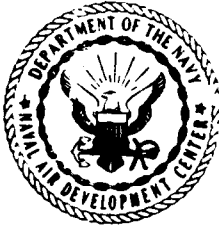


AD-A260 940



2

Report No. NADC-91074-60



DTIC
ELECTE
FEB 5 1993
S C D

DEVELOPMENT OF AN INERTIALLY-LOADED TEST FIXTURE FOR IN-FLIGHT EVALUATION OF AN ACOUSTIC EMISSION MONITORING SYSTEM

P.J. Kulowitch
Air Vehicle and Crew Systems Technology Department (Code 6063)
NAVAL AIR DEVELOPMENT CENTER
Warminster, PA 18974-5000

JULY 1991

FINAL REPORT
Task No. 5.2
Project No. RS34A50
Work Unit No. ZP110
Program Element No. 62234N

Approved for Public Release; Distribution is Unlimited

Prepared For
Naval Airborne Materials Block
NAVAL AIR DEVELOPMENT CENTER
Warminster, PA 18974-5000

93-02113



93 2 4 006

NOTICES

REPORT NUMBERING SYSTEM — The numbering of technical project reports issued by the Naval Air Development Center is arranged for specific identification purposes. Each number consists of the Center acronym, the calendar year in which the number was assigned, the sequence number of the report within the specific calendar year, and the official 2-digit correspondence code of the Command Officer or the Functional Department responsible for the report. For example: Report No. NADC-88020-60 indicates the twentieth Center report for the year 1988 and prepared by the Air Vehicle and Crew Systems Technology Department. The numerical codes are as follows:

CODE	OFFICE OR DEPARTMENT
00	Commander, Naval Air Development Center
01	Technical Director, Naval Air Development Center
05	Computer Department
10	AntiSubmarine Warfare Systems Department
20	Tactical Air Systems Department
30	Warfare Systems Analysis Department
40	Communication Navigation Technology Department
50	Mission Avionics Technology Department
60	Air Vehicle & Crew Systems Technology Department
70	Systems & Software Technology Department
80	Engineering Support Group
90	Test & Evaluation Group

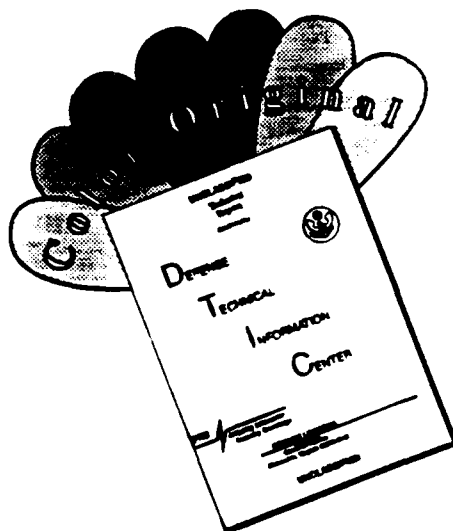
PRODUCT ENDORSEMENT — The discussion or instructions concerning commercial products herein do not constitute an endorsement by the Government nor do they convey or imply the license or right to use such products.

Reviewed By: J. Waldman Date: 9/12/91
Branch Head

Reviewed By: P. Shuford Date: 9/19/91
Division Head

Reviewed By: N. Smith Date: 9/23/91
Director/Deputy Director

DISCLAIMER NOTICE



THIS DOCUMENT IS BEST QUALITY AVAILABLE. THE COPY FURNISHED TO DTIC CONTAINED A SIGNIFICANT NUMBER OF COLOR PAGES WHICH DO NOT REPRODUCE LEGIBLY ON BLACK AND WHITE MICROFICHE.

REPORT DOCUMENTATION PAGE			Form Approved OMB No. 0704-0188	
Public reporting burden for this collection of information is estimated to average 1 hour per response, including the time for reviewing instructions, searching existing data sources, gathering and maintaining the data needed, and completing and reviewing the collection of information. Send comments regarding this burden estimate or any other aspect of this collection of information, including suggestions for reducing this burden, to Washington Headquarters Services, Directorate for Information Operations and Reports, 1215 Jefferson Davis Highway, Suite 1204, Arlington, VA 22202-4302, and to the Office of Management and Budget, Paperwork Reduction Project (0704-0188), Washington, DC 20503.				
1. AGENCY USE ONLY (Leave blank)		2. REPORT DATE July 1991	3. REPORT TYPE AND DATES COVERED Final	
4. TITLE AND SUBTITLE Development of an Inertially-Loaded Test Fixture for In-flight Evaluation of an Acoustic Emission Monitoring System			5. FUNDING NUMBERS TA 5.2 PN RS34A50 WU ZP100 PE 62234N	
6. AUTHOR(S) P.J. Kulowitch				
7. PERFORMING ORGANIZATION NAME(S) AND ADDRESS(ES) Air Vehicle and Crew Systems Technology Department (Code 6063) NAVAL AIR DEVELOPMENT CENTER Warminster, PA 18974-5000			8. PERFORMING ORGANIZATION REPORT NUMBER NADC-91074-60	
9. SPONSORING/MONITORING AGENCY NAME(S) AND ADDRESS(ES) Airborne Materials Block NAVAL AIR DEVELOPMENT CENTER Warminster, PA 18974-5000			10. SPONSORING/MONITORING AGENCY REPORT NUMBER	
11. SUPPLEMENTARY NOTES				
12a. DISTRIBUTION/AVAILABILITY STATEMENT Approved for Public Release; Distribution is Unlimited			12b. DISTRIBUTION CODE	
13. ABSTRACT (Maximum 200 words) An Inertially-loaded test fixture was designed for use with an in-flight acoustic emission monitoring system. The fixture provides a controlled method of simulating structural cracking during flight. It will be used to determine if acoustic emission monitoring is a viable method for monitoring crack growth in aircraft structures which are difficult to access and inspect. The inertially-loaded fixture was designed to use mechanical levers combined with the accelerations of the aircraft to produce the force necessary to cause crack propagation in a test specimen. The fixture minimizes the volume of the device by using a cantilevered specimen along with a loading lever arm which crosses back over the specimen to generate an equivalent lever length of 90 inches within the 21 x 4.5 x 5 inch fixture. The fixture and encasement weigh 23 pounds. The cantilever grip can accommodate specimens of various materials, widths, and thicknesses with minimal alterations to the fixture.				
14. SUBJECT TERMS Acoustic Emission, In-flight Crack Monitoring, Fatigue of Aircraft Structures.			15. NUMBER OF PAGES	
			16. PRICE CODE	
17. SECURITY CLASSIFICATION OF REPORT Unclassified	18. SECURITY CLASSIFICATION OF THIS PAGE Unclassified	19. SECURITY CLASSIFICATION OF ABSTRACT Unclassified	20. LIMITATION OF ABSTRACT SAR	

ABSTRACT

An Inertially-loaded test fixture was designed for use with an in-flight acoustic emission monitoring system. The fixture provides a controlled method of simulating structural cracking during flight. It will be used to determine if acoustic emission monitoring is a viable method for monitoring crack growth in aircraft structures which are difficult to access and inspect. The inertially-loaded fixture was designed to use mechanical levers combined with the accelerations of the aircraft to produce the force necessary to cause crack propagation in a test specimen. The fixture minimizes the volume of the device by using a cantilevered specimen along with a loading lever arm which crosses back over the specimen to generate an equivalent lever length of 90 inches within the 21 x 4.5 x 5 inch fixture. The fixture and encasement weigh 23 pounds. The cantilever grip can accommodate specimens of various materials, widths, and thicknesses with minimal alterations to the fixture.

i

DTIC QUALITY INSPECTED 3

Accession For	
NTIS	☒
DTIC TAB	☐
Unannounced	☐
Justification	
By	
Distribution/	
Availability Codes	
Dist	Avail and/or Special
A-1	

TABLE OF CONTENTS

ABSTRACT	i
TABLE OF CONTENTS	ii
LIST OF FIGURES	iii
NOMENCLATURE	iv
1. SUMMARY	1
1.1 Problem	1
1.2 Objectives of this Study	1
1.3 Results	1
2. INTRODUCTION	4
2.1 Purpose of Report	4
2.2 Scope of Report	4
3. BACKGROUND INFORMATION	5
4. TECHNICAL APPROACH	7
4.1 Design Constraints	7
4.2 Test Fixture Design Concept	7
4.3 Specimen Design	7
4.4 Test Fixture Sizing and Final Design	9
4.5 Acoustic Emission Laboratory Test Data	10
5. CONCLUSIONS	18
6. REFERENCES	19
APPENDIX A: SPECIMEN PREPARATION PROCEDURES	20
APPENDIX B: FRACTURE MECHANICS ANALYSIS FOR PRELIMINARY DESIGN OF THE IN-FLIGHT INERTIAL-LOADING FIXTURE	25
APPENDIX C: CALCULATIONS OF THE APPLIED LOADING WEIGHT	29
APPENDIX D: LABORATORY TEST PROCEDURES	31

LIST OF FIGURES

<u>Figure</u>	<u>Page</u>
1. The Crossed Cantilever Loading Configuration	2
2. The In-Flight Fixture and AE Monitor	3
3. In-Flight Inertial-Loading Fixture	8
4. Cantilever Grip Assembly	9
5. In-Flight AE Monitoring Specimen (Al 7075-T6)	11
6. AE Pinducer Arrangement	11
7. Alternate AE Transducer Arrangement for In-Flight Testing ...	13
8. Typical 0.5mm Pencil Lead Break AE Signal (Calibration)	14
9. Typical Electrical Noise AE Signal	15
10. Typical Epoxy/Aluminum Delamination AE Signal	16
11. Typical Crack Propagation AE Signal	17
12. Specimen Preparation - Required Materials	22
13. Specimen Preparation - Reinforcing the Specimen	23
14. Specimen Preparation - Notching and Drilling	23
15. Specimen Preparation - Attaching Pinducers	24
16. Geometric Correction Factor for 3 or 4 Point Bending	26
17. Moment Diagram for Pivot Arm Assembly at Design Point	30
18. Set-up for Precracking Specimens on Krause Machine	33
19. Close-up of Specimen Mounted in Krause Machine	34
20. Setting the Load on the Krause Machine	35
21. Illustration of Connection of Specimen and Loading Lever Arm	37
22. Schematic Representation of Set-up for Verifying Failure Load Using Laboratory Test Machines	37
23. Failure Load Verification Using Instron Test Machine	38
24. Detailed View of In-Flight Fixture Mounted on Instron Test Machine	39

NADC-91074-60

NOMENCLATURE

<u>Abbreviation</u>	<u>Name</u>
a	Crack Length, inches
K_I	Mode I Stress Intensity Factor, $\text{psi}(\text{inch})^{0.5}$
$K_{I \text{ crit}}$	Mode I Fracture Toughness, $\text{psi}(\text{inch})^{0.5}$
M	Moment Generated at the Notch by the Applied Load
P	Load Applied to the Specimen 10 inches from the Notch
t	Specimen Thickness, inches
S	Span Between Supports for 3-Point Bending
W	Specimen Width, inches
Wt	Loading Weight Designed into Lever Arm
Y	Geometric Correction Factor
σ_{YS}	Yield Stress, psi

1. SUMMARY

1.1 Problem

The design of an inertially-loaded test fixture is part of a project to evaluate the potential using an Acoustic Emissions (AE) Monitoring System to identify crack initiation in aircraft structures that are difficult to access (e.g. fuselage bulkheads near wing attachment lugs).

It is anticipated that by using AE sensors to monitor inaccessible structures that have a high probability of cracking, crack initiation can be detected and premature replacement of the structures can be eliminated. Additionally, significant monetary savings and reduced aircraft down-time could be realized as a result of less frequent inspection of these structures.

