AFFDL-TR-76-30

AD-A025795

F-IIIA WING FATIGUE TEST PROGRAM

EXPERIMENTAL BRANCH
STRUCTURES DIVISION

APRIL 1976

TECHNICAL REPORT AFFDL-TR-76-30
FINAL REPORT FOR PERIOD 24 MARCH 71 - 10 JULY 74

Approved for public release; distribution unlimited

20071128124

AIR FORCE FLIGHT DYNAMICS LABORATORY
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This technical report has been reviewed and is approved for publication.

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AIR FORCE - 4 JUNE 76 - 100

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SECURITY CLASSIFICATION OF THIS PAGE *(When Data Entered)*

REPORT DOCUMENTATION PAGE		READ INSTRUCTIONS BEFORE COMPLETING FORM
1. REPORT NUMBER AFFDL-TR-76-30	2. GOVT ACCESSION NO.	3. RECIPIENT'S CATALOG NUMBER
4. TITLE *(and Subtitle)* F-111A Wing Fatigue Test Program		5. TYPE OF REPORT & PERIOD COVERED Final Report 24 Mar 71 to 10 Jul 74
		6. PERFORMING ORG. REPORT NUMBER
7. AUTHOR(s) Robert L. Schneider		8. CONTRACT OR GRANT NUMBER(s)
9. PERFORMING ORGANIZATION NAME AND ADDRESS Air Force Flight Dynamics Laboratory (AFFDL/FBT) Wright-Patterson Air Force Base, OH 45433		10. PROGRAM ELEMENT, PROJECT, TASK AREA & WORK UNIT NUMBERS
11. CONTROLLING OFFICE NAME AND ADDRESS Air Force Flight Dynamics Laboratory (AFFDL/FBT) Wright-Patterson Air Force Base, OH 45433		12. REPORT DATE April 1976
		13. NUMBER OF PAGES 46
14. MONITORING AGENCY NAME & ADDRESS *(if different from Controlling Office)*		15. SECURITY CLASS. *(of this report)* Unclassified
		15a. DECLASSIFICATION/DOWNGRADING SCHEDULE

16. DISTRIBUTION STATEMENT *(of this Report)*

Approved for public release; distribution unlimited

17. DISTRIBUTION STATEMENT *(of the abstract entered in Block 20, if different from Report)*

18. SUPPLEMENTARY NOTES

19. KEY WORDS *(Continue on reverse side if necessary and identify by block number)*

Fatigue life
F-111
Swing wing aircraft

20. ABSTRACT *(Continue on reverse side if necessary and identify by block number)*

An F-111A wing was fatigue tested in the full stores configuration for evaluation of the wing, wing hard points, and fixed pylons. The spectrum consisted of nine test conditions with loads applied at three wing sweep angles: 25, 50, 72.5°. The repeated application of these test conditions simulated 24,000 hrs of service (6 test lives). Upon completion of the six test lives, the spectrum was modified and cycling continued for an additional four test lifetimes (16,000 hrs). After completion of the 10th fatigue test lifetime, 65% limit load constant amplitude cycles were applied until failure occurred. The F-111A wing has a satisfactory fatigue life for the spectrum imposed upon it.

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FOREWORD

This report was prepared by the Air Force Flight Dynamics Laboratory as a formal record of the F-111 Full Store Configuration ("Iron Bombs") Wing Fatigue Test conducted under Project No. 324A0501, by the Structures Test Branch (FBT). The program was directed by Mr. Robert L. Schneider, Project Engineer, assisted by Mr. Harold D. Stalnaker. Mr. John E. Pappas and Mr. Joseph R. Pokorski were responsible for all instrumentation and data reduction. Mr. James H. Specht and Mr. Ronald E. McQuown were responsible for the operation and maintenance of the automatic loading and programming equipment, respectively.

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SECTION I

INTRODUCTION

1. BACKGROUND AND TEST SUMMARY

As a result of South East Asia experience, the F-111 SPO requested a fatigue test of the F-111A movable outer wing in order to certify it as an all weather strike fighter carrying extensive ordinance on the wing's pivoting and fixed pylons. The test goal was initially four test lives, but was extended to ten fatigue lifetimes to correlate the full stores configuration test to the "clean wing" fatigue test conducted by General Dynamics at San Diego. The clean wing had been tested to ten fatigue lifetimes plus 10,151 constant amplitude cycles before failure occurred in the transition section. Failure of the full stores configuration occurred in the pivot to outer wing splice (Figure 15) after ten fatigue lifetimes (40,000 flt. hrs) plus 7308 constant amplitude cycles.

Testing began on 24 March 1971. The end of the first fatigue life was reached on 23 July 1971. At this time all fuel flow holes in the wing pivot fitting were polished and a boron/epoxy composite doubler (Figure 1) was installed on the lower surface of the pivot fitting. Both procedures were part of the F-111 Recovery Program Fleet Retrofit. On 26 November 1973, the tenth fatigue test lifetime was completed. On 10 July 1974, constant amplitude testing was stopped when large cracks appeared in the lower skin transition area (Figure 15).

SECTION II
DESCRIPTION OF TEST ARTICLE

The test article was composed of a left hand outer wing box with pivoting pylon support structure and two fixed pylons attached. Such items as the wing tip, slats, fixed leading edges, and trailing edges were not included with the test structure, although provision was made for introducing the slat, flap, and aileron loads into the wing structure (Figure 2).

SECTION III

TEST METHOD

1. TEST FIXTURE

The wing was attached to a dummy carry-through fitting that was bolted to the large "H" frames in the northeast section of the AFFDL Structures Test Facility (Figure 3). This dummy fitting simulated the stiffness of the actual F-111 carry-through so as not to bias the test results.

The wing was placed in a zero "G" condition through use of a dead weight counterbalance system that attached to the test linkage and applied a load equal to that of the linkage weight and the weight of the structure.

2. SIMULATION OF TEST CONDITIONS

The F-111 is a swing wing aircraft; its wing sweeps from a 16° maximum forward position to a 72.5° maximum aft position. The full stores configuration has two fixed pylons and two pylons that pivot so as to maintain the same air stream characteristics independent of the wing sweep angle. To simulate the pylon loads at the various sweep conditions, the pylon sweep mechanism and the resultant loads were rotated but the dummy pylons remained in the 26° sweep orientation (Figure 4). A special dummy pylon with nine loading points was needed to accomplish this rotation of test loads (Figure 5).

To simulate the effect of the sweep on the wing airloads, the test loads were changed both spanwise and chordwise to furnish the correct load distribution for each sweep position. The wing airloads were distributed into the wing by means of whiffle trees and load bolts attached directly into the spars through the top skin (Figure 6).

The above procedure allowed the wing to be kept in a fixed position throughout the test and this greatly simplified the loading procedures.

3. TEST EQUIPMENT

a. Load Programming Equipment

Load programs for the test were furnished from the Control Load Programmer (CLP) in the master control room and transmitted through patch panels, cabling panels, cabling runs, etc. to the load controllers located on the test platform near the F-111 wing. The CLP, itself, is a magnetic drum-based, digital function generator, used to supply a 0 to 50 volt program to the individual closed loop load controllers, which are located on a platform near the test article and remote from the CLP. The CLP has five variable time-based tracks which are used to adjust the cyclic rate applied to this structure. The 0 to 50 volt program signal level of the CLP is divided into 200 equal intervals.

The test structure was loaded by 47 hydraulic loading cylinders, each of which requires a load program signal to be furnished by the CLP. One program channel (Load History) was sent through a precision power amplifier to boost the current output sufficiently to drive 14 paralleled load channels that simulate the air load distribution on the wing box.

The output from thirty three channels went directly to the individual load controllers that regulated the various side, drag, and inertia loads to the four pylons which constitute the external store loads. Another program channel provided pulses for driving a counter system located in the test console, which indicated the status of the structure being loaded.

b. Load Controllers

The 50 channel load system (Figure 9) consists of 50 vacuum tube type, rack-mounted analog servo-controllers, each of which can function as an individual system, controlling tension or compression loads, according to a desired program, when incorporated in an electro-hydraulic closed loop.

A dc signal from a programmer is applied to one or more servo-controllers where it is scaled and directed to a hydraulic servo-valve. The resulting hydraulic flow from the servo-valve supplying a loading cylinder applies a load to a structure. A load cell transducer, in series with the loading cylinder, furnishes an electrical output proportional to the load. This voltage feedback to the controller is compared with the scaled program signal. Any difference above or below the program signal is considered an error voltage, which is amplified to the servo-valve for directing fluid in the proper direction to correct the load and reduce the error to zero. This operation is a continual process, with the controller providing instantaneous correction for each increment of load change.

c. Redundant Dump

A backup protection against overloads was designed for test safety and constructed for the F-111 Wing Fatigue Test Program to provide a simple means for reducing the possibility of damage to the wing due to failure of the primary control feedback line or electronics. Failures in this area, though infrequent, do occur and in some cases escape detection by the normal safety networks. If, for example, the load cell power supply should fail or an open circuit occur in the feedback line to the controller, the controller would increase the load demand and, as no feedback would exist, the normal overload detection networks would not function. In addition, if the program were of a small value, the indicated error would also be small and could prevent detection by the error protection network, leaving the structure unprotected.

A redundant safety capability was established by utilizing the second bridge of the multi-bridge load cell as a separate bridge, a separate bridge balance network, and a separate power supply to produce a backup system. This reduced the possibility of structural overloads due to the aforementioned electrical failures. The redundant dump system is shown in Figure 7.

The overload backup consists of individual chassis, each having ten channel capability. Simultaneous, individual adjustable tension and compression limits can be set on the front panel of each chassis for each of the ten channels accommodated in the chassis. These controls can be seen in Figure 8. Exceedance of any one of the limit values actuates immediate abort action for the entire 47 channel load system.

Extreme care was taken at the beginning of the test to establish proper error bands and internal filtering to prevent unintentional aborts due to noise pickup in the feedback lines, power-line spikes, and sluggishness in the hydraulics. An acceptable compromise was secured and the total system setup was put in operation in the F-111 wing tests.

d. Hydraulic System

The hydraulic course for the test program was a 50 GPM pump operating at 3000 psi. Because of the large number of servo-valves used and their normal internal oil leakage, a 24 GPM flow was needed to maintain a steady load and 48 GPM while cycling.

The system was designed with a low and high pressure setting. The low pressure system (800 psi) was used during startup to take out initial linkage slop and to guarantee control of all load channels before high pressure was applied.

e. Test Monitoring and Control

All load channels were continuously monitored by five Sanborn Model No. 350 channel direct writing oscillographs (Figure 10). This system was used primarily for control system trouble-shooting.

The test was conducted from a control console (Figure 10) wnich housed the manual dump, hold, start, and program attenuation controls, high and low hydraulic pressure controls, and various other equipment needed to monitor the loading and programming systems.

SECTION IV

INSPECTION TECHNIQUES

In the early stages of testing, x-ray techniques were attempted which included one serious attempt at Neutron Radiography. The results did not warrant further use of these techniques.

