DREP Contractor's Report Series 93-17

# ACOUSTIC EMISSION MONITORING OF AGING AIRCRAFT STRUCTURES

UNLIMITED

by

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March 1993

Contract Serial No. W7708-6-8277

# ACOUSTIC EMISSION MONITORING OF AGING AIRCRAFT STRUCTURES

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## INTRODUCTION

During the past 20 years acoustic emission has been applied successfully to numerous materials degradation and failure problems. These applications are catalogued in Drouillard's bibliography of acoustic emission [1,2]. There have also been many disappointing results due to inadequate care and technique for discriminating between signals resulting from the materials degradation or failure process of interest and those resulting from environmental noise.

In the case of acoustic emission monitoring of airframes particular care has to be taken to discriminate between fatigue crack sources and noise. Routine measurements at randomly selected airframe locations show that the number of noise signals similar to crack advance signals during simulated flight test cyclic loading is typically in excess of $10^6$ per hour of testing while the number of crack advance signals (if a crack is present) would be less than 1 per hour of testing. This is due to the presence of distributed noise sources throughout the structure which have signal features not unlike those of crack advance signals. I refer here to rubbing and fretting of bolted structural components, fasteners or crack faces. To solve this problem we have developed equipment which screens every detected signal by on-line windowing of a number of selectable signal parameters. The specific choice of on-line windowing parameters is made by the operator to take into account differences in acoustic behavior of the structure at different regions. Off-line windowing can also be applied on all recorded parameters after downloading the data to a portable computer directly or via modem to a computer in a remote location. Using this equipment and methodology crack face rubbing and crack advance signals can be discriminated from airframe noise provided the structure at the location of interest is acoustically calibrated using routine procedures and parametric windowing is then properly applied.

Acoustic emission can be used on-line as a continuous monitoring technique or alternatively as a periodic proof test. In either case the structure is required to be stressed to levels comparable to the highest in-service stress level. By this method it is possible to locate and identify sources such as fatigue cracks in a structure which is subject to complex loading during service. Indeed, acoustic emission can be used to determine the loading conditions and loading sequences under which fatigue crack growth takes place at each location in a large structure as well as providing a history of crack growth at each site during the monitoring period. In addition to providing such a diagnostic capability acoustic emission can detect fatigue cracks which cannot be detected readily and reliably by conventional NDT due to proximity of other interfaces. For example, the problem of detecting cracks in fastener holes can be solved using acoustic emission to monitor many fastener holes simultaneously without the removal of fasteners. Reliable NDT confirmation of these defects requires the removal of fasteners. Indeed a major controversy in using acoustic emission in aircraft structures at this time is the inadequacy of current, practical NDT techniques to confirm the presence of fatigue cracks detected by acoustic emission. These inadequacies have been found for radiography, LPI, ultrasonics and eddy current as they are currently used in practise.

This work, which is part of a larger program to address the application of acoustic emission to the detection of cracks during flight [3,4,5,6,7], will describe the use of acoustic emission to detect fatigue cracks in a full-scale aircraft structure during ground durability and damage tolerance testing. It will be shown that we have succeeded in using acoustic emission monitoring to detect fatigue cracks inside fastener holes without the removal of fasteners. The fatigue cracks detected by acoustic emission were confirmed by NDT and measured to be about 1 mm deep.

## EXPERIMENTAL

### The Data Acquisition System

The 16 channel instrumentation system is based on the dual-channel acoustic emission system developed at the Royal Military College of Canada. This data acquisition system (commercially available from AEMS, Acoustic Emission Monitoring Services Inc., Kingston, Ontario, Canada) was designed and constructed specifically for the recording and interpreting of acoustic emission data in the laboratory and during flight. The design is based on criteria derived from the RMC work of almost a decade in the area of acoustic emission monitoring during flight [3,4,5,6,7]. These studies established the importance of the difference in arrival time of an event at different locations, signal risetime, and the magnitude and variation of the applied stress at the time of occurrence of the event. All of these parameters are necessary to isolate crack-related events from other noise sources during dynamic loading of a large, dispersive structure and are recorded by the data acquisition system used here. To provide maximum flexibility, the data acquisition system is designed to use either an available 115/230V, 47-440 Hz electrical supply or internal batteries.