Flight-testing the AE Monitoring System is extremely important since a large part of this program revolves around being able to distinguish the noises produced by crack propagation from the other noises present on board the aircraft. The test fixture is necessary for testing this AE Monitoring System, since it would not be safe to flight-test the AE System using aircraft structures known to contain cracks.

1.2 Objectives of this Study

The objective of this study was to develop a test fixture that could crack specimens of a material similar to those in use on Navy aircraft structures (7075-T6 Aluminum) to aid in the evaluation of an in-flight AE Monitoring System. An additional requirement was that the test fixture should be able to accommodate testing of specimens of different materials and dimensions without major modifications. Finally, the fixture had to be as compact as possible to make it easy to install in an F-18 for flight-testing.

1.3 Results

An inertially-loaded test fixture was designed and built for use in upcoming flight tests to determine the feasibility of using an AE Monitoring System to monitor crack growth in structural materials such as 7075-T6 Aluminum. Minimal volume of the fixture and encasement along with maximum flexibility in test capabilities were emphasized in the design.

The design of the test fixture utilizes a cantilevered specimen with a levered loading arm crossed over the specimen to generate an equivalent lever length of 90 inches within a 21 x 4.5 x 5 inch encasement. Figure 1 shows the "crossed cantilever" configuration used.

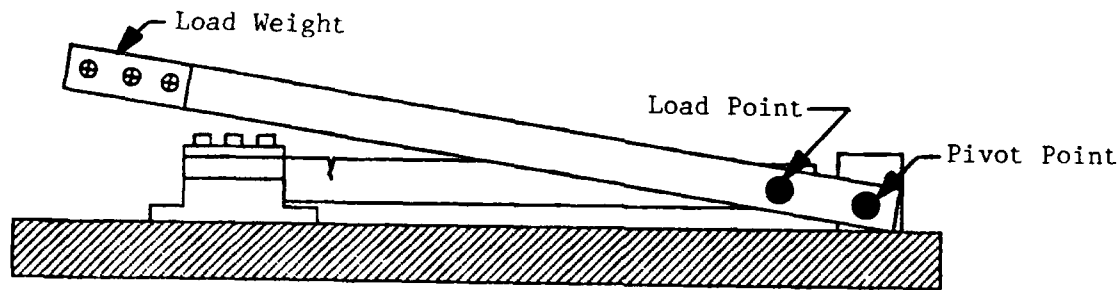


FIGURE 1. The Crossed Cantilever Loading Configuration.

The design of the grip which holds the specimen in a cantilevered position was tailored to facilitate testing of bar-shaped specimens (up to 1.25 inches in width) in either an edgewise or a flat position. The flatwise feature will be especially important to future work when the AE Monitoring System will be applied to composite material.

The fixture holds a notched, reinforced 7075-T6 aluminum specimen in a cantilevered configuration. The load is applied to the end of the cantilevered specimen through a weighted lever arm. The load on the specimen is then multiplied by the accelerations of the aircraft to produce sufficient force to fracture the specimen.

An acoustic emissions monitoring system will be used in conjunction with this fixture (Figure 2) in future flight tests to record the acoustic waveforms produced by crack advances. The waveforms will then be analyzed to identify features which will distinguish the acoustic waveforms caused by crack growth from other noise sources in the aircraft environment.

Procedures were developed to prepare specimens and to test the specimens in the laboratory to estimate how much acceleration the test pilot will have to generate in the aircraft to cause the specimen to fail.



Figure 2. The In-Flight Fixture and AE Monitor

2. INTRODUCTION

2.1 Purpose of Report

The purpose of this report is to detail the design of the inertially-loaded test fixture and the associated hardware which will be used to flight-test the Acoustic Emissions Monitoring System. This report also provides a detail description of how the test fixture operates and how it will be used on the aircraft. Lastly, this report provides step-by-step procedures for using the inertially-loaded in-flight test fixture.

2.2 Scope of Report

This report discusses the design and use of the inertially-loaded test fixture. This report begins with a brief discussion of relevant background information. A description of the design restraints imposed for the inertially-loaded test fixture and a discussion on optimizing the fixture design are contained in the Technical Approach section. Following the conceptual design discussion is a brief summary of the design of the specimen. A discussion of laboratory tests results and how these results impacted the final design of the test fixture is also included. Lastly, procedures for preparing and testing specimens are given in the appendices.

3. BACKGROUND INFORMATION

Many parts of an aircraft which are particularly susceptible to fatigue are inaccessible without extensive disassembly of the aircraft. Included among these parts are the aircraft bulkheads. Many of the bulkheads have a history of cracking at points of high stress. This is particularly true near the wing attachment lugs on many fighter aircraft. Unfortunately, examination of these high stress areas requires near-complete disassembly of the aircraft followed by intensive nondestructive evaluation (NDE) of typical crack locations. Even after aircraft disassembly and NDE, many flaws go undetected because the crack is closed when the part is not loaded. Additionally, many cracks are not detected because the evaluator fails to apply the highly localized NDE technique in the exact area in which the crack is located.

Much emphasis is presently underway to develop techniques that would permit NDE of typical problem areas without significant disassembly of the aircraft structure. The method under investigation at NADC is to monitor the problem areas using a small sensor package that can be installed in the aircraft to monitor cracking throughout the aircraft's service life. In order to evaluate potential NDE techniques, however, it may be necessary to monitor these candidates during flight to determine if these techniques can be effective during aircraft operation. Since any bulkhead which is known to be cracked would be replaced because of safety considerations, the only realistic approach to testing these NDE candidates during actual flight tests is to simulate cracking of the aircraft component with a test fixture capable of producing controlled crack propagation.

The in-flight inertially-loaded test fixture will first be used to demonstrate the feasibility of monitoring crack growth in an aircraft environment using an acoustic emission technique. Acoustic emission monitoring systems operate on the premise that acoustic energy is released when a crack in a material propagates. In general, acoustic emission monitoring is most effective in brittle, high strength materials which have high strain energy release rates. Acoustic emission monitoring systems utilize acoustic transducers (made from piezoelectric materials) which convert the minute displacements caused by the acoustic energy into a voltage differential. The voltage signal is then processed by a computer to discriminate between signals produced by crack propagation from those produced by other sources. The transducers are sensitive to acoustic energy in the frequency bandwidth of 100 to 1000 kHz and can be glued or clamped to the material to be monitored as long as the transducer is acoustically coupled to the material. The test fixture and the acoustic emission monitoring system are to be flown on an F-18.

An important attribute of the acoustic emission monitoring system is its capability to monitor a large area (ie. a bulkhead) using a few 1/2" diameter sensors. Through proper placement of three sensors, the location of the crack source could be identified using triangulation techniques incorporated into system software. The source localization software operates on the difference in arrival times between the three sensors and the speed of sound in that material. Using the difference

in arrival times and the speed of sound in the particular material, the monitoring system calculates the relative distances which the acoustic emission occurred from each sensor. Three sensors permit pinpointing the crack source location in two dimensions (which is adequate for plate-like structures).

In recent years the Canadian Defense Research Establishment has put significant effort into developing in-flight acoustic monitoring capabilities. Two years ago, McBride et al. [1] demonstrated that it is possible to detect crack growth in aluminum 7075-T651 specimens during aircraft flight. The specimens were loaded by an inertial-loading fixture similar to the one described in this paper. The tests were conducted on both a Canadian C-5 and a British Tornado. Later tests conducted by McBride et al. [2] demonstrated crack growth detection and source localization capabilities during wing ground durability and damage tolerance tests. Crack presence was then verified by eddy current and liquid penetrant NDE techniques.

4. TECHNICAL APPROACH

4.1 Design Constraints

Many constraints were imposed on the design of the in-flight inertially-loaded test fixture because the fixture had to be certified as flightworthy for fighter aircraft. For such a fixture to be permitted to fly on an aircraft such as an F-18 or F-14, the fixture must remain contained under 20 g acceleration (crashworthy). Additionally, fighter aircraft have little space for locating additional equipment, which dictates that the volume of the test fixture be kept to a minimum.

In addition to the constraints imposed because the fixture will be flown on a fighter aircraft, other restrictions were imposed to insure that the tests would closely approximate the actual conditions of crack propagation in bulkheads. The specimens were fabricated from 7075-T6 Aluminum alloy because a number of the F-18 bulkheads which have been reported to have cracking problems are also made of this material. Also, the loading mechanisms were designed of steel in order that the deformation (deflection) would be concentrated in the specimen.

4.2 Test Fixture Design Concept

The design of the test fixture had to utilize a large mechanical advantage to achieve enough force to propagate a crack in the 7075-T6 specimen while keeping the volume of the fixture small. An inertial-loading scheme was concluded to be the simplest and least likely to interfere with other systems on the aircraft. The fundamental method for achieving a large mechanical advantage is through the use of levers or pulleys. Levers were considered to be the better choice since they offer a more constrained method of loading. To maximize lever advantages, the specimen was mounted in a cantilever configuration while the loading lever arm was positioned to cross back over the specimen. The final design of the inertially-loaded test fixture is shown in Figure 3. The length of the test fixture was to be limited to 21.5 inches which is the length of the portable computer that will accompany the test fixture during flight. The cantilever grip (Figure 4) was fabricated from mild steel and was designed to accommodate a variety of specimen dimensions and orientations. Calculation of the applied load required to produce crack propagation could not be accomplished until the specimen width, thickness, orientation, and initial crack length were selected.

4.3 Specimen Design

Design of the specimen used 7075-T6 Aluminum. The only available thickness for specimen fabrication was 0.1 inches. The width of the specimen was cut to 1.0 inch to create a specimen similar to that used in previous Canadian tests [1]. The specimen was positioned edgewise to permit greater crack



FIGURE 3. In-Flight Inertial-Loading Fixture.

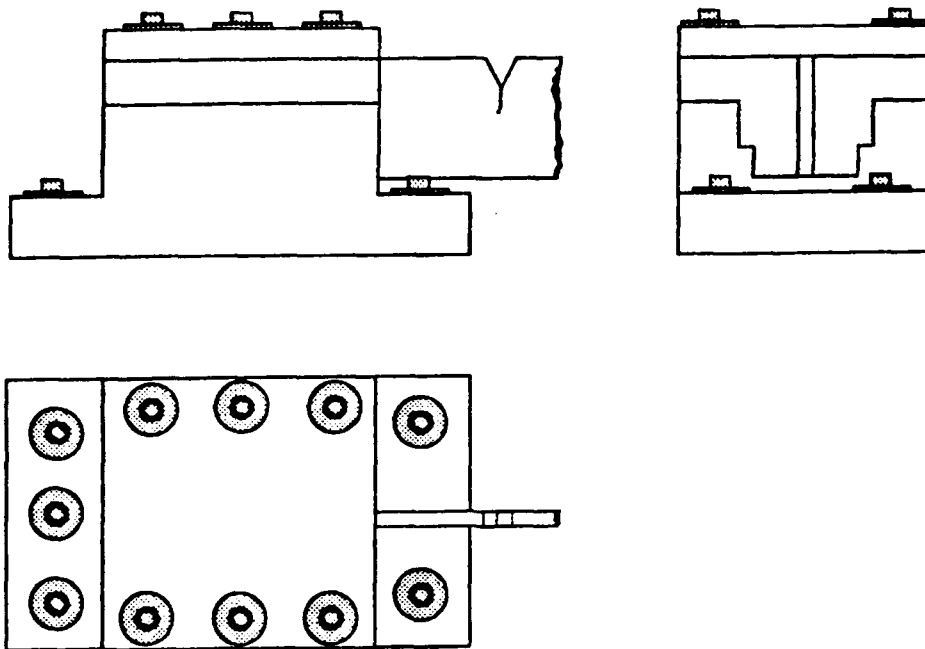


FIGURE 4. Cantilever Grip Assembly.

propagation prior to the onset of fast fracture and to provide greater rigidity to the specimen. This reduced the deflection of the specimen away from the precrack location. The specimen was also reinforced with two more pieces of 0.1 inch thick 7075-T6 aluminum from the load point to a point $3/4$ inch from the precrack to concentrate deformation at the precrack. The procedure for constructing specimens is given in Appendix A.