The techniques which proved to be the most feasible from the standpoint of accuracy, resolution, and detectability, proved to be magnetic rubber inspection, dye penetrant, and visual inspections. Magnetic rubber inspection was used for the fuel flow hole and pylon attach points.

Magnetic rubber is room temperature vulcanizing rubber with powdered iron in suspension. Casts are made by pouring this solution into molds around the area to be inspected and then applying a magnetic flux. This flux causes a flux field to be set up across the crack or flaw opening, which, in turn, attracts the suspended iron powder. This concentration of iron particles gives a permanent replica of the crack (Figure 13 and 14).

Dye penetrant was used for inspection of the pylon fittings and related hardware. The rest of the wing was inspected visually.

SECTION V

TEST CONDITIONS AND TEST SPECTRUM

Nine test conditions were run on the F-111A wing as shown in Table I. Positive symmetrical and asymmetrical maneuvers, and one taxiing, were accomplished at wing sweep angles of 26°, 50°, and 72.5°; six stores positions were used.

Additional loads were applied to the F-111A wing, but not as part of the test spectrum.

2.4G negative and 7.33G positive proof loads were applied before the fatigue cycling began and, again, after the 5th test life. These tests were designed to ensure that no flaw or crack (of critical crack size) was present and that at the end of the 5th test life no original flaw or cracks generated by the fatigue cycling had propagated to this critical size.

Constant amplitude cycles to failure were 65% of limit load and were applied at the end of the 10th fatigue test lifetime to correlate this test with the clean wing fatigue test conducted by General Dynamics at San Diego. (A complete breakdown of all pylon and wing loads is given in General Dynamics Test Loads Spectra Development Report FZS 12-269, dated 10 March 1970.)

The test spectrum was broken into blocks (Table II), each simulating 400 flight hours. This spectrum was used for the first six lifetimes. Each condition contained one or more levels of load with the higher numbered levels corresponding to higher load magnitudes. The loads were applied in a blocked spectrum manner. After six lifetimes were completed, the spectrum was modified (Table III) to speed up the test program. Each of the accelerated blocks represented 800 simulated flight hours.

TABLE I

TEST CONFIGURATION* AND CONDITIONS OF THE TEST SPECTRUM

SYMBOL DESIGNATION	MANEUVER	STORES CONFIGURATION	WING SWEEP ANGLE
W-1	Pos. sym.	24 M-117 stores on all 4 wing pylons	26°
W-2	Pos. asym.	Same as W-1	26°
W-3	Taxi, to represent ground operations	Same as W-1	26°
W-4	Pos. sym.	12 M-117 stores on the 2 pivoting pylons (2 inboard)	26°
W-7	Pos. sym.	12 M-117 stores on the 2 pivoting pylons (3 per pylon)	50°
W-8	Pos. asym.	Same as W-7	50°
W-9	Pos. sym.	4 MK-43 stores on the 2 pivoting pylons (1 per pylon)	72.5°
W-10	Pos. asym.	Same as W-7	72.5°
W-11	Pos. sym.	Without stores	50°

*F-111A with or without "stores" as indicated.

TABLE II

BLOCK SPECTRUM

Condition	Cycles/Level				
	1	2	3	4	5
W-1	11	8	2		
W-3	19				
W-4	30	6	1 △1		
W-2	14	5	3	2	1 △2
W-3			1 △1		
W-9	40	9	6	3	2
W-10	8	3	1 △2		
W-7	525	135	48	15	
W-8	27	10	15	5	
W-3		8			
W-11	215	274	45	8	1

Total Cycles - 1506

Legend:

△1 Applied in odd numbered blocks.

△2 Applied in even numbered blocks.

TABLE III

MODIFIED BLOCK SPECTRUM

Condition	Cycles/Level				
	1	2	3	4	5
W-1	22		6		
W-4	60		8		
W-2	28		16		2
W-3			2		
W-9	80		30		10
W-10			2		
W-7	525				
W-11	215	274	45	8	1
W-7	525	270		70	
W-8	52			40	
W-11	215	274	45	8	1

SECTION VI

INSTRUMENTATION

The wing was initially instrumented with 255 strain gage channels, 17 of which were on the inboard fixed pylon and 17 on the outboard fixed pylon. As the program progressed many of these strain gages became inoperable, so that by the end of the test program only 123 strain gage channels were operable. (Detailed information on strain gages and their locations may be found in the General Dynamics Report FZS-12-260, dated 10 April 1969.) Ten of these strain gages were designated as critical and were monitored continuously during the fatigue cycling.

Deflections were measured along the front and rear spar at ten locations. A total of 47 load transducers (load cells) were used; 14 on the wing bays and 33 on the pylons. A CDC 1604B computer in conjunction with two transmitter-multiplexer units and external signal conditioning modules and bridges was the primary data acquisition system. (Detailed specifications are available in WADD Technical Report 61-163.)

Data were obtained a total of 170 times. The data runs are enumerated in Tables IV and V as a function of test condition and block number.

Appendix B details the response of fatigue-experienced sensors that were attached to the wing structure.

SECTION VII

TEST RESULTS

The aft pylon attachment bolt of the outboard fixed pylon failed at 95 percent of the first fatigue lifetime (Figure 11). The fracture occurred in the threaded portion of the attachment bolt. A metallurgical evaluation was conducted by Mr. Alan Gunderson of the Air Force Materials Laboratory to determine the cause and mode of failure. Based on the predominant intergranular fracture, longtime to failure after loading, and the deep crevices between grains, it was concluded that stress corrosion cracking was the primary cause of failure.

The pylon attachment bolt's original heat treat of 260-280 ksi was changed to 200-220 ksi to reduce the susceptibility to stress corrosion cracking and the machining of the threads was changed to rolled threads. After these changes, a satisfactory fatigue life was obtained.

A crack in upper surface fuel flow hole No. 13 was discovered at the end of the first fatigue life. The crack was in the compression surface of the wing (top) and was caused by the large compressive stress (above yield) creating a high tension residual stress in the fuel flow hole. The theory of this type of crack propagation is explained in "The Load Interaction and Sequence on the Fatigue Behavior of Notched Coupons," ASTM STP519 pp 109-132, by J.M. Potter. This crack was polished out after the first life and again after the second life. The crack history is plotted in Figure 12. This plot was generated from magnetic rubber casts taken throughout the program. Typical casts are presented in Figure 13 and 14.

A crack was discovered in upper surface fuel flow hole No. 14 after the second lifetime. The crack was polished out at the end of the second lifetime. A crack was discovered in upper surface fuel flow hole No. 11 after the 6th fatigue lifetime; it propagated as shown in Figure 12. A crack was again found in fuel flow hole No. 14 after the 10th fatigue lifetime; it propagated as shown in Figure 12.

During the constant amplitude testing, after the 10th lifetime, major cracking occurred in the lower skin transition section. The cracks were discovered after cycle 7308. The transition section is where the D6AC steel pivot place is joined to the outer aluminum skin. A crack extended from bolt hole No. 233 to bolt hole No. 215 and another started at bolt hole No. 238 and extended to the chordwise stiffener as indicated in Figure 15. This concluded the fatigue testing.

SECTION VIII

CONCLUSIONS

1. The test demonstrated that the F-111A full stores configuration wing has a satisfactory fatigue life for the applied test spectrum, with the exception of the outboard fixed pylon attachment bolt. Changes to the attachment bolt heat treatment and thread fabrication, as previously described, provide for a satisfactory fatigue life of this part.

2. Cracks in the upper surface fuel flow holes were found not to be critical, and satisfactory fatigue life can be demonstrated even without corrective action.

3. The "full stores" configuration fatigue wing has essentially the same fatigue life as the clean configuration wing.

TABLE IV

SERIAL NUMBER* OF STRAIN AND DEFLECTION DATA
TAKEN DURING FATIGUE BLOCK ONE

CONDITION	LEVEL L-1	L-2	L-3	L-4	L-5	PROOF LOAD
-2.4 G Proof						801
+7.3 G Proof						802
W-7	803			804		
W-1	805	805	802			
W-3	807					
W-4	808	809	809			
W-2	812	811	810	813		
W-3						
W-9	815	816	817	818	819	
W-10	820	821				
W-7		822	823	824		
W-9	825	826	827	828		
W-3		827				
W-11	830	830	831	831	831	

*Detailed strain and deflection data is stored at AFFDL on magnetic tape under the above serial number. It may be reconstructed upon request.

TABLE V

STRAIN AND DEFLECTION DATA BY SERIAL NUMBER, TAKEN DURING REMAINDER OF TEST

SPECTRUM BLOCK	CONDITION/LEVEL								
	W-1 L-3	W-4 L-3	W-2 L-5	W-3 L-3	W-9 L-5	W-10 L-3	W-7 L-4	W-8 L-4	W-11 L-5
2	832	--	833	--	--	834	--	835	836
3	837	838	--	839	840	--	841	842	843
4	--	--	844	--	--	845	--	--	--
5	846	847	--	848	849	--	850	851	852
6	--	--	--	--	--	--	--	--	--
7	--	853	--	854	--	--	--	--	--
8	855	--	856	--	857	858	859	860	861
9	--	862	--	863	--	--	--	--	--
10	864	--	865	--	866	867	868	869	870
11	871	--	--	872	873	--	874	875	876
12	--	--	877	--	--	--	--	--	--
13	--	--	--	--	--	--	--	--	--
14	--	--	878	--	--	--	--	--	--
15	879	880	--	881	882	--	883	884	885
19	--	886	--	887	--	--	--	--	--
20	889	--	890	--	891	892	893	894	895
21	896	--	--	--	--	--	--	--	--
24	--	--	897	--	--	--	--	--	--
25	898	899	--	900	901	--	902	903	904
26	905	--	--	--	--	--	--	--	--
29	--	906	--	--	--	--	--	--	--
30	907	--	908	--	909	910	911	912	913
31	914	--	--	--	--	--	--	--	--
34	--	--	915	--	--	916	--	--	--
35	--	917	--	918	919	--	--	--	--
36	922	--	--	--	--	--	--	--	--
39	--	923	--	924	--	--	--	--	--
40	925	--	926	--	937	978	979	930	931

TABLE V (CONT)

SPECTRUM BLOCK	CONDITION/LEVEL W-1 L-3	W-4 L-3	W-2 L-5	W-3 L-3	W-9 L-5	W-10 L-3	W-7 L-4	W-8 L-4	W-11 L-5
41	932	--	--	--	--	--	--	--	--
45	--	--	--	--	--	--	--	--	933
49	--	--	--	934	--	--	--	--	--
50	925	--	936	--	937	--	938	--	939
	-24 G Proof	--	940	+7.3 G Proof		--	941	--	--
51	942	--	--	--	--	--	--	--	--
54	--	--	--	--	--	943	--	--	--
55	944	--	--	945	946	--	947	--	948
60	949	--	950	909	951	952	953	--	955
61	956	--	--	--	--	--	--	--	--
65	--	--	957	--	--	--	958	--	959
70	960	--	--	--	--	--	--	--	--
75	--	--	961	--	--	--	963	--	962
85	--	--	964	--	--	--	966	--	965
95	--	--	967	--	--	--	969	--	968

