With less of a contact arc due to increased clearance, the radial stress has to be larger over the contact region to react the total load. The effect of clearance on the hoop stress is interesting. Even though the peak value is not influenced, the peak is spread over a larger circumferential region with increased clearance. With an increased volume of material at a high stress level, there is an increased likelihood of material failure. Also, with increased clearance, the region of maximum hoop stress moves toward $\theta = 0$ deg. It is expected that clearance levels would strongly influence where around the circumferential location damage or failure would occur in an actual joint.

Concluding Remarks

From the results of this study, it can be concluded that pin elasticity is not an important variable. A rigid-pin assumption is accurate. The cosinusoidal assumption can be inaccurate. However, friction and clearance effects must be included in any realistic analysis.

Acknowledgments

The work reported on here was supported by the NASA—Virginia Tech Composites Program, Cooperative Agreement NAG-1-343 with the NASA Langley Research Center.

References

[1]Hyer, M.W. and Klang, E.C. "Contact Stresses in Pin-Loaded Orthrotropic Plates," *International Journal of Solids and Structures*, Vol. 21, No. 9, 1985, pp. 957-975.

[2]Chang, F.K., Scott, R.A., and Springer, G.S., "Failure of Composite Laminates Containing Pin Loaded Holes-Method of Solution," *Journal of Composite Materials*, Vol. 18, No. 3, 1984, pp. 255-278.

[3]Oplinger, D.W., "On the Structural Behavior of Mechanically Fastened Joints in Composite Structures," *Fibrous Composites in Structural Design*, edited by E.M. Lenoe, D.W. Oplinger, and J.J. Burke, Plenum Press, New York, 1980, pp. 575-602.

[4]Hyer, M.W. and Liu, D., "Photoelastic Determination of Stresses in Multiple-Pin Connectors," *Experimental Mechanics*, Vol. 23, No. 3, 1983, pp. 249-256.

[5]Hyer, M.W. and Liu, D., "Stresses in a Quasi-Isotropic Pin-Loaded Connector Using Photoelasticity," *Experimental Mechanics*, Vol. 24, No. 1, 1984, pp. 48-53.

Impact of Loads Recording Methodology on Crack-Growth-Based Individual Aircraft Tracking

Robert M. Engle Jr.*
Flight Dynamics Laboratory
Wright-Patterson Air Force Base, Ohio
and
Thomas F. Christian Jr.†
Damage Tolerance Analysis Laboratory
Robins Air Force Base, Georgia

Introduction

SINCE its inception,[1] the Air Force Structural Integrity Program has used a fatigue analysis as the basis for its life calculations and the full-scale fatigue test to identify "hot spots" and establish aircraft inspection intervals. The dominant parameter in any fatigue analysis is the magnitude of the loads. Hence, the recording devices chosen for aircraft tracking programs have historically been loads recorders. Counting accelerometers and multichannel (V-G-H) recorders have been used for years to collect loads data in terms of magnitude only.[2] V-G-H recorders measure velocity, load factor, and altitude. The Air Force is currently changing the individual aircraft tracking system from one based on fatigue to one based on fracture mechanics and crack growth. The fracture-mechanics-based crack growth analysis is much more sensitive to load sequence effects which lead to the crack growth retardation phenomenon. The transition to the crack-growth-based tracking programs gives rise to the question of whether the magnitude of the loads is a sufficient parameter for crack growth tracking. The objective of this study was to investigate the effects on crack growth life of various truncation patterns which simulate the manner in which certain typical tracking devices gather data. The baseline stress history chosen was a 200 flight sequence generated by the fighter aircraft loading standard for fatigue (FALSTAFF) program.[3] Two stress levels were evaluated. Analytical life predictions for all cases were performed using both fatigue and crack growth analyses. The sequences were then tested to determine experimental lives and the life variations were correlated.

Presented as Paper 84-2410 at the AIAA/AHS/ASEE Aircraft Design Systems and Operations Meeting, San Diego, CA, Oct. 31-Nov. 2, 1984; received Nov. 20, 1984; revision received Sept. 6, 1985.

*Aerospace Engineer.
†Aerospace Engineer. Senior Member AIAA.