The precrack was oriented near the cantilever grip to maximize the mechanical advantage. A fracture mechanics analysis was performed to determine the load required to propagate a crack of given length. The geometric correction factors for a 3-point bending configuration were used to approximate those of a cantilever configuration. The results showed that between 97 and 117 pounds applied at a load point 10 inches from the precrack would be required to propagate a precrack ranging from 0.15-0.40 inches. The details of the analysis are given in Appendix B. Based on this analysis a precrack length of $1/4$ inch (load = 1120 inch-pounds) was chosen to allow for significant stable crack growth prior to fast fracture.

Three specimens with initial precrack lengths of $1/4$ inch were loaded in a cantilever configuration using an Instron Test Machine to determine the load necessary to propagate the precrack. The test showed that the onset of crack propagation occurred at 830 inch-pounds and fast fracture occurred at 890-900 inch-pounds. The 830 inch-pounds load was

used to calculate the loading weight required at the design point (acceleration = 4 g's). The calculation of the design point load weight is shown in Appendix C.

4.4 Test Fixture Sizing and Final Design

The 21.5 inch restriction on the overall length of the fixture permitted the loading lever arm to be no longer than 19 inches. The specimen and the grip had to fit between the loading weight and the load point. The weight was not to extend more than 2 inches along the length of the lever arm. Also, the load point was chosen to be 2 inches from the pivot point. The resulting load configuration yielded approximately a 9:1 mechanical advantage in the lever arm (see Appendix B for moment balances). The specimen and grip had to fit in the remaining 15 inches of unused space. The maximum specimen length was limited to 11 inches plus the 3 inches to be held in the grip. This yielded approximately a 10:1 mechanical advantage for the specimen. Hence, the effective lever length of the fixture was 90 inches.

The load weight was selected so the precrack would propagate under an acceleration of 4 g's (see Appendix C for calculations). It was calculated that a load weight of approximately 1.6 pounds would produce the desired 830 in-pound load (load at which cracking occurred in preliminary tests) at 4 g's acceleration. The 1.6 pound weight produced a static load at the load point of 20.7 pounds.

Three tests were run on an Instron using the in-flight test fixture to determine if the applied load would cause the precrack to propagate under an equivalent 4 g acceleration. An additional 5 lbs was added to the load weight to simulate 4 g's (1.6x3). The loading lever arm was observed to deflect to the base of the fixture with minimal crack propagation as detected by visual inspection and acoustic emission sensors. The failure of the crack to grow was attributed to excessive bending of the specimen. As a result a number of minor modifications were made to the specimen design. The precrack was moved from a point 1 inch from the cantilever grip to a point 1/4 inch from the grip. Also, the length of the precrack was increased to 5/16 inch. Lastly, the length of the reinforcement on the specimen was increased 1/4 inch. The specimen design is illustrated in Figure 5. This new specimen design resulted in a static moment at the notch of 220 inch-pounds.

Additional tests were run using the new specimen design, and the results showed crack propagation occurred readily. It was noted, however, that the limited area around the precrack made it impossible to mount three small sensors in that area without placing at least one sensor on the crack. The result was that the information from the sensor placed on the crack became distorted. It was attempted, therefore, to use pinducers (very small acoustic emissions transducers which are shaped like a pin) instead of the standard AE transducers.

The three pinducers were bent to 90 degree angles to fit within the test fixture. The pinducers were then positioned in #41 holes which were drilled in one of the reinforcing pieces of Al 7075-T6 (Figure 6).

Venting holes (#55) were then drilled through the specimen and the second reinforcing piece. The pinducers were then attached to the specimen using a vulcanizing silicone rubber adhesive (see Appendix A for procedure details). The pinducer arrangement was designed to detect acoustic emissions on very compact specimens and to enhance reproducibility of sensor positioning for all tests. The reproducibility created by this fixed sensor arrangement permitted the same equipment settings to be used for all test and made the results of each test directly comparable. Additionally, the specimens can be prepared days in advance of the test. Lastly, the pinducers are an inexpensive alternative to the standard acoustic emission transducers. The arrangement of the pinducers is shown in Figure 6.

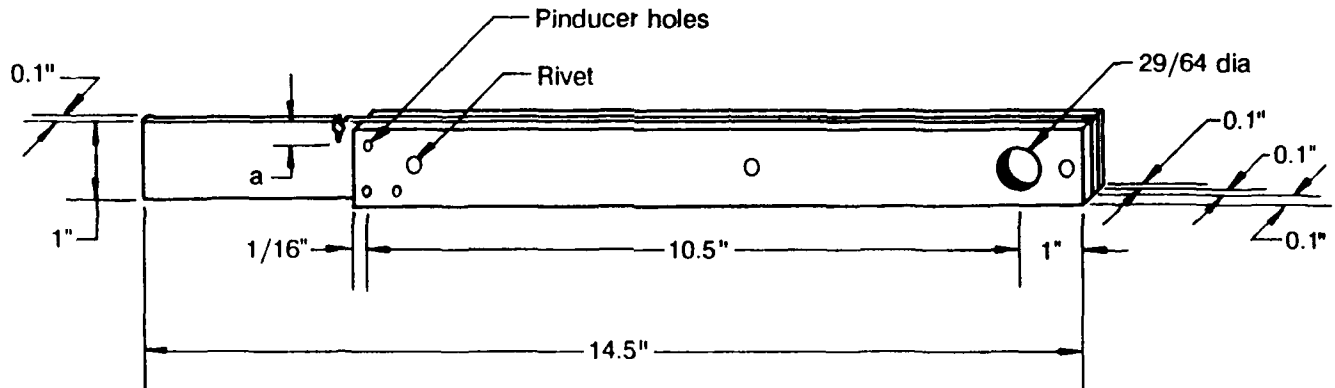


FIGURE 5. In-Flight AE Monitoring Specimen (7075-T6 Al).

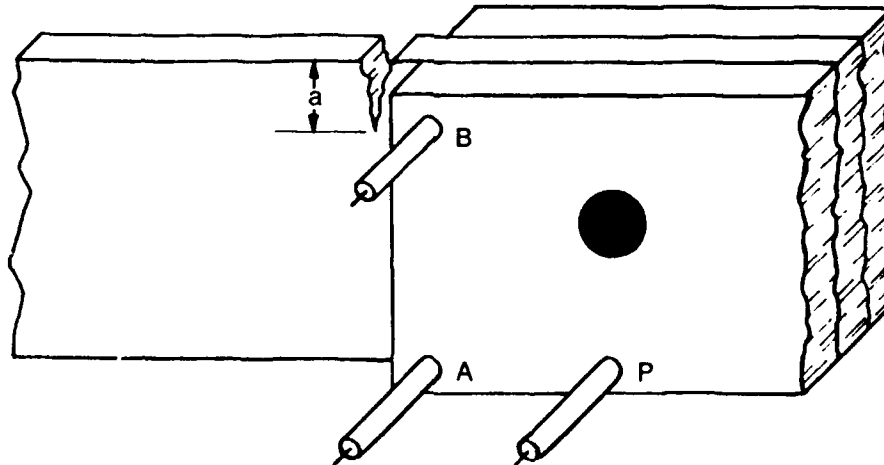


FIGURE 6. AE Pinducer Arrangement.

An alternate acoustic emissions transducer arrangement was prepared as a backup for the pinducers. This arrangement utilizes two - 1/2 inch diameter transducers which are placed within the cantilever grip. Small springs within the grip assure that the transducers remain in contact with the specimen. This alternate transducer arrangement is illustrated in Figure 7.

4.5 Acoustic Emission Laboratory Test Data

The AE test data gathered during the Instron loading of the modified in-flight inertial-loading test fixture development provided data for differentiating the AE signals from noises produced by other physical phenomena. Figures 8, 9, 10, and 11 show characteristic AE signal associated with a 0.5mm pencil lead break (a standard calibration technique), electrical noise, epoxy/aluminum interface delamination, and crack propagation, respectively. In addition to providing reference data for known acoustic phenomena, the acoustic test data also provided correlations between acoustic signals and applied load. These data will be used for developing a flight plan during future flight tests.

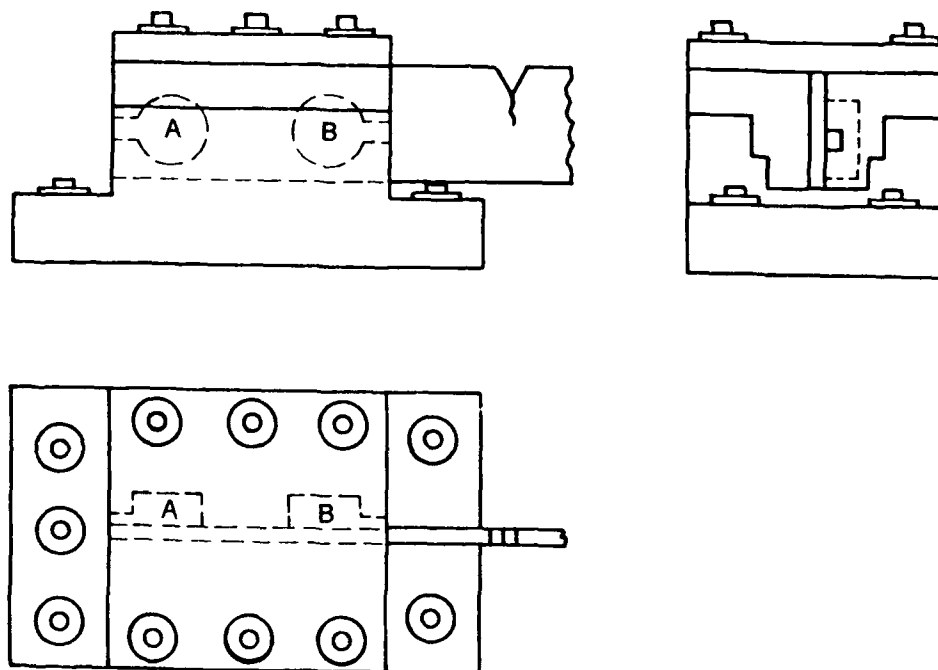


FIGURE 7. Alternate AE Transducer Arrangement for In-Flight Testing.