TABLE VI

TYPE AND RESPONSE OF FATIGUE SENSORS INSTALLED ON THE F-111A WING DURING THE FATIGUE TEST

FM[5] NO.	M.F.[6]	WING STATION	INSTALLATION (SAFE-LIFE)	END OF USABLE LIFE- (SAFE-LIFE)	TYPE
24	1.0	214.7	1.0	No Response	204-DA-STE
25	1.0	"	1.0	"	"
26	1.0	"	1.0	"	"
16	2.0	212.7	0.5	1.0	SAP[1]
19	2.0	242.7	0.5	4.0	SAP
28	2.0	214.7	1.0	--	EXP'MTL[4]
29	2.0	214.7	1.0	--	"
15	2.5	212.7	0.5	1.0	SAP
27	2.5	214.7	1.0	1.7	SAP
32	2.5	242.7	4.0	--	FM[2]
17	3.0	212.7	0.5	1.0	SAP
20	3.0	242.7	0.5	4.0	SAP
23	3.0	245.2	1.0	4.0	SAP
33	3.0	242.7	4.0	7.0	FM
34	3.0	242.7	4.0	10. + C.A.[3]	FM
36	3.0	245.2	6.0	10. + C.A.	SAP
37	3.0	245.2	6.0	10. + C.A.	SAP
31-A	3.0	242.7	8.0	10. + C.A.	FM
33-A	3.0	242.7	8.0	10. + C.A.	FM
22	3.5	245.2	1.0	4.0	SAP
18	4.0	242.7	0.5	2.7	SAP
21	4.0	245.2	1.0	4.0	SAP
30	4.0	242.7	2.5	5.3	SAP
31	4.0	242.7	2.5	7.0	FM
18-A	4.0	242.7	6.0	10. + C.A.	SAP

TABLE VI (CONT.)

FM[5] NO.	M.F.[6]	WING STATION	INSTALLATION (SAFE-LIFE)	END OF USABLE LIFE- (SAFE-LIFE)	TYPE
21-A	4.0	245.2	6.0	8.8	SAP
30-A	4.0	242.7	6.0	10. + C.A.	SAP
35	4.0	245.2	6.0	8.4	SAP

NOTES:
1. SAP 204-DA-STE, Dentronics, Inc.
2. FM-S/N Fatigue-life gage, micro measure.
3. C.A. = constant amplitude.
4. Experimental plastic multiplier, 204-DA sensor.
5. FM = fatigue monitor.
6. M.F. = mechanical strain multiplication factor.

APPENDIX A

F-111A WING FATIGUE TEST HISTORY

24 March 1971	- Proof tests of 2.4G negative and 7.33 positive.
13 April 1971	- Strain survey and start of Block No. 1.
14 June 1971	- Failed after mounting bolt in outboard fixed pylon.
23 July 1971	- Completed 1st fatigue life. Crack found in fuel flow hole No. 13 was polished out (approx. 0.125 inches in depth).
3 August 1971	- Installed boron patch on lower plate of pivot fitting.
13 August 1971	- Rework of all fuel flow holes completed.
18 October 1971	- Started 2nd life.
28 November 1971	- Block 15 Crack in fuel flow hole No. 13.
11 January 1972	- Completed 2nd fatigue life. Crack in fuel flow hole No. 13 now 3/16 inch in depth. Discovered crack in hole No. 14.
8 February 1972	- General Dynamics personnel polished out cracks. Pivot bushings were also replaced.
3 March 1972	- Started 3rd fatigue life.
28 April 1972	- Completed 3rd life. Crack reinitiated in fuel flow No. 13.
27 July 1972	- Completed 4th fatigue life. Crack in fuel flow hole No. 13 was approx. 15/32 inch surface length and 5/32 inch in depth.
2 August 1972	- Started 5th fatigue test life.
9 September 1972	- Requested by F-111 SPO to extend test to 10 lifetimes to be consistent with recommendations presented in the 30 April 1971 F-111 Structural Audit Report.
6 October 1972	- Completed 5th lifetime.
12 October 1972	- Negative 2.4G proof test.
13 October 1972	- 7.33G positive proof test. Hole No. 13 crack has propagated to 1/4 inch in depth, and 21/32 inch in surface length.
27 February 1973	- Replaced wing in test fixture after 5 month delay due to TACT wing test and calibration.
12 April 1973	- Started 6th lifetime.

29 May 1973	-	Completed 6th lifetime. Discovered crack in fuel flow hole No. 11. Crack in fuel flow hole No. 13 had propagated very little (Figure 12).
June 1973	-	Modified spectrum to speed-up test program for 7 through 10 fatigue test lifetimes.
5 July 1973	-	Started 7th life.
27 July 1973	-	Completed 7th lifetime. Crack in fuel flow hole No. 13 has propagated to a depth of 7/16 inch; no change in No. 11.
17 August 1973	-	Started 8th lifetime.
13 September 1973	-	Completed 8th lifetime. Crack in fuel flow hole No. 11 has propagated to approx. 1/32 inch in depth and No. 13 has not changed significantly.
21 September 1973	-	Started 9th lifetime.
25 October 1973	-	Completed 9th lifetime. Crack in fuel flow hole No. 13 has not propagated to any significant degree. Crack in fuel flow hole No. 11 has propagated to a depth of 5/64 inch and 3/16 inch in surface length.
30 October 1973	-	Started 10th lifetime.
26 November 1973	-	Completed 10th lifetime. Crack in fuel flow No. 13 is now 13/32 inch in depth. A crack was found in fuel flow hole No. 14 and is approx. 1/16 inch in depth.
10 June 1974	-	Started constant amplitude cycles to failure.
9 July 1974	-	Loud noise occurred @ 6890 cycles. - Continued cycling.
10 July 1974	-	Very loud noise occurred. Testing was stopped. Total constant amplitude cycles: 7308.

The wing was shipped to Sacramento ALC for a complete teardown and inspection similar to the inspection completed on the clean configuration wing tested by General Dynamics of San Diego.

APPENDIX B

FATIGUE EXPERIENCE SENSORS

Both regular and mechanical strain-multiplied types of fatigue experience sensors were attached to the lower wing skin near fatigue critical areas during various portions of the ten safe-life fatigue tests (Table VI). The sensors were monitored at the completion of each 10% of each safe-life simulation until sensor failure. Sensors responded to the load experiences with a relatively constant (linear) permanent resistance change after an initial early rapid change (Figure 16). The changes were in a direct relation to the strain spectra of the attachment areas and the devised sensitivities (M.F.) of each sensor. The sensors, in general, performed with predictable responses after the steady state response was established until the onset of fatigue cracking or failure of the sensor. At that time the resistance change increased at a relatively rapid rate until continuity ceased (open circuit) or the sensor was replaced.