The output of each of the piezoelectric sensor elements is amplified by a preamplifier with nominal gain of 40 dB. The resulting signal is buffered, logarithmically amplified, envelope followed and peak detected. These operations are accomplished using signal conditioning modules custom-made for the purpose (Figure 1a). The output of each envelope follower is separately fed into the digital data acquisition system where the times of pre-selected amplitude threshold crossings (6 dB apart) are recorded (Figure 1b). The output of the peak detectors and strain gauge are digitized by an A/D convertor and stored in memory.

All of the above data are compressed into an event record which includes the time of occurrence of the event at each sensor, the difference in arrival times at two sensors ($\Delta t$), event risetimes for 6 dB change in amplitude, event durations, event decay times and event peak amplitudes. The resulting data set is then extracted from the data acquisition system via an RS-232 interface and stored on disk on an external portable computer. By using a data multiplexer, several 2-channel systems are operated in parallel to achieve multi-channel capability. Since each 2-channel system has a CPU there is no decrease in data acquisition rate as the number of channels is increased. Extensive screening of data, field analysis and interpretation can be carried out immediately on the host monitoring computer. Final analysis and interpretation are accomplished using spreadsheet software. Table 1 lists the general specifications of the apparatus.

### Wing Loading

Fatigue cycling loads were applied at various loading points to simulate the known flight load spectra measured on flying aircraft. The actual loadings used are defined in terms of the in-flight acceleration of the aircraft (g) or the corresponding transverse strain measured by strain gauges attached to the lower wing skin. Table 2 lists the simulated flight load spectrum in terms of g and the normalized percent transverse strain relative to the value at 7g measured on the lower wing skin. The output of the strain gauge is recorded by the data acquisition system as the measure of the wing loading conditions at the time of occurrence of each acoustic emission signal. The highest strain manoeuvres (6.5g and 7g) are of particular interest for the acoustic emission monitoring since they provide the highest loads, and hence, the most probable circumstances for crack advance acoustic emissions.

Table 1. General specifications for the AEMS digital data acquisition system for in-flight acoustic emission monitoring applications.

| | |
|---|---|
| 2 Channels AE | 60 dB dynamic range |
| 2 Analog Channels | 10 V full-scale |
| Power Supply | 115/230V, 47-440 Hz or battery powered |
| Power Consumption | 10 Watts maximum |
| Data Storage Capacity | 192 or 384 Kbyte RAM with battery back-up |
| Dimensions | 23 cm x 13.5 cm x 25 cm |
| Weight | 2 kg |
| Mass Data Storage | RS232 transfer to external computer |
| Windowing on all recorded parameters | Available on-line and during post analysis |

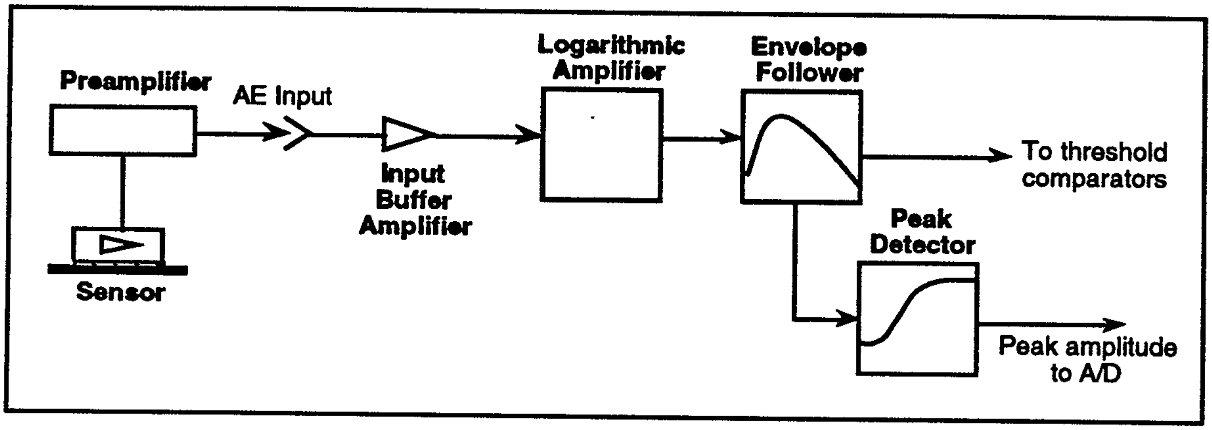


Fig. 1a. Schematic diagram of the AEMS acoustic emission signal conditioning.