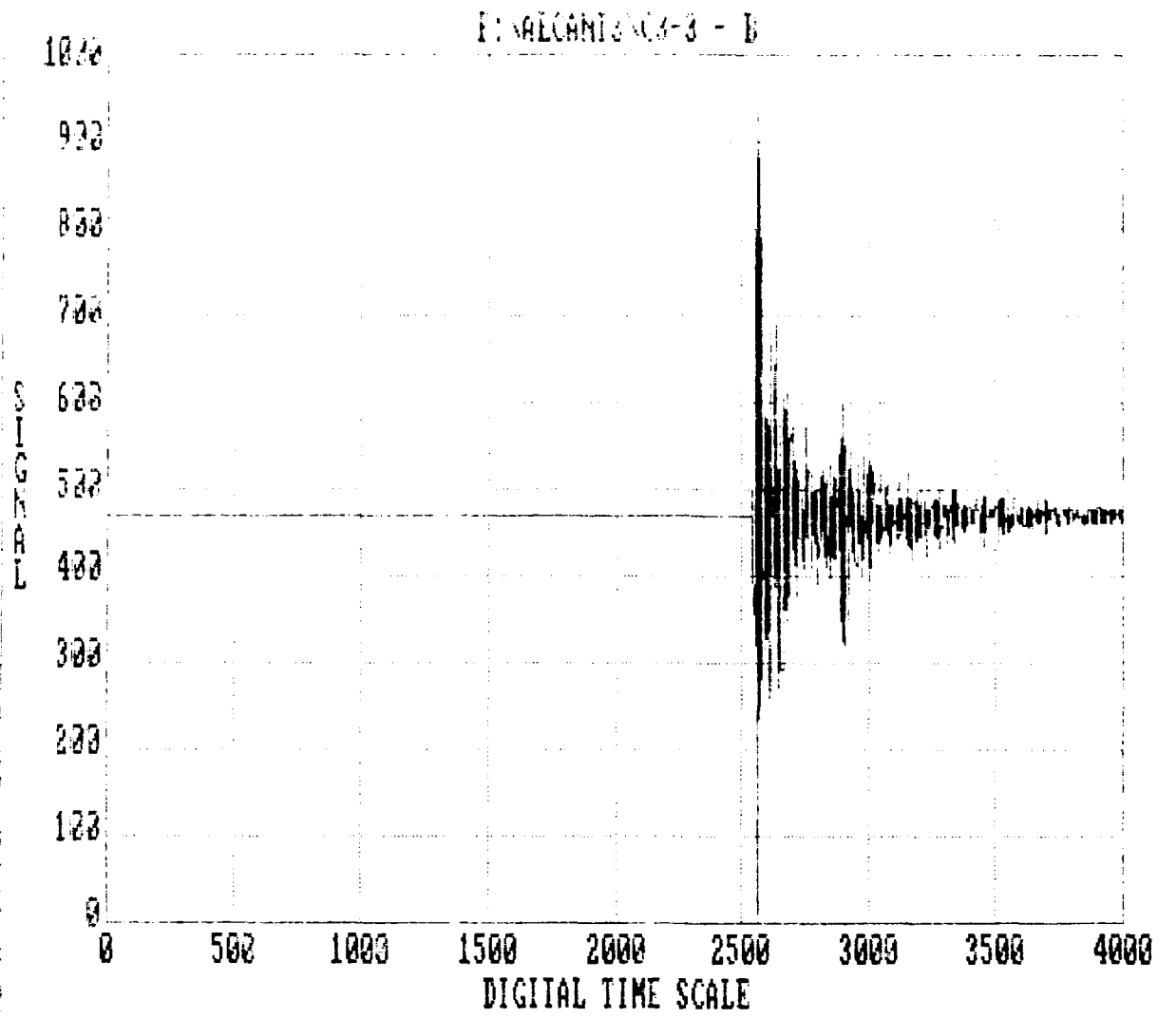


FIGURE 8. Typical 0.5mm Pencil Lead Break AE Signal (Calibration).

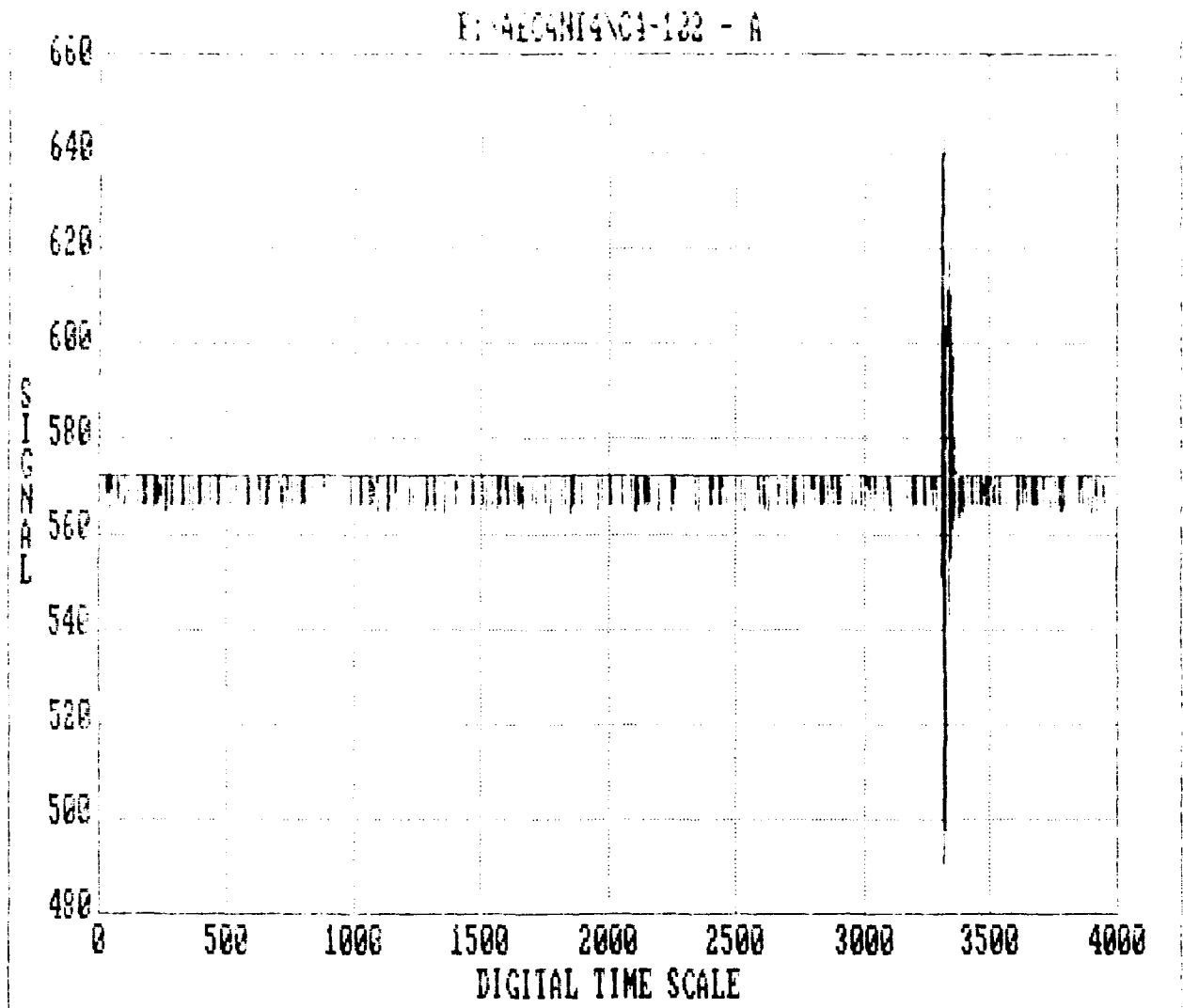


FIGURE 9. Typical Electrical Noise AE Signal.

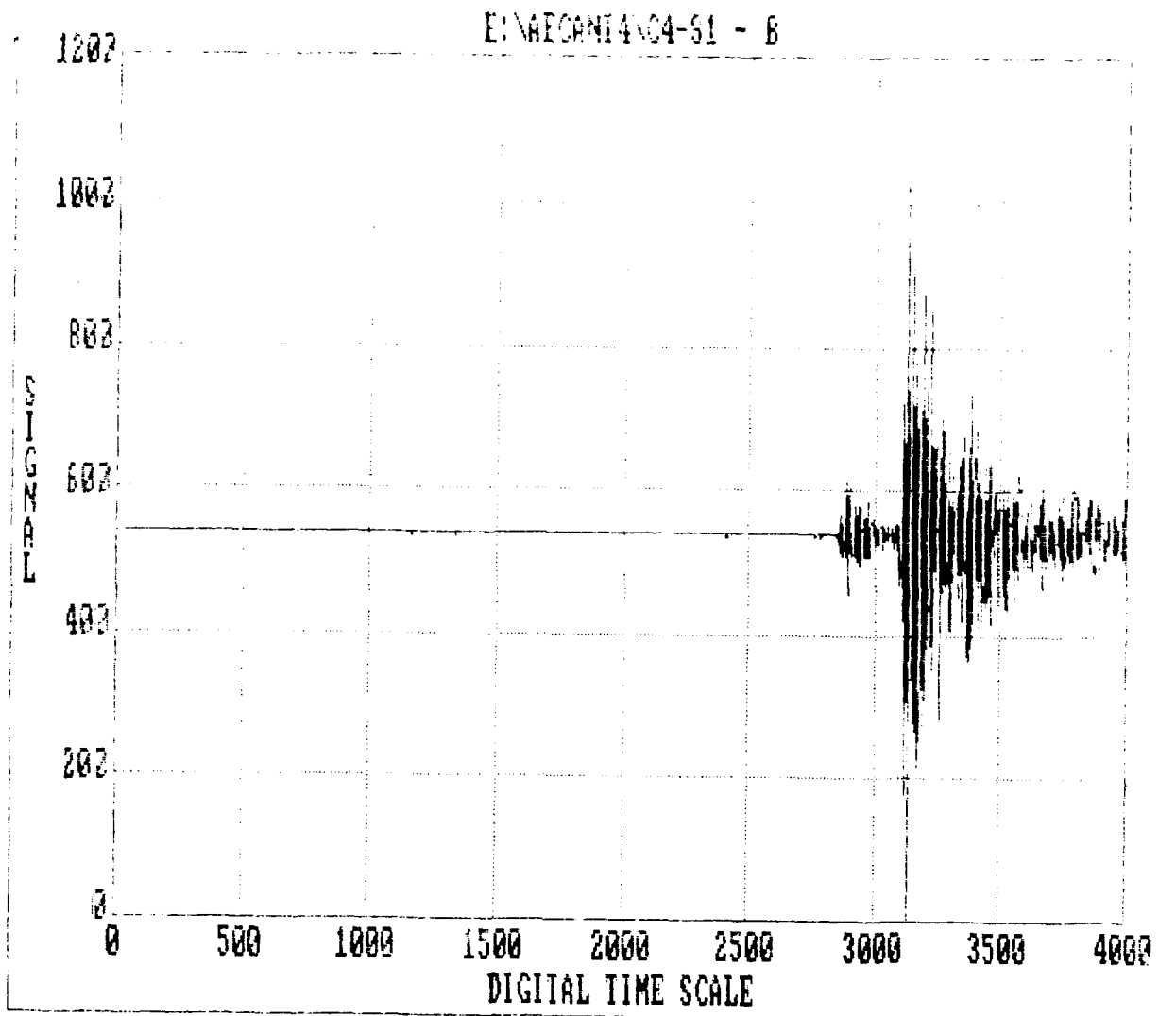


FIGURE 10. Typical Epoxy/Aluminum Delamination AE Signal.