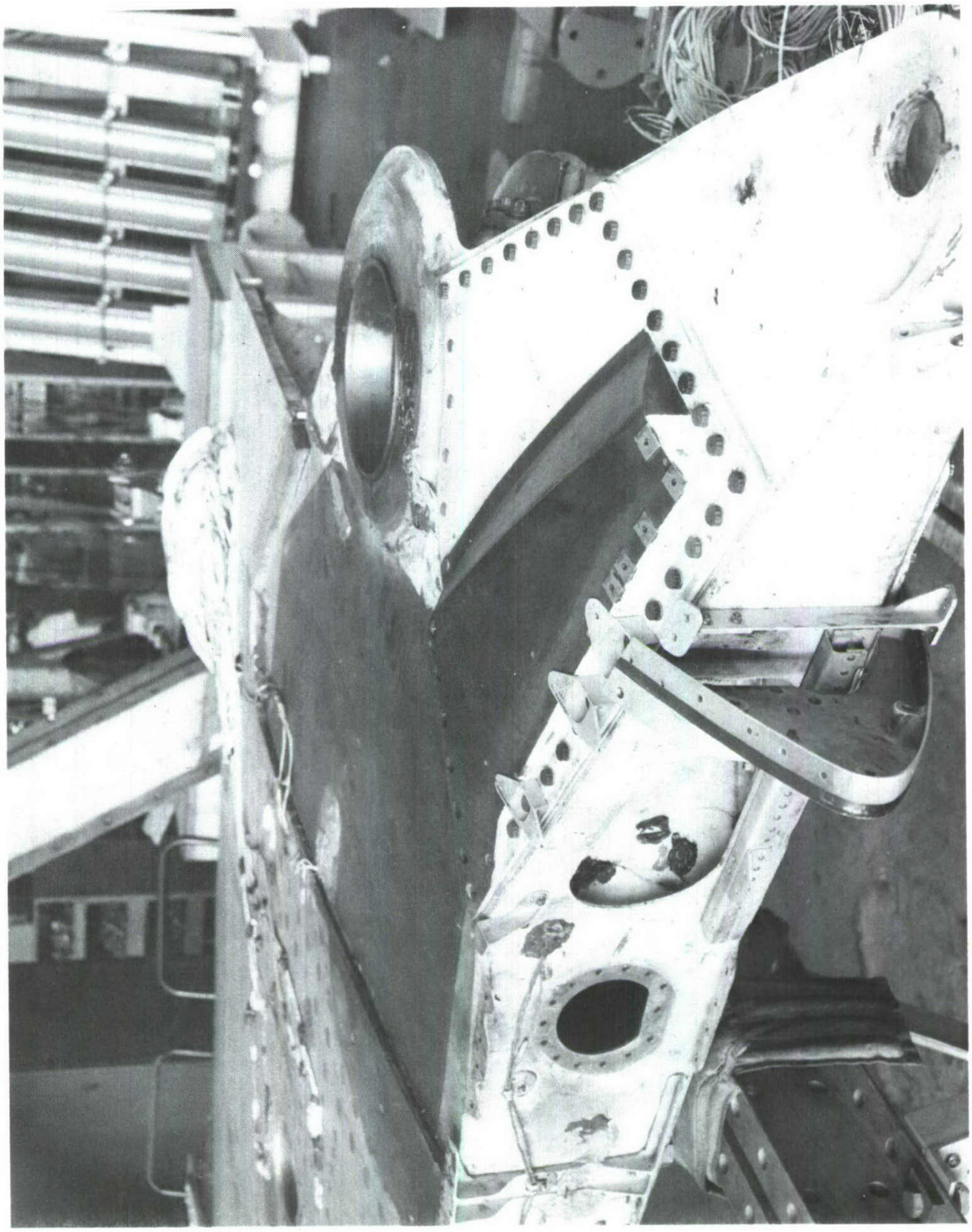


Figure 1. Boron Composite Doubler

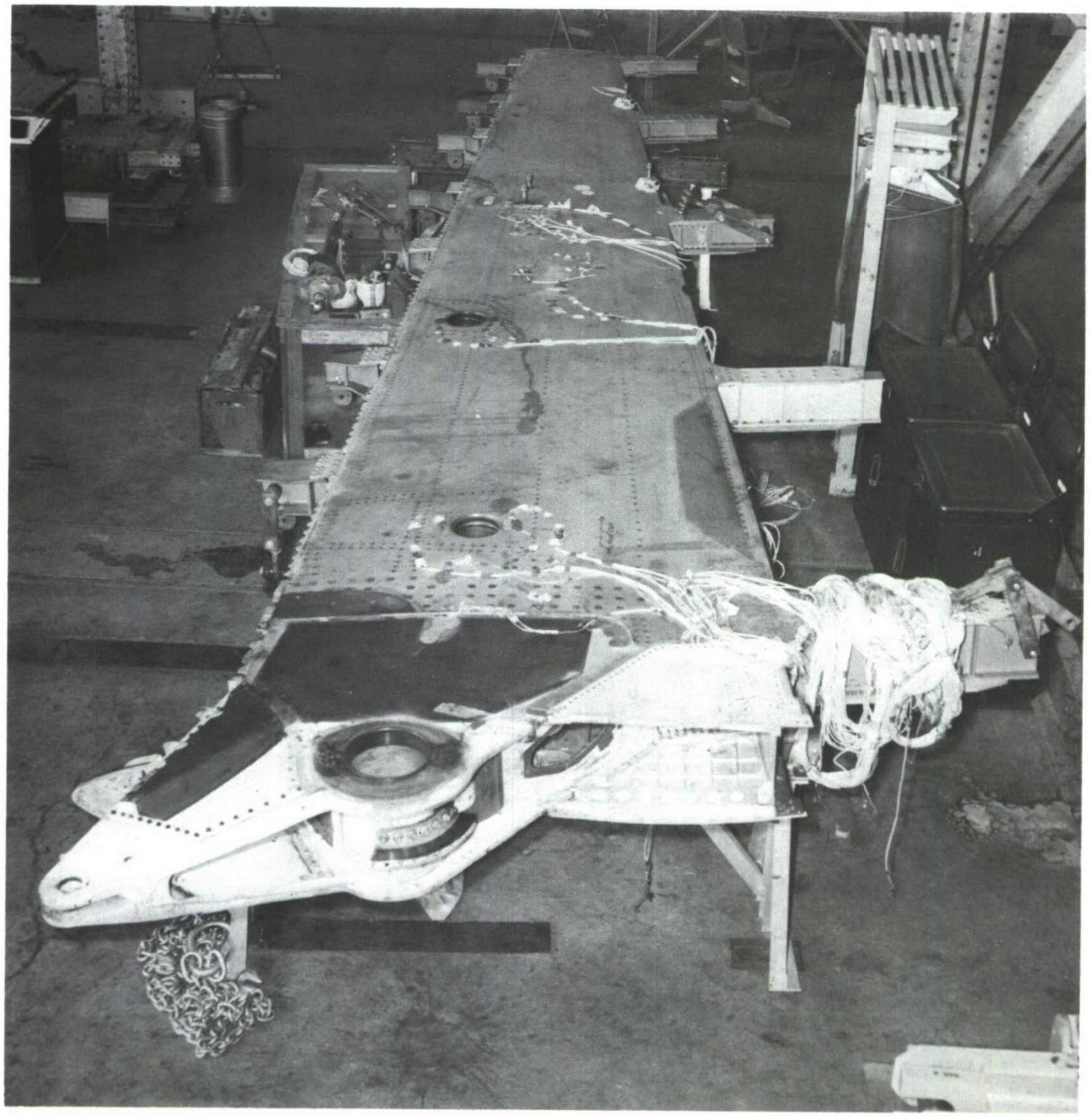


Figure 2. Test Specimen

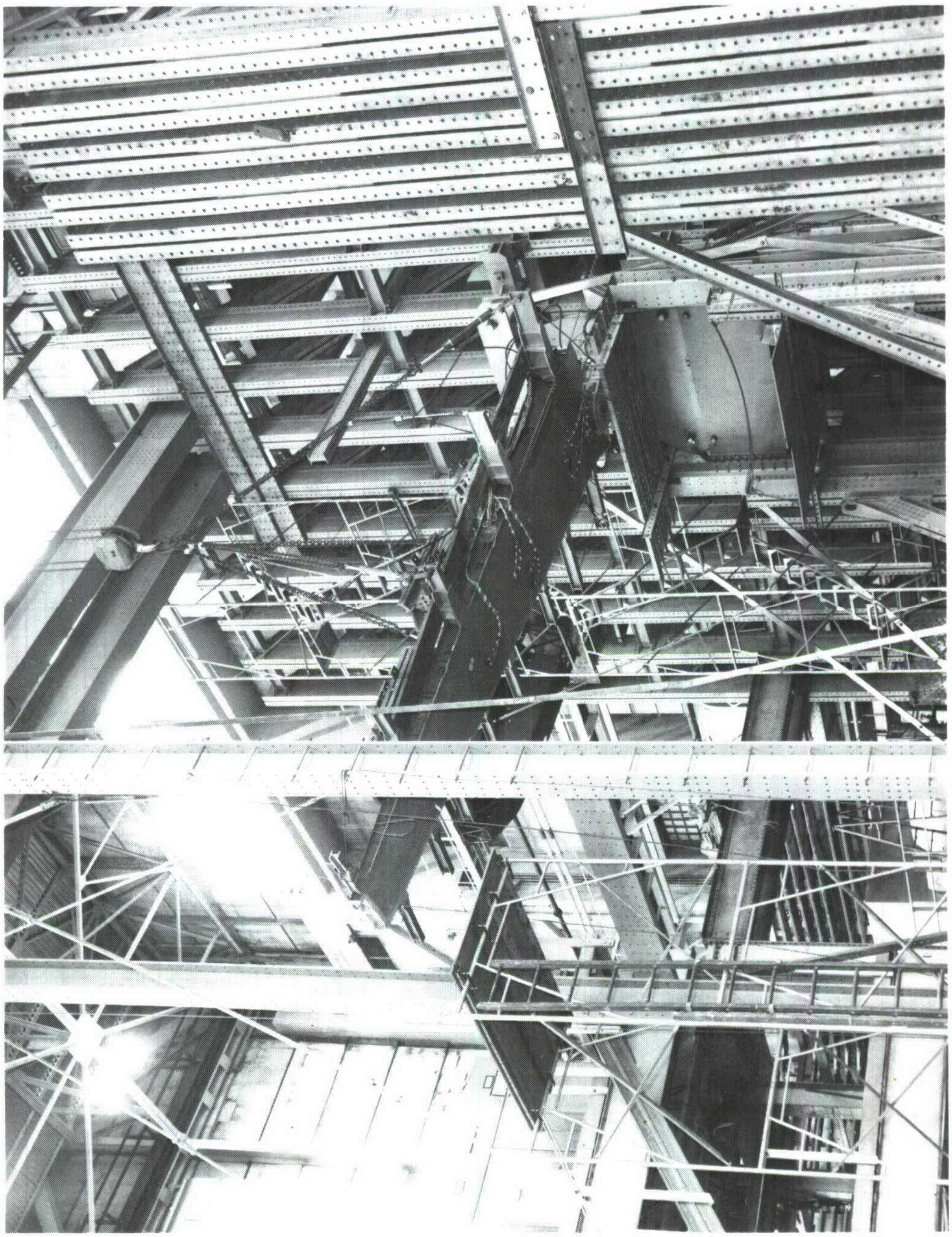


Figure 3. Wing Attached to Dummy Carry Through

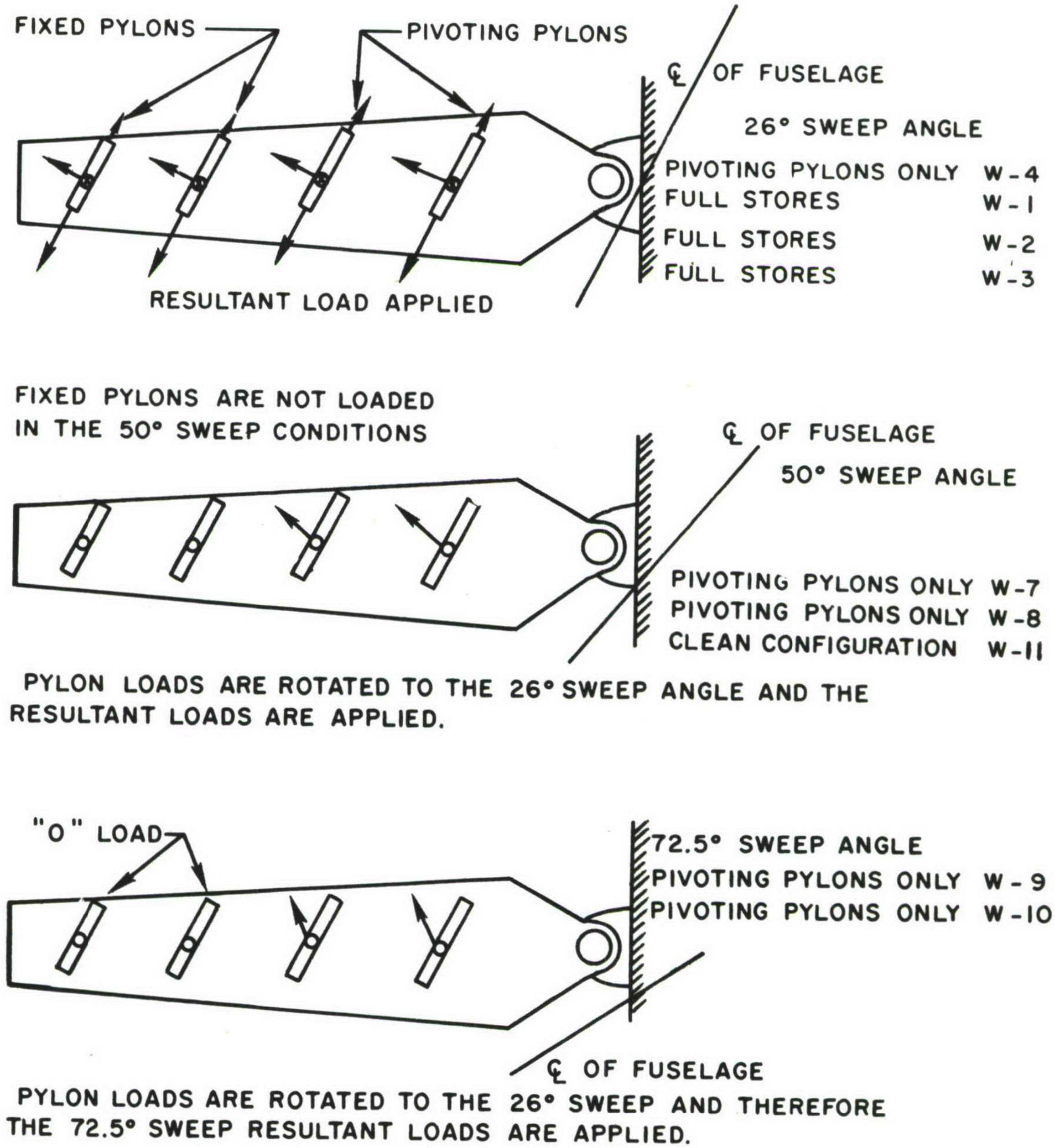