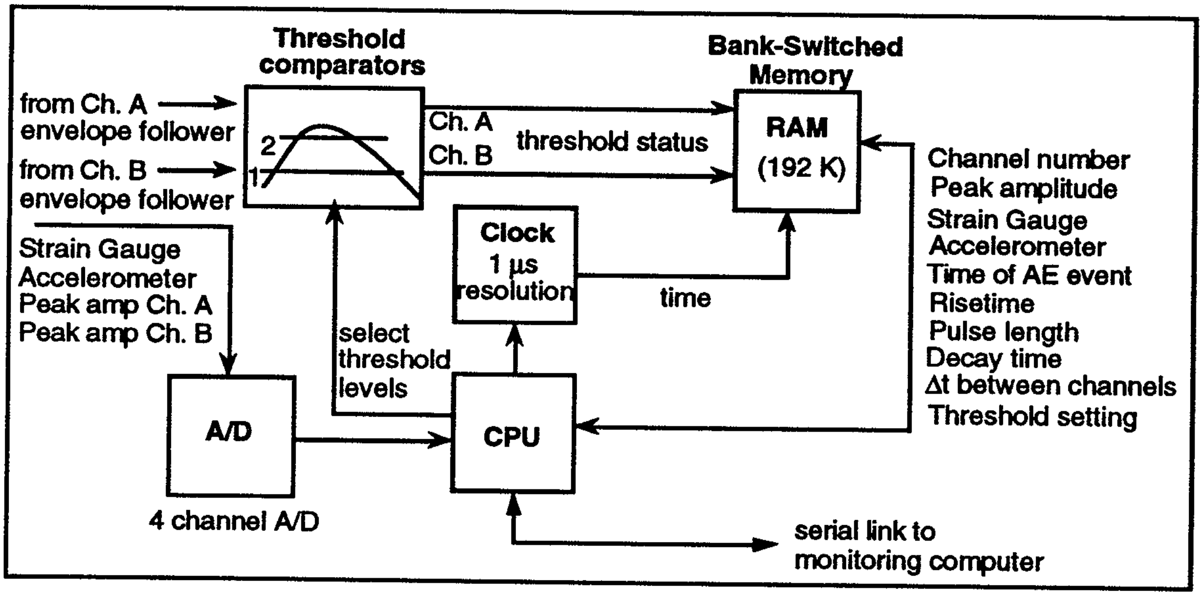


Fig. 1b. Schematic diagram of the AEMS acoustic emission data acquisition computer.

Table 2. Number of occurrences of maximum g levels per thousand flying hours derived from measurements during typical flight conditions. Also shown is the % strain in the lower wing skin relative to the value at 7g.

| g | Strain % | Loadings/$10^3$ Flying Hrs |
|---|---|---|
| 4.5 | 64.3 | 1770 |
| 5.0 | 71.4 | 2828 |
| 5.5 | 78.6 | 1589 |
| 6.0 | 85.7 | 198 |
| 6.5 | 92.9 | 45 |
| 7.0 | 100.0 | 90 |

## Acoustic Emission System Calibration

Pencil lead fracture was used to obtain the area calibration of the various acoustic emission parameters ($\Delta t$, pulse length, risetime, etc). It was found that, close to the array, the difference in arrival time at neighboring sensors is linear and results from an acoustic wave velocity of 2 km/s. The measured risetime of the signals detected by the Dunegan Endevco D9202A sensors was less than 3μs and was essentially independent of source position between the sensors provided no abrupt changes in material thickness occurred. Such changes in the thickness usually involved additional fasteners to connect the wing skin to the substructure, and hence, introduced additional reflections. The amplitude of the detected signal could be reduced by as much as 20 dB as a result of such acoustic scattering.