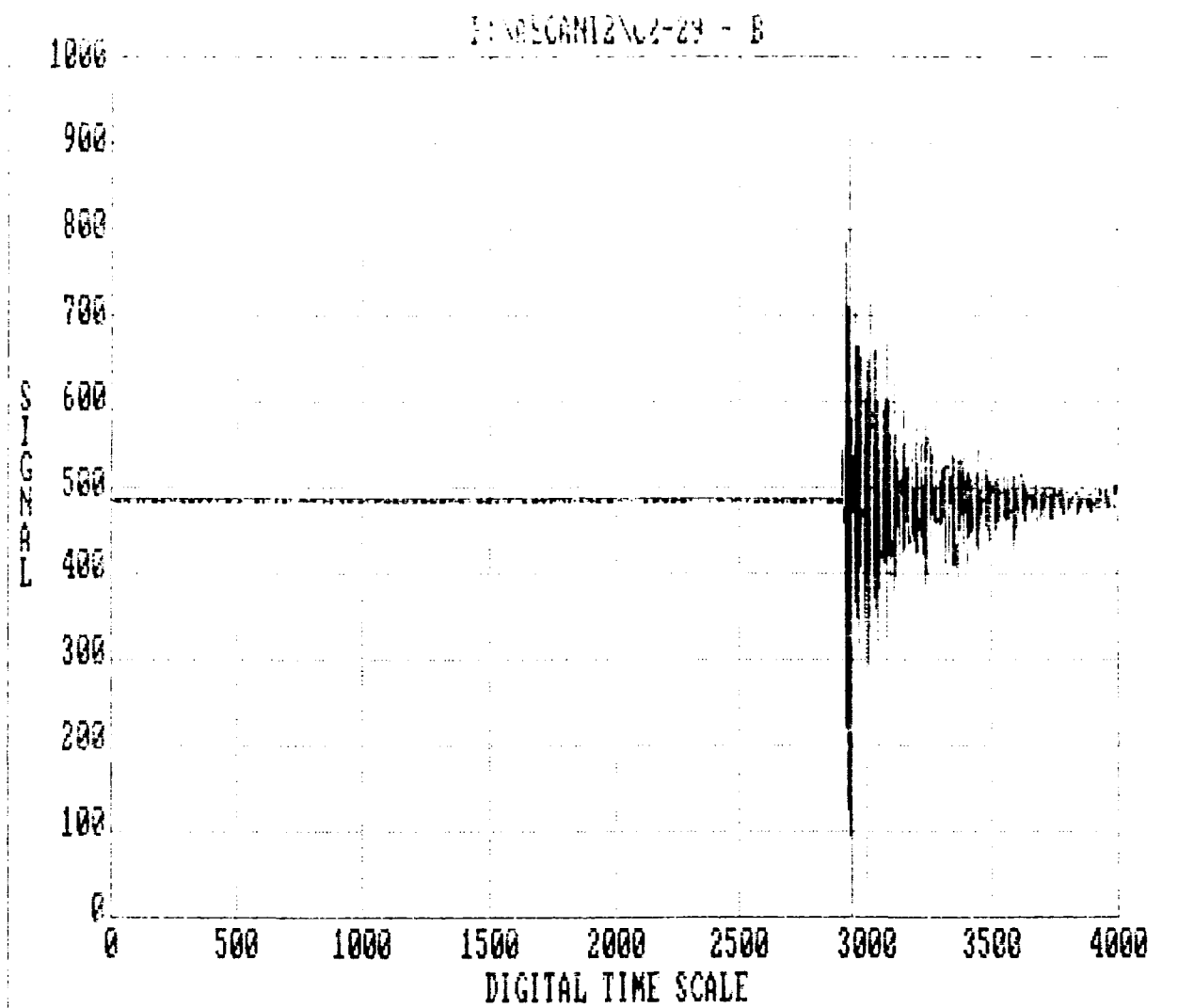


FIGURE 11. Typical Crack Propagation AE Signal.

5. SUMMARY AND CONCLUSIONS

An in-flight inertially-loaded test fixture was developed to evaluate the feasibility of using acoustic emissions monitoring to detect crack growth in difficult-to-access aircraft structures.

The grip was designed to hold a notched, reinforced 7075-T6 aluminum specimen in a cantilevered configuration. It holds aluminum specimens (nominally 0.1 x 1.0 x 14.25 inch) in an edgewise position, although other specimen geometries and orientations can be accommodated.

The specimen design consisted of a 0.1 x 1.0 x 14.25 inch piece of 7075-T6 Aluminum bar reinforced with two pieces of 0.1 x 1.0 x 11 inch pieces of 7075-T6 Aluminum bar epoxied and riveted to the portion of the specimen between the load point and the notch. The load point is 1 inch and the notch is 11.75 inches from the free end of the specimen (10.75 inch moment arm). Prior to testing the specimen, a 5/16 inch fatigue crack was grown from the notch.

The load is applied to the end of the cantilevered specimen through a lever arm. At the end of the lever arm is a 1.6 pound weight. The mechanical advantage of the lever arm is 18:2, and it is constructed of steel to concentrate deformation in the specimen. As a result of this mechanical advantage, the applied moment at the notch is 220 inch-pounds. This applied moment on the specimen is then multiplied by the accelerations of the aircraft to produce sufficient force to fracture the specimen. The design point of the fixture is to produce crack propagation at accelerations greater than 4 g's.

An acoustic emission monitoring system will be used in conjunction with this fixture in future flight tests to record the acoustic waveforms produced by crack advances. The waveforms will then be analyzed to identify features which will distinguish acoustic waveforms caused by crack growth from other noise sources in the aircraft environment.

Test procedures were developed to prepare for flight testing. Additionally, procedures were developed to test the specimens in the laboratory to estimate how much acceleration the test pilot will have to generate in the aircraft to cause the specimen to fail. These procedures include recommendations for adjusting the load required to produce failure of the specimen and, therefore, the aircraft acceleration required for specimen failure.

6. REFERENCES

1. S. L. McBride, M. D. Pollard, J. D. MacPhail, P. S. Bowman, and D. T. Peters, "Acoustic Emission Detection of Crack Presence and Crack Advance During Flight," J. of Acoust. Emissions, Vol. 8, 1989, pp. 4-7.
2. S. L. McBride, M. R. Viner, and M. D. Pollard, "Acoustic Emission Monitoring of a Ground Durability and Damage Tolerance Test," TTCP: Minutes of 1990 Meeting, Subgroup P, Materials Technology, Technical Panel 5, pp. 110-116.

APPENDIX A

SPECIMEN PREPARATION PROCEDURES

MATERIALS

The specimens used for this project were made from 7075-T6 Aluminum because the F-18 bulkheads which are prone to cracking are made from this material. 7075-T6 Aluminum bar was not available at the time of this study; therefore, one inch wide strips of this material were cut from a 0.100 inch thick sheet. Figure 12 shows the materials required to construct a specimen. A list of the materials required follows:

<u>Item Description</u>	<u>Quantity</u>
7075-T6 Aluminum bar- 18.0 x 1.00 x 0.10	1
7075-T6 Aluminum bar- 11.75 x 1.00 x 0.10	2
Rivets- 0.125 inch diameter, 0.375 inch length	3
5 minute epoxy	as required
Pinducer Cables and Pinducers	as required

SPECIMEN PREPARATION PROCEDURES

1. File chevron notch in 18.0 x 1.0 x 0.1 inch piece of aluminum at a point 6 inches from one end. Scribe a mark 5/16 inch from the chevron notch to identify when the precrack has grown to the desired length.
2. Mount specimen edgewise in Krause fatigue machine with chevron notch facing down under load point. Set load to 10 and reset cycle counter. Start machine and tighten load bolt by hand until there is no vibration at the specimen load point. Monitor precrack growth and stop the machine when the precrack has grown to 5/16 inch mark (see Appendix D for more details).
3. Clamp aluminum pieces as shown in Figure 13.
4. Drill 3 holes as shown in Figure 13 using #21 drill bit (or just large enough to put the rivets through if different diameter rivets are used).
5. Drill 29/64 inch diameter hole where designated in Figure 14.
6. Unclamp pieces of aluminum.
7. Drill 3 holes as shown in Figure 14 using #41 drill bit in one of the two reinforcing pieces.
8. Mix 5 minute epoxy and apply an even coat to the surfaces of the aluminum which will be in contact.

9. Place pieces together. Push rivets through the holes to align the pieces.
10. Squeeze rivets in a vice to secure the pieces of aluminum.
11. Let dry overnight.
12. Redrill #41 holes to remove epoxy which may have entered the holes during bonding.
13. Drill #55 holes all the way through the specimen in the center of the #41 holes to allow the RTV to vent.
14. Cut 3.5 inches from end of specimen with precrack and 1/4 inch from the opposite end.
15. Put high temperature RTV in the #41 holes and insert the pinducers in the holes as shown in Figure 15. Let dry overnight.

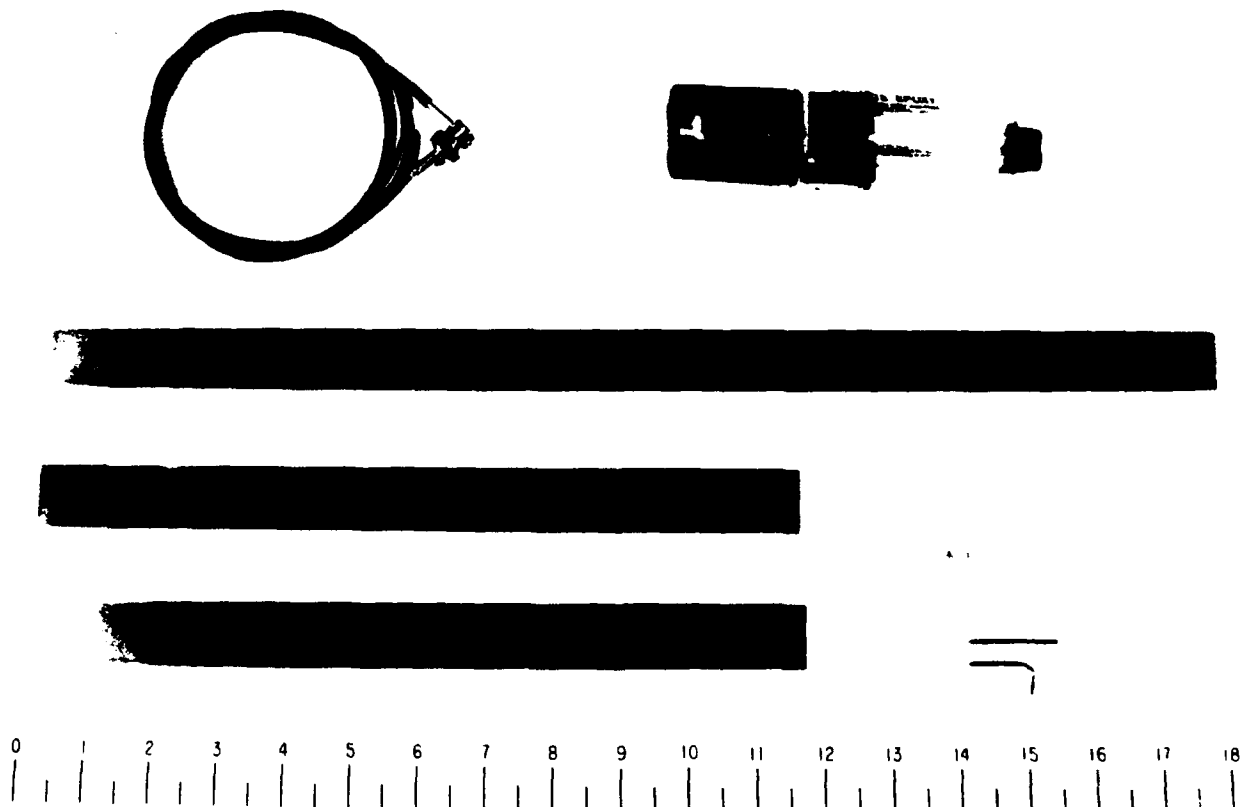


Figure 12. Specimen Preparation - Required Materials.