Experimental Procedure

The material used for this study was 2024-T3 aluminum. Test specimens were 4 in. wide × 16 in. long with a 0.25-in.-diam centered hole. All specimens were 0.25-in. thick. Crack growth specimens were precracked with a 0.02-in. radial through-crack on each side of the hole using the electrodischarge machining method. The fatigue specimens had no precrack. All testing was done in laboratory air. Each specimen was tested individually in a servocontrolled axial loading frame. The load sequences were programmed using a PDP 11/34 computer. Loading rate was 5 Hz. Crack lengths were measured using an optical microscope with an accuracy of 0.001 in.

Stress Sequences

Baseline

The baseline stress sequence chosen was a 200 flight sequence generated using the FALSTAFF program. This 200 flight block contained 17,000 cycles in a random flight-by-flight sequence. The FALSTAFF sequence consists of integers from 0 to 32 that correspond to load levels determined by the input design limit stress.

V-G-H Simulation

Once the baseline sequence was obtained, an attempt was made to simulate the load magnitude data that would be obtained using a V-G-H recorder yet still retaining the sequencing information needed for a crack growth analysis. This was accomplished by filtering the 17,000-cycle baseline through 12 windows with a 1-*g* minimum threshold. The windows selected gave stress levels comparable to those on load/environment spectra survey (L/ESS) recorders on fighter aircraft. This gave a stress history that retained the sequence of the baseline FALSTAFF but was reduced from 32 to 12 levels and from 17,000 to 13,500 cycles.

Accelerometer Simulation

A similar filtering approach was taken to simulate a counting accelerometer while retaining sequencing information. The number of levels was adjusted to four, which is typical of the number of windows on a counting accelerometer. Again the minimum stresses were set to 1-*g*. Typical rise-fall criteria were applied to define the levels to be recognized. This truncation level gives the minimum information, reducing the 17,000-cycle baseline to 3550 cycles.

Stress Level Variations

Two stress levels, 35 and 40 ksi, were chosen for both test and analysis. These maximum stress levels are typical of average and severe usage for this material on a fighter aircraft. The 40-ksi stress level was such that the failure points of the test were controlled by two large loads in the FALSTAFF sequence. Consequently, the 40-ksi tests exhibit very little scatter; scatter is more evident in the 35-ksi tests.

Analysis

Both the fatigue and crack growth analyses were performed on all three stress sequence variations for two different stress levels (35 and 40 ksi). The fatigue life predictions were made using the Sequence Accountable Fatigue Analysis.[4] This analysis accounts for sequence effects on fatigue life by modeling the residual stress relaxation after high loads. The residual stress properties used in this study were obtained from Ref. 5. The crack growth predictions were made using the CRACKS computer program.[6] Sequence effects were accounted for by the Willenborg model.[7] The crack growth rate relationship used was the Forman equation.[8] The analytical and test lives for both stress levels for both fatigue and crack growth specimens are summarized in Table 1.

Fatigue Results

The Sequence Accountable Fatigue Analysis predicts the baseline stress history to have the shortest life with the counting accelerometer simulation (Four-level) having the longest life. This occurs because the simulations are basically low load truncations of the baseline FALSTAFF stress sequence. The magnitude of the differences in the predicted lives indicates that the levels of truncation are significant in terms of fatigue damage. Table 1 shows that the fatigue analysis is capable of predicting the trends in the lives with the change in recorder simulation but overpredicts the sensitivity to truncation effects.

Crack Growth Analysis Results

The crack growth predictions match the trends of the test data very well. CRACKS with the Willenborg model does a more than adequate job of predicting truncation effects. Since no attempt was made to obtain constant-amplitude crack growth rate data for the actual production lot of material being used, average data from the literature were chosen. Based on these data, the prediction of the truncation effects is exceptional. When all of the test lives were normalized to the average of the baseline tests, CRACKS predicted the truncation effects to within 15.5%. These results are presented in Table 2.