## The Acoustic Emission Measurements

Repairs were carried out on the port (left) wing prior to continuous monitoring. These involved enlarging some fastener holes to remove cracks detected by conventional NDT as well as the repairs to the wing substructure. The starboard (right) wing passed inspection carried out using rotoscan ultrasonics to examine the fastener holes. The regions monitored were selected by the fatigue test engineers and are located on the lower wing skin of a fighter aircraft. The particular areas of interest are the 39% and 44% spar regions within the linear array of sensors shown in figure 2. The precise location of each sensor is constrained by the presence of the pads which are used to transmit the simulated flight loads to the wing. With the sensors positioned between the 39% and 44% spars in the linear array shown in figure 2, it was shown by acoustic emission monitoring during a static wing loading that the 44% spar defects on the port side could be detected and located by acoustic emission. During this static test defects were also detected on the starboard side at 1 meter from the wing centre line. NDT inspection failed to confirm these defects. Dynamic testing commenced and acoustic emission data was recorded for 700 equivalent flying hours and analyzed at intervals for each of the sensor pairs shown in figure 2. A histogram of these acoustic emission data is also shown in figure 2.

After 700 equivalent flying hours of testing it was clear from the acoustic emission data that significant crack growth was occurring in the starboard wing skin particularly at 1 meter from the wing centre in the 39% and 44% spars. This result substantiated the earlier static loading observation and testing was stopped to perform an inspection of the structure. Fasteners were removed to inspect fastener holes in the damage area identified by acoustic emission. Fatigue cracks were detected in several fastener holes using eddy current only after the structure was loaded to open the cracks. The surface-breaking cracks were found to be a few millimetres long with depths measured to be in the range 0.3 mm to 1.6 mm. <u>It is important to note that these defects were present in the structure prior to our acoustic emission monitoring and were undetected during NDT inspections</u>.

As a result of the above success of acoustic emission monitoring a major NDT inspection was carried out for all fastener holes in the 39% and 44% spars. The resulting defect histogram is compared with the acoustic emission histogram in figure 2. The identified defects were quantified by using progressive reaming followed by rotating coil eddy current inspection. The crack depths thus obtained are shown in figures 3(a) and 3(b) which also include the acoustic emission results for comparison. Note also that the defect in the critical radius (Example A, Figure 3a) was also found first by acoustic emission monitoring and later confirmed by eddy current and LPI.

Following the detection of crack sites the acoustic emission data was further analyzed to reconstruct the history of defect growth (Figure 4) and to separate the crack growth acoustic emission data from crack face rubbing data (Figure 3). This latter separation involved examination of the occurrence of acoustic emission signals within a given manoeuvre or manoeuvre sequence.

Figure 4(a) shows the acoustic emission history resulting from flight-simulated structural testing for a single defect (Example A, Figure 3a). Note that the crack advances very rapidly in a short period (15, 7g loadings, 150 equivalent flying hours) while the crack face rubbing noises were relatively small in number. This result suggests that <u>continuous monitoring</u> of this defect (which had been undetected by a series of periodic NDT inspections) is appropriate to avoid potential catastrophic failure of the wing.

Figure 4(b) shows the acoustic history of the cluster of bolt hole defects (Example B, 1 meter from wing centre, Figure 3b) which resulted in termination of the test. It is noted that significant crack advance occurred as the result of about 20, 7g loadings (200 equivalent flying hours) while crack face rubbing noises were monotonic and relatively small in number. Again the advantages of continuous monitoring are evident since periodic NDT inspections had failed to detect these rapidly growing defects.