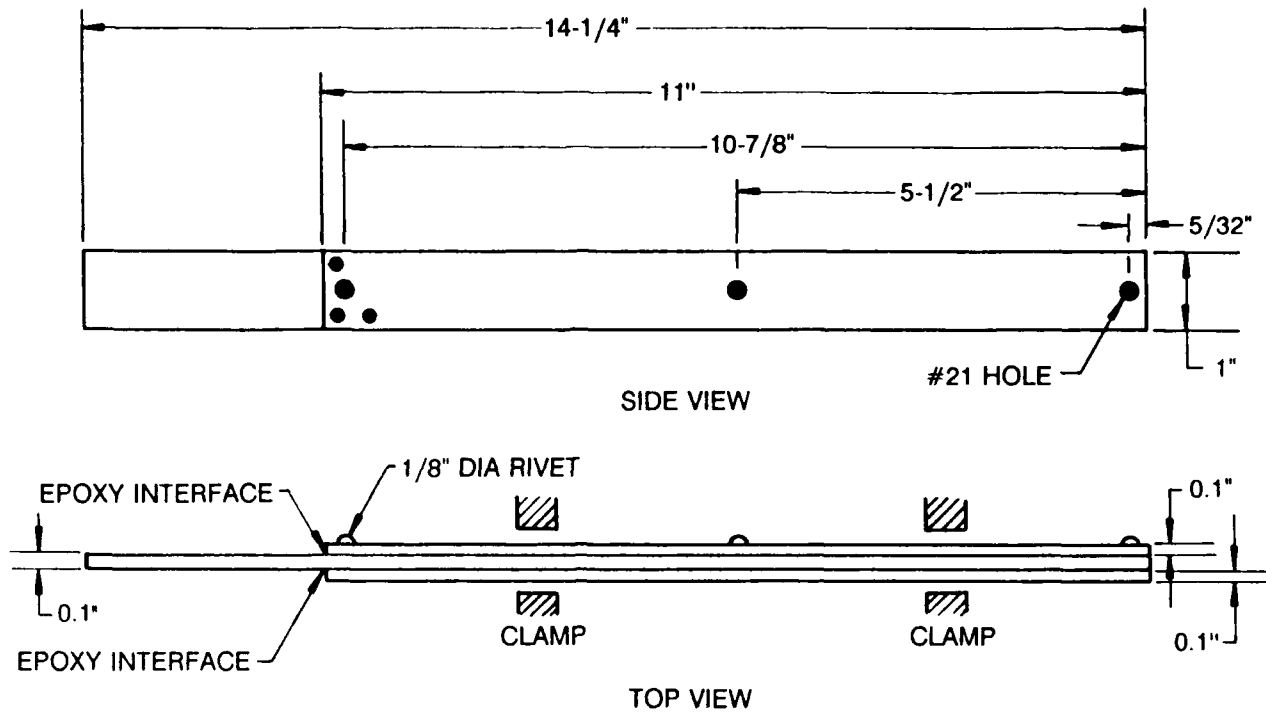


FIGURE 13. Specimen Preparation - Reinforcing the Specimen.

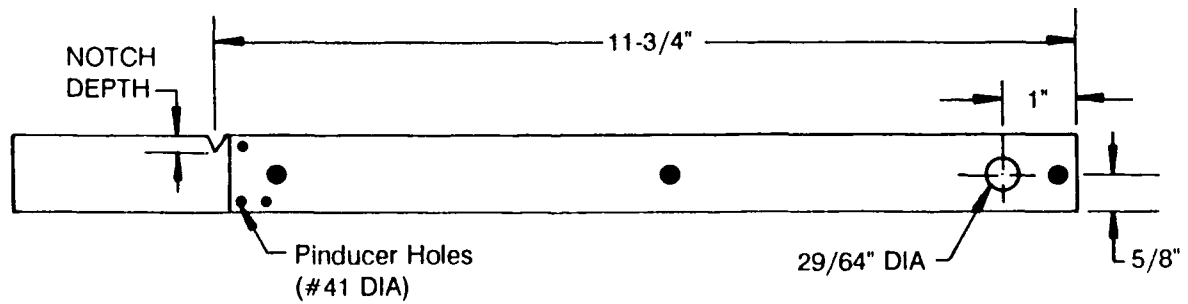


FIGURE 14. Specimen Preparation - Notching and Drilling.



Figure 15. Specimen Preparation - Attaching Pinducers.

APPENDIX B

FRACTURE MECHANICS ANALYSIS FOR PRELIMINARY
DESIGN OF THE IN-FLIGHT INERTIAL-LOADING FIXTURE

Analysis

In the preliminary designing of the in-flight test fixture a number of base design decisions were made. Aluminum 7075-T6 was chosen as the optimum specimen material for the in-flight acoustical monitoring system because it has a high energy release rate and because it is commonly used as a structural material on combat aircraft. A cantilever loading geometry was chosen because it offers a large mechanical advantage in a minimal amount of space and because the Canadian fixture, which used this design, was successful. Once the base decisions were made, a fracture mechanics analysis was performed to size the components to produce controlled crack propagation in an Al 7075-T6 specimen.

The analysis which follows utilizes fracture mechanics relations that were established for three point bending of bars as an approximate solution for cantilever bending of a bar. The analysis assumes that a precracked beam subjected to a load, P , at the midpoint of the distance between the supports of span, S , would behave similarly to a cantilevered beam with a length, S , (span), between the crack and the load point.

The Mode I stress intensity factor for a beam in 3-point bending has been shown to follow the relation:

$$(1) \quad K_I = Y * (6Ma^{0.5})/(tW^2)$$

$$(2) \quad K_I = \sigma * (3.14159 * a)^{0.5}$$

where M is the moment at the crack ($M=f(P,S)$), a is the initial crack length, W is the height of the beam, t is the thickness of the beam, σ is the stress at the crack tip, and Y is a geometric correction factor taken from Figure 16.

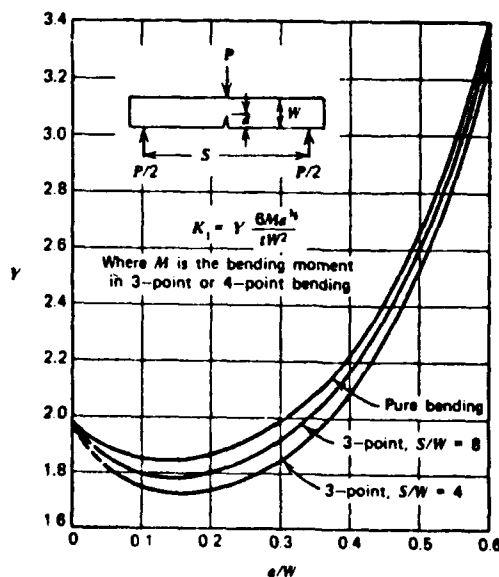


FIGURE 16. Geometric Correction Factor for 3 or 4 Point Bending.

Since the fixture was designed to propagate the crack, the condition where $K_I = K_{I \text{ crit}}$ was examined. Under this condition the critical stress intensity factor is defined by the following relation:

$$(3) \quad K_{I \text{ crit}} = \sigma_{YS} * (3.14159 * a)^{0.5}$$

where σ_{YS} is the yield stress for the material. Similarly this relation could be used to calculate the length of the crack when fast fracture occurs by substituting the ultimate stress for the yield stress. Since the specimen dimensions were chosen based on material availability, the values of W and t were fixed at 1.0 inches and 0.1 inches respectively. Overall limitations on the length of the fixture resulted in the distance between the precrack and the load point being limited to approximately 10 inches. The moment at the crack was a function of the span and the applied load.

$$(4) \quad M = S * P$$

The initial crack length was then varied and the corresponding value of load necessary to produce crack growth was calculated from the following relation which was obtained by combining equation (1)-(4):

$$(5) \quad \sigma_{YS} * (3.14159 * a)^{0.5} = Y * (6 * (10P) * a^{0.5}) / ((0.1) * (1.0)^2)$$

which can be simplified to equation (6) by substituting 73000 psi for σ_{YS} .

$$(6) \quad P = 216/Y$$

Recalling that Y is a function of a and that $S/W = 10$ is very close to pure bending Table 1 was created.

TABLE 1. Predicted Load to Produce Crack Propagation as a Function of Initial Crack Length

Crack Length, a inches	Correction Factor, Y	Load, P pounds	Moment, M in-pounds
0.05	1.89	114	1140
0.10	1.86	116	1160
0.15	1.85	117	1170
0.20	1.86	115	1150
0.25	1.92	112	1120
0.30	1.99	108	1080
0.35	2.09	103	1030
0.40	2.23	97	970

From Table 1 it can be seen that the load required to produce crack propagation increases until a/W exceeds 0.15 and it remains relatively constant for a/W between 0.15 and 0.22. An a/W ratio of 0.25 was, therefore, chosen to assure the crack would not arrest in subsequent loadings at the same load amplitude. This resulted in a design point load of 112 pounds (at 4 g's) applied at a point 10 inches from the precrack location.

APPENDIX C

CALCULATIONS OF THE APPLIED LOADING WEIGHT

Analysis

The pivot arms were designed as long as possible (19 inches from pivot point) within the 21.5 inch overall length limitation. The two pivot arms were made from 1/8" x 1" stainless steel and weighed 0.67 pounds each.

Three tests were performed to experimentally confirm the load required to produce crack propagation and ultimate failure. The results showed that a specimen of the previously stated geometry with a 0.25 inch precrack will fail under a bending moment of 890 inch-pounds (somewhat less than the 1120 inch-pounds estimated in the fracture mechanics analysis). Crack propagation was noted to begin under a bending moment of 830 inch-pounds at the precrack.

The design target for this specimen was to produce crack propagation at aircraft accelerations of 4 g's. Since the load will be applied at a distance of 10 inches from the precrack, the load necessary for crack propagation was determined to be 83 pounds (830 in-lbs/10 in).

A moment balance was performed to determine the amount of weight which needed to be added to the end of the pivot arms to produce the required design load 20.75 pounds (83/4 g's, based on test results) at 1 g with the load point positioned 2 inches from the pivot point. The width and the length of the added weight were fixed at 2.875" and 2" respectively while the height was left variable. Figure 17 shows a diagram of the moment balances for the pivot arm.

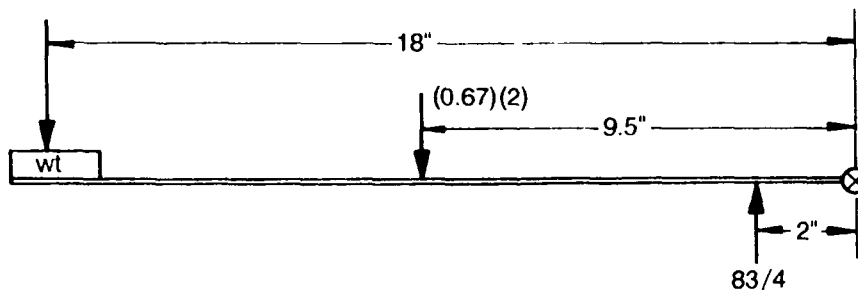


FIGURE 17. Moment Diagram for Pivot Arm Assembly at Design Point.

Computation of the weight required to balance the moments at the design point results in $Wt = 1.59$ pounds. Since the density of steel is 0.283 lbs/in³, the height of the loading weight can be computed to be 0.98 inches. For convenience, however, a height of 1 inch was chosen. Back-calculations showed that this value change the design point acceleration to 3.95 g's. The 1.59 pound weight produced a 20.7 pound static load at the load point.

APPENDIX D

LABORATORY TEST PROCEDURES

I. Precracking Specimens

1. Mount notched specimen edgewise in Krause machine with chevron notch facing down (see Figures 18 and 19 for illustration).
2. Turn loading fixture so the specimen is pinned against the vertical supports.
3. Zero cycle counter.
4. Set load to 10 (see Figure 20 for illustration).
5. Rotate the loading cam until the loading arm is at maximum deflection and adjust deflection stop so a piece of paper will just slide between the stop and the loading arm.
6. Start the test machine and tighten the load bolt until there is no vibration between the load bolt and the specimen.
7. Monitor crack growth with a magnifying glass and a flashlight until the fatigue crack grows to 5/16 inches (Figure 19). If number of cycles becomes excessive (greater than 10,000), increase the maximum load by 5%. If number of cycles is very small (less than 2,000), decrease the maximum load by 5% on future specimens.