Figure 4. Simulation of Pylon Loads Due to Sweep Angle Change

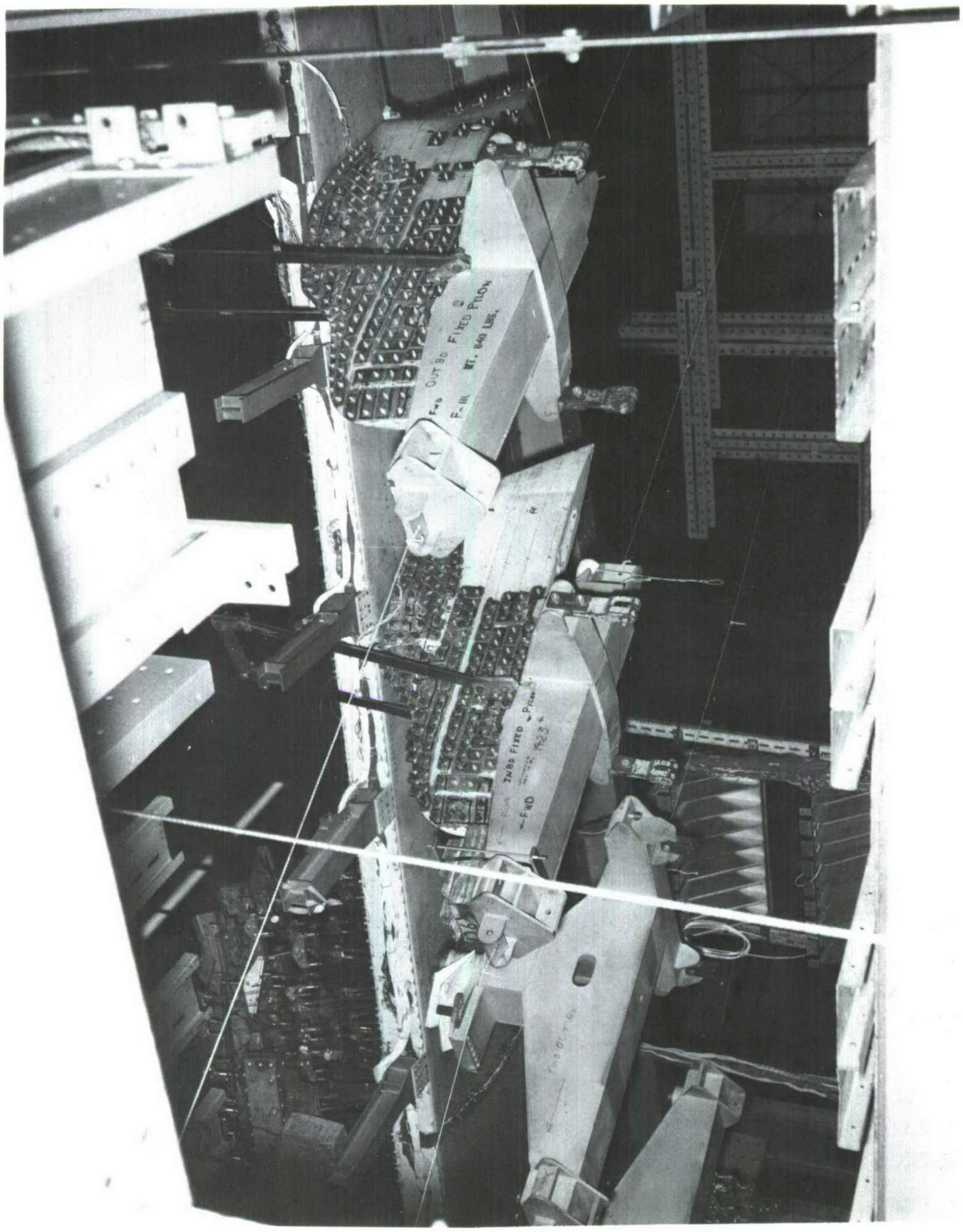


Figure 5. Dummy Pylons

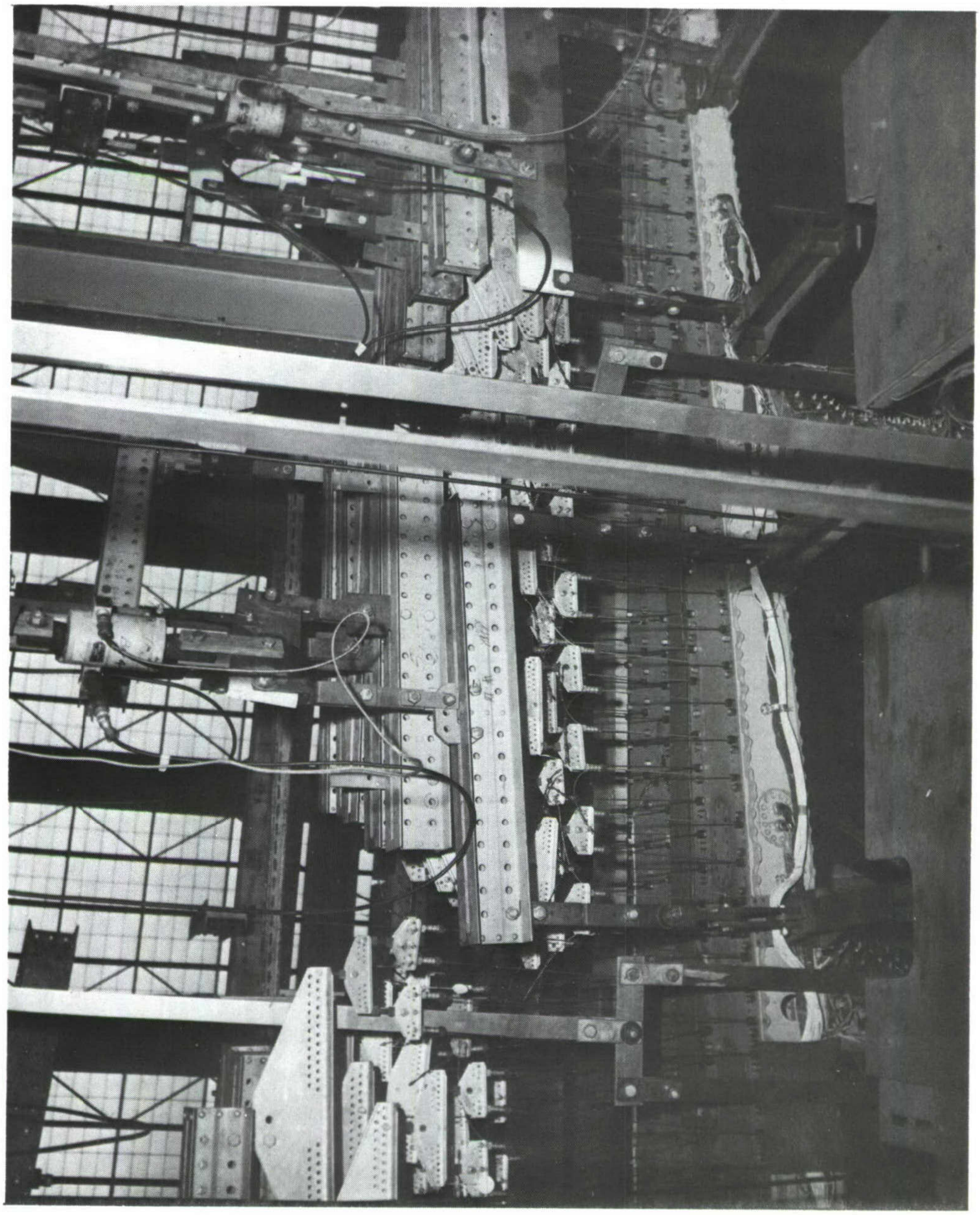


Figure 6. Application of Wing Air Loads

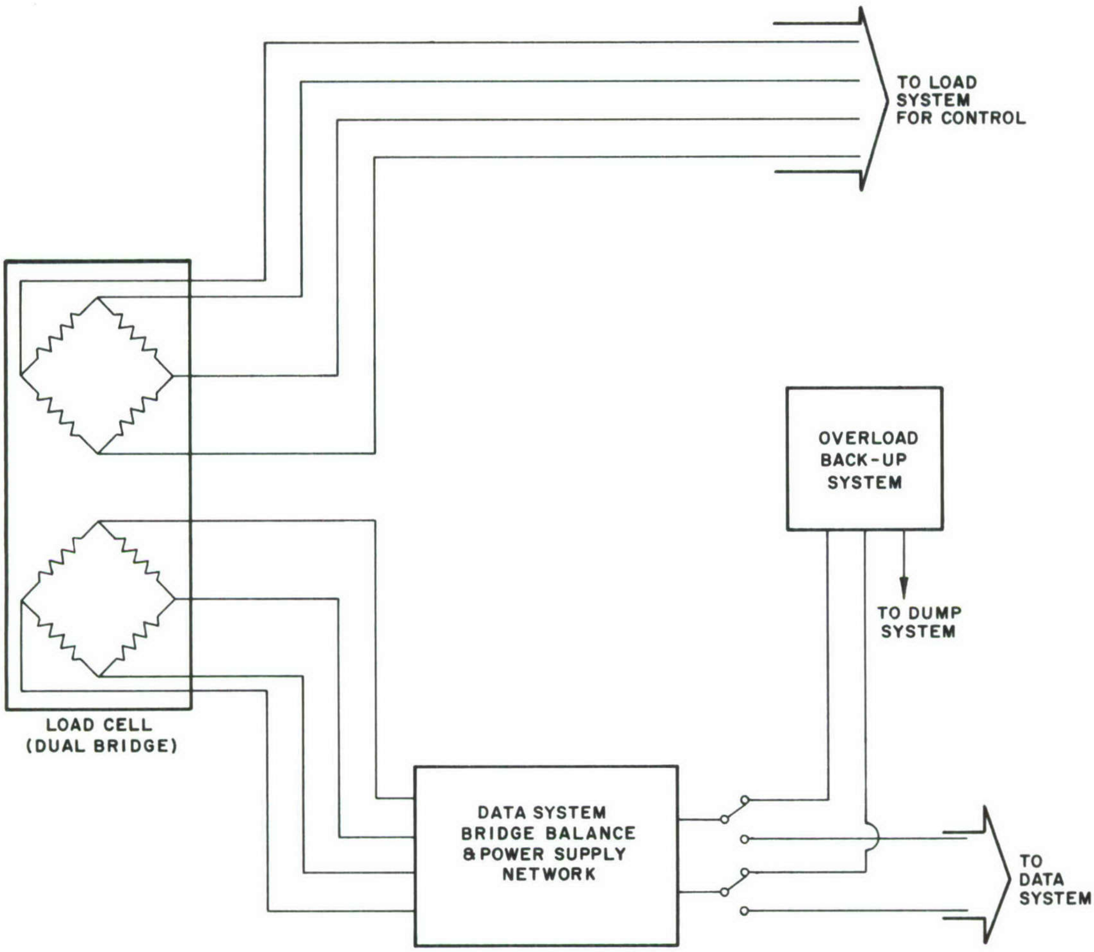


Figure 7. Redundant Dump Schematic