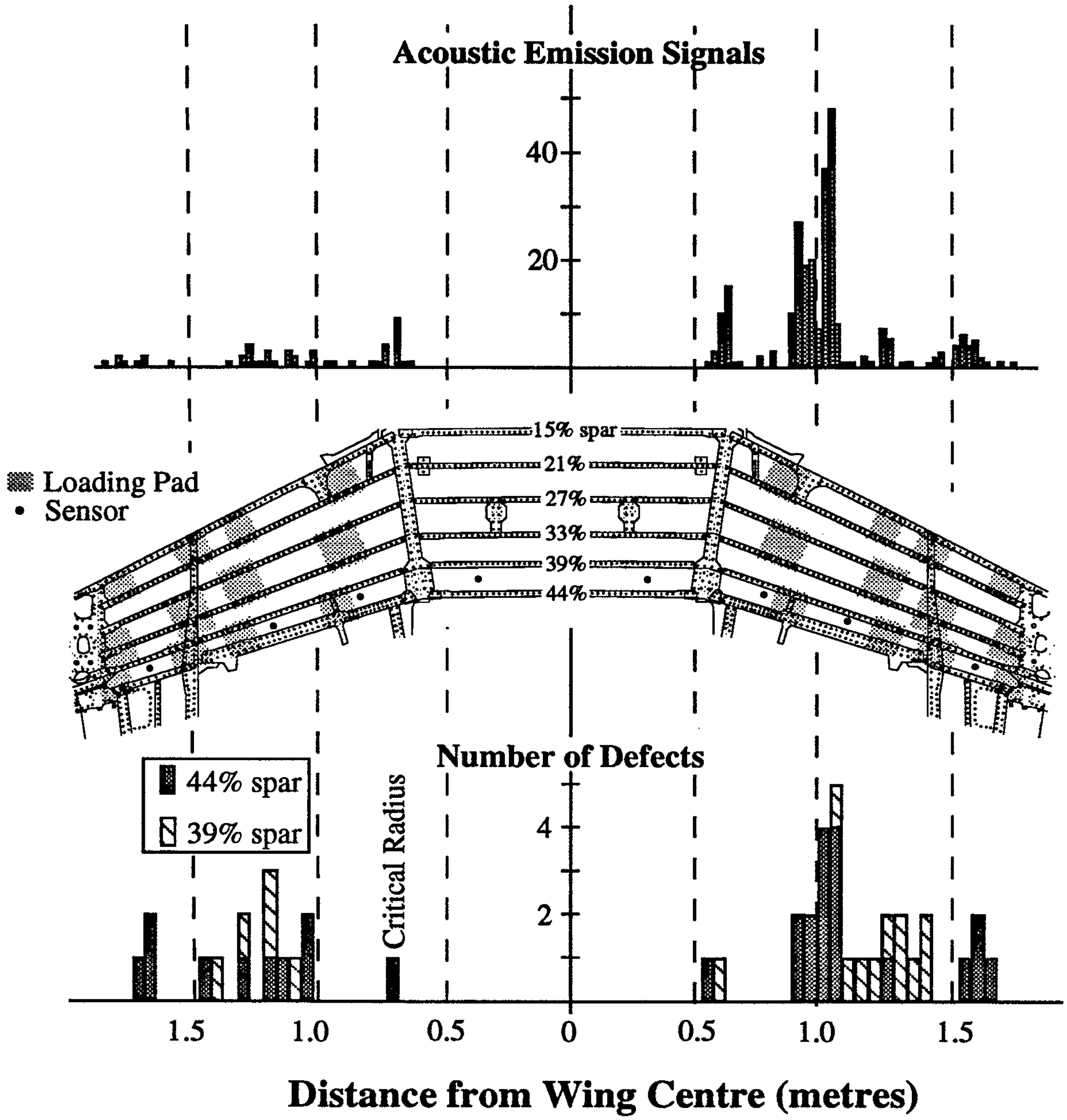


Fig. 2. Shows a schematic diagram of the monitored wing section with loading pads and sensor locations indicated. Also included are histograms of the detected acoustic emissions and the fatigue cracks confirmed by rotating coil eddy current measurements after removal of all fasteners in the 39% and 44% spars.