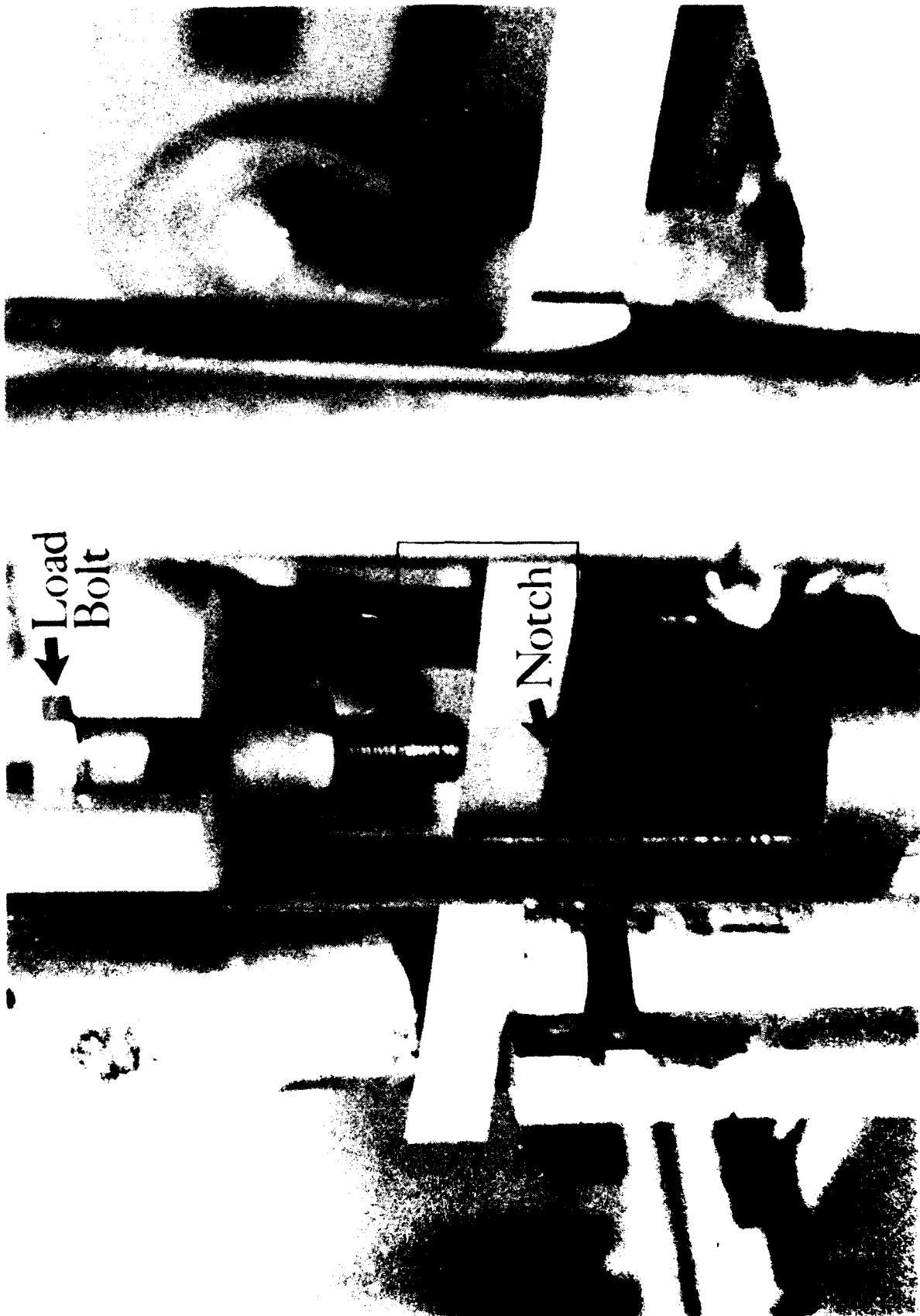


Figure 18. Set-up for Precracking Specimens on Krause Machine.

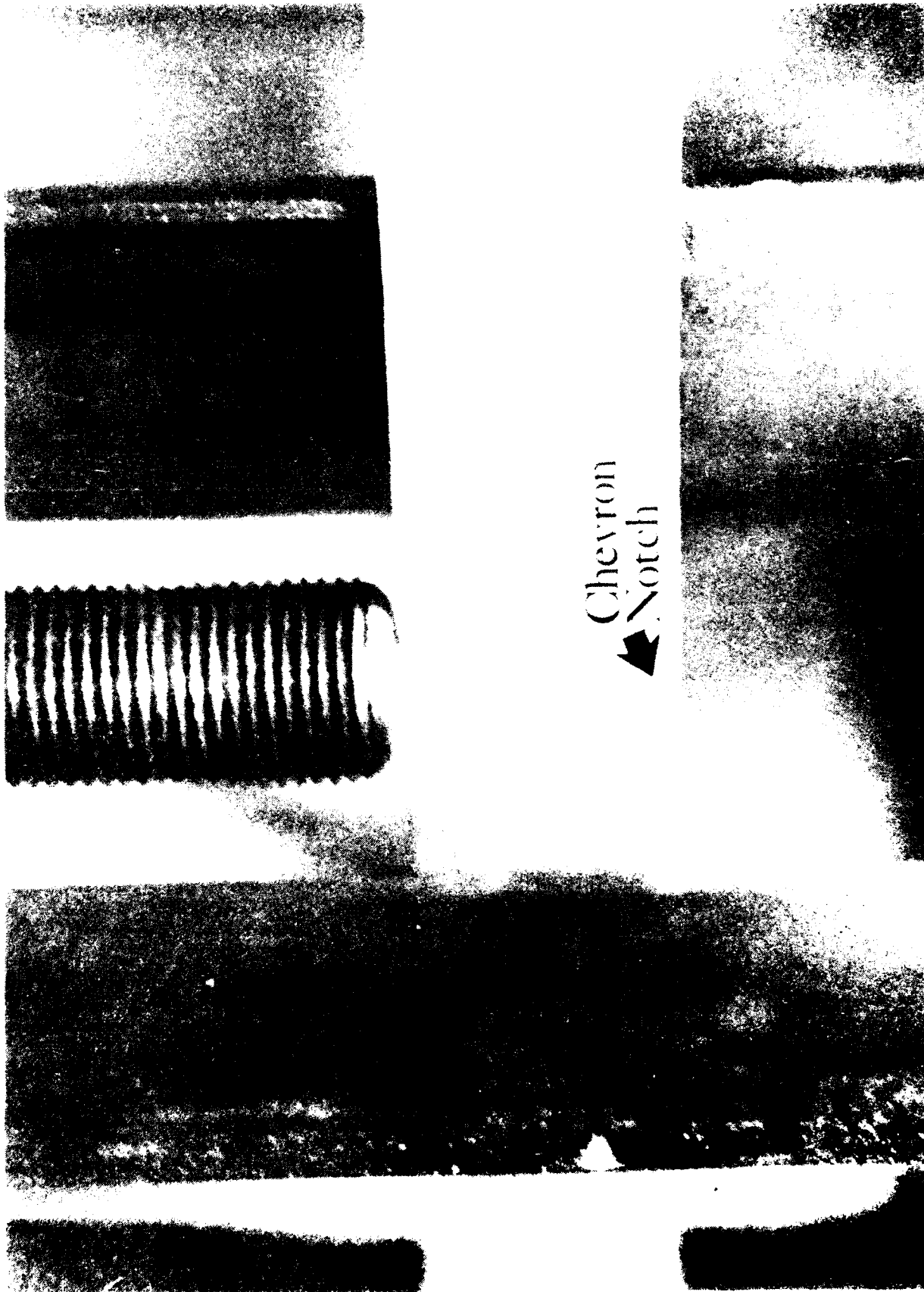


Figure 19. Close-up of Specimen Mounted in Krause Machine.

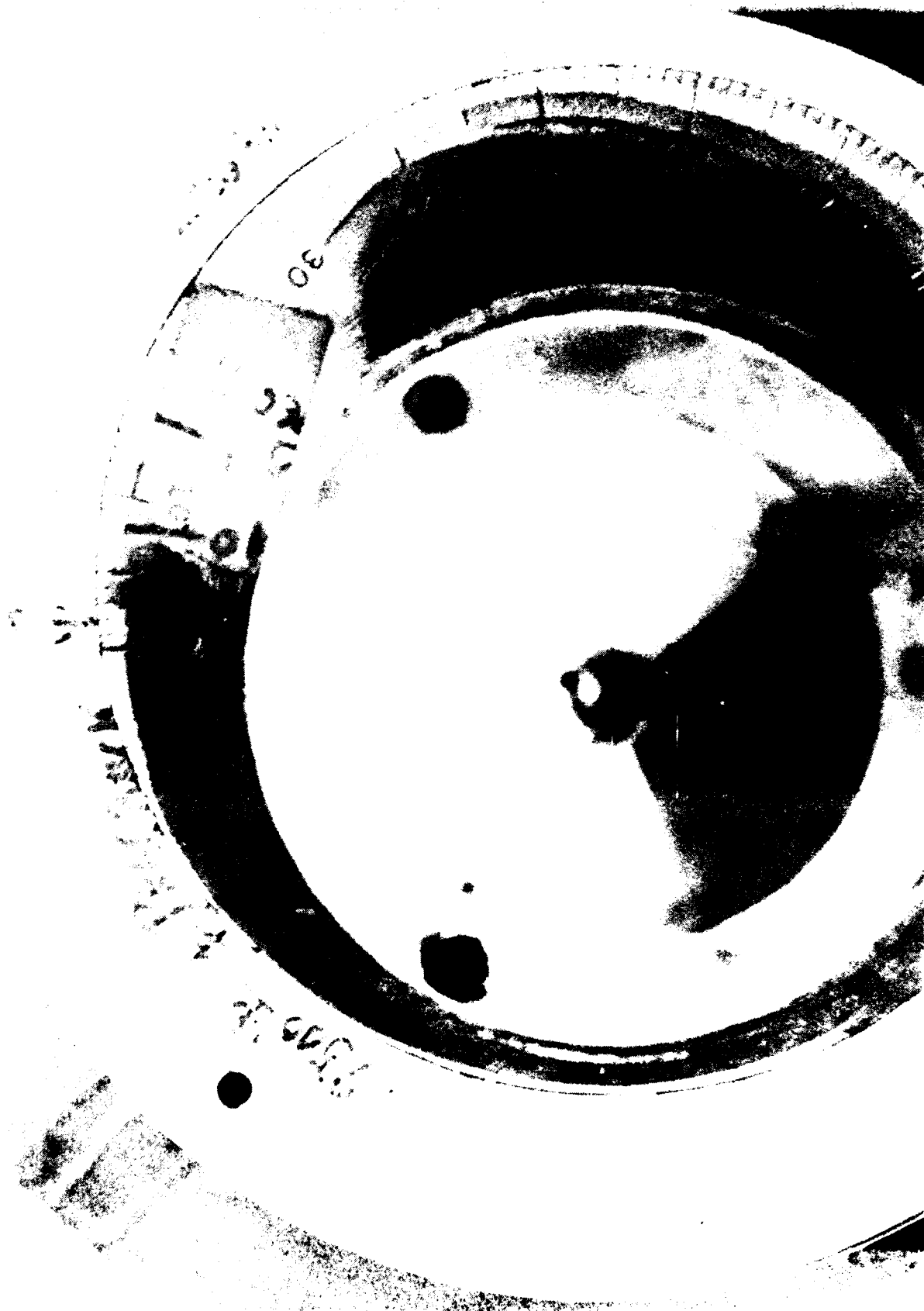


Figure 20. Setting the Load on the Krause Machine.