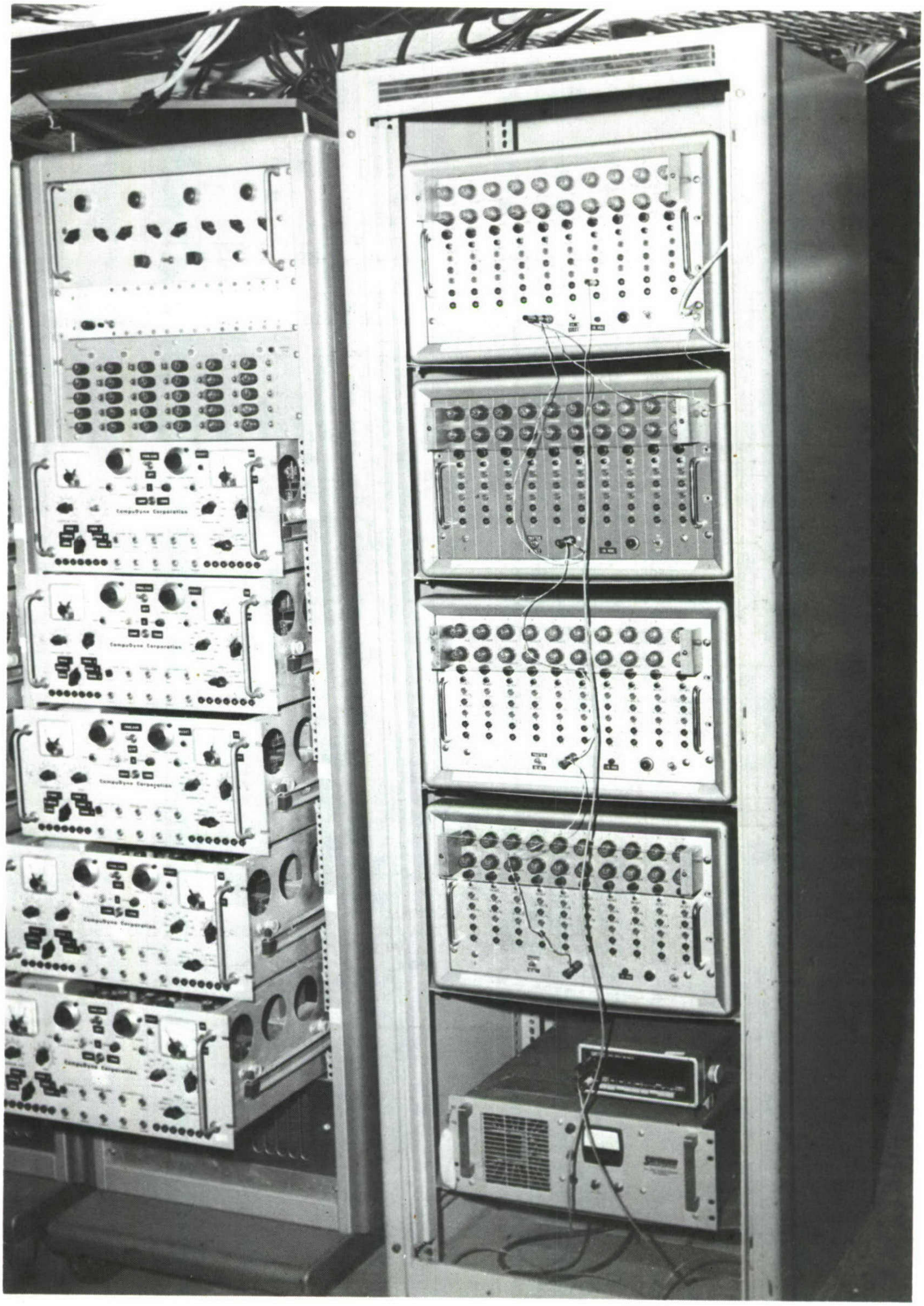


Figure 8. Redundant Dump

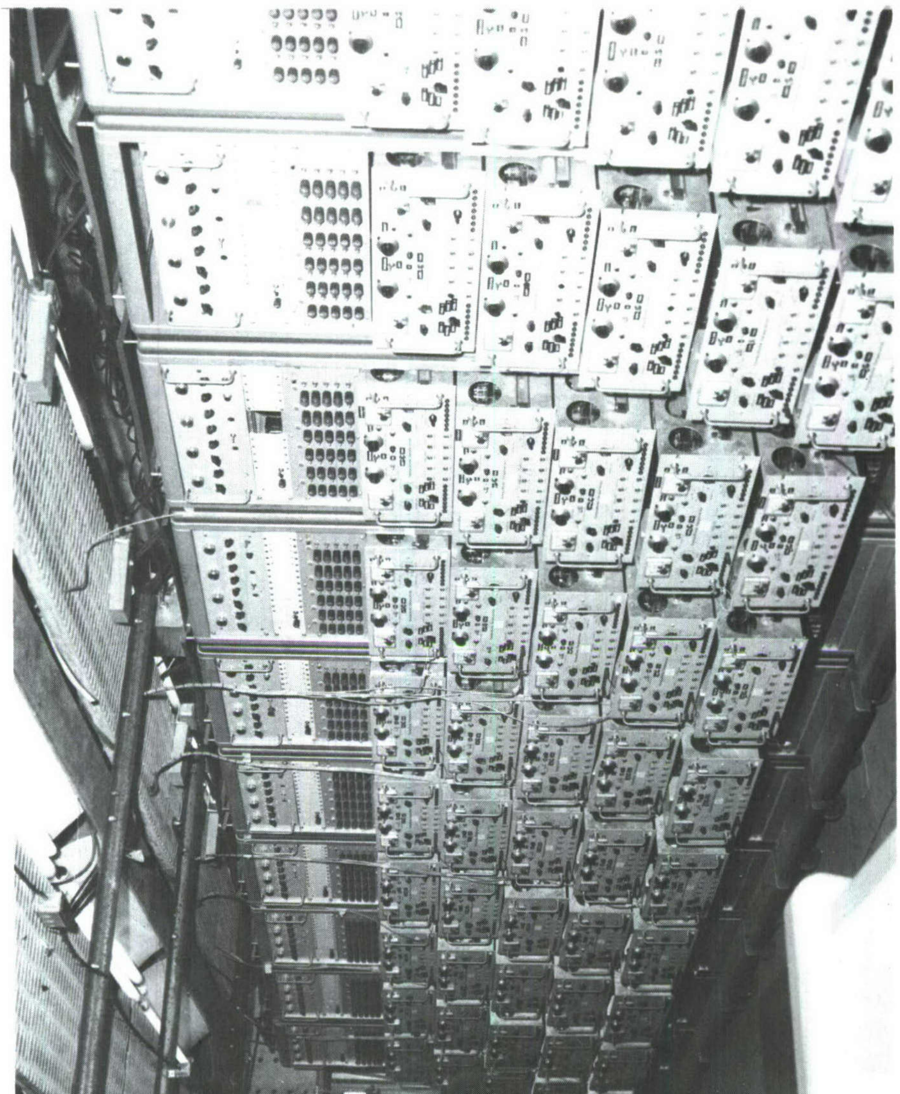


Figure 9. 50 Channel Loading System

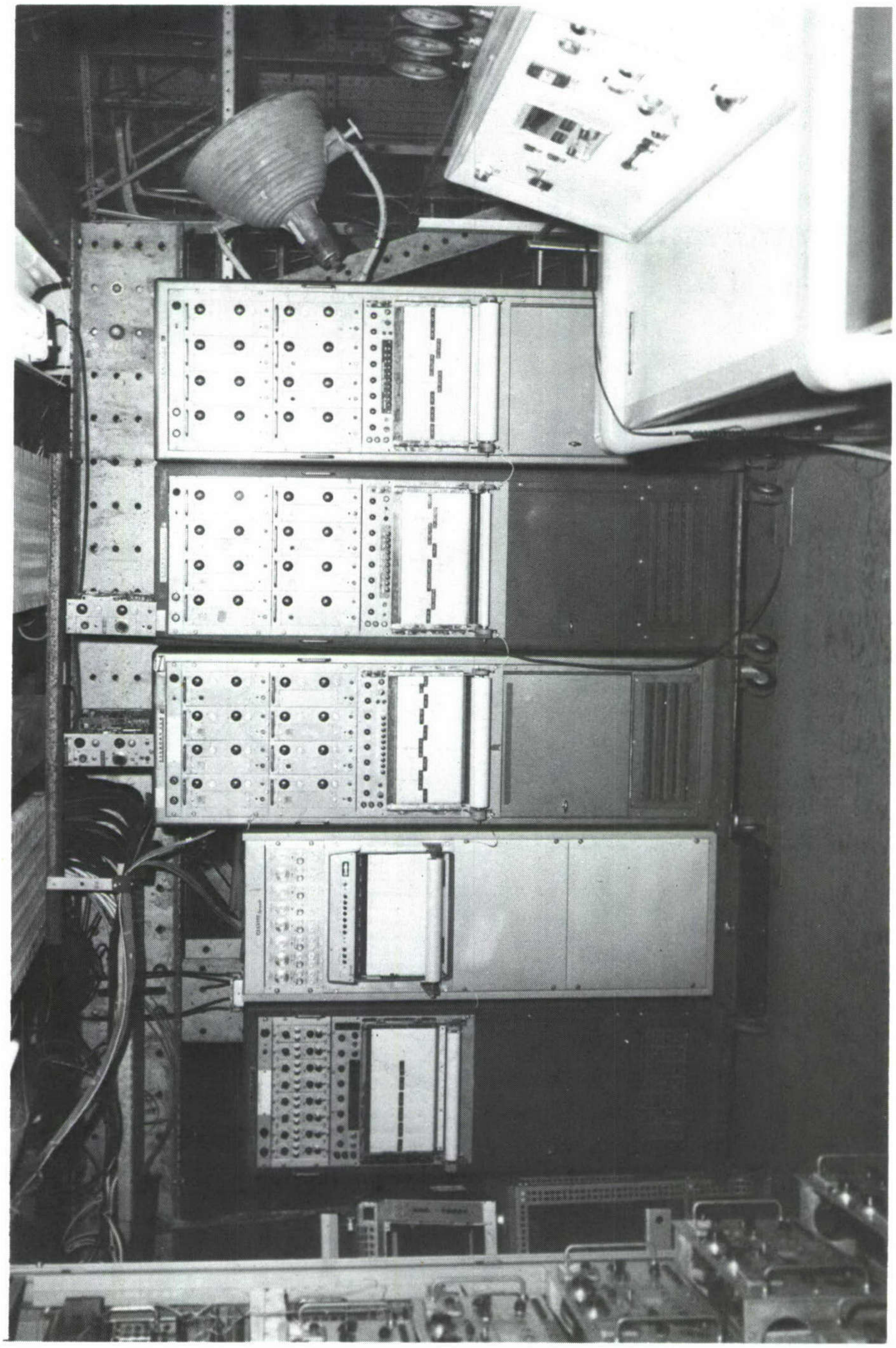


Figure 10. Sanborn and Control Console

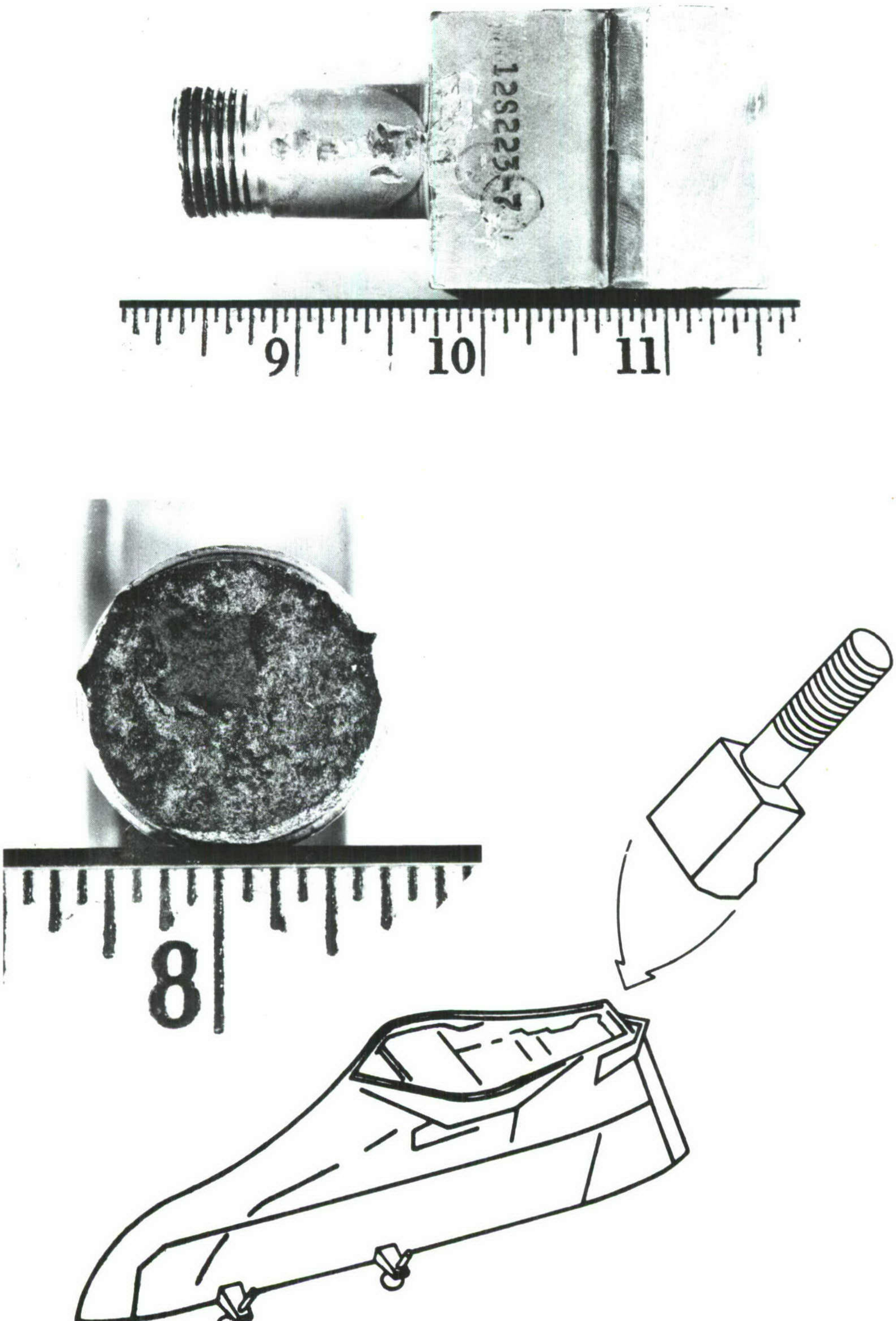


