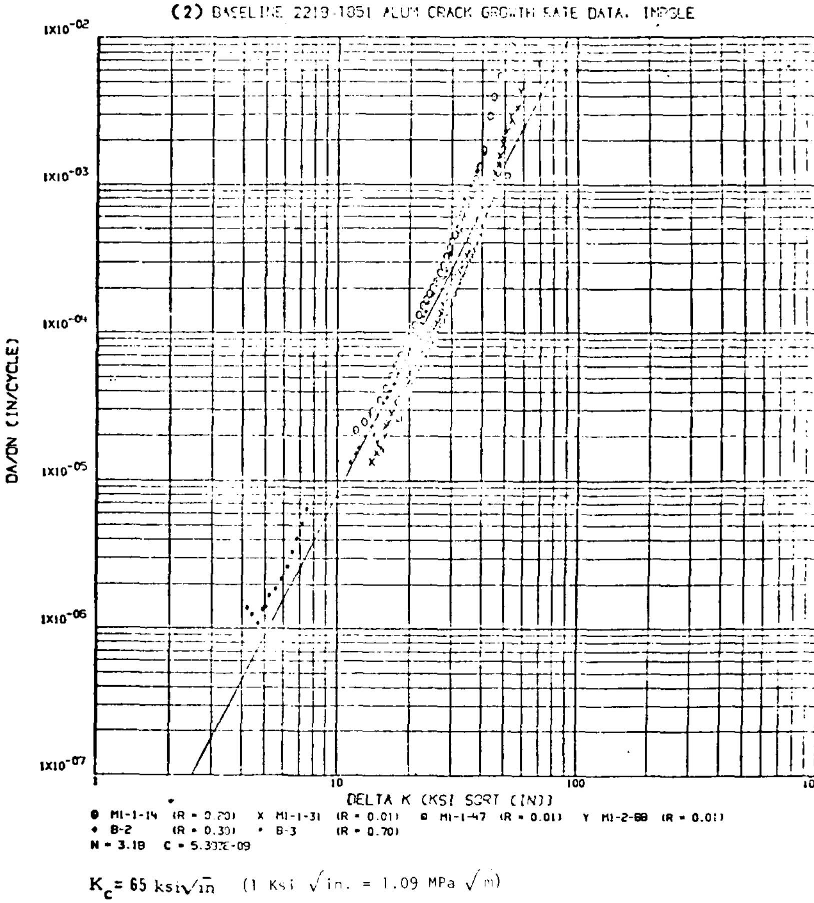

FIG. 3—*2219-T851 aluminum baseline fatigue crack growth rate data and fracture toughness data.*

Random Spectrum Data

Random load spectra furnished to members of the Task Group were presented in tabular format. The spectrum tables presented in Tables 5 to 8 are the Air-to-Air (A-A), Air-to-Ground (A-G), Instrumentation and Navigation (I-N), and Composite missions of a typical fighter aircraft. Numerical values in these tables are in the form of percentage of the design limit stress (DLS). Three levels of DLS were tested: DLS = 138 MPa (20 ksi), 207 MPa (30 ksi), and 276 MPa (40 ksi). Table 9 is the composite mission of a typical transport aircraft. Numerical values in the table are tensile or compressive stresses.

The fighter spectra were generated with the aid of a stress history simulation computer program, SPECGN 1 [11], using the baseline load spectra of a typical fighter aircraft as inputs. SPECGN 1 generates stress histories employing a random noise theory. It has been shown that the measured flight load factor histories can be simulated very well by the load factor time histories developed using random noise theory [12]. The random noise theory approach requires that the random load factor time history be generated possessing a specific power spectral density (PSD) shape, and mean and root-mean-square (RMS) level. An advantage of this approach is that both the desired exceedance content and frequency content of the process can be preserved. The preservation of the frequency content assures that proper coupling of peaks and valleys is attained.