NADC-91074-60

II. Lab Testing Precracked Specimens to Verify Failure Load

1. Connect specimen to weighted lever arm using load pin, spacers, and washers (shown in Figure 21).
2. Mount precracked specimen in cantilever grip.
3. Clamp fixture to Instron or MTS compression table with load weight under machine cross-head (illustrated in Figures 22, 23, and 24).
4. Zero load on test machine.
5. Connect acoustic emissions (AE) sensors to specimen using clamps or tape. Be sure to use a coupling agent between the sensor and the specimen.
6. Prepare AE equipment for monitoring a test (see AE monitoring system manuals for procedure).
7. Run a displacement controlled test using a slow cross-head speed. Monitor and record the applied load. Manually control the maximum and minimum cross-head displacements (and applied loads) to simulate the reaction of the specimen to various accelerations on board the aircraft. The following conversion factor may be used to convert applied loads to accelerations:

**Addition of 2 lbs Applied - Addition of 1 g Acceleration
to Loading Weight**

Therefore, if the specimen fractures when the test machine applies 10 lbs. of load to the loading weight, it can be expected that fracture on the aircraft will occur at 6 g's ($10/2+1$).

8. If the specimen does not fail under 12 lbs. applied load, it will not crack during flight. If this occurs, a number of solutions can be pursued. A longer precrack could be grown or the loading weight could be increased. Additionally, the specimen material or geometry may be changed to reduce its strength.

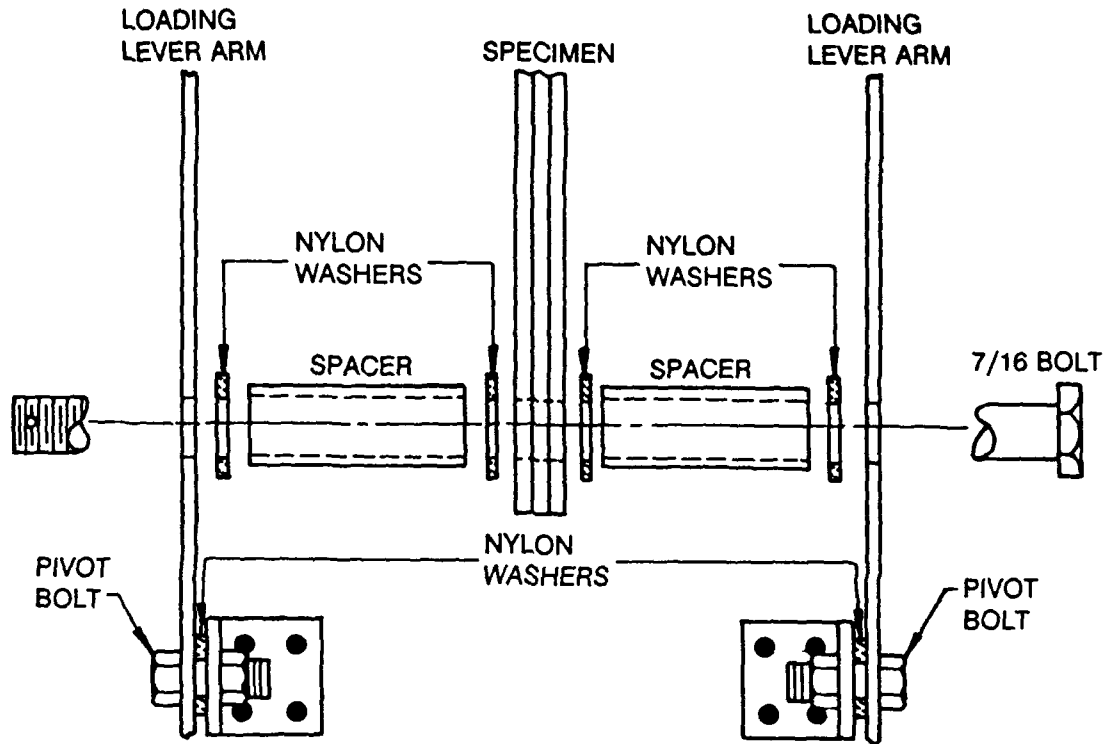


FIGURE 21. Illustration of Connection of Specimen and Loading Lever Arm.

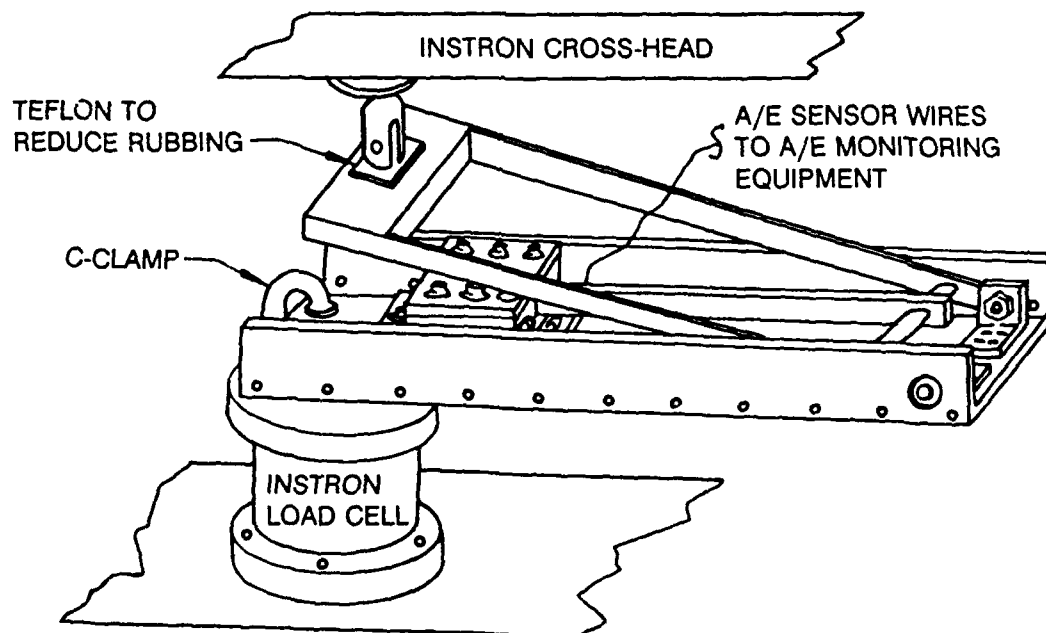


FIGURE 22. Schematic Representation of Set-up for Verifying Failure Load using Laboratory Test Machines.

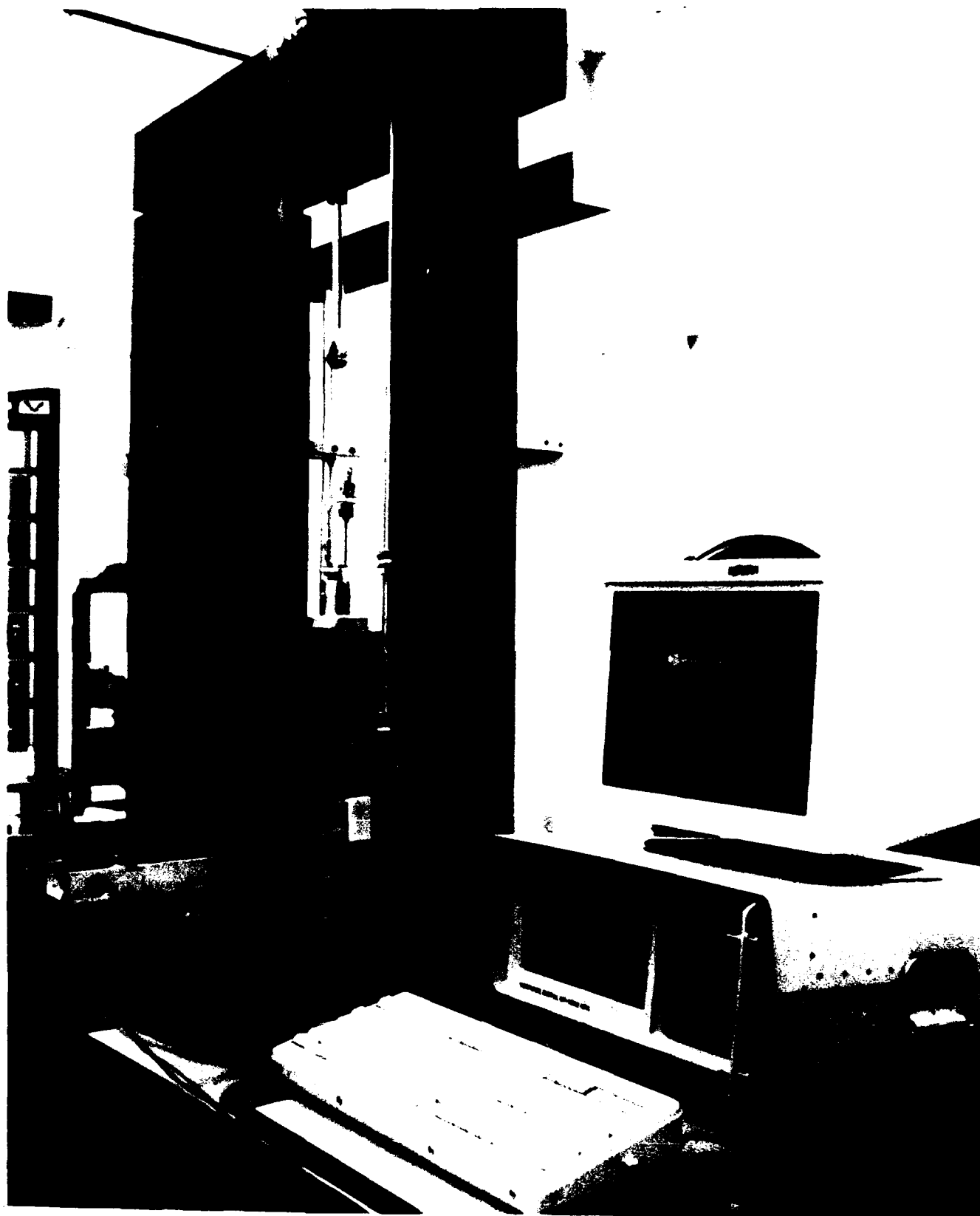


Figure 23. Failure Load Verification Using Instron Test Machine.

NADC-91074-60



Figure 24. Detailed View of In-Flight Fixture
Mounted on Instron Test Machine.

DISTRIBUTION LIST
Report No. NADC-91074-60

	No. of Copies
Naval Air Systems Command	6
Washington, DC 20351-0001	
(1 for AIR-530T; L. Slotter)	
(1 for AIR-5302; M. Dubberly)	
(1 for AIR-5304; J. Collins)	
(1 for AIR-5304B; J. Stevenson)	
(1 for AIR-5304B; S. Povinelli)	
(1 for AIR-5304C; D. Moore)	
 Naval Air Development Center	 23
Warminster, PA 18974-5000	
(1 for Code 60C; M. Donnellan)	
(1 for Code 606; I. Shaffer)	
(1 for Code 604; S. Haung)	
(2 for Code 6063; J. Waldman, W. Frazier)	
(2 for Code 6043; J. Alper, E. Kavtz)	
(10 for Code 60633; P. Kulowitch)	
(4 for Code 60633; W. Scott, J. Boodey, D. Granata, G. McConnell)	
(2 for Code 8131)	
 Naval Postgraduate School	 1
Monterey, CA 93943	
(1 for M. Gorman)	
 Defense Advanced Research Projects Agency	 1
Defense Sciences Office	
1400 Wilson Blvd.	
Arlington, VA 22209	
(1 for W. Baker)	
 Defense Technical Information Center	 2
ATTN: DTIC-FDAB	
Cameron Station BG5	
Alexandria, VA 22304-6145	
 Center for Naval Analysis	 1
4401 Fort Avenue	
P.O. Box 16268	
Alexandria, VA 22302-0268	