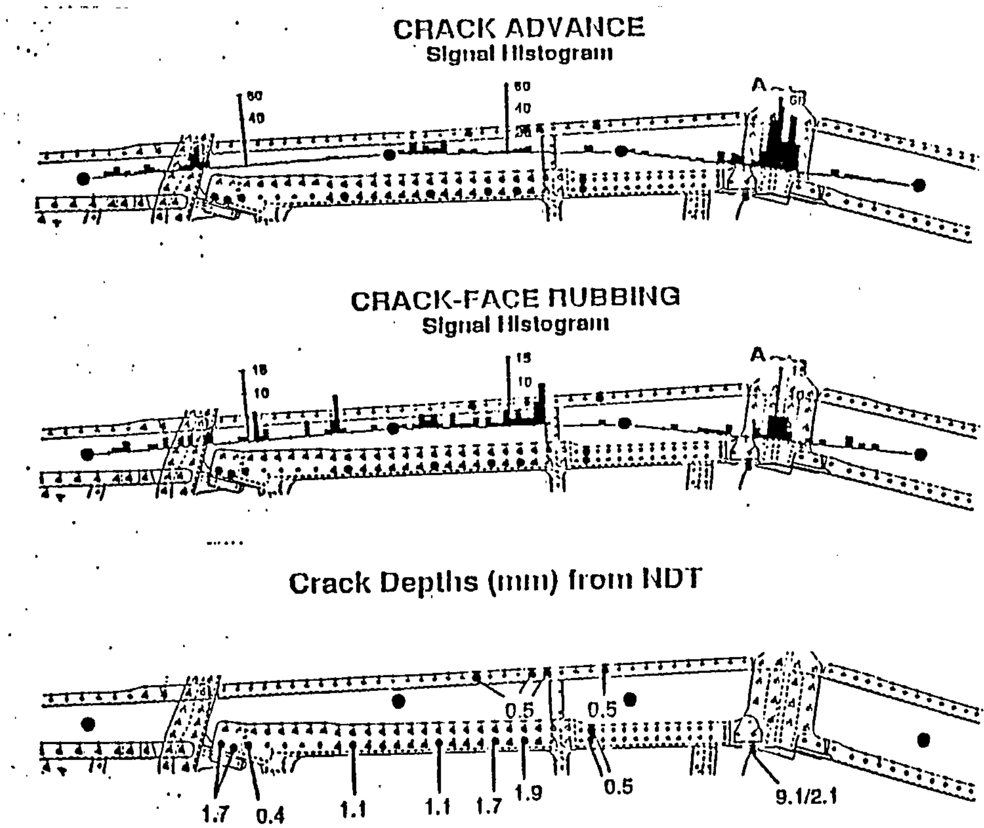


Fig. 3a. Comparison of crack depth measurements with acoustic emission detection of crack advance and crack face rubbing for the port (left) wing.

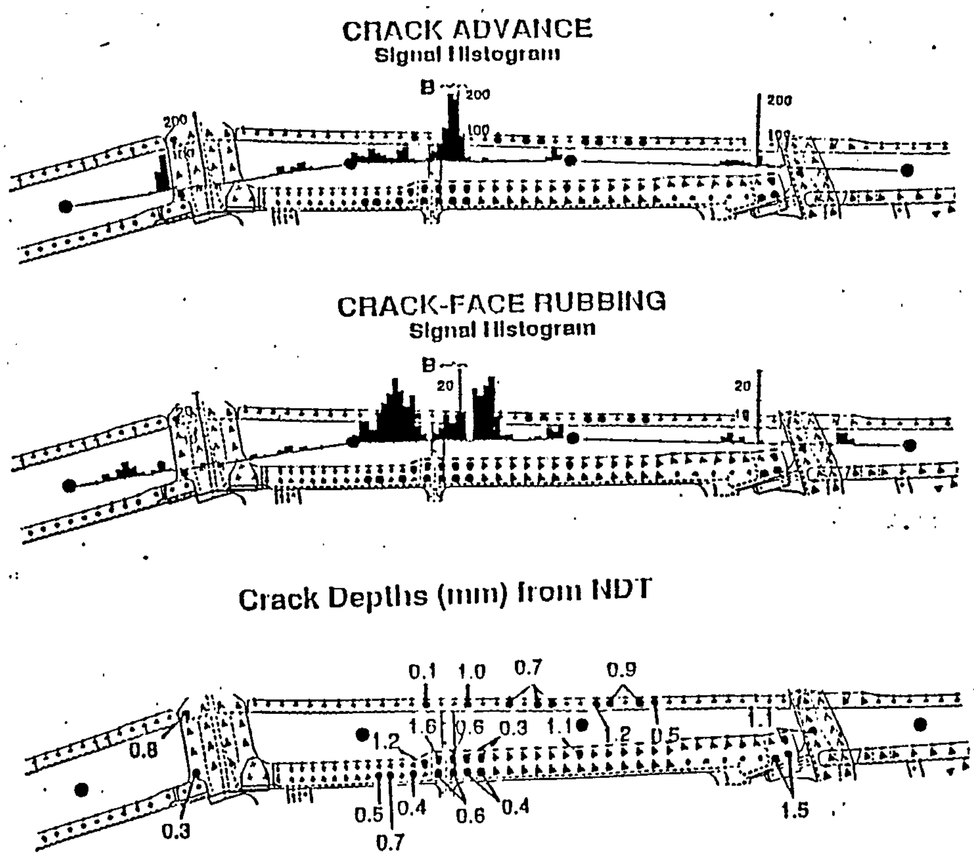


Fig. 3b. Comparison of crack depth measurements with acoustic emission detection of crack advance and crack face rubbing for the starboard (right) wing.