Figure 11. Aft Pylon Attach Bolt Failure

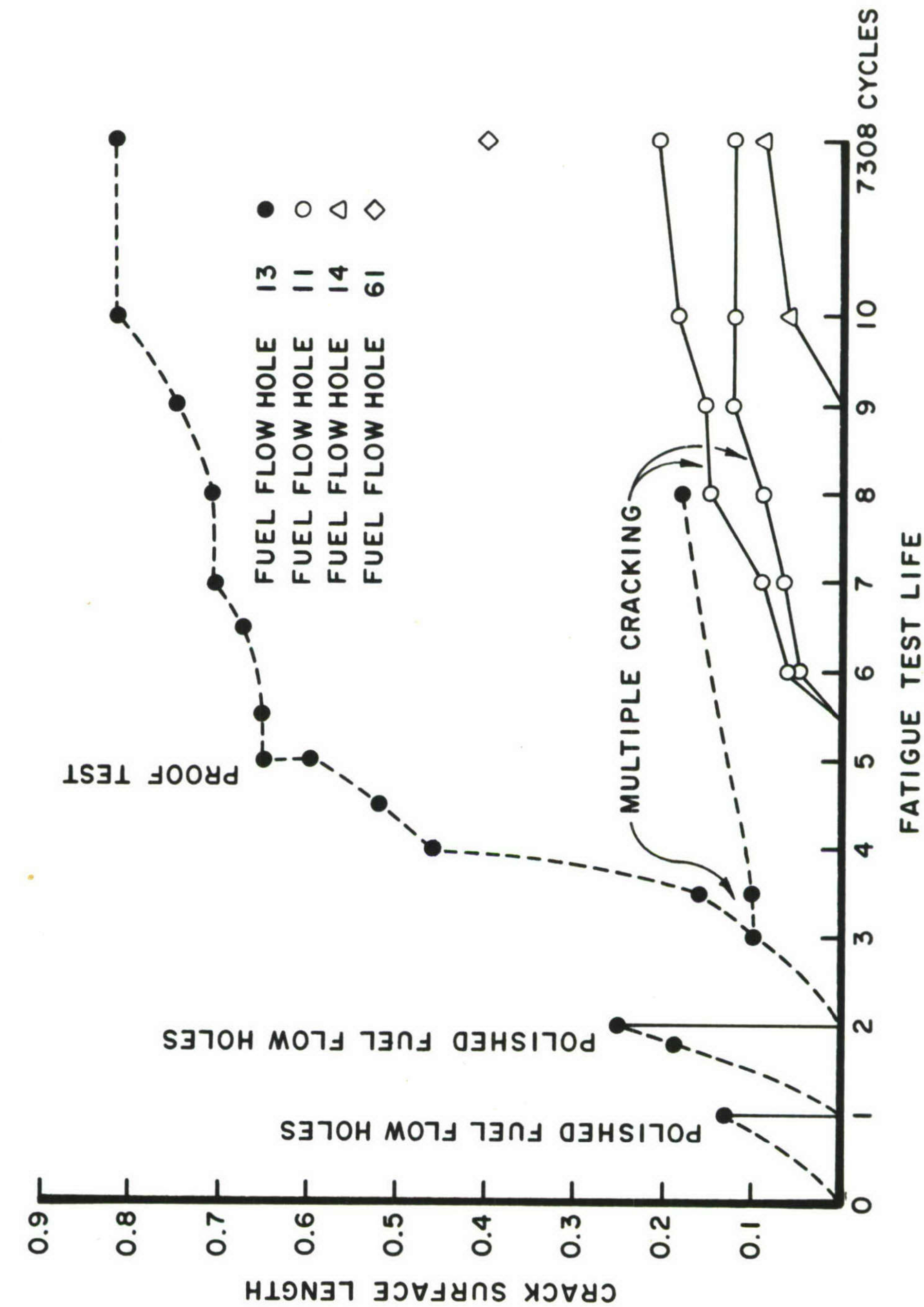
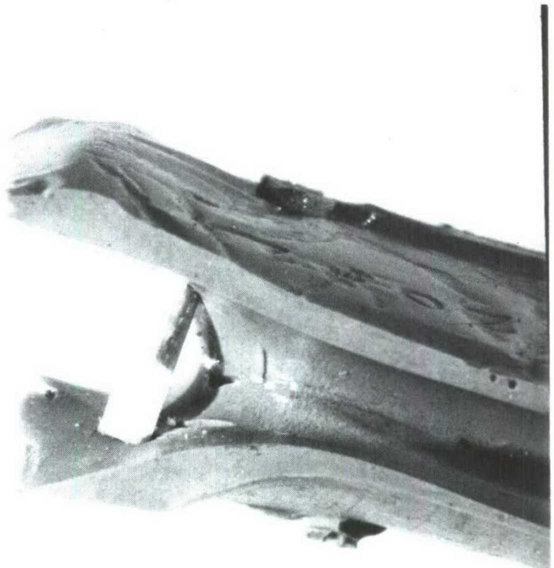


Figure 12. Fuel Flow Hole Crack Length vs. Fatigue Life



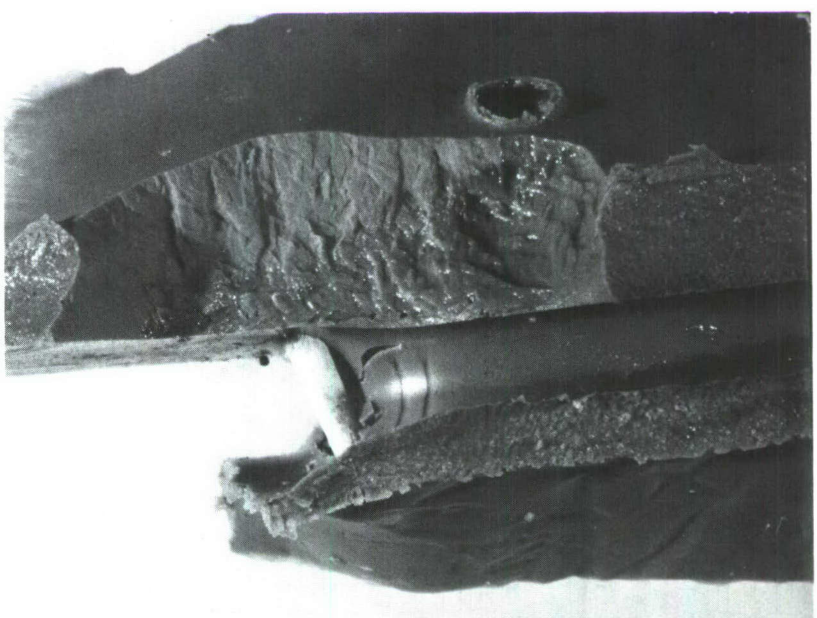
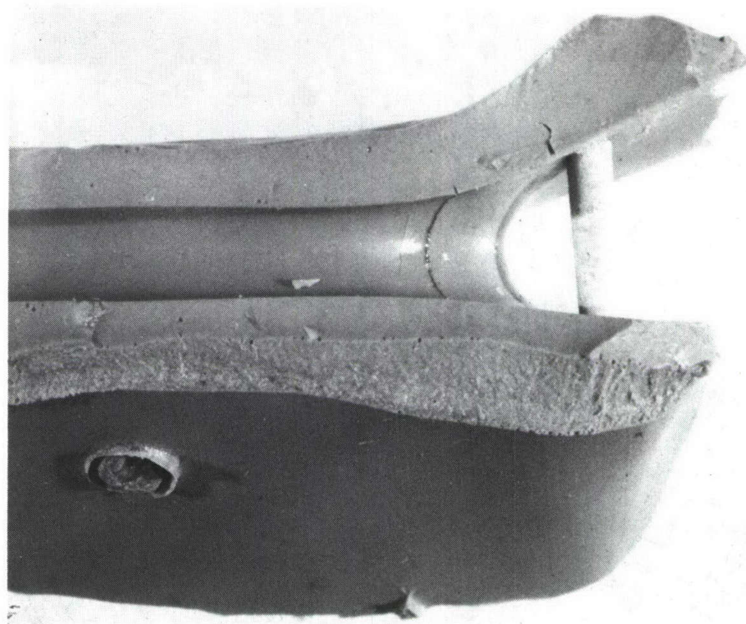