Baseline load spectra for a typical fighter aircraft were used to generate random cycle-by-cycle stress histories in terms of peaks and valleys for the A-A, A-G, I-N, and Composite missions. The generated peaks and valleys are in the form of percentage of DLS as shown in the spectrum tables. The baseline PSD corresponding to A-A, A-G, and I-N missions are shown in Figs. 4, 5, and 6, respectively. A portion of the simulated sequence of peaks and valleys of each of these missions is shown in Figs. 7, 8, and 9. Using these simulated stress histories, a unitblock was constructed. The unitblock is a block of flights considered to be most representative of the mission throughout the life time of the aircraft. A 50 flight unitblock for each mission was constructed by selecting 50 stress segments, each corresponding to a duration of one flight and each beginning with a valley stress and ending with a peak stress. The ground-air-ground (G-A-G) cycles were inserted at the beginning and end of each flight in such a way that in each flight, the first valley was replaced by and the last peak was followed by the ground load. The exception is that, for the first flight of the unitblock, the first simulated

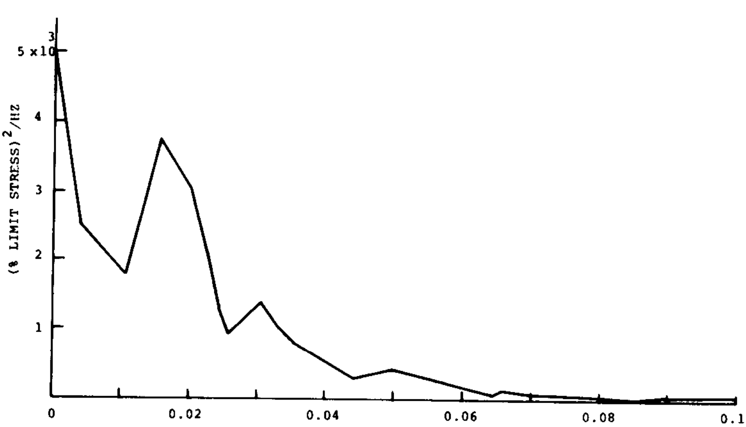

FIG. 4—*Power spectral density for a Fighter A-A mission.*

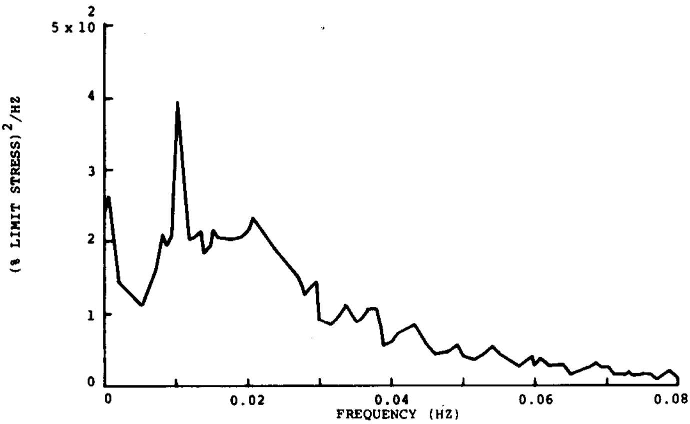

FIG. 5—*Power spectral density for a Fighter A-G mission.*

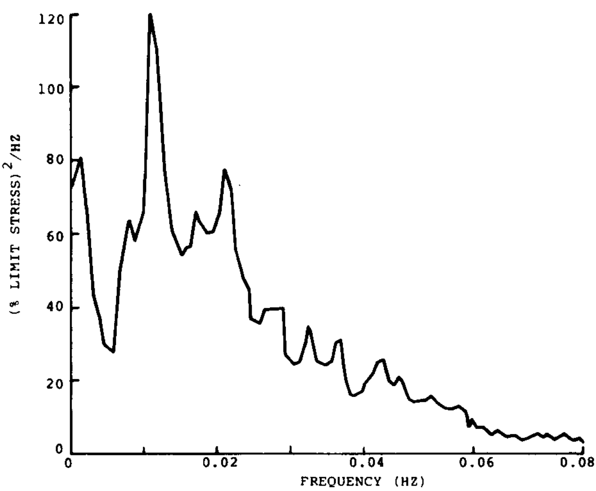

FIG. 6—*Power spectral density for a Fighter I-N mission.*

stress rise was replaced by a stress rise consisting of the ground load and 70 percent of DLS. The ground loads were taken to be −5 percent of DLS for A-A and I-N missions, and −10 percent of DLS for A-G mission.

The fighter composite unitblock was constructed in the following form:

$$11\,(A\text{-}A)_{1\text{-}11} + 11\,(A\text{-}G)_{1\text{-}11} + 3(I\text{-}N)_{1\text{-}3} + 11\,(A\text{-}A)_{12\text{-}22} + 11(A\text{-}G)_{12\text{-}22} + 3(I\text{-}N)_{4\text{-}6}$$

where $11(A\text{-}A)_{1\text{-}11}$ designates 11 flights of A-A missions taken from the first flight to the 11th flight of the A-A mission baselines.