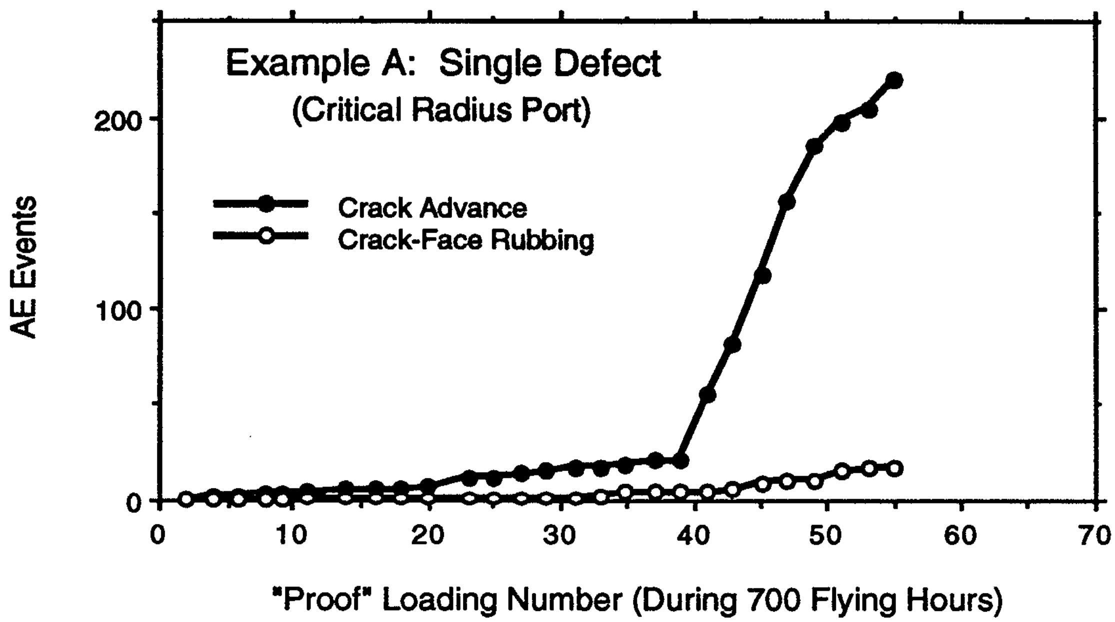


Fig. 4a. Acoustic emission history of crack growth from a single defect.

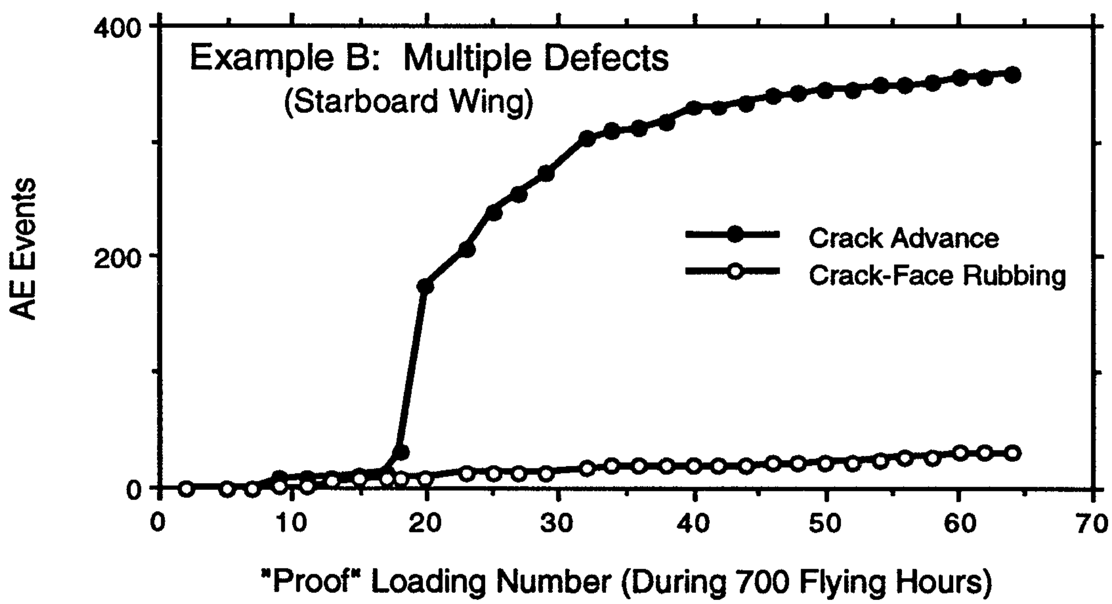


Fig. 4b. Acoustic emission history of crack growth from a cluster of defects.