Figure 13. Magnetic Rubber Casts



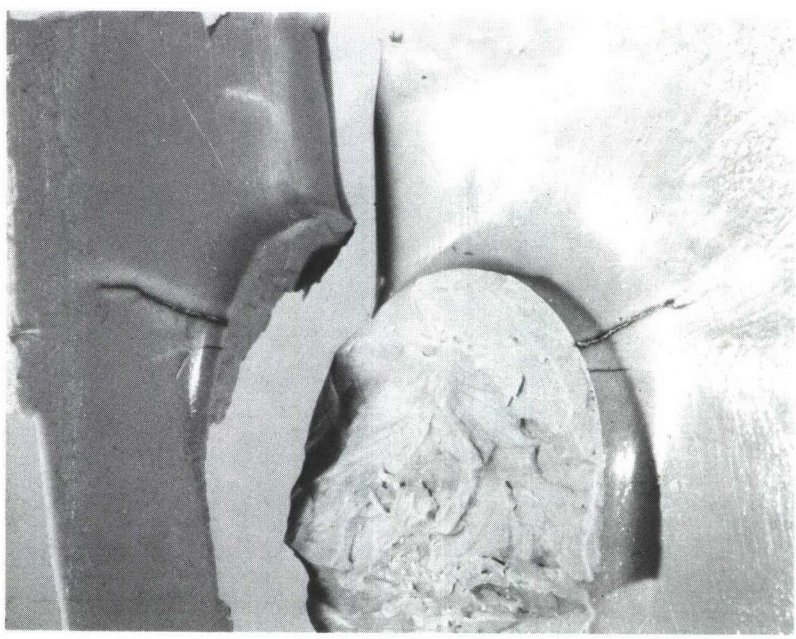


Figure 14. Magnetic Rubber Casts

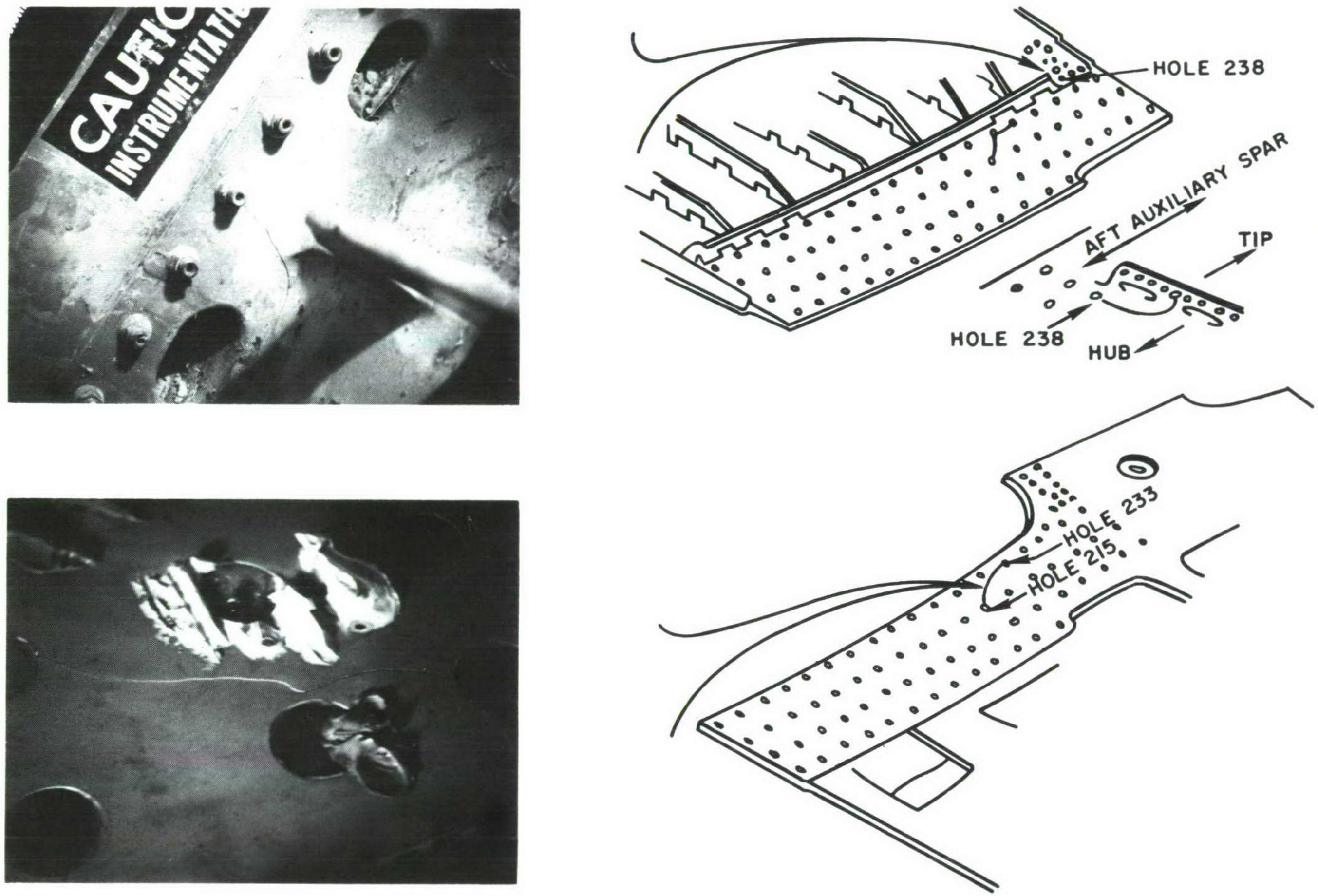


Figure 15. Wing Failure

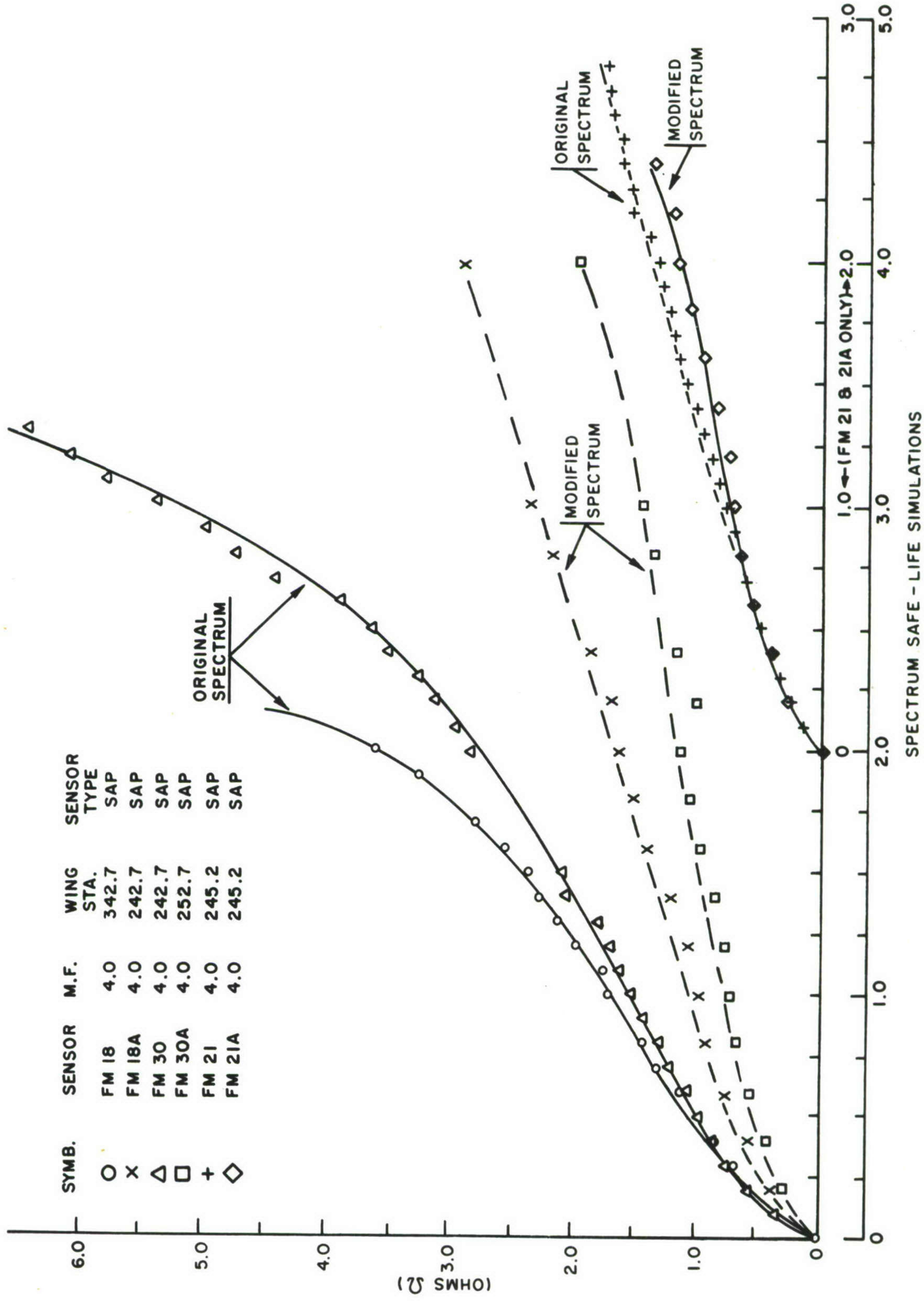


Figure 16. Fatigue Sensor Response As a Function of Life

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