Conclusions

The study described herein, while not all-inclusive, does show that the fracture mechanics discipline upon which crack growth tracking programs are based can function using only the parameters typically collected by fatigue-based programs. In fact, Table 2 indicates that the crack growth modeling actually does a better job of predicting the changes in life due to the variations in usage. Coupling these data with sensitivity studies performed on crack growth analysis capabilities in general by other investigators,[9-12] leads to the conclusion that maintenance impact of the transition to crack-growth-based tracking should be minimal. Further, the fracture mechanics model exhibits much less scatter than the fatigue model and can be "tuned" to match both material properties and load history in a more consistent manner. Thus, the force management engineer can always elect to force his tracking program to be on the conservative side if he is willing to pay the price in increased inspection costs and aircraft downtime.

Table 1 Recorder sensitivity study summary[a]

	40 ksi				35 ksi			
	Crack growth		Fatigue		Crack growth		Fatigue	
Simulation	Pred	Test	Pred	Test	Pred	Test	Pred	Test
Baseline	2561	4600	3000	2360	4332	6200	5200	9620
		4600		3160		5760		7560
		4600		3560		6400		
		4800						
V-G-H	2001	3600	4150	3360	3170	5200	6700	5365
		3800		2880		4000		7100
		3600		4340		5000		
						4600		
Accelerometer	1899	4000	5500	3800	2990	4400	8350	9240
		3600		3920		4800		
				4240				

[a]All results in flights.

Table 2 Life truncation ratios, N_{var}/N_{bl}

	40 ksi				35 ksi			
	Crack growth		Fatigue		Crack growth		Fatigue	
Variation	Pred	Test	Pred	Test	Pred	Test	Pred	Test
Baseline	1.0	1.0	1.0	1.0	1.0	1.0	1.0	1.0
V-G-H	0.78	0.79	1.38	1.17	0.73	0.78	1.29	0.73
Accelerometer	0.75	0.82	1.83	1.32	0.69	0.75	1.61	1.08

References

[1]"Air Force Aircraft Structural Integrity Program: Airplane Requirements," Aeronautical Systems Div., U.S. Air Force, Wright-Patterson AFB, OH, ASD-TR-66-57, May 1970.

[2]Negaard, G.R., "The History of the Aircraft Structural Integrity Program," Aerospace Structures Information and Analysis Center, Wright-Patterson AFB, OH, ASIAC Rept. 680.1B, June 1980.

[3]van Dijk, G.M. and de Jonge, J.B., "Introduction to a Fighter Aircraft Loading Standard for Fatigue Evaluation—FALSTAFF," National Aerospace Laboratory, the Netherlands, NLR MP 75017, May 1975.

[4]Potter, J.M. and Noble, R.A., "A Users Manual for the Sequence Accountable Fatigue Analysis Computer Program," AFFDL-TR-74-23, May 1974.

[5]Potter, J.M., "An Experimental and Analytical Study of Spectrum Truncation Effects," AFFDL-TR-73-123, Dec. 1973.

[6]Engle, R.M. Jr., "CRACKS II—A User's Manual," AFFDL-TM-74-173-FBE, Aug. 1974.

[7]Willenborg, J.D., Engle, R.M. Jr., and Wood, H.A., "A Crack Growth Model Using an Effective Stress Concept," AFFDL-TM-71-1-FBR, Jan. 1971.

[8]Forman, R.G., Kearney, V.E., and Engle, R.M., "Numerical Analysis of Crack Propagation in Cyclic Loaded Structures," *Journal of Basic Engineering, Transactions of ASME,* Ser. D, Vol. 89, No. 3, 1967, pp. 459-464.

[9]Dill, H.D. and Saff, C.R., "Effects of Fighter/Attack Spectrum on Crack Growth," AFFDL-TR-76-112, March 1977.

[10]Parker, G.S., "Generalized Procedure for Tracking Crack Growth in Fighter Aircraft," AFFDL-TR-76-133, Jan. 1977.

[11]Abelkis, P.R., "Effect of Transport/Bomber Loads Spectrum on Crack Growth," AFFDL-TR-78-134, Nov. 1978.

[12]Lambert, G.L. and Bryan, D.F., "The Influence of Fleet Variability on Crack Growth Tracking Procedures for Transport/Bomber Aircraft," AFFDL-TR-78-158, Nov. 1978.