## SUMMARY AND CONCLUSIONS

Acoustic emission monitoring of a full-scale ground durability and damage tolerance test clearly demonstrates the potential value of acoustic emission for determining "defect zones" in large aircraft structures. Such information can then be used to focus NDT resources more effectively on damage areas.

We have shown that acoustic emission monitoring is superior to conventional NDT in locating fatigue cracks, and, in addition, can be used to determine the loading conditions and loading sequences under which fatigue crack growth occurs.

We have shown that cracks originating inside fastener holes in aircraft structures can be detected easily by acoustic emission without the removal of fasteners. The size of cracks in such locations which were detected in this work range from 0.5 mm to 1.5 mm in depth and have similar surface breaking dimensions. Reliable NDT confirmation of these acoustic emission results required removal of fasteners and loading the structure to open the cracks. Confirmation of acoustic emission sources as cracks is often severely limited by the inadequacies of practical NDT techniques as they are currently applied to aircraft structures. These inadequacies are evident for radiography, LPI, ultrasonics and eddy current when they are applied to a complex structure such as fastener hole cracks in a wing skin containing several hundred fasteners.

The time has come for serious consideration to be given to the implementation of acoustic emission monitoring in aircraft structures both on the ground and during flight. No other known technique has the capability to find fatigue cracks reliably in such a complex situation involving a multiplicity of distributed crack initiation sites and hence assure the safety of aging aircraft structures.

## ACKNOWLEDGEMENTS

Supporting funds were provided by the Department of National Defence, Canada (ARP 3610-208, CRAD 144690RMC01 and DAS Eng 84779ACCE01). The project was monitored by Mr. W.R. Sturrock of the Materials Section, Defence Research Establishment - Pacific, Victoria, British Columbia, and Major W.J. Miller, National Defence Headquarters, Ottawa, Ontario. The ground durability and damage tolerance test was conducted by Canadair, Montreal, under the supervision of Mr. G. Deziel of National Defence Headquarters, Ottawa.

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| | |
|---|---|
| REPORT NUMBER: | DREP Contractors' Report 93-17 |
| TITLE: | Acoustic Emission Monitoring of Aging Aircraft Structures |
| AUTHOR(S): | S.L. McBride, M.R. Viner and M.D. Pollard |
| DATED: | February 1993 |
| SECURITY GRADING: | UNCLASSIFIED |

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2. SECURITY CLASSIFICATION (overall security classification of the document including special warning terms if applicable)

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3 TITLE (the complete document title as indicated on the title page. Its classification should be indicated by the appropriate abbreviation (S,C or U) in parentheses after the title.)

Acoustic Emission Monitoring of Aging Aircraft Structures

4 AUTHORS (Last name, first name, middle initial)

McBride, Stuart, L., Viner, Melvyn, R., and Pollard, Michael, D.

5 DATE OF PUBLICATION (month and year of publication of document)

March 1993

6a. NO. OF PAGES (total containing information. Include Annexes, Appendices, etc.)

11

6b. NO. OF REFS (total cited in document)

7

7 DESCRIPTIVE NOTES (the category of the document, e.g. technical report, technical note or memorandum. If appropriate, enter the type of report, e.g. interim, progress, summary, annual or final. Give the inclusive dates when a specific reporting period is covered.)

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8. SPONSORING ACTIVITY (the name of the department project office or laboratory sponsoring the research and development. Include the address)

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9a PROJECT OR GRANT NO. (if appropriate, the applicable research and development project or grant number under which the document was written. Please specify whether project or grant)

9b. CONTRACT NO. (if appropriate, the applicable number under which the document was written)

W7708-6-8277

10a. ORIGINATOR'S DOCUMENT NUMBER (the official document number by which the document is identified by the originating activity. This number must be unique to this document.)

DREP CR 93-17

10b. OTHER DOCUMENT NOS. (Any other numbers which may be assigned this document either by the originator or by the sponsor)

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